NC STATE UNIVERSITY

Tacho Lycos

2024 NASA Student Launch

Preliminary Design Review

High-Powered Rocketry Club at NC State University
1840 Entrepreneur Drive
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Common Abbreviations and Nomenclature

AGL = Above Ground Level
AIAA = American Institute of Aeronautics and Astronautics
APCP = Ammonium Perchlorate Composite Propellant
ASME = American Society of Mechanical Engineers
AV = Avionics
BEMT = Blade Element Momentum Theory
BP = Black Powder
CDR = Critical Design Review
CG = Center of Gravity
CP = Center of Pressure
ECD = Electronics, Communication, & Data
EIT = Electronics and Information Technology
FAA = Federal Aviation Administration
FEA = Finite Element Analysis
FMEA = Failure Modes and Effects Analysis
FN = Foreign National
FRR = Flight Readiness Review
HEO = Human Exploration and Operations
HPR = High-Power Rocketry
HPRC = High-Powered Rocketry Club
L3CC = Level 3 Certification Committee (NAR)
LCO = Launch Control Officer
LRR = Launch Readiness Review
MAE = Mechanical & Aerospace Engineering
MSDS = Material Safety Data Sheets
MSFC = Marshall Space Flight Center
NAR = National Association of Rocketry
NCSU = North Carolina State University
NFPA = National Fire Protection Association
PDR = Preliminary Design Review
PLAR = Post-Launch Assessment Review
PPE = Personal Protective Equipment
RF = Radio Frequency
RFP = Request for Proposal
RSO = Range Safety Officer
SAIL = STEMnauts Atmosphere Independent Lander
SL = Student Launch
SLS = Space Launch System
SME = Subject Matter Expert
SOW = Statement of Work
STEM = Science, Technology, Engineering, and Mathematics
TAP = Technical Advisory Panel (TRA)
TRA = Tripoli Rocketry Association
VTOL = Vertical Take-Off and Landing

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1.1 Team Summary

1.1.1 Team Name and Mailing Address

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1.1.2 Mentor Information

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1.1.3 Social Media Accounts

X	FaceBook	Instagram	TikTok	YouTube	LinkedIn	Website
@ncsurocketry	/TachoLycos/	@ncsurocketry	@ncsurocketry	ncsurocketry	/tacholycos/	ncsurocketry.org

1.1.4 Time Spent on PDR Milestone

The team has spent approximately 480 hours on the PDR milestone.

1.2 Launch Vehicle Summary

1.2.1 Official Target Altitude

The official target apogee of the launch vehicle is 4050 ft. AGL.

1.2.2 Motor Selection

The current leading motor choice is the Aerotech L1520T.

1.2.3 Vehicle Size and Mass

The launch vehicle leading design is 105 in. in length and the estimated weight is 42.6 lbs.

1.2.4 Recovery System

The leading recovery system design incorporates an RRC3 altimeter, a dual-deploy altimeter and tracking device known as the Quasar, and a Big Red Bee 900 for the nose cone tracker. The drogue parachute is a Fruity Chutes 18” Classic Elliptical, the main parachute is a Fruity Chutes 84” Iris Ultra Compact, and the nose cone/payload parachute is a Fruity Chutes 48” Classic Elliptical. The main parachute will be deployed at 800 ft, so the nose cone and payload can completely separate from the launch vehicle and have time to stabilize before payload deployment.

1.3 Payload Summary

The SAIL’s descent will be controlled using two co-axially mounted contra-rotating rotors. The SAIL will contain data logging devices to prove human survivability and land in a vertical orientation. The method for deploying the SAIL involves the use of a radio command to open a custom-designed latch. A transmission will be sent from the ground, once the launch field RSO gives permission, to an RF receiver in the nose cone of the launch vehicle. This receiver will be connected to an Arduino which will power a servo to open the latch and release the SAIL into free fall.

2 Changes Made Since Proposal

2.1 Changes Made to Launch Vehicle

Table 2.1: Changes Made to Launch Vehicle

Change Description	Justification	Affected Subsystem(s)
Leading recovery design was changed from double AV bay method to deployment bag method which changed the leading vehicle design dimensions, payload ejection method, and separation points.	The deployment bag method is preferred for its simplicity (see Section 3.6).	Recovery, Structures, Aerodynamics, Payload

2.2 Changes Made to Payload

Table 2.2: Changes Made to Payload

Change Description	Justification	Affected Subsystem(s)
Priority shifted from auto-rotation to developing a motorized rotor design.	Auto-rotation proved to be a difficult concept to justify with FEA simulations. Powered flight is more appealing in the sense that it is a common method for drone flight and is more predictable when it comes to aerodynamic performance.	Payload Systems, Payload Structures, Payload Electronics
SAIL deployment method changed from release directly from payload bay to release from the nose cone.	Deploying from the nose cone will allow the SAIL to be pulled out of the payload bay by the nose cone parachute. This led to a simpler recovery design compared to formulating a method for getting the payload bay to be open above the ground.	Payload Systems, Recovery, Structures

2.3 Changes Made to Project Plan

Table 2.3: Changes Made to Project Plan

Change Description	Justification	Affected Subsystem(s)
Added Team Derived Requirements and are enforcing them in the design.	Required per NASA SL 2024 Competition requirements.	Project Management, Vehicle, Payload
Added a structures build schedule for the sub-scale launch vehicle.	To ensure timely completion of sub-scale construction before the sub-scale flight scheduled for November 18th-19th.	Project Management, Structures
Added a back up sub-scale launch window December 16th-17th (Depicted in Figure 6.2).	To ensure the completion of sub-scale launch before CDR milestone in the event of unforeseen conflicts with the original window.	Project Management
Scheduled PDR presentation on November 9th at 2pm ET (Depicted in Figure 6.2).	Required per NASA SL 2024 Competition requirements.	Project Management

3 Vehicle Criteria

3.1 Launch Vehicle Mission Statement and Success Criteria

The mission of the launch vehicle is to safely house all payload structures and electronics as it ascends to a declared apogee of 4050 ft. and then descends under parachute until payload release is authorized. The launch vehicle will be designed to be reusable, reliable, and safe while aligning with all NASA and team-derived vehicle requirements.

The vehicle will be declared successful in its mission if it accomplishes the above mission statement. Some further guidelines of the vehicle success criteria are included in Table 3.1 below.

Table 3.1: Success Criteria for Launch Vehicle

Success Level	Vehicle Criteria
Success	Nominal takeoff and ascent; Reaches within $\pm$ 250 ft. of declared apogee; Follows recovery timeline; Payload ejected without damage; Recovered without any damage; Can be relaunched the same day
Partial Success	Nominal takeoff and ascent; Reaches within $\pm$ 500 ft. of declared apogee; Some minor damage upon landing that can be repaired at the field; Payload tangled during ejection but retains all functions
Partial Failure	Nominal takeoff and ascent; Reaches within $\pm$ 750 ft. of declared apogee; Damage upon landing that would prevent another launch within the same day; Payload damaged upon ejection
Failure	Catastrophe at takeoff; Nominal takeoff and ascent, but fails to get over 3000 ft. or manages to exceed 6000 ft.; Irreparable damage upon landing; Payload destroyed upon ejection

3.2 Alternative Launch Vehicle Designs

3.2.1 Airframe Material Selection

Two materials were under consideration for the construction of the airframe: Blue Tube and G12 fiberglass. Both of these materials are commonly used in high-power rocketry. Given that the airframe will take on all of the loads applied to the launch vehicle, it is important that it is capable to withstand these loads with a high factor of safety. Preliminary calculations of the force experienced by the airframe can be found in Section 3.3.3.

Blue Tube

Blue Tube is an airframe material that is manufactured by Always Ready Rocketry LLC. It is a spirally-wrapped vulcanized fiber that has a density of 0.751 oz/in^3, making it 28% more dense than phenolic tubing and 36% lighter than fiberglass [13]. Originally used for tank ammunition, Blue Tube is extremely resistant to blunt impact forces, abrasion, shattering, cracking, tearing, etc. Manufacturer testing revealed that Blue Tube has a maximum compressive strength of 3000 lb, which is far greater than the loads expected on the launch vehicle. Furthermore, there is less of a health risk associated with sanding Blue Tube than with sanding fiberglass. The only downside to Blue Tube is that it is not water resistant, which is problematic given that the team's home launch field has multiple irrigation ditches that the launch vehicle could potentially land in.

G12 Fiberglass

G12 fiberglass is a material specifically designed for tubing in high-power rocketry that uses spirally wound roving fiberglass and epoxy. Stronger than Blue Tube, G12 fiberglass is also highly resistant to blunt impact forces, shattering, abrasion, and cracking. Furthermore, this material is water resistant, making it preferable over Blue Tube. The downsides to G12 fiberglass are that it is significantly heavier than Blue Tube, with a density of about 1.05 oz/in^3, and that it poses a greater health risk when sanded or cut [2]. Such health risks can be mitigated by wearing proper PPE and only sanding and cutting the fiberglass in well-ventilated areas. Despite the increase in the weight of the launch vehicle if G12 fiberglass is chosen, the motors we have to choose from will easily provide enough thrust for the mission. A breakdown of the weight of the launch vehicle is shown in Table 3.4.

3.2.2 Nose Cone

The nose cone is used to reduce drag by providing a smooth contour for flow to change direction along the front surface of the launch vehicle. This reduces the overall dynamic pressure experienced by the launch vehicle by providing small angle tangency to the oncoming flow velocity. Many nose cones follow predefined geometries, such as the Von Karman, which is an optimized Haack series profile, the conic profile formed by the revolution of a right triangle around the vehicle's central axis, and an ogive profile formed by segmenting a large diameter circle into a section radius of the launch vehicle. Each of these profiles is commercially available for the launch vehicle. Previous team experience with the ogive profile governed the nose cone selection into two distinct profiles.

4:1 Ogive

The tangent ogive nose cone comes in two varieties, 4:1 and 5:1, where the ratio is length to diameter (e.g. a 4:1 nose cone with a six-inch diameter will be 24 in. long). The nose cone is made from G12 fiberglass and has a screw-on anodized aluminum tip. The shape of the tangent ogive nose cone allows enough space for ballasts or electronics to be inserted into the nose cone while retaining aerodynamic performance for subsonic flights. The advantage of the 4:1 nose cone over the 5:1 nose cone is that it is lighter and reduces the overall length of the launch vehicle.

5:1 Ogive

The 5:1 tangent ogive nose cone is a longer version of the 4:1 nose cone (in this case, 30 in. long instead of 24 in.). The 5:1 nose cone is also G12 fiberglass and has a screw-on anodized aluminum tip. While this does allow for more space for ballasts or electronics to be inserted into the nose cone, the current launch vehicle design should not require the addition of such ballasts for stability and should only need a few small electrical components for a latch.

3.2.3 Nose Cone Bulkhead

The nose cone bulkhead is used to mount recovery hardware so that the shock cord can connect the nose cone to the rest of the launch vehicle. The nose cone bulkhead can be used to mount ballasts or electronics inside the nose cone. Two design considerations for the nose cone bulkhead are discussed in further detail below.

Fixed

The fixed nose cone bulkhead is the most common and easiest to manufacture. The bulkhead will have two holes for a U-bolt to be inserted into. The entire bulkhead assembly is epoxied to the inside of the nose cone shoulder. If any ballasts needs to be added to the nose cone, it must be attached to the bulkhead before it is epoxied into the nose cone. The ballasts is then permanently inaccessible, harming the reusability of the launch vehicle. Furthermore, this permanent inaccessibility means that no electronics can be effectively stored inside the nose cone.

Removable

A removable nose cone bulkhead assembly consists of a centering ring with four screw holes, which will be epoxied into the nose cone shoulder, as well as a slightly smaller removable bulkhead that has bolt holes that align with those on the centering ring. In the centering ring holes, four threaded 1/4-20 T-nuts are epoxied into place for the bolts to thread into. The smaller removable bulkhead is able to fit a U-bolt for mounting recovery hardware and a latch for payload deployment on one side, and threaded rods to mount an avionics sled on the other side since the inside of the nose cone is accessible in this design. The removable bulkhead can then be secured to the centering ring with 4 1/4-20 steel round head bolts.

3.2.4 Fin Material Selection

Aircraft-Grade Birch Plywood

Aircraft-grade birch plywood is the first and simplest option for the construction of the fins. This plywood is relatively lightweight with a density of 0.362 oz/in^3 and is easy to work with. This plywood comes in 1/8 in. thick sheets. Two layers of the plywood can be epoxied together to make 1/4 in. thick fins that are strong enough to withstand the aerodynamic forces the launch vehicle will be subjected to in subsonic flight. The only downside is that, on rare occasions, the force of the landing or the dragging of the fin can on the ground by the parachute can severely damage the fins. This material is compatible with the laser cutter that the club uses and no major health risks are posed when sanding the material.

Sandwich Composite

Composite fins can be constructed in a number of different ways. A light core material such as balsa wood or aircraft-grade birch plywood is chosen first, then layers of fiberglass or carbon fiber are added on both sides of the wood using epoxy and compressed by a vacuum. Once the epoxy is cured, the fin is removed from the vacuum, the excess composite material is trimmed from the edges, and the fin is sanded for a smooth finish. Last year, the team completed this process for the fins and achieved 30-40% weight savings overall, but many of the fins needed to be replaced due to poor quality and damage from landings [32]. The club is actively working on new methods (such as the Vacuum-Assisted Resin Transfer Molding (VARTM) process) to create composite fins for the future that will be stronger and easier to manufacture. Given that there is not an urgent need for our current launch vehicle design to lose any weight (especially in the aft region), the benefits of composite fins do not outweigh the cost of the manufacturing process.

G10 Fiberglass

G10 fiberglass is the final material under consideration for the fins. G10 fiberglass is a material that is made from multiple layers of fiberglass cloth soaked in epoxy which is then heat-cured. It is commonly used for fins in high-power rocketry on supersonic rockets due to its high strength and resistance to aerodynamic flutter. Though the removable fin system for our launch vehicle allows for the removal of broken or damaged fins, the ultimate goal is to not have to replace the fins at all. This material will certainly be able to withstand any aerodynamic forces for the launch vehicle's subsonic flight and should also be far more resistant to impacts, abrasion, or cracking compared to birch wood fins or composite fins. The only downside to G10 fiberglass is that it is far heavier than the other two options with a density of 1.15 oz/in^3. Fortunately, the added weight does not harm the stability of the current launch vehicle design, nor does it cause any concern for the overall weight of the launch vehicle.

3.2.5 Fin Design

The fins have been designed in accordance with team-derived requirement LVF 1. The fins are designed to shift the aerodynamic center of the launch vehicle behind the center of gravity to improve stability. Leveraging previous team experience with various fin designs and construction methods has yielded two fin designs under consideration. The fin can will contain four equally spaced fins of the same design.

Symmetric Fins

A symmetric fin design would be made of 0.125 in. thick G10 fiberglass with a root chord of 8 in. and and tip chord of 4 in. The geometry of the fin will be mirrored over the center line which makes fabrication easier. The fins will be secured to the removable fin system by two #8-32 machine screws at a notched extrusion at the base of the fin. This fin profile provides an easy way to determine center of pressure. The mean aerodynamic chord center of pressure will be aligned with the center line of the fins, making aerodynamic calculations and fin placement trivial.

An aerodynamic simulation was conducted to determine the overall drag of the fin profile along with flight characteristics, such as turbulence and pressure gradient, across the fin structure. This simulation was conducted in ANSYS Fluent using a 567.5 ft/s maximum launch vehicle velocity and the SST K-ω fluid model. The results are shown below in Figures 3.1 and 3.2.

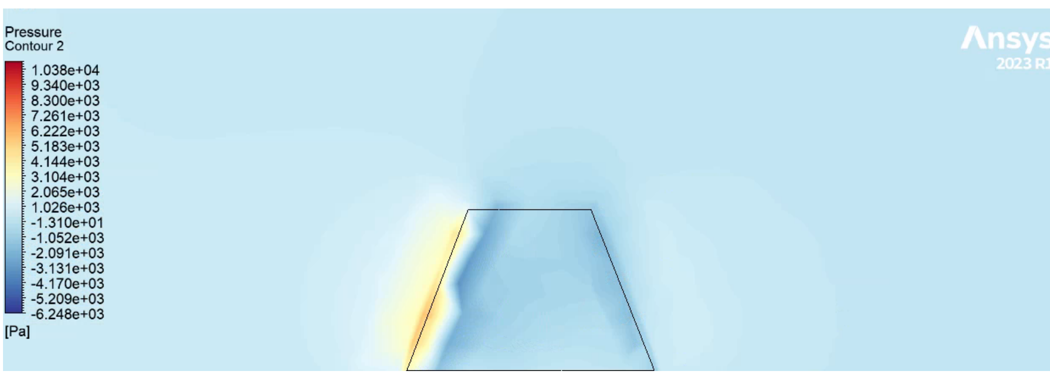


Figure 3.1: Pressure profile of the symmetric fins at 567.5 ft/s.

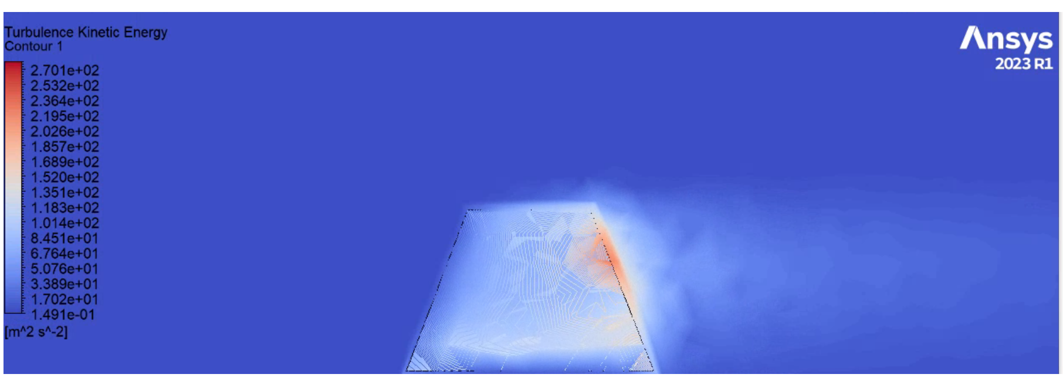


Figure 3.2: Turbulence profile of the symmetric fins at 567.5 ft/s.

From this simulation, the total drag force generated by each fin is 4.66 lbf. The orange and red areas of figure 3.1 highlight areas of high pressure stagnation due to momentum conservation as the fluid velocity is brought to rest at the blunt edge facing the oncoming flow. From the turbulence profile shown in figure 3.2, high kinetic energy due to particle interaction at the aft end of the fin profile is observed. This is caused by particle interaction when the fluid void is filled at the trailing edge of the fin. High turbulence in the wake of the fin is also observed, leading to the conclusion of aerodynamic inefficiencies in the symmetric fin profile design.

Swept Fins

An alternative to the symmetric fin profile, that still conforms to the team derived launch vehicle requirement LVF 1, is a swept fin profile. This profile will contain the same root chord of 8 in. and tip chord of 4 in. The difference is that the tip chord will be swept back from the root chord. This swept design shifts the center of pressure of the launch vehicle further aft than the symmetric fin profile. This fin profile will also be secured via the removal fin system, such that in the event of fin damage, fins can be swapped using simple tools. While this fin profile contains aerodynamic advantages, the introduction of a trailing edge tip to the profile increases the likelihood of damage since the trailing edge would extend past the end of the fin can. There may be instances where the weight of the rocket may rest on the fin profile, such as in certain launch setup and landing scenarios. Another disadvantage of this profile is the introduction of a more complex geometry to manufacture. A jig or computerized cutting machines will be required to ensure that the exact sweep measurement is kept between fins to minimize the risk of non-symmetric aerodynamic forces on the launch vehicle during flight.

Following the same simulation methodology as the symmetric fin analysis, at a simulated flow velocity of 567.5 ft/s, yielded plots for pressure, turbulence, and total drag in Figures 3.3 and 3.4. The same range of values were used in the analysis of the symmetric fins and the swept fins.

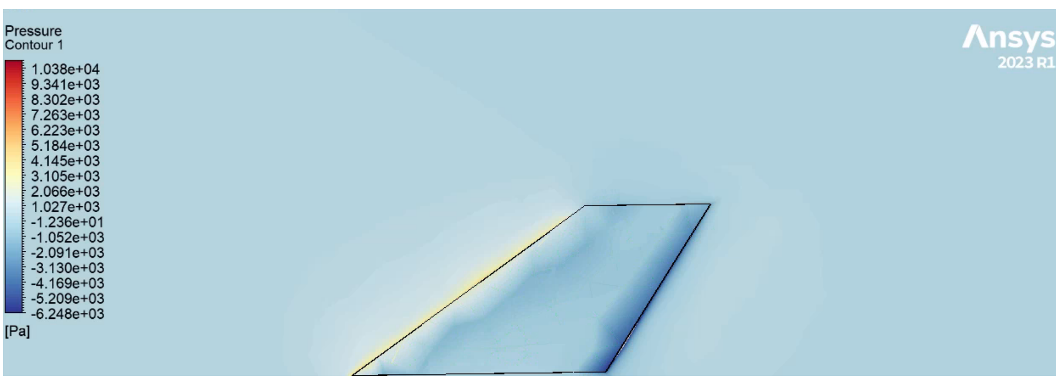


Figure 3.3: Pressure profile of the swept fins at 567.5 ft/s.

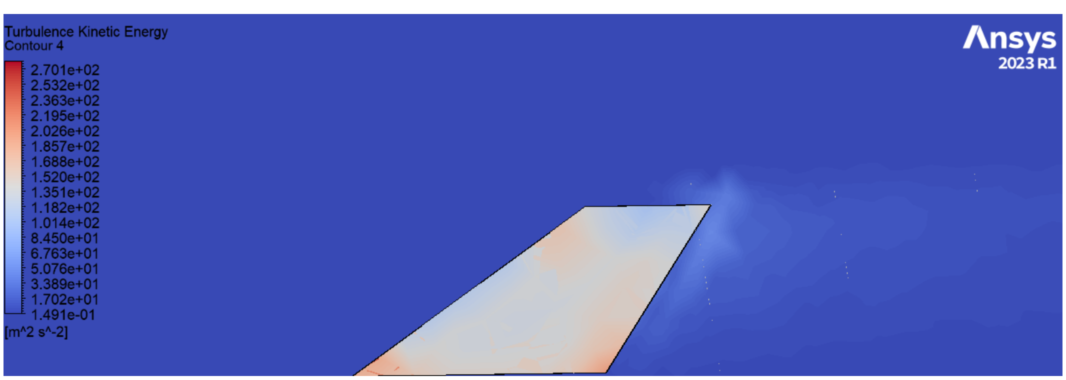


Figure 3.4: Turbulance profile of the swept fins at 567.5 ft/s.

From the simulation force analysis, the total drag force of each swept fin is 3.53 lbf. which is a 24.3%

reduction in total drag. This drag reduction correlates with a reduction in the leading edge pressure along each fin and an increase in negative pressure along the trailing edge of each fin. This pressure on the trailing edge acts in the opposite direction to the coordinate axis defined in the simulation, counteracting the force generated by the leading edge pressure. The turbulence study provided a more uniform turbulence profile across the faces of the swept fins, with a reduction in trailing edge turbulence effects and a less pronounced turbulent wake aft of the fins. Overall the results of the aerodynamic study reinforce the conclusions made that a swept fin profile is more advantageous for the final launch vehicle design.

3.2.6 Fin Can Design

The fin can is designed to house the motor, motor tube, and a method of attachment for the fins. Two methods of attaching the fins to the fin can are described below.

Fixed-Fin Design

Traditionally, the fins are permanently fixed to the fin can by sandwiching the fin tabs between two centering rings that surround the motor tube. The motor tube and centering rings are epoxied into the fin can and slots are cut into the airframe between the centering rings to accept the fins. The edges of the fin tabs are coated with epoxy so that when the fins are inserted they will bond with the motor tube and the centering rings. Once the epoxy has time to cure, fillets will be added to the fins with filler epoxy. This fixed-fin design has been used by the team in previous years and is widely used in high-power rocketry for strong fin connections. The main drawback of this design is that it harms the reusability of the launch vehicle. If a fin breaks, the entire fin can would have to be replaced which would cost lots of time and money. A motor catastrophe would likely cause damages that are not accessible to fix.

Removable Fin Design

The removable fin system design aims to improve the reusability of the launch vehicle by enabling the replacement of fins and/or addition of ballasts as needed. It is made from three centering rings that hold the motor casing in place and pairs of plywood runners that hold the fins. Two of the centering rings have slots cut into them to accept plywood runners which will hold the centering rings and the fins in place. The runners are epoxied permanently to the slots in the centering rings. Each runner has two holes to accept two #8-32 round head screws which will hold each fin in place. Two 1/4 in. diameter threaded rods are placed between the two centering rings to provide more support for the assembly. Between each set of runners on the assembly, there are two L-brackets (one for both the top and bottom centering rings) with nuts welded onto them so that the entire assembly can be secured to the airframe with a total of 8 #8-32 machine screws. The final centering ring is attached to the aft of the assembly and serves as a thrust plate. The thrust plate has a small section of motor tube epoxied to it to add a motor retainer. The thrust plate is attached to the rest of the assembly using the aforementioned 1/4 in. threaded rods that extend through the entire assembly. Once fin slots are cut into the airframe, the entire removable fin system can slide into the fin can and is secured with the #8-32 machine screws. While the removable fin system design is far more complex to manufacture, the benefits that it offers for the reusability of the launch vehicle are worth it. This design was developed and implemented by the team last year [32] and retained the reusability of the launch vehicle after suffering a CATO.

3.2.7 Tail Cone Design

The tail cone of the launch vehicle is used to decrease the overall drag profile of the launch vehicle by providing a smooth transition along the external contour of the launch vehicle. This can be leveraged to increase the overall apogee of the launch vehicle. While initial assumptions may lead to the conclusion that a tail cone should be a requirement for the launch vehicle, design considerations for a launch vehicle with and without a tail cone have been supplied.

Flat Base

The launch vehicle's main purpose is to securely house the payload for deployment in flight. Due to constraints with the payload location for separation and deployment, the payload has been placed towards the forward end of the launch vehicle. This means that stability concerns arise when attempting to modify the aft geometry of the launch vehicle for aerodynamic purposes. A flat aft section of the launch vehicle has the advantage of moving the center of pressure of the launch vehicle farthest aft, such that the required ballasts and overall weight of the launch vehicle are minimized, leading to less motor total impulse required to reach the target apogee. This tail design also requires no additional manufacturing or design, so that other sections of the launch vehicle can be focused on during manufacturing and testing. The sharp edge of the flat tail cone design increases the base drag profile of the launch vehicle, but with careful motor selection and aerodynamic simulations, the target apogee of the launch vehicle is still obtainable. The lack of a tail cone also increases the risk of fin damage if the swept fin design is chosen.

Boat-Tail

Traditionally, launch vehicles developed by the team have utilized a boat tail for the many aerodynamic advantages it creates during flight. The smooth transition between the airframe and motor of the launch vehicle decreases the total pressure drop required for the flow to fill the void of air behind the launch vehicle during flight. This decreases the total drag and aerodynamic forces on the aft section of the launch vehicle. A tail cone can also be used to add mass to the aft end of the launch vehicle for stability. This would shift the center of gravity of the launch vehicle aft. In certain scenarios, such as in the case of a launch vehicle that requires ballasts in the nose of the launch vehicle for stability, the center of gravity shift in the launch vehicle may not be advantageous. Ballasts would have to be added to compensate for the aerodynamic conditions that result from a boat tail, which would increase the total mass of the launch vehicle and decrease its target apogee.

3.2.8 Alternative Separation Points

Two layouts for separation points on the launch vehicle have been considered. The details and advantages/disadvantages of each layout are described below.

Double AV Bay Design

The design using two AV bays requires three separation points. AV bay 1 is located between the nose cone and drogue parachute 1/payload bay. The AV bay 1 is attached to the former with four nylon rivets and to the latter with four shear pins. AV bay 2 lies between drogue parachute bay 2 and the main parachute bay/fin can. It is attached to the former using four nylon rivets and to the latter using four shear pins.

The first separation point separates drogue parachute bay 2 from the drogue parachute 1/payload bay. This separation is initiated via an ejection charge in AV bay 2. The second separation point separates the nose cone from the drogue parachute 1/payload bay via an ejection charge in AV bay 1. The final separation point separates the main parachute bay from AV bay 2 via an ejection charge in AV bay 2. This design results in four separate sections and, with redundant charges, six ejection charges. The most concerning aspect of this design is the number of ejection charges needed to execute the mission. Furthermore, the number of separation points and independent sections increases the likelihood of failure for the recovery events.

Single AV Bay Design

This design of the launch vehicle uses only one AV bay and has only two separation points. The AV bay lies between the main parachute/payload bay and the drogue parachute bay/fin can. It is secured to the former using nylon rivets and to the latter with shear pins. The first separation point separates the AV bay from the drogue bay/fin can via an ejection charge to release a drogue parachute. The second separation point separates the nose cone and the main parachute/payload bay via an ejection charge, releasing both the main parachute and the payload. This design reduces the number of separation points to two and the number of ejection charges to four including redundant charges. This design is much safer and is less likely to fail at recovery events.

3.2.9 Motor Alternatives

Three motors are currently under consideration. Each motor being considered satisfies NASA Vehicle Requirement 2.1, which specifies a predicted launch vehicle apogee between 4000 and 6000 feet AGL. Only Aerotech motors are being considered for the launch vehicle due to the team's experience with the motor manufacturer and its flight-proven use. Specifications for each motor have been supplied in the Table 3.2 below.

Table 3.2: Leading Motors for 2024 Launch Vehicle

Motor	Propellant Mass (slug)	Total Mass (slug)	Total Impulse (lb•sec)	Average Thrust (lb)	Maximum Thrust (lb)	Burn Time (sec)	Casing	Length (in)
L1390G	0.1351	0.2657	887.77	313.48	376.55	2.6	RMS-75/3840	20.86
L1520T	0.1270	0.2501	835.16	352.45	396.85	2.4	RMS-75/3840	20.39
L1256WS	0.1345	0.2573	850.90	282.60	339.12	3.0	RMS-75/3840	22.08

The first motor under consideration is the L1390G. This motor has the highest total impulse of the three motors, which may be needed, depending on the final payload mass of the vehicle, to reach the declared apogee. This motor uses barium nitrate in the propellant mix which gives the exhaust a green color during burn. The high mass of this motor will also shift the center of gravity of the launch vehicle further aft, decreasing the stability of the launch vehicle. A thrust profile for the L1390G motor has been provided in Figure 3.5 below.

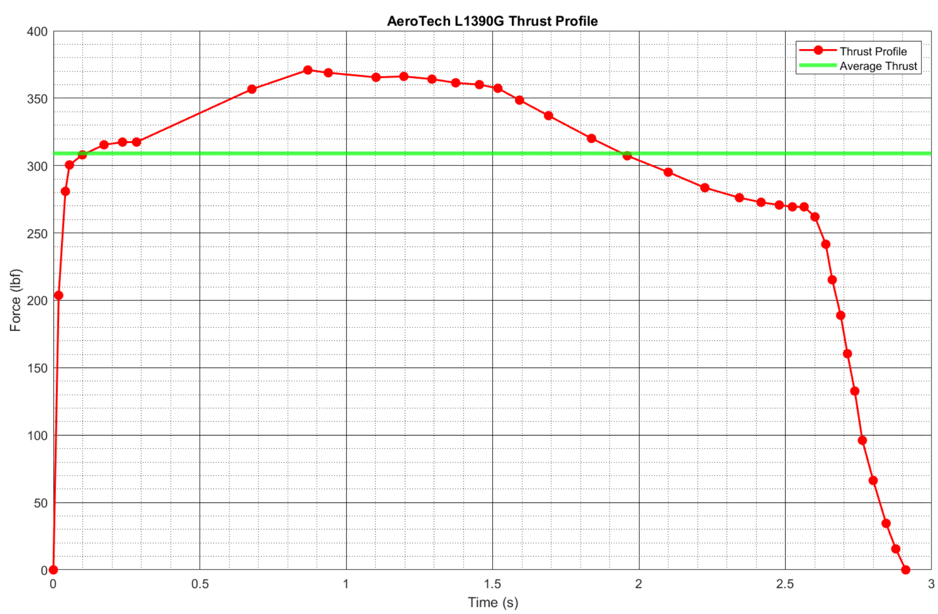


Figure 3.5: AeroTech L1390G thrust profile versus burn time.

The next leading motor under consideration is the L1520T. This motor has been flight-proven by the team in past years. This motor has the least total impulse and the lowest mass of the motors under consideration, which may be useful in the event that payload weighs less than predicted. The blue exhuast of this motor is achieved by the use of copper compounds in the propellant mixture. A thrust profile for the L1520T has been provided in Figure 3.6 below.

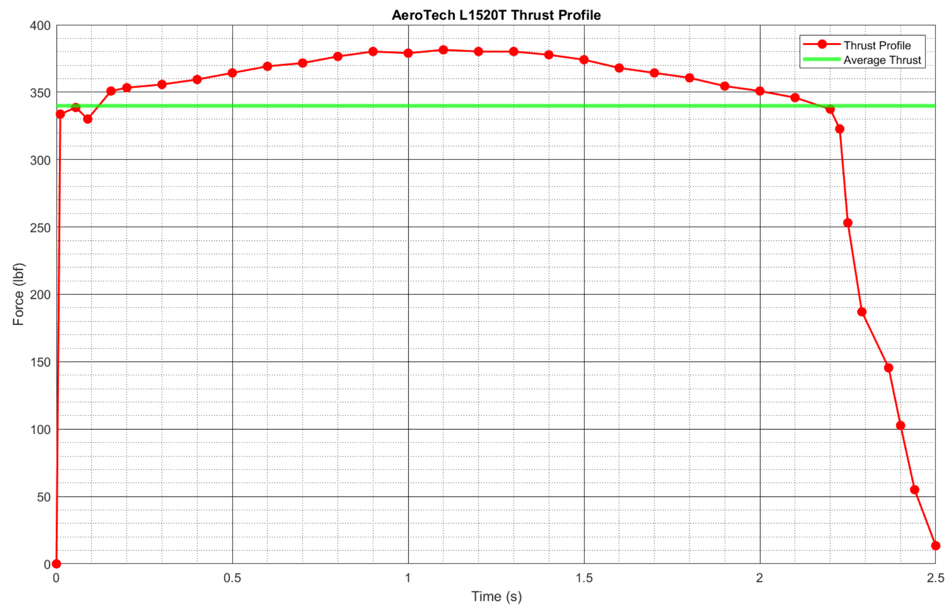


Figure 3.6: AeroTech L1520T thrust profile versus burn time.

The final motor under consideration is the AeroTech L1256WS. This motor has a total impulse between that of the AeroTech 1390G and AeroTech 1520T. This motor profile trails in thrust towards the end of the burn which will reduce aerodynamic stress on the launch vehicle during ascent. The white color of the exhaust is due to magnesium compounds in the propellant mixture. A thrust profile for the L1256WS has been provided in Figure 3.7 below.

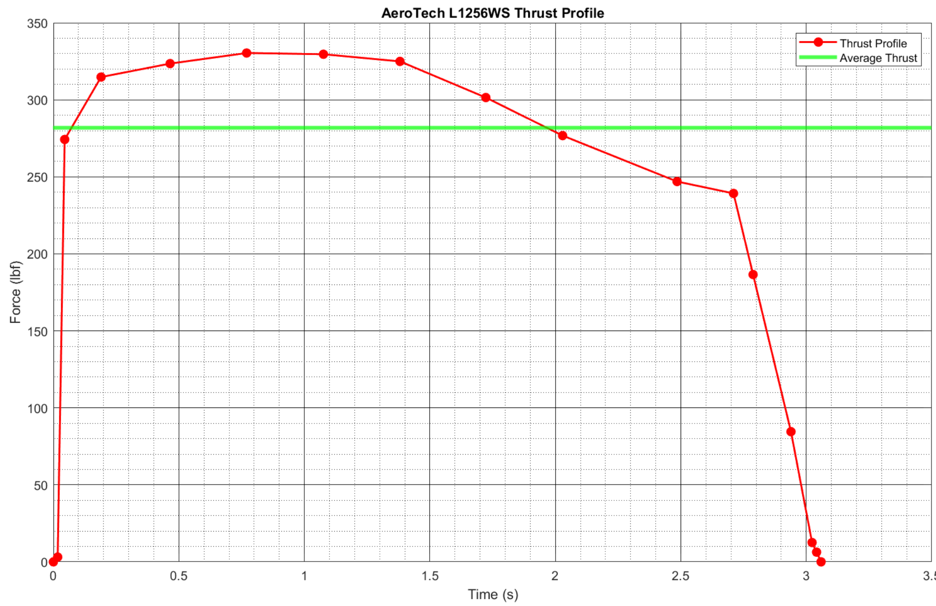


Figure 3.7: AeroTech L1256WS thrust profile versus burn time.

3.3 Leading Launch Vehicle Design

3.3.1 Launch Vehicle Sections and Layout

The leading launch vehicle design consists of four sections, two of which have more than one purpose. The four sections of the launch vehicle (from left to right, as seen in Figure 3.8) are the nose cone, the main parachute/-payload bay, the avionics (AV) bay, and the drogue parachute bay/fin can. The length of the launch vehicle is 105 in., with a weight of approximately 41.4 lbs. This places the CG of the launch vehicle at 65.2 in., measured from the tip of the nose cone. With the current four-fin configuration, the CP is 77.9 in., measured from the tip of the nose cone. The diameter of the airframe is 6.17 in. The stability margin is calculated to be 2.06. A diagram of the launch vehicle design and components is shown below in Figure 3.8.

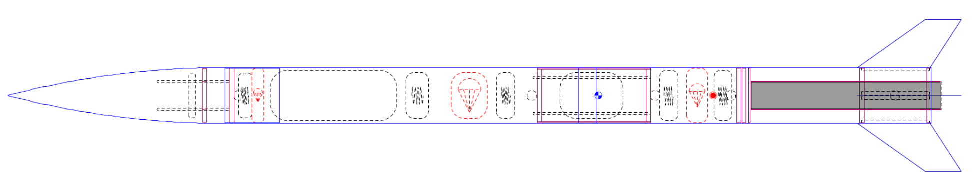


Figure 3.8: Diagram of the leading launch vehicle design and components.

The main parachute/payload bay and the AV bay are connected with four 4-40 nylon rivets so that these sections will remain together in flight. The nose cone and the drogue parachute bay/fin can are connected to the main parachute/payload bay and the AV bay each with four 4-40 shear pins. This current configuration will leave the launch vehicle in only three independent sections at its descent. The dimensions of the launch vehicle are illustrated below in Figure 3.9.

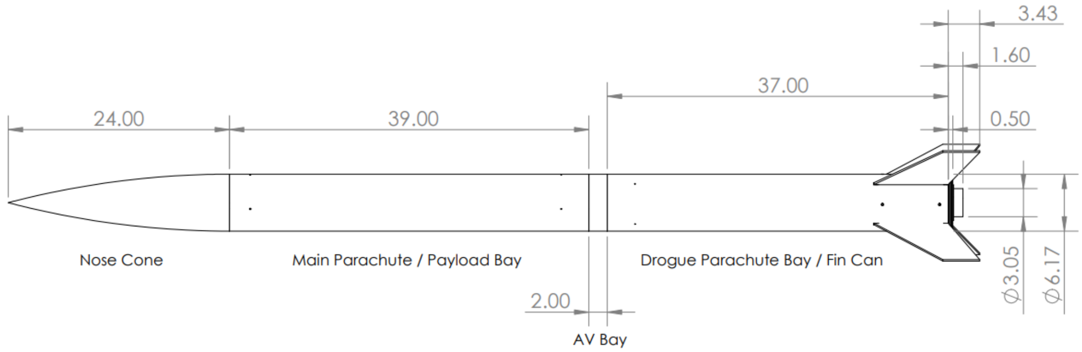


Figure 3.9: Dimensions of the launch vehicle. The overall length of the launch vehicle is 105.43 in. All sections of airframe have a diameter of 6.17 in.

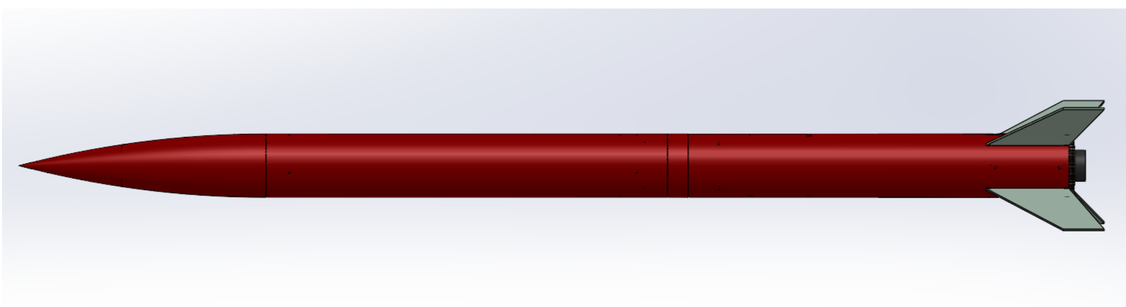


Figure 3.10: CAD Model of the assembled launch vehicle.

3.3.2 Separation Points

The current launch vehicle design has two separation points. The first separation occurs between the AV bay and the drogue parachute bay/fin can and the second occurs between the nose cone and the main parachute/-payload bay. Energetics will be located on both sides of the AV bay to initiate these separations. The nose cone is attached to the main parachute/payload bay with four 4-40 nylon shear pins. Likewise, the drogue parachute bay/fin can is attached to the AV bay with four 4-40 nylon shear pins. A diagram of the separation points of the launch vehicle is shown below in Figure 3.11.

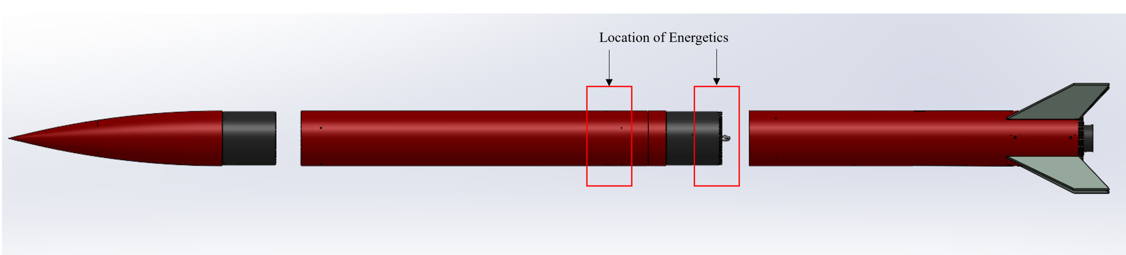


Figure 3.11: Diagram of the separation points on the launch vehicle including the locations of energetics.

3.3.3 Airframe Material Selection

The material selected for the airframe is G12 fiberglass. This material is chosen because of its high impact strength and its resistance to abrasion, shattering, cracking, and, most importantly, water. These characteristics contribute substantially to the reusability of the launch vehicle. With the airframe and couplers being made entirely from G12 fiberglass, the current launch vehicle weight estimate is about 42.5 lb. A complete mass breakdown of the launch vehicle is available in Table 3.5. Simulations using the current motor choices indicate that the launch vehicle should reach an apogee greater than 4,000 ft. G12 fiberglass offers sufficient protection for the launch vehicle while still allowing it to reach altitudes within range for the competition.

Preliminary Force Analysis

To ensure that the airframe will not fail, it is necessary to consider the maximum loads experienced by the launch vehicle. To do so, a free-body diagram of the launch vehicle in flight is shown below in Figure 3.12.

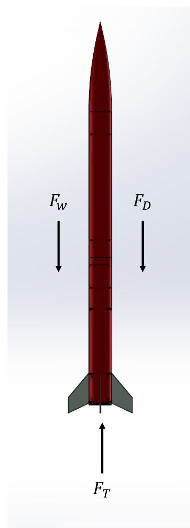


Figure 3.12: A free body diagram showing the primary forces acting on the launch vehicle.

From the free-body diagram, the three main forces experienced by the launch vehicle are thrust, weight, and drag. The maximum thrust force is assumed to be the maximum thrust of the motor which will occur just after the launch vehicle leaves the rail. The net force on the launch vehicle is shown below in Equation 1.

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

The net force, F_{net}, is the sum of the thrust force of the motor, F_T, the launch vehicle weight, F_W, and the force of drag, F_D. The drag force is calculated using Equation 2.

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

The force of drag, F_D, is a function of the air density ρ, the velocity v, the coefficient of drag C_D, and the frontal area A. Estimates for each of these values are tabulated below in Table 3.3.

Table 3.3: List of Variable Values

Variable	Value	Source
g	32.2 ft/s^2	Known constant
ρ	0.00237 $slug/ft^3$	Known constant
F_W	42.5 lb	OpenRocket and Measurements
A	0.2076 ft^2	Hand Calculation
v	567.6 ft/s	OpenRocket Simulation
F_T	396.86 lb	Motor Manufacturer Datasheet
C_D	0.47	OpenRocket Simulation

Using the parameters from the table, it is found that the maximum force of drag, F_D, is approximately 37.3 lb. Using Equation 1, the net force on the airframe, F_{net}, is about 317.1 lb. Thus, the airframe must be able to support this load with an appropriate margin of safety. The margin of safety is determined via Equation 3 below.

$$M.S. = \frac{\text{Allowable Load or Stress}}{\text{Applied Load or Stress} * F.S.} - 1 \tag{3}$$

Structural FEA simulations were conducted on the longest section of airframe on the launch vehicle (the main parachute/payload bay) assuming that it will experience the greatest stresses. A minimum factor of safety of 2 is considered for all structural components. The results from the FEA are shown below in Figure 3.13.

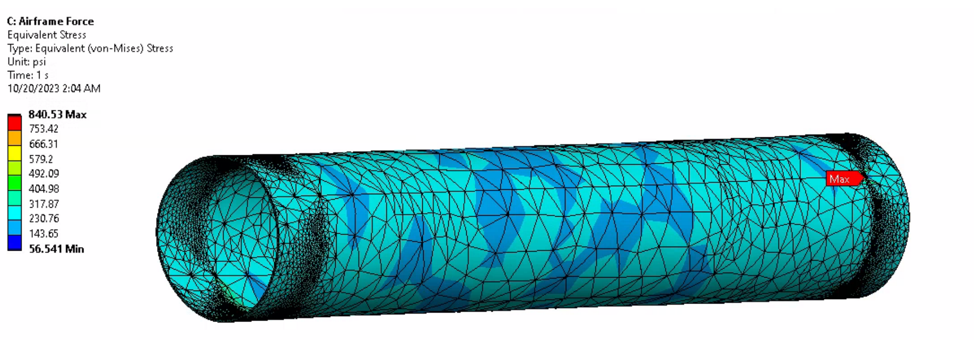


Figure 3.13: Results of FEA conducted on the main parachute/payload bay.

From Figure 3.13, it is shown that the maximum stress occurs at the holes where the main parachute/-payload bay connects to the AV bay. This maximum stress is about 840.5 psi. Additionally, the ultimate compressive strength of the section is roughly 65 ksi. A positive safety margin of 49.56 is achieved. Given that G12 fiberglass is not available for most FEA softwares, the analysis above was conducted on G10 fiberglass. G12 fiberglass is proven to be stronger than G10 fiberglass, therefore, if G10 has a safety margin of 49.56 it can be assumed that the safety margin for G12 fiberglass is greater.

3.3.4 Bulkhead Thickness

Preliminary Force Analysis

Historically, the team has used bulkheads made from multiple plies of 1/8 in. thick aircraft-grade birch plywood epoxied together with West Systems' 105 resin and 205 slow hardener. Such bulkheads are usually 3/4 in. thick, however, 1/2 in. bulkheads have also been used and were equally as effective [32]. As such, 1/2 in. bulkheads will be considered to ensure the structural integrity and reduce the weight of the launch vehicle.

The force from the main parachute on the AV bay bulkhead will be considered for structural FEA purposes. Results from such FEA simulations are shown below in Figure 3.14.

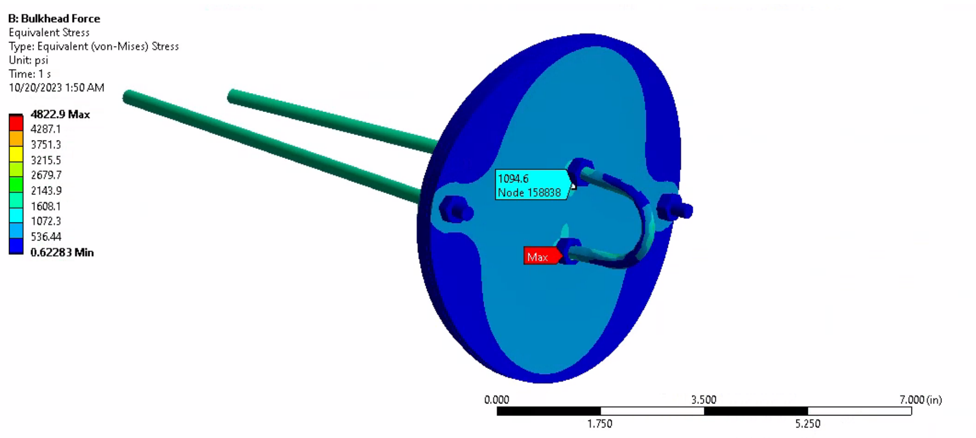


Figure 3.14: Results of FEA conducted on the AV bay bulkhead from main parachute shock cord force.

From the FEA simulation, the maximum stress on the bulkhead is 1094.6 psi, while the ultimate tensile stress is 4757 psi. This yields a positive safety factor of 1.173. Thus, 1/2 in. thick bulkheads provide sufficient strength for use in the launch vehicle.

3.3.5 Nose Cone

A 4:1 tangent ogive nose cone with an anodized aluminum tip will be used for the launch vehicle. This shape allows for electronics and/or ballasts to be placed inside the nose cone for payload, recovery, and stability purposes. A 4:1 is preferred over a 5:1 for the purpose of saving weight. The 5:1 nose cone does provide more space for ballasts and electronics, but the space provided by the 4:1 nose cone will serve the current design well. The 4:1 nose cone option is widely available through retailers and will be easy to obtain. As the ratio implies, the nose cone will be 24 in. long. A coupler will be pre-installed three inches into the nose cone, leaving a six-inch section of coupler to mount the next piece of the airframe to. A dimensioned drawing of the selected nose cone is included below in Figure 3.15.

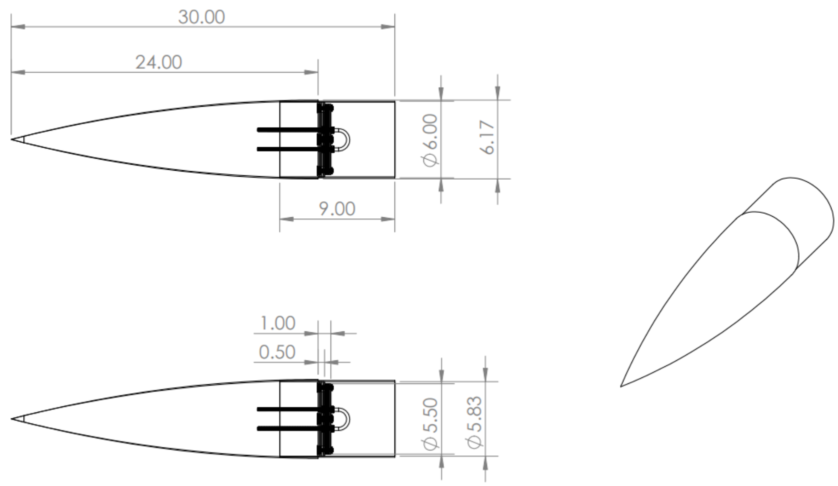


Figure 3.15: Dimensions of the assembled nose cone.

3.3.6 Nose Cone Bulkhead

The nose cone bulkhead will be comprised of two parts: the permanent nose cone centering ring and the removable nose cone bulkhead. Given that electronics and possibly ballasts will be stored within the nose cone, it is necessary to be able to remove the nose cone bulkhead. A half-inch thick centering ring with four holes to accept four 1/4-in. tee nuts will be permanently epoxied into the nose cone so that it is flush with the aft end of the nose cone inside the coupler. A 1/2-in. thick bulkhead with four holes that line up with those on the permanent nose cone centering ring can be screwed into place with four 1/4-in. round head bolts. Figure 3.16 and Figure 3.17 illustrate where the permanent nose cone centering ring and removable nose cone bulkhead lie within the nose cone. A U-bolt will be attached to the removable nose cone bulkhead to attach a shock cord for recovery. Additionally, two stainless steel threaded rods will be attached to the removable nose cone bulkhead to mount a sled for electronics.

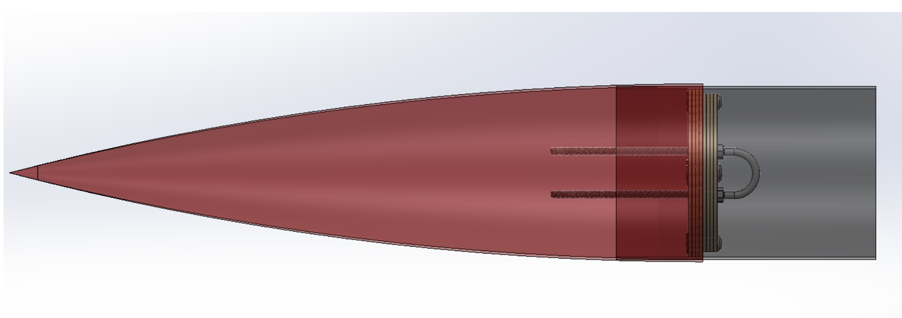


Figure 3.16: Position of nose cone bulkhead inside the nose cone.

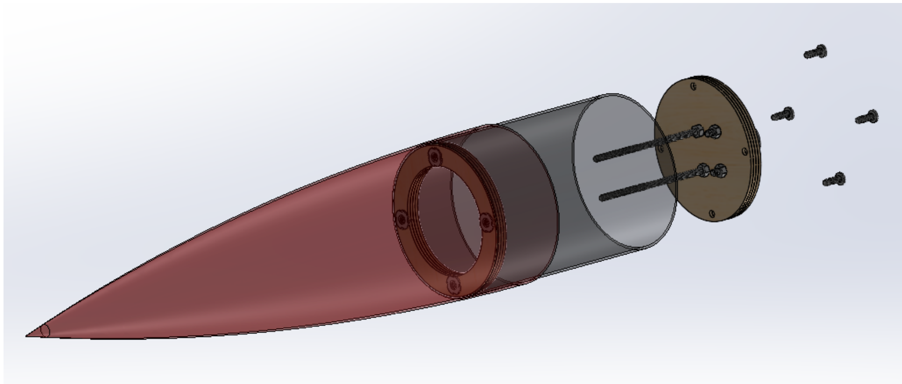


Figure 3.17: Removable nose cone bulkhead and permanent nose cone ring.

A dimensioned drawing of both the permanent nose cone centering ring and the removable nose cone bulkhead is shown below in Figure 3.18.

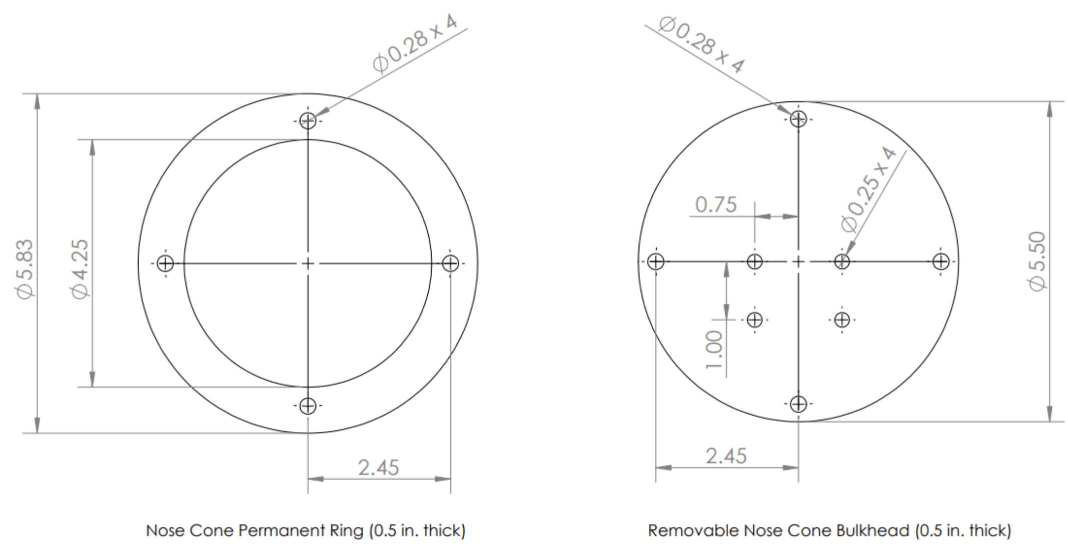


Figure 3.18: Dimensions of removable nose cone bulkhead and permanent nose cone ring. Both will be constructed from four layers of 1/8 in. thick aircraft-grade birch plywood.

3.3.7 Main Parachute and Payload Bay

The main parachute/payload bay will be made out of a single section of G12 fiberglass airframe that is 39 in. long. Dimensions of the main parachute/payload bay are shown below in 3.19. This bay is located between the nose cone and the avionics bay. It will be connected to the avionics bay with four 4-40 nylon rivets and to the nose cone with four 4-40 nylon shear pins. The main parachute/payload bay will hold the main parachute and its shock cord, along with the payload assembly.

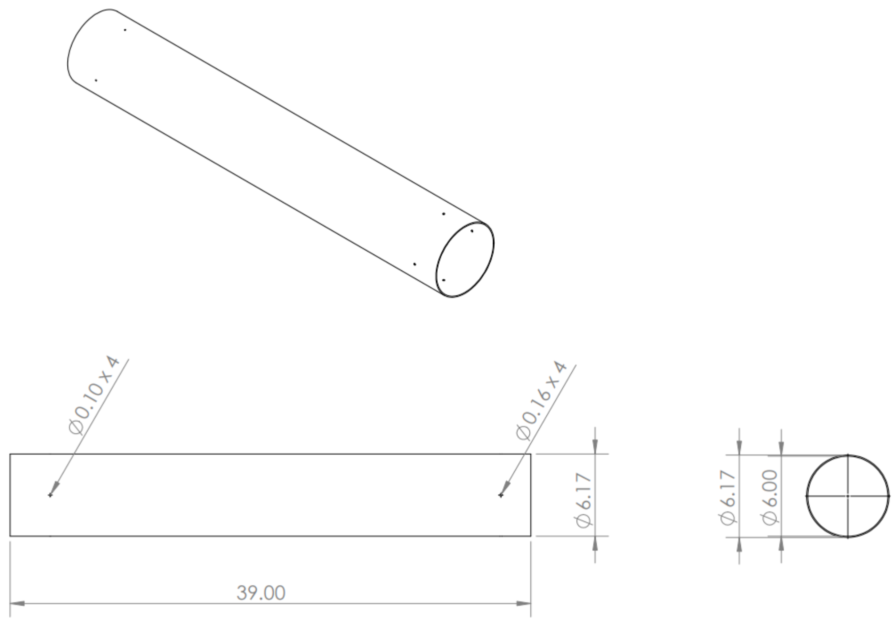


Figure 3.19: Dimensions of the Main Parachute/Payload Bay. This section of airframe will be constructed from G12 fiberglass.

3.3.8 Avionics Bay

The avionics bay will be made out of a 2 in. section of G12 fiberglass airframe and a 12.5 in. section of G12 fiberglass coupler. The airframe is epoxied to the coupler such that there is 4.5 in. of coupler forward of the 2 in. airframe band and 6 in. of coupler aft of the 2 in. airframe band. This airframe band will have one hole drilled into it for the altimeter pull-pin switch. Additional holes will be drilled into the coupler for altimeter pressure ports. A CAD model of the avionics bay is shown in Figure 3.20 and dimensions of the are shown in Figure 3.21. The avionics bay is located between the main parachute/payload bay and the drogue parachute bay/fin can. It will be connected to the main parachute/payload bay by four 4-40 nylon rivets and to the drogue parachute bay/fin can by four 4-40 nylon shear pins. The avionics bay will house all of the recovery electronics.

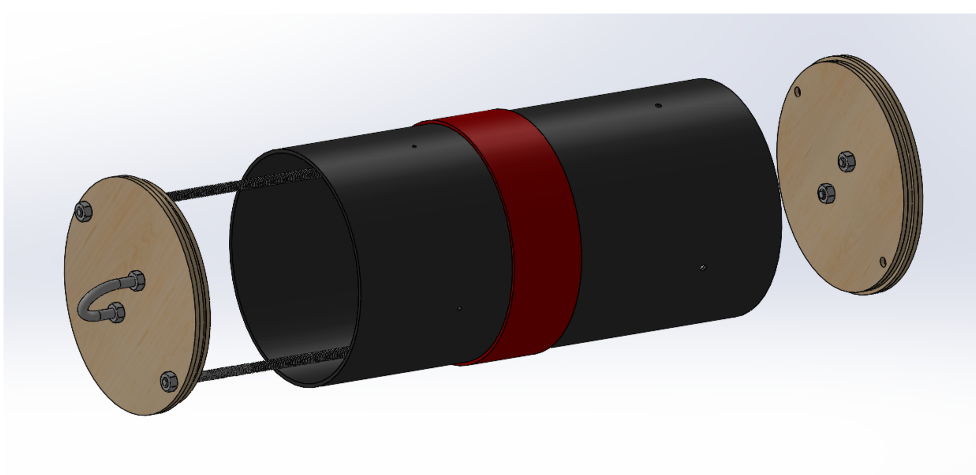


Figure 3.20: CAD model of the disassembled avionics bay.

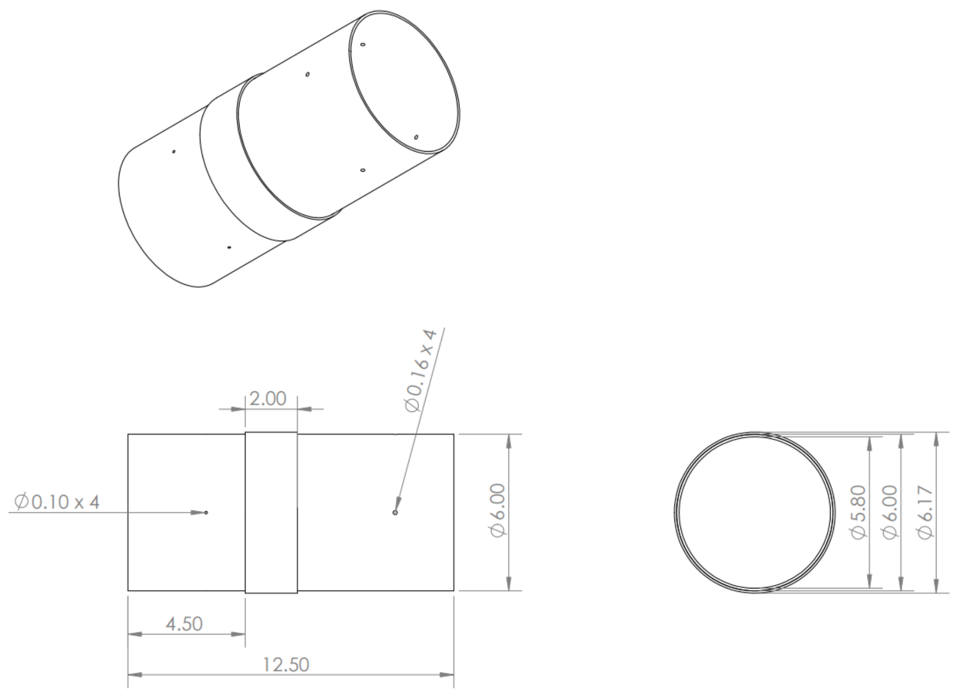


Figure 3.21: Dimensions of Avionics Bay. This section will be created by permanently epoxying a section of airframe to a coupler section. Both will be made of G12 fiberglass.

There will be one 1/2 in. thick bulkhead on either side of the Avionics Bay. Each bulkhead will be made out of four layers of 1/8 in. thick aircraft-grade birch plywood. These 1/8 in. thick layers will be epoxied together in order to withstand the parachute deployment forces with an acceptable factor of safety, verified with simulations and calculations (see Section 3.3.4), and tested with a universal testing machine. These bulkheads will be connected with two 1/4 in. diameter threaded rods with washers and nuts on the outward-facing sides of the bulkheads. The outward-facing side of each bulkhead will have two blast caps and the inward-facing side of each bulkhead will have two terminal blocks for parachute deployment. The outward-facing side of each bulkhead will also have one U-bolt which will be used to attach the shock cord to the Avionics Bay. The dimensions for the avionics bulkheads are shown below in 3.22.

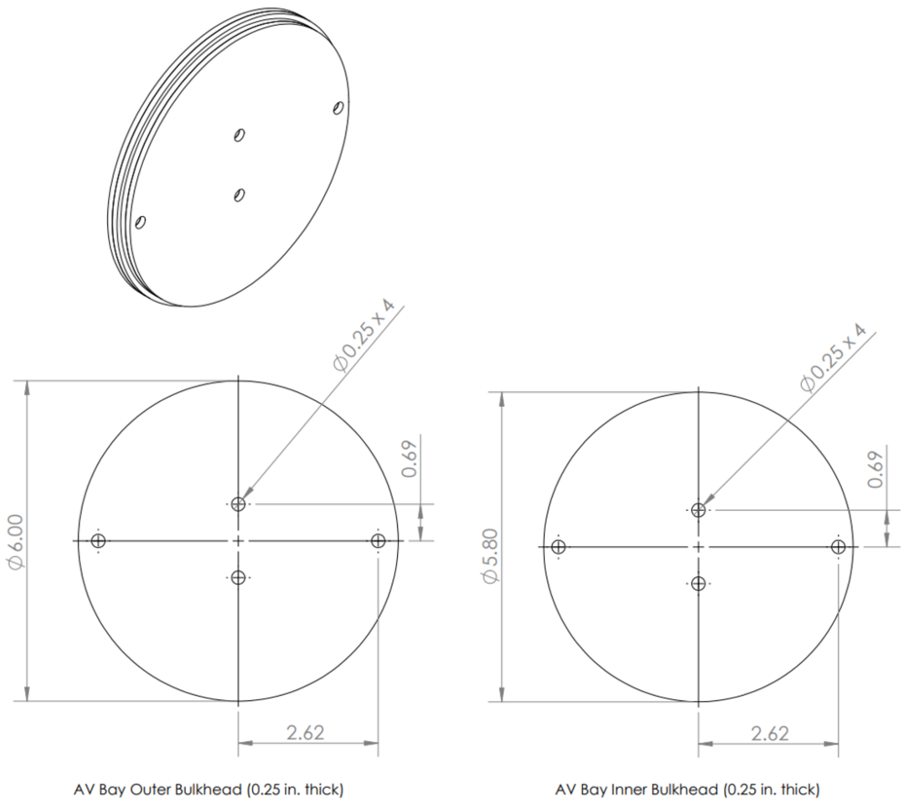


Figure 3.22: Dimensions of AV Bay bulkheads. These bulkheads will be made from four layers of 1/8 in. aircraft-grade birch plywood.

3.3.9 Drogue Parachute Bay and Fin Can

Design

The drogue parachute bay/fin can will be made of a single 37 in. long piece of G12 fiberglass airframe. This section will house the drogue parachute along with its shock cord, the removable fin system, and the motor along with its casing. This section will be connected to the AV Bay by four 4-40 nylon shear pins. There are four 1/8 in. thick slits extending 8.10 in. from the aft end of the fin can that will allow for the fins in the removable fin system to slide into place. The dimensions of the drogue parachute bay/fin can and the fin slits are shown in 3.27. The removable fin system will occupy the bottom 9 in. of the airframe and will be separated from the drogue parachute bay by a 1/2 in. thick bulkhead composed of four 1/8 in. thick plies of aircraft-grade birch plywood. The dimensions for this bulkhead are shown in 3.28. This bulkhead will be epoxied together, then epoxied into the launch vehicle 25 in. from the aft end of the fin can airframe. The drogue parachute is attached to a U-bolt on the drogue parachute bulkhead by a shock cord, which occupies the remaining space in the airframe aft of the avionics bay.

The removable fin system is composed of two 6 in. diameter bulkheads, each made of four 1/8 in. thick plies of aircraft-grade birch plywood that are epoxied together. The inner diameter of each bulkhead is 3.05 in., which allows room for the motor casing. The bulkheads are separated by 7.07 in. and connected with two 1/4 in. diameter threaded rods. As shown in 3.26, there are four pairs of 1/8 in. plywood runners, each at 90°angle from one another. All eight runners are epoxied into the bulkheads at the ends of the removable fin system in 0.13 in. slots. Each runner has two holes for #8-32 screws, 4.57 in. from each other that allow for securing the fins in the removable fin system. There are also eight L-brackets between the two bulkheads that allow for the removable fin system to be screwed into the airframe of the fin can. These L-brackets are positioned midway between the runners for the fin slots, and are at 90° angles from one another. The thrust plate is screwed into the aft bulkhead of the removable fin system. This bulkhead is composed of a 1/8 in. sheet of 60/61 aluminum, connected to three 1/8 in. thick pieces of aircraft-grade birch plywood. This bulkhead is also attached to the retaining ring for the motor, as seen in Figure 3.25.

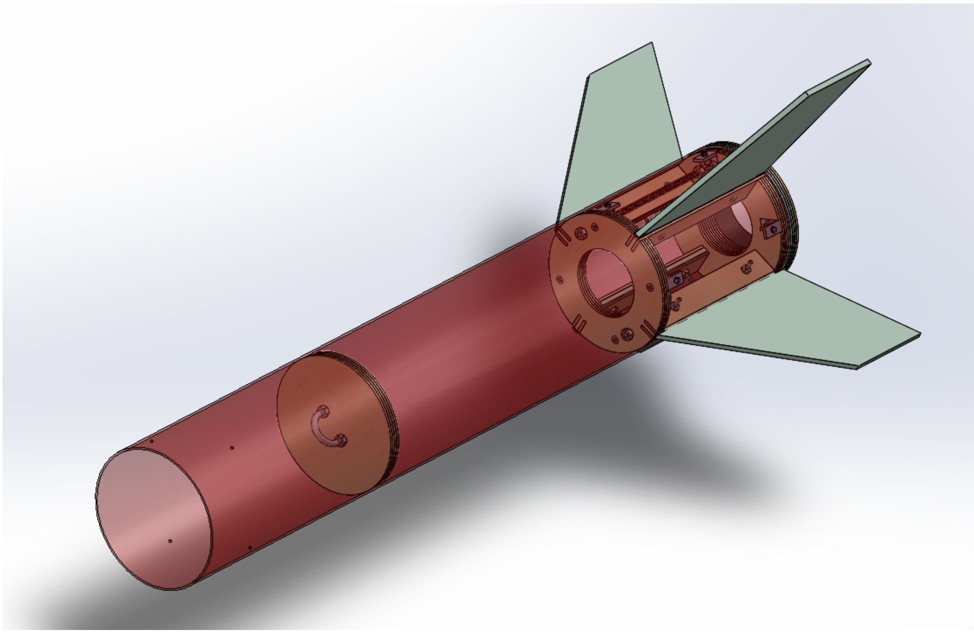


Figure 3.23: Complete drogue parachute bay/fin can assembly.

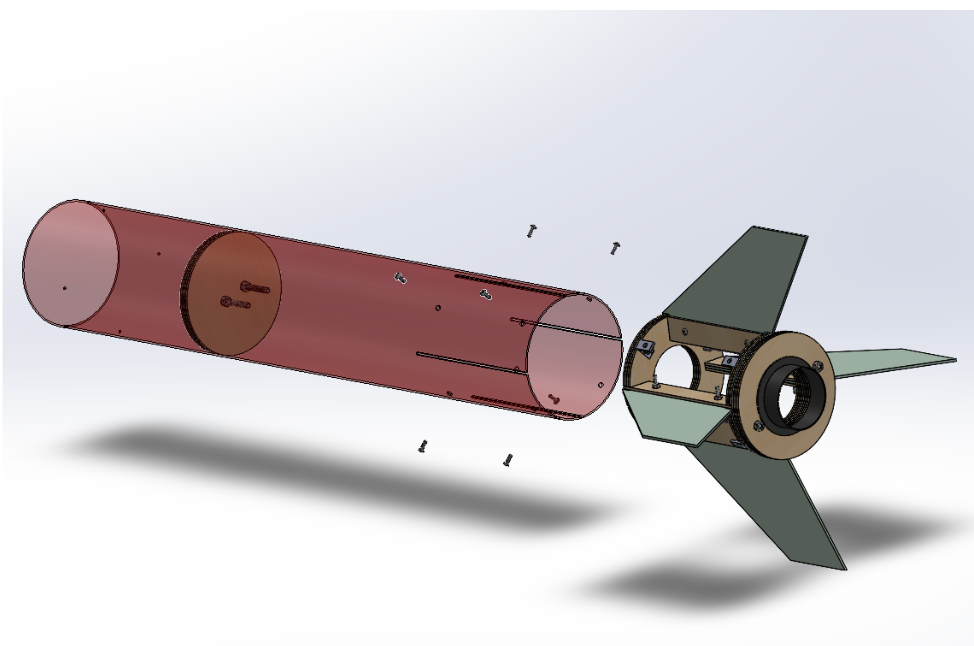


Figure 3.24: Drogue parachute bay/fin can with fin system removed.

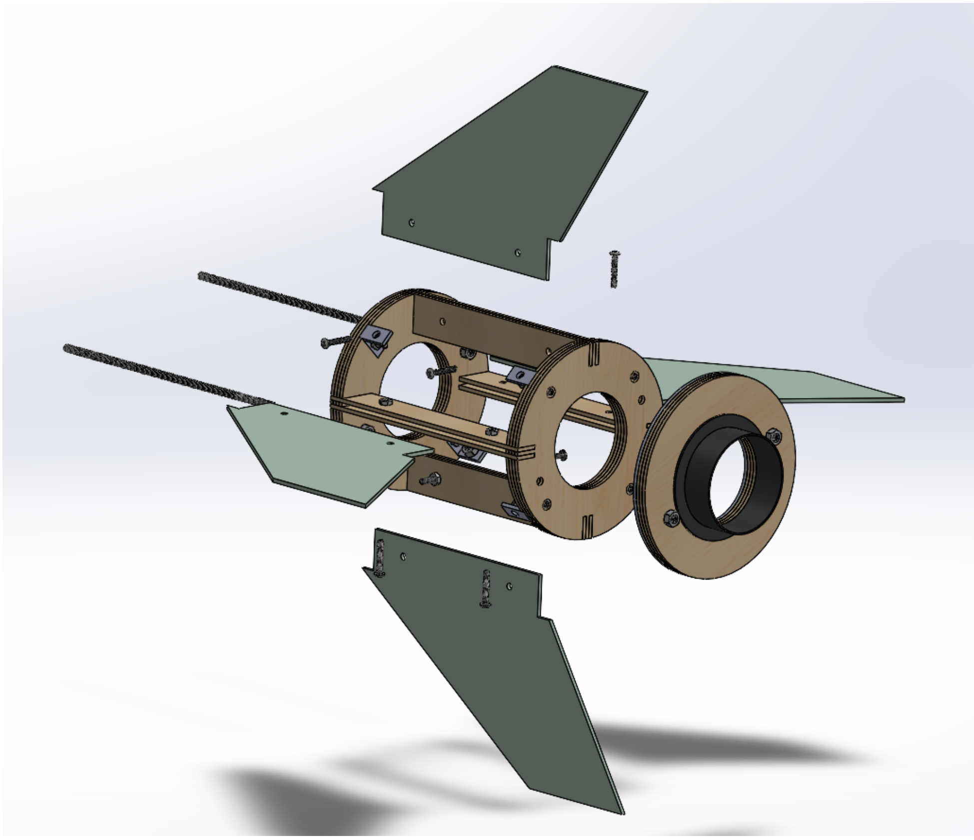


Figure 3.25: Exploded view of the removable fin system.

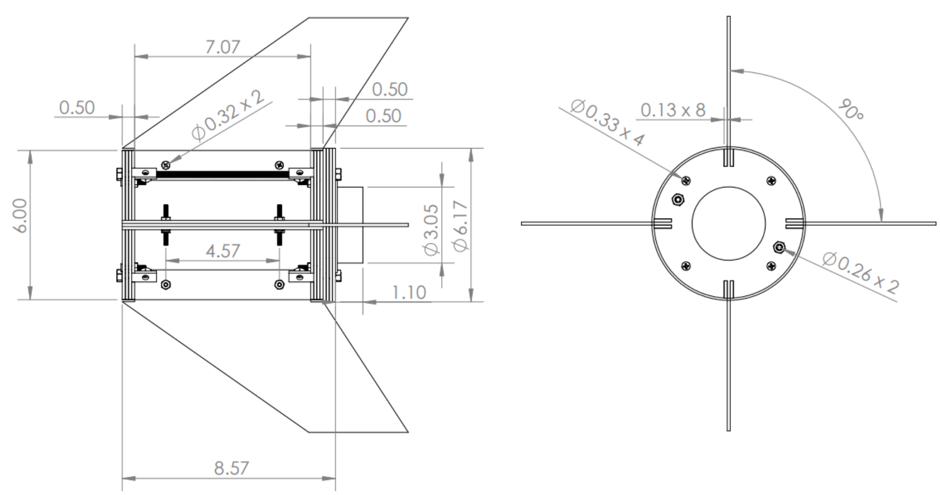


Figure 3.26: Dimensions of the removable fin system. The bulkheads and runners will be fabricated from aircraft-grade birch plywood, while the hardware and thrust plate will be made from stainless steel and 60/61 aluminum, respectively.

Construction

The length of the drogue parachute bay/fin can section of the airframe will be measured, marked, and then cut by staff in NC State's Senior Design Lab. This is to ensure that the ends of the airframe are level and that members of the team are safe from large amounts of fiberglass particulates. The fin slots will also be created in this space. The holes for the removable fin system as well as shear pin holes will be drilled in a well-ventilated area while wearing proper PPE. Dimensions of the drogue parachute bay/fin can are shown below in Figure 3.27.

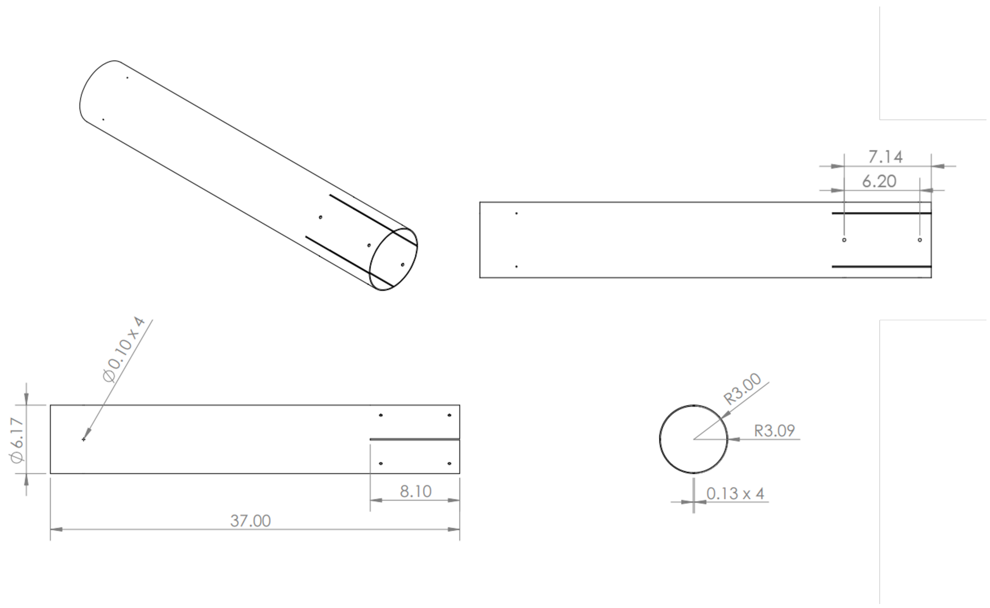


Figure 3.27: Dimensions of drogue parachute bay/fin can. This section of airframe will be made from G12 fiberglass.

The fin can bulkhead is constructed from four plies of 1/8 in. thick aircraft-grade birch plywood. These layers are epoxied together with West Systems 105 epoxy resin and 205 slow hardener. The bulkhead layups are then held under a vacuum for 24 hours for the epoxy to cure. After the epoxy has cured, a U-bolt is added to hold recovery hardware. Dimensions of the fin can bulkhead are shown below in Figure 3.28.

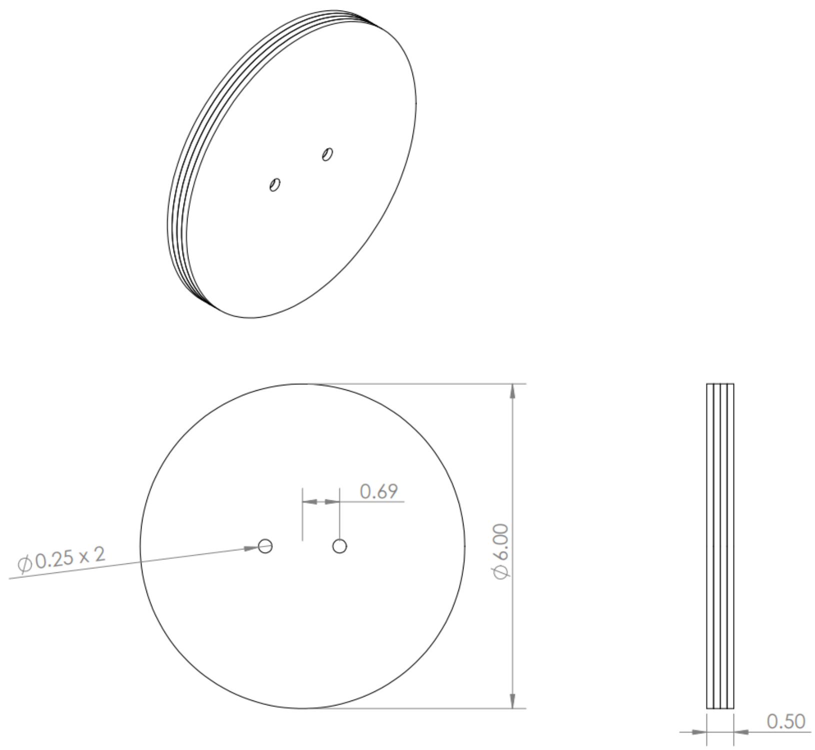


Figure 3.28: Dimensions of drogue parachute bay/fin can bulkhead. This bulkhead is made from four layers of 1/8 in. thick aircraft-grade birch plywood.

The bulkheads and runners of the removable fin system, as well as the rest of the components made from aircraft-grade birch plywood included in the launch vehicle, will be laser-cut at NC State's Entrepreneurship Garage, which requires training and certification to access. The bulkheads for the removable fin system are constructed the same way as the rest of the bulkheads in the launch vehicle. Each of the runners are attached to the removable fin system bulkheads with the same epoxy resin and hardener and left to cure. In order for the removable fin system to be secured to the airframe with screws, four L-brackets are screwed into each bulkhead such that two L-brackets are between each fin. Each L-bracket will have one #8-32 nut welded to it to accept the 1/2 in. #8-32 screws which will attach the removable fin system to the airframe. Two 1/4 in. stainless steel threaded rods are added to the removable fin system to help transfer a portion of the force from the motor. Finally, the thrust bulkhead will be constructed from three layers of 1/8 in. aircraft-grade birch plywood which will be epoxied and cured as before. The remaining 1/8 in, will be dedicated to a 60/61 aluminum thrust plate which will be cut out by staff in NC State's Senior Design Lab with a water jet. On the thrust bulkhead, a motor retaining ring that keeps the motor in place will be added. The thrust plate and thrust bulkhead will slide onto the threaded rods on the removable fin assembly and will be secured using two 1/4 in. nuts.

Preliminary Force Analysis

While the removable fin system is secured to the airframe with 8 #8-32 screws, this is not the only way it is held in place under the thrust of the motor. The thrust bulkhead, which consists of three 1/8-in. thick layers of aircraft-grade plywood and the motor retaining ring has the same outer diameter as the airframe. Furthermore, a thrust plate made of 60/61 aluminum that has the same outer diameter as the airframe is secured between the airframe and the thrust bulkhead. Both these components help minimize the stresses on the removable fin system and the screw holes in the airframe.

Furthermore, the threaded rods in the removable fin system will help to redirect the force from the runners that hold the fins in place. A preliminary structural FEA simulation is shown below in Figure 3.29.

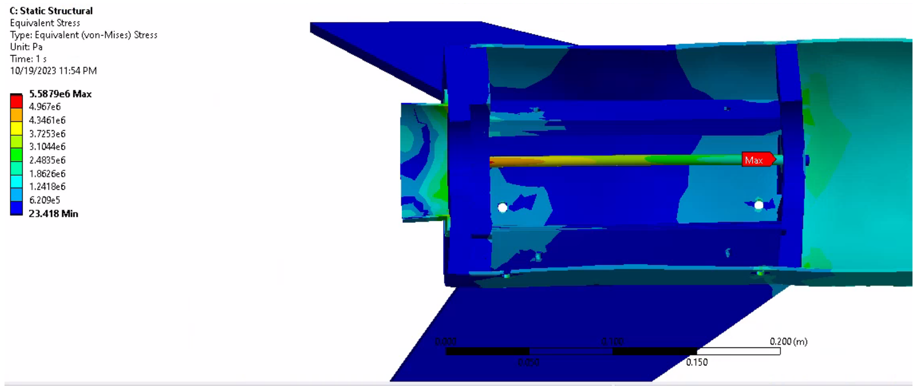


Figure 3.29: Structural FEA simulation performed on the removable fin system.

The results from the simulation concluded that the maximum stress which occurs on the 1/4-in. threaded rods in the removable fin system was 810.45 psi. The ultimate compressive strength was 24.65 ksi. With a factor of safety of 2, a positive safety margin of roughly 14.2 was achieved.

3.3.10 Fins

Design

The launch vehicle will utilize the swept fin profile discussed in Section 3.2.5. This profile will be swept at 55° to produce a leading edge length of 9.15 in. and a trailing edge length of 6.27 in. The root chord of the fins will be 8 in., the tip chord of the fins will be 4 in., and the span of the fins will be 5.25 in. The total mass of the swept fin profile would be 15.77 oz. per fin, therefore the total mass of all four fins would be 3.94 lbs. The leading edge and trailing edge of the fin profile will utilize bevels to reduce the bluff body effects of the frontal area of the fin. Each fin will be sanded on the surface to reduce skin friction drag on the fins during flight. By sweeping the fins aft, the center of pressure of the fins and the overall center of pressure of the launch vehicle is shifted aft, increasing the stability of the launch vehicle. By varying the span of the fins, the center of pressure may be refined to generate a stability value desired for the vehicle. A dimensioned profile of the fin design has been provided in Figure 3.30 below.

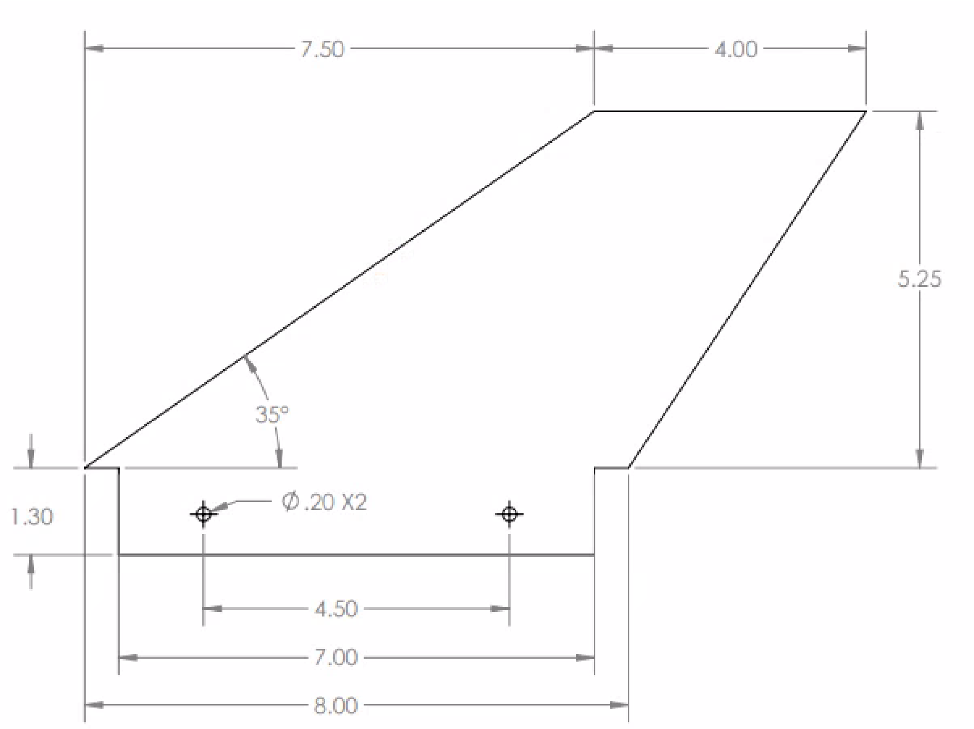


Figure 3.30: Dimensioned drawing of the swept fin profile with securement structure.

Material and Construction

The leading material for the fins is 1/8 in. thick G10 fiberglass. G10 fiberglass is typically used for fins in high-power rocketry for launch vehicles that achieve supersonic speeds, where aerodynamic loads and fin flutter make the fins more prone to failure. In the past, the team has experimented using 1/4 in. aircraft-grade birch plywood fins and 1/8 in. composite fins which had the same plywood as a core material and either carbon fiber or fiberglass as a layup material. While both composite layups are capable of withstanding the forces of subsonic flights, there is always a risk of failure upon the landing of the launch vehicle. Considering that the current design of the fins extend beyond the air-frame, it is important to choose a material that is highly resistant to cracking, shearing, and abrasion. The shape of the fins will be traced onto a sheet of 1/8 in. thick G10 fiberglass and then cut out with a scroll saw and sanded in NC State's Senior Design Lab under proper ventilation.

3.3.11 Tail Cone

The launch vehicle will utilize a flat aft profile. This will allow for the center of pressure to be furthest aft and ensure that the total mass of the airframe is minimized. This decision against a boat tail cone also negates the need for a complex manufacturing process to develop a tail cone.

3.3.12 Launch Vehicle Weight Estimates

The weight estimate for the leading launch vehicle design is approximately 42.5 lb. This weight was determined by a combination of weighing the items that were already available to the team, experimental calculations based on density, and manufacturer data sheets. The weight of each section of the launch vehicle is tabulated below in Table 3.4.

Table 3.4: Launch Vehicle Weight Estimates

Section	Weight (lb)
Nose Cone	6.01
Main Parachute/Payload Bay	14.91
AV Bay	3.25
Drogue Parachute Bay/Fin Can	18.33
Total	42.50

A breakdown of the weights of each of the sections is shown below in Table 3.5.

Table 3.5: Launch Vehicle Section Weight Estimates

Nose Cone	
Component Name	Weight (lb)
Airframe and Coupler	4.1
Removable Bulkhead	0.405
Latch	0.461
Electronics Sled	0.597
Quick Links	0.176
U-Bolt	0.088
Misc. Hardware	0.185
Total	6.01
Main Parachute/Payload Bay	
Component Name	Weight (lb)
Airframe	4.94
Shock Cord	1.10
Main Parachute	0.423
Payload Parachute	0.20
Nomex	0.309
Deployment Bag	0.258
Quick Links	0.176
SAIL	7.50
Total	14.91
AV Bay	
Component Name	Weight (lb)
Airframe and Coupler	1.48
Bulkheads	0.617
Blast Caps	0.106
Electronics Sled	0.597
U-Bolts	0.176
Misc. Hardware	0.272
Total	3.25
Drogue Parachute Bay/Fin Can	
Component Name	Weight (lb)
Airframe	4.62
Fin Can Bulkhead	0.321
Fin Can U-Bolt	0.088
Thrust Bulkhead	0.196
Thrust Plate	0.280
Drogue Parachute	0.139
Nomex	0.150
Fins	3.08
Removable Fin System	0.692
Misc. Hardware	0.723
Loaded Motor	8.04
Total	18.33

3.3.13 Motor Selection

Based on simulation data with the motors discussed in Section **??**, the Aerotech L1520T has been selected. This motor provides an average thrust of 352.45 lbs. yielding a thrust-to-weight ratio of 8.27 at liftoff. Previous successful team experience with this specific motor also makes this it highly desirable for competition selection.

3.4 Recovery Subsystem

3.4.1 Recovery Launch Preparation

Mounted onto the AV sled, which is housed in the AV bay, the launch vehicle contains two independent dual-deploy altimeters. These altimeters will be rigorously tested and carefully programmed in order to confirm successful use on launches. The altimeters will be programmed with their manufacturer-provided computer software. The apogee event, drogue parachute deployment event, main parachute deployment event, and time delay from descent events can be specified in the software. In order to test the altimeters, a pressure chamber created by the team is used to mimic the ascent and descent phases of the launch vehicle by varying the pressure. This ensures the altimeters are correctly identifying events based on the pressure measured detected. Upon successful testing and programming, the altimeters are mounted to the avionics sled.

When the launch vehicle is being assembled, the main parachute, drogue parachute, and nose cone/payload parachute will be attached to their respective shock cords via a metal quick link. The shock cords are attached to their subsequent section bulkheads by metal quick links as well. The nose cone/payload parachute and drogue parachute will be wrapped in a Nomex cloth to protect them from the black powder ejection charges. The main parachute will be in a deployment bag that protects it from the black powder charge. All of the shock cords will be insulated using biodegradable fire resistant blow-in insulation in order to protect the Kevlar webbing from the black powder ejection chargers.

Upon successful assembly of the launch vehicle, the altimeters and trackers will be armed on the launch pad. These devices are armed on the launch pad right before launch to reduce the risk of prematurely setting off ejection charges. The altimeters must be armed before the motor igniter is installed so that the launch vehicle will separate and descend under a parachute in the case of a premature motor ignition. Once the altimeters are armed, they will output an audible signal in the form of beeps. This will confirm the operation of the recovery system prior to flight. If an error is encountered during arming, launch will be halted until functionality of the altimeters is confirmed.

3.4.2 Description of Recovery Events

Once the items mentioned above are completed, the launch vehicle is ready to be launched. When the primary altimeter detects an apogee event, a signal will be sent to the primary drogue parachute charge. One second after the apogee event, the secondary altimeter will send a signal to the secondary drogue parachute charge in order to ensure separation in the event of the failure of the primary charge. After the ejection charges are fired, the 4-40 nylon shear pins connecting the AV bay and the drogue bay/fin can are broken. This will separate aft end of the launch vehicle, between the AV bay and the drogue bay/fin can. Upon separation, the drogue parachute is ejected from the bay attached to a shock cord.

The coupled AV bay, payload/main parachute bay, and nose cone will descend under drogue configured 5 ft above the top of the fin can. This ensures the separated sections do not hit each other and cause damage during descent. The descent velocity (discussed in Sections 3.7.7) is not high enough to cause structural damage from sudden deceleration when the main parachute is deployed (see Section 3.7.10). Once the primary altimeter senses that the launch vehicle has hit 800 ft AGL, a signal will be sent to the primary main parachute charge. One second later, the secondary altimeter will send a signal to the secondary main parachute charge. After the ejection charges are fired, the 4-40 nylon shear pins connecting the nose cone with the main parachute/payload bay and the forward bulkhead of the AV bay will break. This will separate the nose cone from the rest of the launch vehicle. Tethered to the nose cone are the payload, the nose cone parachute, and the deployment bag for the main parachute. The nose cone will begin to descend under the parachute independently. The payload will remain attached to the nose cone bulkhead via a shock cord. Once the separation occurs, the main parachute will come out after the payload, but it will remain attached to the AV bay bulkhead to aid the descent of the rest of the launch vehicle. This main deployment will allow the launch vehicle to slow down and meet the NASA SL Requirement 3.3. The nose cone will continue to descend independently until the command to drop the payload is given. After this, the payload will drop and the nose cone will continue to fall by itself under parachute. Shown below is a diagram depicting the major recovery events for the launch.

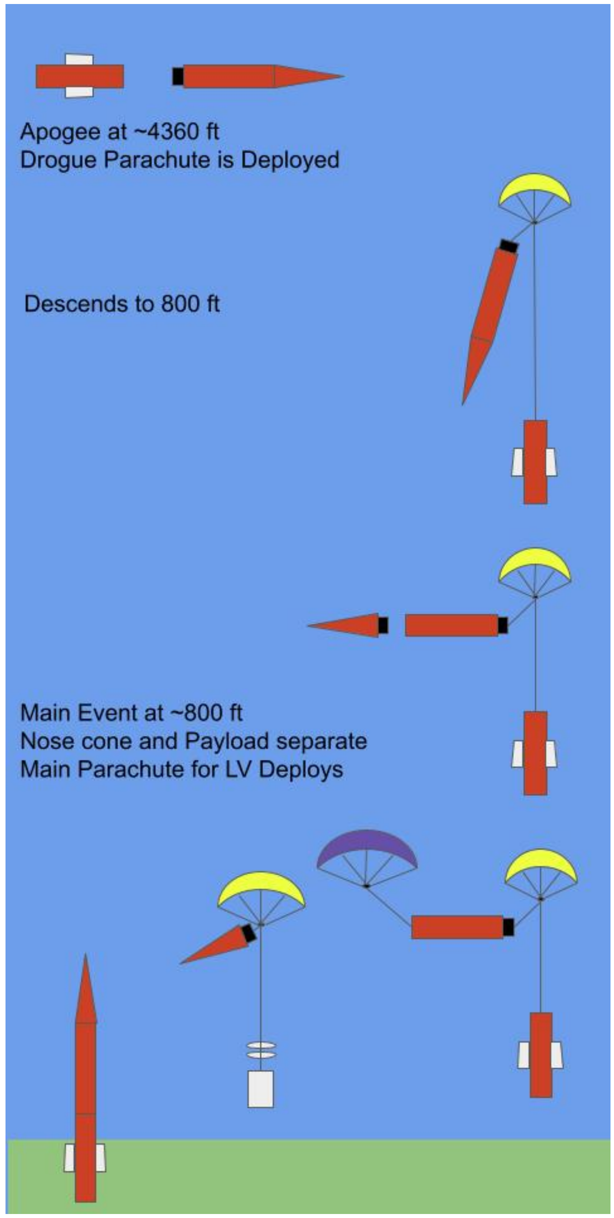


Figure 3.31: Diagram for Recovery Timeline

3.5 Alternative Recovery Components

3.5.1 Alternative Altimeters

In order ensure redundancy in the recovery system of the launch vehicle, two dual-deploy altimeters will be on board. These will ignite the black powder charges located on the outer surfaces of the AV bay to separate the launch vehicle at drogue and main events. Shown below in Table 3.6 are the altimeter options in consideration for the launch vehicle.

Table 3.6: Alternative Altimeter Options

Altimeter	Main Deployment Variability	Delay After Apogee	Altitude Logging Resolution	Dimensions	Data Recorded	Sampling Rate	Price	Owned by Club
Stratologger CF	100 - 9999 ft Increment: 1 ft	0 - 5 s Increment: 1 s	1 ft	L: 2” W: .84” H: 0.5”	Altitude, Velocity, Temperature	20/s	$69.95	Yes
RRC3	300 - 3000 ft Increment: 100 ft	1 - 30 s Increment: 1 s	1 ft	L: 3.92” W: .925”	Altitude, Velocity, Temperature, Time to Apogee	20/s	$101.33	Yes
Eggtimer Quasar	100 - 3000 ft	0 - 3 s Increment: .1 s or 3 - 30 s Increment: 1 s	1 ft	L: 3.816” W: 1.09” H: .5”	Altitude, Velocity, Milestone Events	20/s	$100.00	Yes
Entacore AIM	100 - 100,000 ft Increment: 1 ft	Available	1 ft	L: 2.56” W: 0.98” H: 0.59”	Altitude, Velocity, Temperature	10/s	$121.15	Yes
EasyMini	100 - 100,000 ft Increment: 100 ft on ascent, Increment: 10 ft on descent	Available	8 in	L: 1.5” W: 0.8” H: 0.6”	Altitude, Velocity, Acceleration, Voltage, Time to Apogee, Total Flight Time	100/s Ascent, 10/s Descent	$101.78	No

An important device that is necessary for the launch vehicle is the altimeter, as it controls the ejection charges needed to separate the launch vehicle and deploy its parachutes. Desired qualities for these devices such as reliability, precision, form factor, and ease of use are sought after when choosing the altimeters for the launch vehicle. Presented below is some basic research for each altimeter option along with their pros and cons.

Stratologger CF

The Stratologger CF is known for its ease of use as well as its smaller form factor compared to the other options such as the RRC3 and Quasar. It also has a larger main deployment altitude variation and smaller increments of 1 ft. A smaller form factor results in less space taken on the avionics sled, an added benefit especially for the sub-scale avionics sled. On the other hand, this club has had previous experience with this model failing to deploy parachutes accurately in the past and is less precise than the RRC3 sport altimeter. The Stratologger CF is shown in Figure 3.32 below.

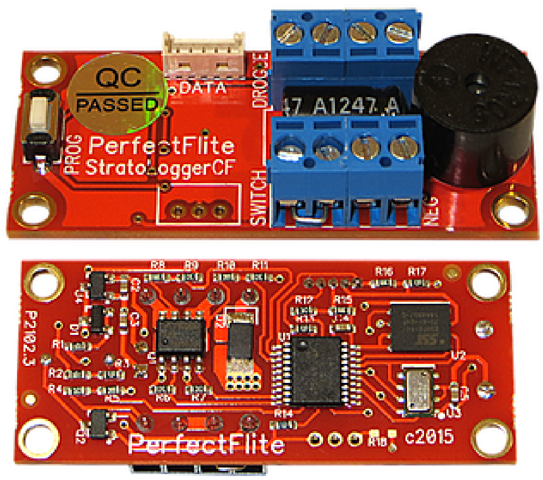


Figure 3.32: Strattologger CF by PerfectFlite

RRC3 Sport Altimeter

Known for its precision and ease of use, the RRC3 ”Sport” altimeter is an appealing option. It has several deployment modes that can be activated through both push button switches and dip switches on the device in addition to the typical programming through its dedicated computer software ”mDACS”. One notable disadvantage is its large form factor which can take up significant space on an AV sled. This is a problem specifically for the sub-scale avionics sled, although there are options to negate this variable. Though it has a smaller main deployment variability and larger increment, this is not a concern for competition requirements. The advantages, practicality, and experience in the club with this altimeter make it very favorable. Presented in Figure 3.33 below is an image of an RRC3 sport altimeter.

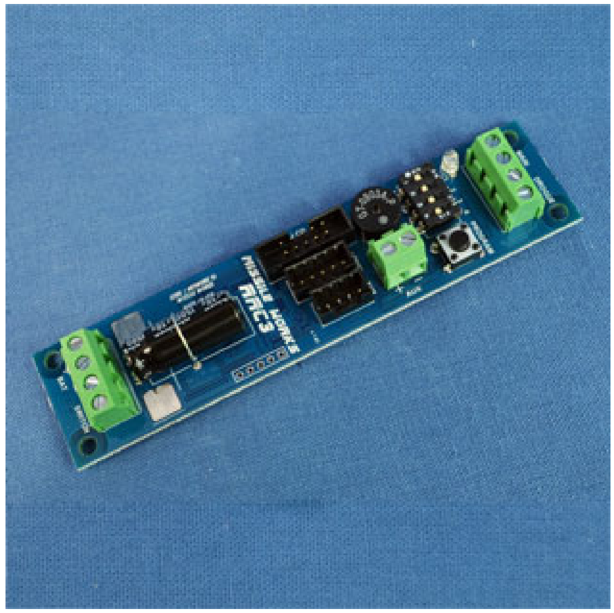


Figure 3.33: RRC3 Sport Altimeter by Missile Works

Eggtimer Quasar

The Eggtimer Quasar functions as a dual-deploy altimeter and a GPS tracker. It has roughly the same deployment variability as the RRC3 and offers similar precision as well. A small disadvantage is its form factor, as it is almost the same size as the RRC3 which can clutter the AV sled. The only real drawback is the difficulty of assembly as it comes in a kit of several components and must be assembled after delivery. However, the club already owns an assembled and functioning Quasar. Due to its practicality and efficiency of design, this altimeter is highly favored by the team. Shown below in Figure 3.34 is an image of an Eggtimer Quasar.

Figure 3.34: Quasar Altimeter/Tracker by Eggtimer

Entacore AIM

The Entacore AIM is another option for the altimeter, as it is easy to use. The main drawbacks consist of a lack of precision compared to other options and its form factor. Additionally, it has failed on occasion for parachute deployment, and the club has had trouble with the wiring of these devices. Due to the wiring complexity, precision drawbacks, and safety concerns, this device is not preferable. Presented in Figure 3.35 below is an image of an Entacore AIM.

Figure 3.35: Entacore AIM 3.0 by Entacore

EasyMini

The last option is the EasyMini dual-deploy altimeter. The advantages consist of its increased precision, and smaller form factor. On the other hand, there is no club experience with this device, and the wiring is stated to be complex. Furthermore, the club does not already own this altimeter, making it the most expensive option. Shown below in Figure 3.36 is an image of an EasyMini altimeter.

Figure 3.36: EasyMini by Altus Metrum

3.5.2 Tracking Device Alternatives

In order to adhere to the NASA SL Requirement 3.13.1, there will be a tracking device in each independent section of the launch vehicle so the sections can be located after landing. Due to the recovery system that has been selected, there will be one tracker secured in the nose cone which will descend separately from the other portion of the launch vehicle. The other launch vehicle sections (main parachute/payload bay, AV bay, and drogue parachute bay/fin can) will remain together via shock cord and will have a tracker in the AV bay. There are four trackers under consideration. Two of the options requiring an amateur radio license to operate. The tracker details are listed below in Table 3.7.

Table 3.7: Alternative Tracker Options

Tracker	License Required	Transmitter Power	Transmitter Frequency	Range	Owned by Club	Price	Comments
Big Red Bee 900	No	250 mW	900 MHz	6 Miles	Yes	$209.00	The simplest option.
Big Red Bee Beeline	Yes	100 mW	420 - 450 MHz	40 + Miles	No	$359.00	Various modes, too expensive.
Eggtimer Quasar	Yes	100 - 250 mW	420.250 MHz	11 + miles	Yes	$100	Functions as a GPS Tracker, and a dual-deploy altimeter.
Feather-weight GPS Tracker	No	60 mW	915 MHz	26 miles	No	$165.00	Can use with smartphone, though there will be difficulty setting up a pull-pin switch.

Big Red Bee 900

The first tracker option under consideration is the Big Red Bee 900. It is the simplest of the options due to ease of use, and no amateur radio license is required to operate at its frequency. The 900 MHz transmitter is attached to a microcontroller board with a GPS antenna on the board. It is powered by a single LiPo battery and a handheld receiver will simultaneously receive real-time longitude-latitude coordinates as well as display battery voltage. Once interfaced with a laptop, the transmitter locations can be displayed on Google Maps. Despite its limited range, it is the favored candidate for the independent nose cone section and is already owned by the club. Presented below in Figure 3.37 is an image of a Big Red Bee 900 tracker.

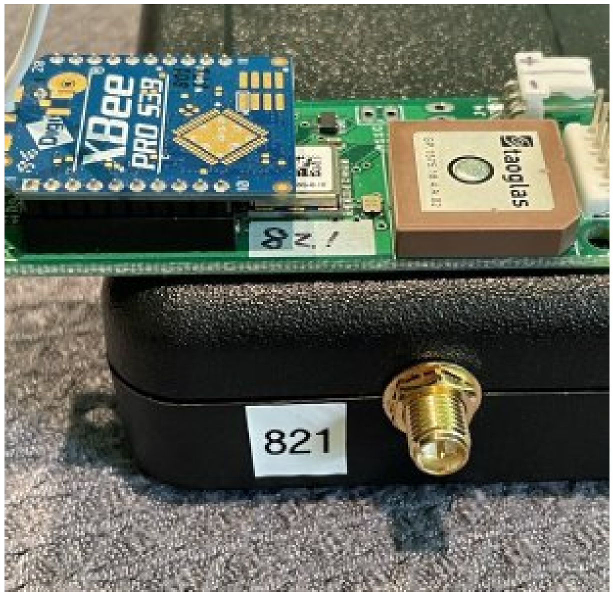


Figure 3.37: Big Red Bee 900 Tracker

Big Red Bee Beeline

The next tracker under consideration is the more advanced version of the Big Red Bee 900. The Big Red Bee Beeline has a much larger range and operates on an amateur radio frequency. This tracker emits a homing signal that would have to be found using a receiver and a directional antenna instead of using a laptop. Despite it being more accurate, the complexity is not worth the trouble, especially due to the price tag since as this model is not currently owned by the club. Shown below in Figure 3.38 is an image of the Big Red Bee Beeline tracker.

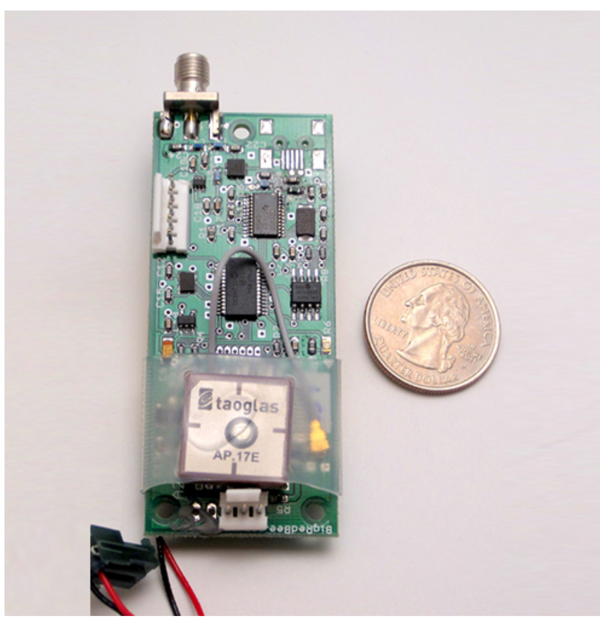


Figure 3.38: Big Red Bee Beeline Tracker

Eggtimer Quasar

As mentioned previously, the Eggtimer Quasar functions as a dual-deploy altimeter and a tracker. It transmits signals on a 70 cm band and requires an amateur radio license to operate. This tracker is paired with a handheld Eggfinder LCD receiver that will pick up the transmitted location from the tracker. This device automatically will detect launch due to its dual altimeter functionality, at which point the GPS is armed. After the flight it will transmit its location five seconds after it senses no movement of the launch vehicle and will continue to transmit to the receiver until it is deactivated. It can be programmed by using its WiFi functionality to connect to a nearby smart phone. This tracker has been used by the team before and is already owned by the club. Due to its functionally as an altimeter and tracker, along with previous club experience, this tracker is favored for the launch vehicle. In Figure 3.39 below is an image of the Eggtimer Quasar.

Figure 3.39: Quasar Altimeter/Tracker by Eggfinder

Featherweight GPS

The last tracker under consideration is the Featherweight GPS tracker, as it is one of the simpler options. A single LiPo battery is used to power the unit which consists of an antenna and a GPS tracker. There is a dedicated receiver for this tracker, and it can be connected to a smartphone. The location data can be imported into Google Maps to analyze flight paths. It operates on a 900 MHz frequency so no amateur radio license is required. A drawback is that it is not already owned by the club and the club has no prior experience using this device, therefore it is not favored by the team for competition. Shown below in Figure 3.40 is an image of a Featherweight GPS tracker.

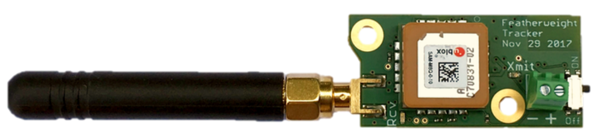


Figure 3.40: Featherweight GPS Tracker

3.5.3 Altimeter Arming Alternatives

In order to comply with NASA SL Requirements 3.6 and 3.7, screw switches and pull-pin switches are being considered for the design. Both of these switches can be accessed from the exterior of the launch vehicle and can be locked in the on position.

Screw Switches

Screw switches are easy to assemble and they are cheap. A small PCB is used as a break in the circuit where two wires are soldered onto the board and a 3/16 in. screw can be screwed in and out of the terminal. The terminal is designed so that the raising and lowering of this screw will complete or break the circuit. The switch will be located on the avionics sled, where a drilled hole in the air-frame of the avionics bay is made directly on top of the switch. A screwdriver will be inserted into this machined hole and can tighten the screw so the circuit completes. This is an appealing option for altimeter arming and has been used by the team in the past. The only disadvantages are that it is difficult to accurately machine the hole over the switch and it is difficult to get the screwdriver in the switch hole as the board is not parallel to the flat surface of the sled. As a result, it is not the favored option for altimeter arming. Figure 3.41 is an image of a screw switch.

Figure 3.41: 6/32” Screw Switch by Missile Works

Pull-Pin Switches

The other option being considered is the pull-pin switch. A pin is placed into a limit switch that keeps the circuit open. This limit switch is screwed into the sled firmly. A hole is machined into the AV bay airframe aligned to the switch so the pin can be inserted into the switch from the exterior of the launch vehicle. Upon removal of the pin, the circuit completes and the altimeters and trackers will be powered on. Due to the ease of access for arming the altimeters on the launch pad, this is the favored method of altimeter arming. Figure 3.42 is an image of a pull-pin switch.

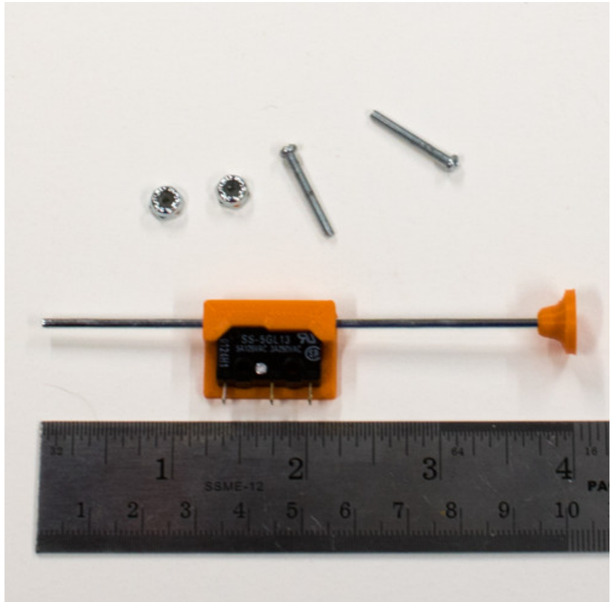


Figure 3.42: Pull-pin switch by Lab Rat Rocketry.

3.5.4 Avionics Sled Material

The altimeters, tracker and their respective batteries are secured to the AV sled. For this reason, the sled must be capable of securing these devices firmly while also being able to sustain up to 15 Gs of force during both launch and recovery events. In addition, these devices must be easily accessible before and after the launch. With these constraints in mind, there are two the material options to consider.

Birch Plywood

Birch plywood has been a popular choice for sled material over the years due to the ease of laser cutting and epoxying. This material for a sled makes it able to withstand the large forces the launch vehicles experiences without damaging the avionics. It is fabricated by laser cutting the wood into jigsaw shapes with epoxy being used to secure each half of the jigsaw. Some advantages of this include the sled being lightweight and easy to drill into so the avionics can be mounted.

PETG 3D Printing Filament

There are many benefits to a PETG sled design. It is easy to model and 3D print, eliminating the complexity of the jigsaw design required to fabricate a plywood sled. This material is also easy to drill into with minimal issues and is lighter than the birch plywood design. Shown below in Figure 3.43 is a CAD model of the full scale AV sled design.

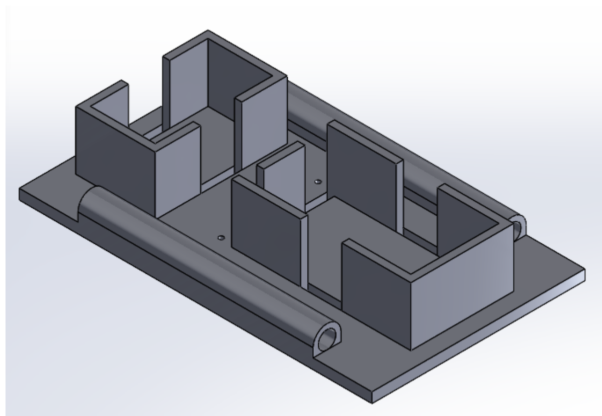


Figure 3.43: Full scale Avionics Sled

3.5.5 Drogue Parachute Alternatives

The main goal of the drogue parachute is to slow down the launch vehicle such that the sudden shock from the main deployment does not snap the shock cord or damage the launch vehicle's structure. As such, the significant factor considered is the descent velocity of the vehicle under a drogue parachute. The Team-Derived Requirement RF 3 states that the launch vehicle shall not exceed a descent velocity of 120 ft/s for the reasons mentioned above. On the other hand, the drogue parachute cannot be too big either, as the launch vehicle descending at a much slower rate will raise concerns about meeting competition descent time and drift distance from NASA SL Requirements 3.11 and 3.12. There is a fine balance the team tried to find when selecting the drogue parachute of the launch vehicle.

The descent velocity of the launch vehicle under drogue descent can be found using Equation 4 below. Let m be the burnout mass of the launch vehicle, g is earth's gravitational acceleration constant, A is the parachute's area, C_D is the drag coefficient of the parachute, ρ is the density of the air, and V_D is the descent velocity of the launch vehicle.

$$v_d = \sqrt{\frac{2gm}{SC_D\rho}} \tag{4}$$

Listed in Table 3.8 below are some of the options considered for the drogue parachute of the launch vehicle, all of which are listed in the club's parachute catalog. In the table are the descent velocities, descent time to main deployment, and drift distance, which were calculated using the equations in Sections 3.7.8 and 3.7.9.

Table 3.8: Alternative Drogue Parachute Options

Parachute	Drag Coefficient	Descent Velocity	Descent time from Apogee to Main Deployment	Wind Drift from Apogee to Main Deployment (20 mph)	Owned by Club
Fruity Chutes 12” Classic Elliptical	1.339	175.21 ft/s	20.33 s	596.35 ft	No
Fruity Chutes 15” Compact Elliptical	1.5	101.31 ft/s	35.17 s	1031.65 ft	Yes
Fruity Chutes 18” Classic Elliptical	1.428	113.10 ft/s	31.50 s	924.00 ft	Yes
Fruity Chutes 24” Classic Elliptical	1.473	83.53 ft/s	42.66 s	1251.36 ft	Yes

With the desired qualities mentioned above, the favored option for the drogue parachute is Fruity Chutes 18” Classic Elliptical. This is because it stays under the team-derived drogue descent velocity of 120 ft/s, while also having a reasonable descent time. Since the 12” parachute descends at 175.21 ft/s, it is not a viable option. The 24” has too long of a descent time until main deployment, so it will not be selected either. While the 15” may work, the 18” is more favorable only due to its lower descent time.

3.5.6 Main Parachute Alternatives

The goal of the main parachute is to slow the launch vehicle down enough so there is no damage upon landing. It is mainly determined by the kinetic energy of the launch vehicle when landing. This parameter is a factor of the descent velocity of the launch vehicle and its mass. Equation 4 in Section 3.5.5 was used to calculate descent velocity. Additional constraints are that the parachute cannot be too big because it could fail to meet descent time and drift distance requirements.

Listed below are some of the options considered for the main parachute of the launch vehicle, all of which are listed in the club’s parachute catalog. In the table are the descent velocities, kinetic energies for the heaviest section of the launch vehicle (fin can), descent time from main deployment, and drift distance, which were calculated using the equations in Section 3.7.7, 3.7.8, and 3.7.9. It is important to state that the drift distances noted were calculated in the RAS Aero II model.

Table 3.9: Alternative Main Parachute Options

Parachute	Drag Coefficient	Descent Velocity	Kinetic Energy	Descent time from Main Deployment	Wind Drift from Main Deployment (20 mph)	Owned by Club
Fruity Chutes 72” Iris Ultra Compact	2.033	18.95 ft/s	77.96 ft-lb	42.23 s	1238.75 ft	No
Fruity Chutes 84” Iris Ultra Compact	2.135	15.85 ft/s	54.56 ft-lb	50.48 s	1480.75 ft	No
Fruity Chutes 96” Iris Ultra Compact	2.088	14.02 ft/s	43.71 ft-lb	57.048 s	1673.41 ft	No
Fruity Chutes 120” Iris Ultra Compact	2.105	11.17 ft/s	27.11 ft-lb	71.61 s	2100.56 ft	Yes

From the options above, the 72” parachute is not viable due to the kinetic energy being too high. The 120” parachute meets the kinetic energy requirement but has too long of a descent time. For these reasons, the Fruity Chutes 84” Iris Ultra Compact is the favorable candidate for the main parachute, as it meets the kinetic energy requirement for bonus points, and has a better descent time than the 96” parachute. Additionally, the drift distance for the 96” and the 120” is too large.

3.5.7 Nose Cone Parachute Alternatives

It is important to mention that the payload itself does not have a parachute attached to it, as the parachute is connected to the nose cone. The payload is looped on the shock cord connected to the nose cone, and deployed via a latch. It is imperative that the payload and nose cone do not fall too fast because the payload needs to be stable in order to begin its deployment phase. As such, the significant factor considered here is the descent velocity of the nose cone and payload when they are still tethered to each other. The descent velocity is found using the same equation shown in the drogue parachute section, with the mass of the section including the nose cone, parachute, shock cord, and payload.

Listed below are some of the options considered for the parachute of the nose cone, all of which are listed in the club’s parachute catalog. Table 3.10 lists the descent velocities, and descent time from main deployment for the nose cone with the payload still attached, which are calculated using the equations in Sections 3.5.5 and 3.7.8. Table 3.11 lists the descent/landing parameters of just the nose cone since the payload will be dropped at a certain altitude after main deployment. It consists of the nose cone’s descent velocity, kinetic energy, descent time from payload deployment ($\sim$450 ft), and drift distance from main deployment (which is a factor of the descent velocity with and without the payload).

Table 3.10: Alternative Nose Cone Parachute Options with Payload Attached

Parachute	Drag Coefficient	Descent Velocity	Descent time from Main Deployment	Owned by Club
Fruity Chutes 36” Compact Elliptical	1.428	33.97 ft/s	10.30 s	Yes
Fruity Chutes 42” Classic Elliptical	1.421	29.19 ft/s	11.99 s	No
Fruity Chutes 48” Classic Elliptical	1.439	25.38 ft/s	13.79 s	Yes

Table 3.11: Alternative Nose Cone Parachute Options without Payload Attached

Parachute	Drag Coefficient	Descent Velocity	Kinetic Energy	Descent time from Payload Deployment	Wind Drift from Main Deployment (20 mph)	Owned by Club
Fruity Chutes 36” Compact Elliptical	1.428	24.74 ft/s	69.90 ft-lb	18.19 s	835.71 ft	Yes
Fruity Chutes 42” Classic Elliptical	1.421	21.26 ft/s	51.62 ft-lb	21.16 s	972.53 ft	No
Fruity Chutes 48” Classic Elliptical	1.439	18.49 ft/s	39.01 ft-lb	24.344 s	1118.65 ft	Yes

All of these options are viable for the nose cone and payload parachute, as they meet all of the recovery requirements. The deciding factor will be the descent velocity while the payload is still attached. A descent velocity under 29.33 ft/s is preferable for the payload team, as it allows the payload to deploy under a stabilized condition. As such, the favored candidate for the nose cone and payload parachute is the Fruity Chutes 48” Classical Elliptical.

3.5.8 Shock Chord Material

The launch vehicle will be tethered together during descent via shock cords. These need to be able to withstand the sudden shock force when the main parachute deploys and hold the weight of the launch vehicle. Additionally, they must be able to survive the heat, force, and pressure from the black powder ejection charges.

The two options for shock cord material under consideration are the 1 in. nylon webbing, and 5/8 in. tubular Kevlar webbing. The Kevlar webbing is known to be a stronger material, as its maximum load rating is much higher. On the other side, nylon webbing is a much cheaper option, while still being a strong material. This advantage of the nylon webbing is eliminated though due to the club already owning several lengths. As explained in Section 3.7.10, the maximum load the shock cord experiences will be 299.47 lbf.

3.6 Leading Recovery Design

3.6.1 Avionics Bay

The AV bay will be located aft of the main parachute/ payload bay, and forward of the drogue parachute bay in the fin can. There will be a bulkhead on each side of the bay to protect the electronics, and black powder charge canisters will be mounted on the outside of them. Two charges will be on each bulkhead, the aft side for drogue deployment at apogee, and the forward side for main parachute and payload deployment at 800 ft. The bulkheads will be secured at the ends of the bay using threaded rods and nuts.

3.6.2 Avionics Sled

The AV sled will be 3D printed using PETG filament and will be secured into place on the rods via nuts. The sled will house two redundant altimeters, a tracking device, and batteries. A sheet of aluminum foil will be placed on the flat surface of the sled to reduce interference between the tracking system and the altimeters.

3.6.3 Recovery Electronics

Housed in the AV sled will be an RRC3 sport altimeter, an Eggtimer Quasar, and their batteries. As mentioned in Section 3.5, the Quasar functions as an altimeter and a GPS tracker, and both functions will be utilized. The RRC3 is chosen due to its precision, ease of use, and prior use by the team. The Eggtimer Quasar was chosen due to its precise altimeter, its accurate GPS tracker, simplicity in receiving the transmitter's signal on a handheld receiver, and its prior use by the team. For the nose cone tracker, the Big Red Bee 900 shall be the GPS tracker. Known for its simplicity, and small design, it can easily be fit onto the nose cone avionics sled and used with ease. The team also has experience with the Big Red Bee 900, making it the leading tracker for the nose cone and payload independent section.

3.6.4 Proof of Redundancy

Shown below in Figure 3.44 is an electronics flow diagram for the altimeters and tracking devices used in the recovery subsystem of the launch vehicle.

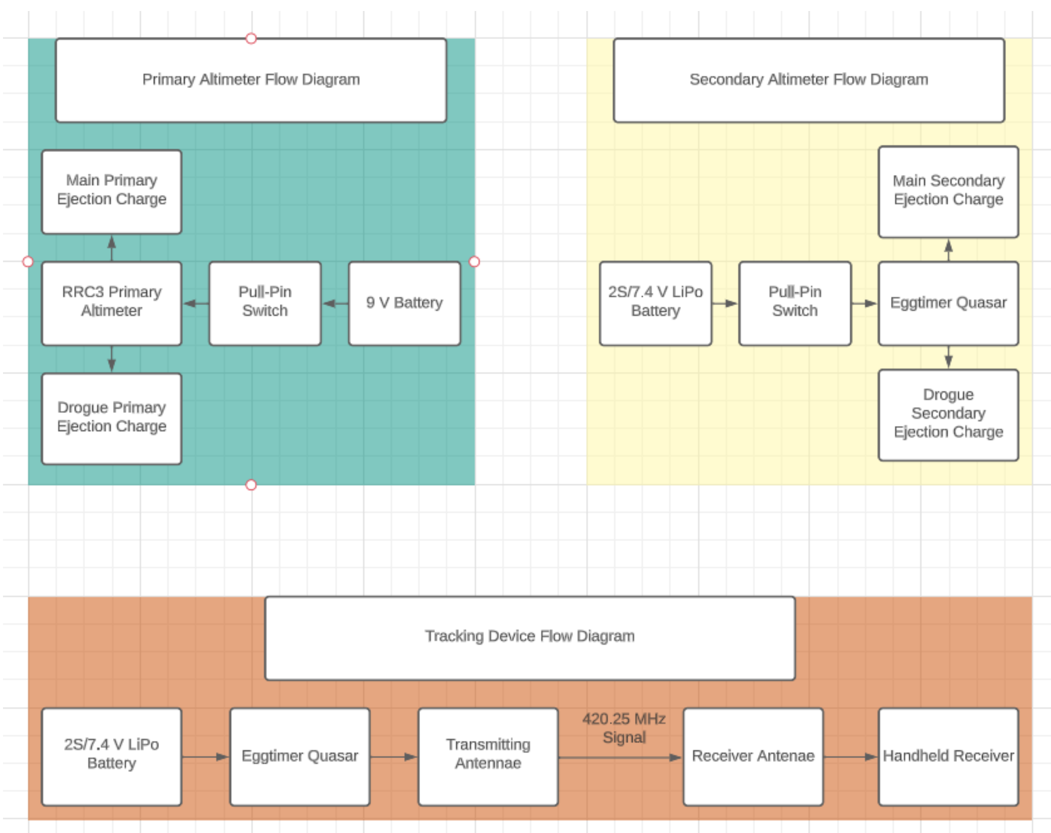


Figure 3.44: Avionics Flow Diagram

The top left figure with the teal background is the electronic flow diagram for the primary altimeter. It includes the altimeter, the battery, the mechanical arming switch, and the ejection charges to separate the launch vehicle. This altimeter will be the competition altimeter. The first charge will separate the fin can and drogue bay from the AV bay that stays coupled with the payload/main parachute bay. Additionally, the second charge will separate the nose cone and payload from the launch vehicle, and the main parachute will deploy for the launch vehicle. Upon successful event 2, the payload parachute attached to the nose cone will also deploy.

The top right figure with the yellow background is for the secondary altimeter. It features the exact components, and operates the same way as the primary altimeter, though the charges are fired with a 1-second delay. This addition brings redundancy to the design, ensuring successful separation during descent.

The figure at the bottom with the red background is the electronic flow diagram of the tracking devices. This diagram is accurate for both the launch vehicle tracker and the nose cone tracker, though the frequency of the transmitter may vary. The components on the vehicle include the battery, the GPS tracker, and the transmitter antennae. For the ground components, there is a receiving antennae and a handheld receiver.

The point of having two independent altimeters is to create redundancy in the system to ensure the launch vehicle separates during flight. If the primary altimeter were to fail, the secondary altimeter powered by a separate source can take over and perform separation successfully. Each altimeter connects to a terminal block on each AV bay bulkhead through soldered connections. The ejection charges are held in PVC canisters on the other side of the bulkhead and are wired through the bulkhead with an e-match. Plumbers putty is used to seal the hole once the wire is through so the avionics are sealed from the ejection charges.

3.6.5 Shock Cord Selection and Sizing

The 5/8 in. Kevlar shock cord is the favorable option for the launch vehicle shock cord material due to its high strength, durability, and flame resistance.

When considering the length of the shock cord for the launch vehicle, a good rule of thumb is to have the shock cord length 3 to 5 times the length of the launch vehicle. As mentioned earlier in the report, the total length of the launch vehicle will be 105.43 in., so the expected range of the shock cord shall be from 316.29 to 527.15 in.. The club already owns enough length of the Kevlar webbing for this constraint so no additional shock cord will need to be purchased. It is important to mention again that the launch vehicle is splitting into two independent sections and there will be a shock cord tether for each independent section. The secondary shock cord will tether the nose cone and the payload together and is ejected from the launch vehicle at the main deployment event. The main shock cord will tether the payload/main parachute bay, the AV bay, and the fin can together. The leading design is to have an 8 ft. secondary shock cord for the nose cone and payload, an 8 ft. shock cord connecting the main parachute with the payload/main parachute bay and AV bay, and an 18 ft shock cord connecting the AV bay to the drogue parachute and drogue bay/fin can.

It is imperative that the tethered sections are not at the same heights when the launch vehicle is descending. If this is the case, the sections will collide with each other and could cause damage to the sections, since the launch vehicle can swing and move around while descending. For this reason, a 5 ft. gap of separation is included in the length calculations.

This system is designed such that the main parachute lies above the AV bay. While the vehicle descends under drogue, the upper half of the launch vehicle will lie above the fin can section. This will prevent anything from impeding the deployment of the main parachute and eliminates the risk of puncturing the parachute as a result of impact with another section. For the independent nose cone, the payload will descend at least 5 ft. below the top of the nose cone so the payload can safely deploy and not hit the nose cone.

Shown below in Figure 3.45 is the leading shock cord length for each independent section of the launch vehicle, along with the parachute placement along the shock cord.

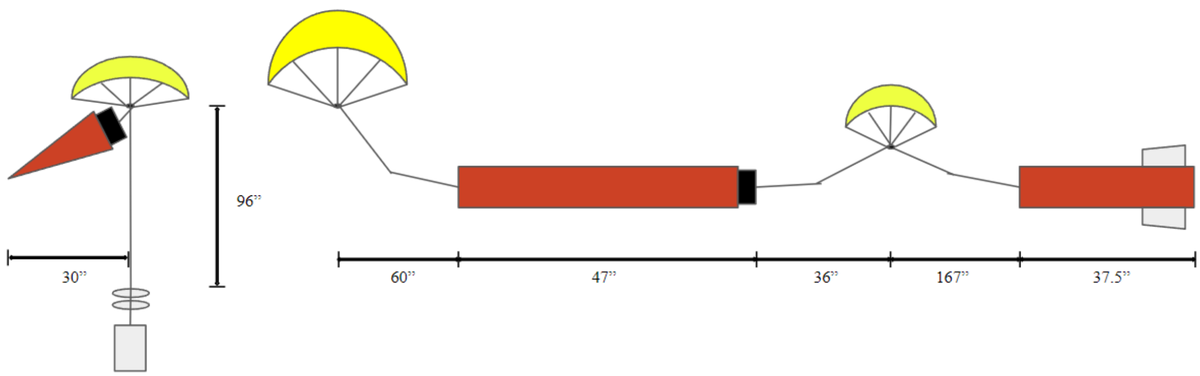


Figure 3.45: Shock Cord and Parachute Placement Diagram

3.6.6 Parachute Selection

The Fruity Chutes 18" Classic Elliptical was chosen for the drogue parachute. It has a descent velocity of 113.10 ft/s, resulting in a descent time to main deployment of 31.50 seconds, and a maximum wind drift to main deployment of 924 ft. These specifications satisfy Team-Derived Requirement RF 3.

The Fruity Chutes 84" Iris Ultra Compact was selected for the main parachute. This parachute has a maximum drift distance from deployment of 1238.15 ft and a descent time of 50.48 s. Additionally, the fin can and payload bay will land with kinetic energies of 54.56 ft-lb and 41.82 ft-lb respectively, satisfying NASA SL Requirement 3.3.

The Fruity Chutes 48" Classic Elliptical was chosen for the nose cone/payload parachute. With the payload attached, this section has a descent speed of 18.49 ft/s, a drift distance of 1118.65 ft and a descent time of 38.134 s. The section will land with a kinetic energy of 39 ft-lb, satisfying NASA SL Requirement 3.3.

The total descent time from apogee is approximately 81.98 seconds, and the maximum drift distance is approximately 2404.76 ft. For the nose cone parachute, the total descent time from deployment with main is 38.134 seconds, and the maximum drift distance from main deployment is 1118.65 ft, resulting in a total drift distance of 2042.65 ft from apogee. Both parachute combinations satisfy NASA SL Requirements 3.11 and 3.12.

3.6.7 Ejection Charge Sizing

When determining the mass of the black powder charge, the varying factor is the empty volume in the compartment that is separating. This empty volume is found by taking the volume of the compartment and subtracting the volume of the shock cord, parachute, and any other components in the section. The other factor to be found is the pressure that will separate the section by breaking the shear pins. Once the empty volume and the pressure are calculated, the ideal gas law equation can be used to determine the mass needed for black powder. The ideal gas law is shown in Equation 5 below.

$$PV = mRT \tag{5}$$

Let P represent the pressure needed to separate the section, V represents the empty volume of the section, m is the mass of the black powder, R is the gas constant of black powder combustion products, and T is the temperature of black powder during combustion. This temperature is known to the 3307 degrees Rankine, the gas constant is 22.16 ft-lb, and the calculated pressure is 10 psi.

The recovery system includes a secondary black powder charge to ensure redundancy for launch vehicle separation. Each secondary charge will have an additional 0.5 grams than the primary charge for that section which ensures separation in case of primary charge failure and it is not large enough to cause damage. Shown below in Table 3.12 are the calculated ejection charge sizes.

Table 3.12: Ejection Charge Sizing for each Separating Section

Point of Separation	Volume of Section	Primary Charge Mass	Secondary Charge Mass
Nose Cone and Main Parachute/ Payload Bay	389.51 cubic in.	2.01 grams	2.51 grams
Avionics Bay and Drogue Bay/ Fin Can	136.21 cubic in.	.70 grams	1.2 grams

The ejection charge itself shall be a 777-grade FFF black powder. Its very fine grains allow for faster combustion. A fast combustion is preferred since it allows for a clean separation and leaves less unignited black powder scattered in the section.

In order to comply with NASA SL Requirement 3.2, a ground ejection test will be performed for both primary charges before launch. A successful ejection test demonstrates that the launch vehicle will be able to separate and the ejection charges are sized appropriately. If the launch vehicle fails to separate during the ejection test, an additional 0.2 grams will be added to the primary charge, and the test will be repeated again. This process will continue until safe, successful separation is ensured.

3.7 Mission Performance Predictions

3.7.1 Launch Day Target Apogee

The target apogee for the launch vehicle is 4050 ft. This value was calculated through the use of multiple simulation suites in accordance with Team-Derived Requirement LVF 3. The analysis methodology used for this determination has been outlined in Figure 3.46 below.

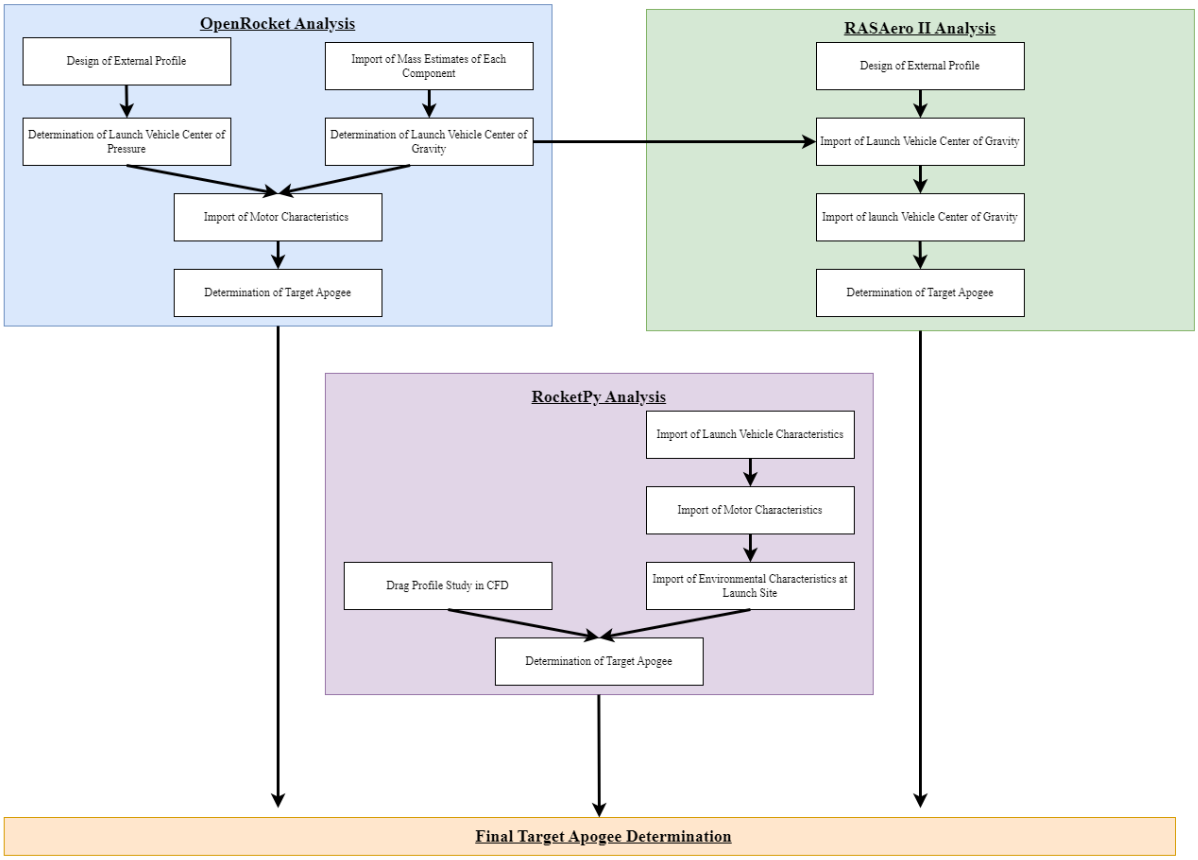


Figure 3.46: Analysis methodology for determination of the launch vehicle target apogee.

Each simulation suite will utilize the conditions outlined in Table 3.13 below for apogee verification unless otherwise stated.

Table 3.13: Defined Launch Conditions

Parameter	Value	Justification
Wind Speed	10 mph.	Enveloping Value
Launch Rail Length	12 ft.	NASA 1.12
Launch Rail Cant	5°	NASA 1.12

OpenRocket Prediction

OpenRocket was used to accurately determine the center of gravity of the launch vehicle given the required mass components and payload. All mass components of the full scale launch vehicle were catalogued by the appropriate subsystem member and added to the launch vehicle in the appropriate location. Motor selection was based on the internal library of AeroTech motors preinstalled within OpenRocket. The complete model of the full scale launch vehicle is shown in Figure 3.47 below.

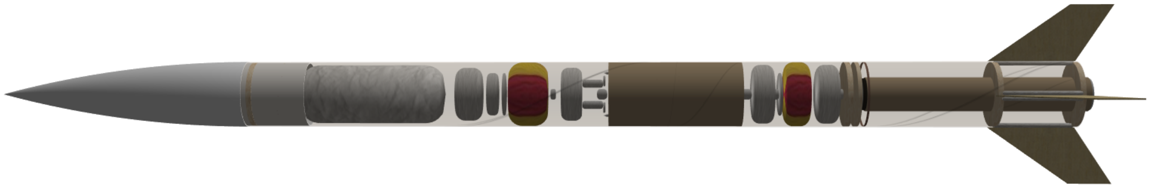


Figure 3.47: 3D figure view of the full scale launch vehicle in OpenRocket.

Given the environmental conditions listed in Table 3.13, the flight profile shown in Figure 3.48 was exported from OpenRocket.

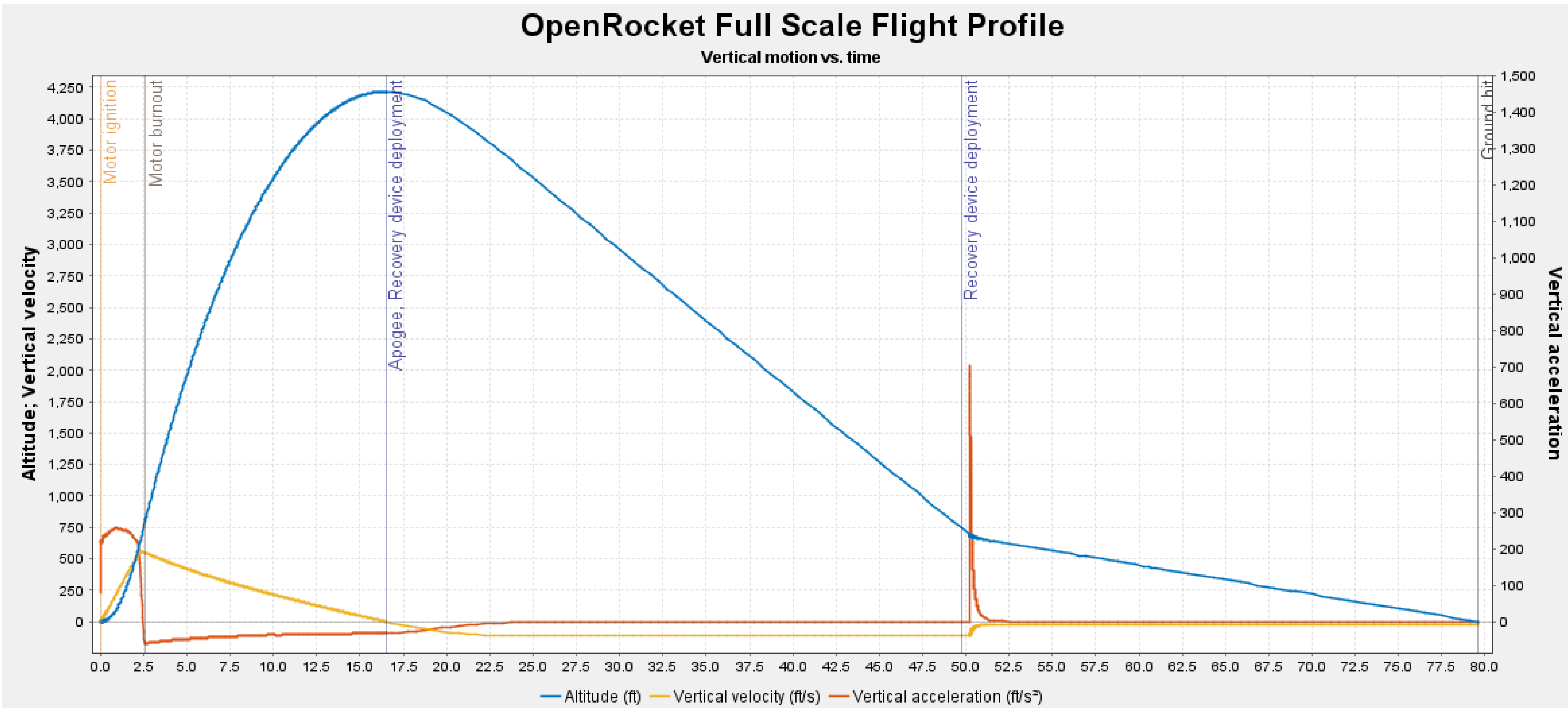


Figure 3.48: OpenRocket flight profile of the full scale launch vehicle using an AeroTech L1520T motor.

From this flight profile, the launch vehicle reaches an apogee of 4215.8 ft with a maximum velocity of 567.6 ft/sec. The maximum acceleration of the launch vehicle is 8.175 G's, which satisfies Team-Derived Requirement LVD 7.

RASAero II Prediction

To confirm the apogee prediction, the use of a compressible atmosphere model is required. The RASAero II simulation suite developed by Charles E. Rogers and David Cooper includes variable drag coefficient and center of pressure across the mach number suite of the launch vehcile's trajectory. These aerodynamic calculations have been calibrated against NACA and NASA sounding rocket and wind tunnel data to improve the accuracy of results.

To build the full scale launch vehicle within RASAero II, the external contour of the launch vehicle must be developed. This can be done using the basic geometry definition within the simulation interface. Once the external contour has been defined, the resulting center of pressure versus mach number can be determined. The center of pressure does not vary significantly for subsonic vehicle flight, but in the cases of transonic and supersonic flight, the center of pressure will shift forward as mach number increases until the vehicle becomes unstable. Once the center of pressure is known, the center of gravity can be manually entered from OpenRocket mass predictions. Finally, the motor thrust definition file and environmental conditions of the launch rail and atmosphere can be defined. Depicted in Figure 3.49 below is the full scale launch vehicle modeled in RASAero II.

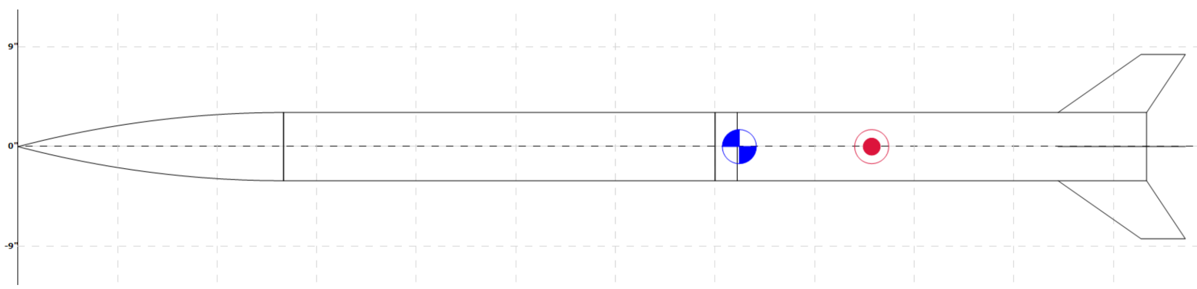


Figure 3.49: 2D planar geometry of the full scale launch vehicle in RASAeroII.

Running the flight profile simulation in RASAero II yields the results depicted in Figure 3.50.

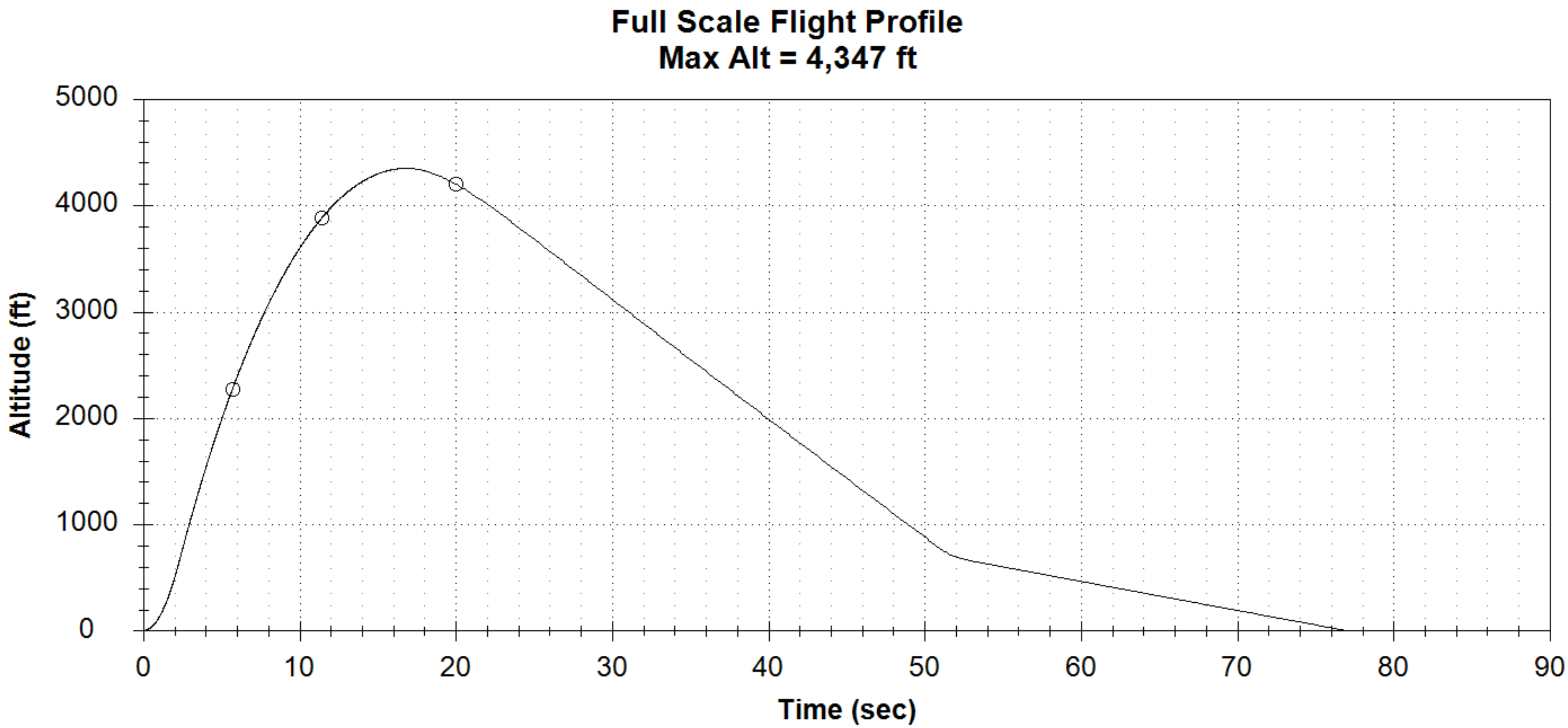


Figure 3.50: RASAero II flight profile of the full scale launch vehicle using an AeroTech L1520T motor.

From this flight profile, the maximum altitude achieved by the launch vehicle has been predicted to be 4347 ft with a maximum velocity of 575.3 feet per second. The maximum acceleration achieved is predicted to be 8.23 G's. All of these values fall within NASA and Team-Derived Requirements for the launch vehicle.

RocketPy Prediction

To further confirm the apogee prediction, modeling the drag of the launch vehicle as a function of velocity is required. To use this drag profile within the apogee prediction, a Python module known as RocketPy can be used. This Python library allows for 6-degree-of-freedom simulation, real-time atmospheric conditions during accent, and Monte Carlo optimization of the launch vehicle. The launch vehicle is built from a class structure which allows for high-fidelity refinement of the launch vehicle's specific characteristics. A simulation is built by first supplying motor grain geometry characteristics. The thrust curve of the motor is then built, including expected losses within the nozzle and variation in the grain structure, causing variation in thrust performance. The launch vehicle is then built using methods that allow for specific integration of nose cone, fins, and airframe moments of inertia and geometry definition. Finally, an atmospheric model can be uploaded, such as the Global Forecast System (GFS) upper-level winds model, and used in simulation. As the day of launch approaches, the specific forecast for the launch field may be used to determine the final ballasts measurements that may be needed to achieve the desired apogee. A block diagram supplied by the developers of the RocketPy library is depicted in Figure 3.51 to further explain the methodology of the launch vehicle trajectory simulation.

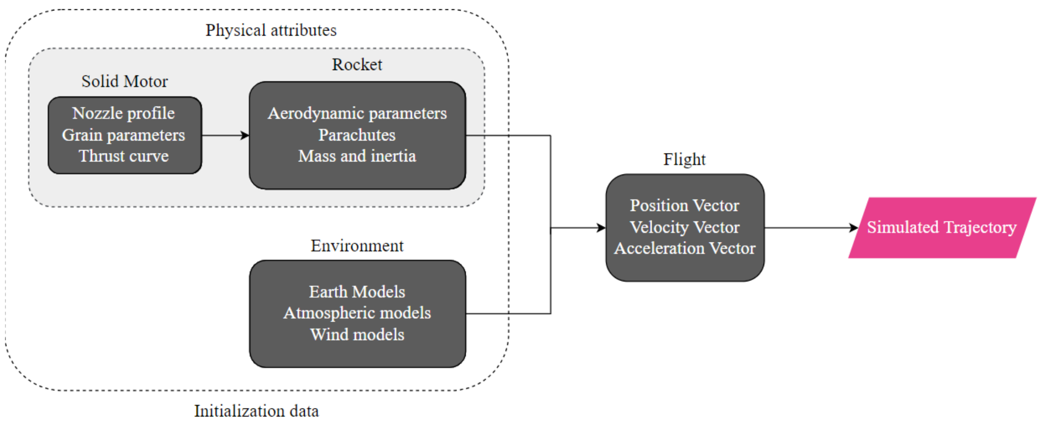


Figure 3.51: Block Diagram of Rocketpy Simulation Methodology.

For the determination of the drag profile of the launch vehicle, ANSYS Fluent was used with the supplied 3D model of the launch vehicle's external structure. The SST k-ω fluid model was used in fluid velocity increments of 45 mph. The external walls of the launch vehicle geometry were then used in a force report to determine the total drag on the launch vehicle as the velocity was increased until the simulation reached the maximum vehicle velocity from the two previous simulations. A depiction of the velocity profile used in the determination of the drag force of the launch vehicle is shown in Figure 3.52 below.

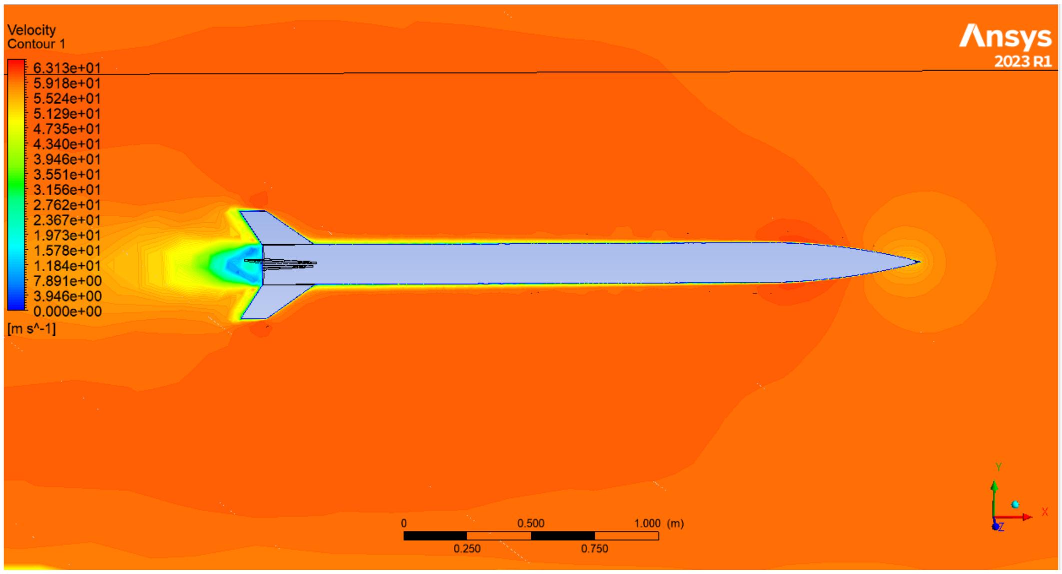


Figure 3.52: Velocity profile of the launch vehicle at 575 ft/sec. using ANSYS Fluent.

A depiction of the iterative approach ANSYS Fluent utilized to determine the drag force on the launch vehicle is shown in Figure 3.53 below.

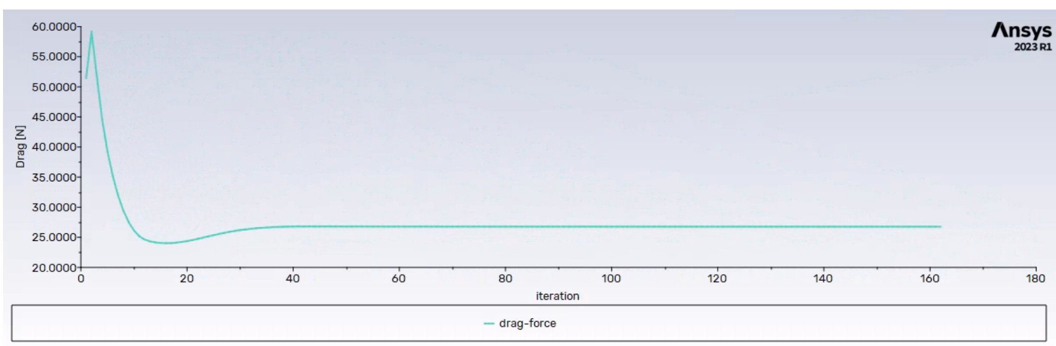


Figure 3.53: Iterative approach to drag force determination using ANSYS Fluent.

From this analysis, the drag force predicted by each of the models can be co-plotted against the velocity of the launch vehicle.

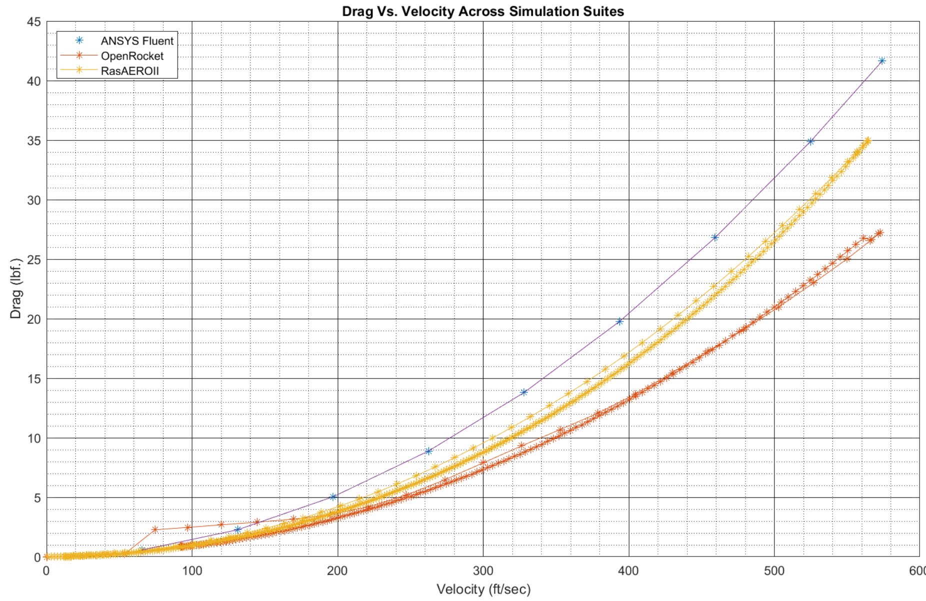


Figure 3.54: Drag profile of the launch vehicle across simulation suites.

This analysis shows that the drag profiles predicted through OpenRocket and RASAero II are less than the drag profile determined numerically through ANSYS Fluent. The team has had difficulty in previous years with the determination of apogee, with the launch vehicle apogee being significantly less than predicted, and this may be a reason for this difference in expected versus actual apogee. Conversion of the drag to a coefficient of drag yields a value 31 greater than the OpenRocket predicted drag and 62 greater than the drag predicted by RASAero II. While the driving force behind apogee is the gravitational force of earth, this discrepancy between drag profiles allows for further refinement of the expected apogee as the launch vehicle reaches CDR maturity. Using this ANSYS CFD-derived drag profile yields the following apogee prediction in RocketPy.

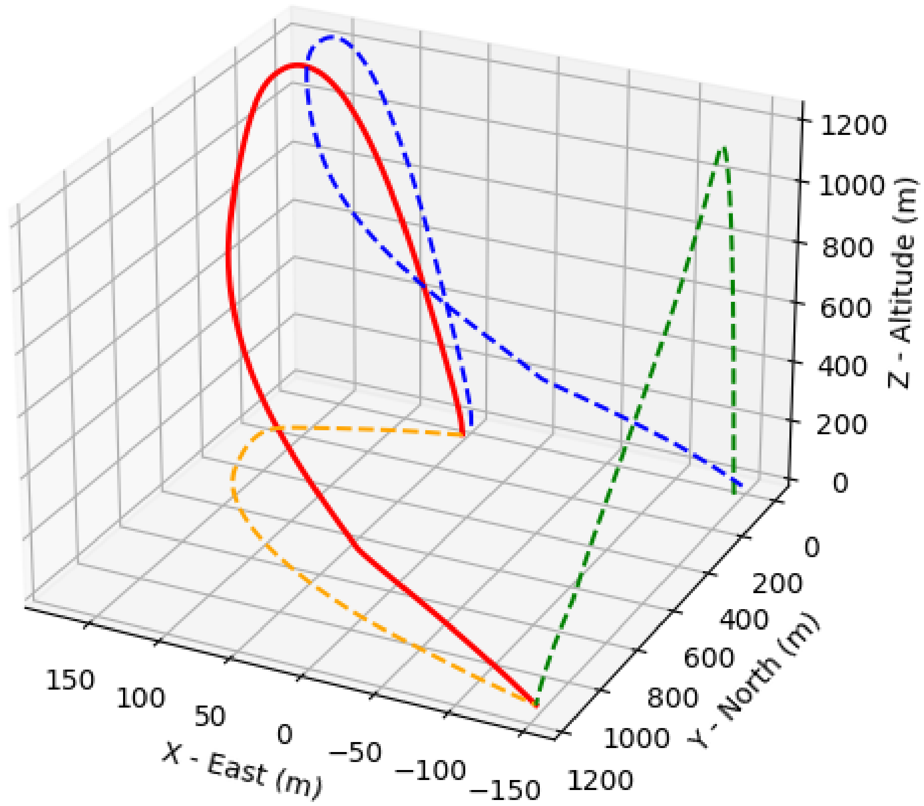


Figure 3.55: Launch vehicle flight profile plotted in 3D Space from Rocketpy.

From the RocketPy analysis, the apogee of the launch vehicle with the refined drag calculations is 4037.2 ft. The maximum velocity is 559.5 ft/s, and the maximum acceleration is 8.125 G's. This is the highest fidelity model that will be used for the determination of the declared apogee.

Verification Calculations

To verify that these calculations are bounded by expected values, the Fehskens-Malewicki equations can be used. These equations were published in 1973 by the MIT Press on the topic of advanced model rocketry. The equations that will be used for this analytical determination of launch vehicle apogee are listed below.

The drag force unit velocity squared can be expressed by the constant K:

$$k = \frac{1}{2}\rho C_d A \tag{6}$$

The empirical factor q, which is a relationship between the thrust, drag, and gravity, can be expressed as:

$$q = \sqrt{\frac{T - Mg}{k}} \tag{7}$$

The empirical factor x, which relates the drag and q per unit mass, can be expressed as:

$$x = \frac{2kq}{M} \tag{8}$$

The maximum velocity of the launch vehicle can then be derived by using Equations 7 and 8:

$$v_{max} = q\frac{1 - e^{-xt}}{1 + e^{-xt}} \tag{9}$$

The altitude of motor burnout, where the force of drag and gravity become the primary forces acting on the launch vehicle, can be determined by:

$$Z_{burnout} = -\frac{M}{2k}\ln\left(\frac{T - Mg - kv_{max}^2}{T - Mg}\right) \tag{10}$$

The total coast distance of the launch vehicle after burnout can be determined by:

$$Z_{coast} = \frac{m\ln\left(\frac{mg+kv^2}{mg}\right)}{2k} \tag{11}$$

Finally, to determine the apogee of the launch vehicle, the coast distance and height of burnout can be summed:

$$Z_{apogee} = Z_{burnout} + Z_{coast} \tag{12}$$

Using Equations 6 - 12, the resulting values can be tabulated.

Table 3.14: Apogee Calculation Constants and Results

Constant	Variable Name	Value	Units
M	Power On Average Mass	1.2605	Slug
m	Power Off Average Mass	1.1970	Slug
g	Gravitational Acceleration	32.174	ft/s^2
t	Motor Burn Time	2.4	s
T	Average Thrust	352.45	lbf
ρ	Air Density	0.002377	$slug/ft^3$
A	Launch Vehicle Frontal Area	0.2076	ft^2
C_d	Drag Coefficent	0.54	NA
Equation	Result		Units
k	0.00013323		slug/ft
q	1530.008		ft^2/s^2
x	0.32344		ft/s^2
v_{max}	565.72		ft/s
$Z_{burnout}$	695.41		ft
Z_{coast}	3348.19		ft
Z_{apogee}	4043.61		ft

From the analytical apogee analysis, the value determined is 6.41 ft. shorter than the RocketPy analysis, which yields a percent difference of 0.158. Given that the analytical solution does not account for launch angle, wind, or other losses, the approach is an extremely useful tool for first-order approximation of the launch vehicle apogee given basic vehicle characteristics.

3.7.2 Flight Profile Simulations

Given the information discussed in Section 3.7.1, a finalized flight profile of the launch vehicle can be determined using the RocketPy position, velocity, and acceleration export tool.

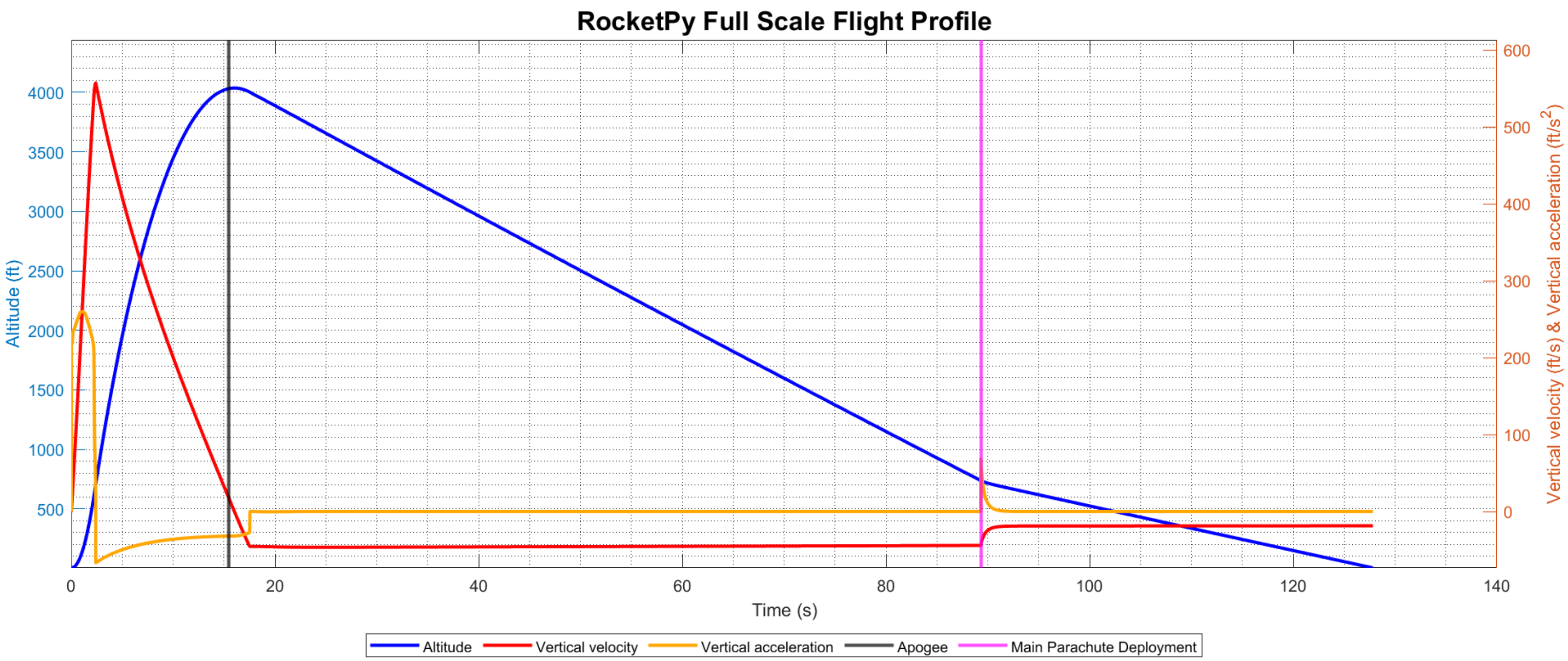


Figure 3.56: RocketPy flight profile of the full scale launch vehicle using an AeroTech L1520T motor.

3.7.3 Payload Weight Affect on Apogee Prediction

It is important to understand how the development of the payload and environmental factors determine the overall apogee of the launch vehicle. Mass modification to the payload or a variation in wind speed for the baseline value will result in the modification of the final apogee and the need for removal or addition of ballasts within the designated sections of the launch vehicle. To determine the variation of launch vehicle apogee with payload mass modification, the model was run with payload masses varying from the baseline 5 lb. payload up to a mass of 7 lbs. Specific modification of the launch vehicle center of gravity in the model was taken to verify the accuracy of simulation results.

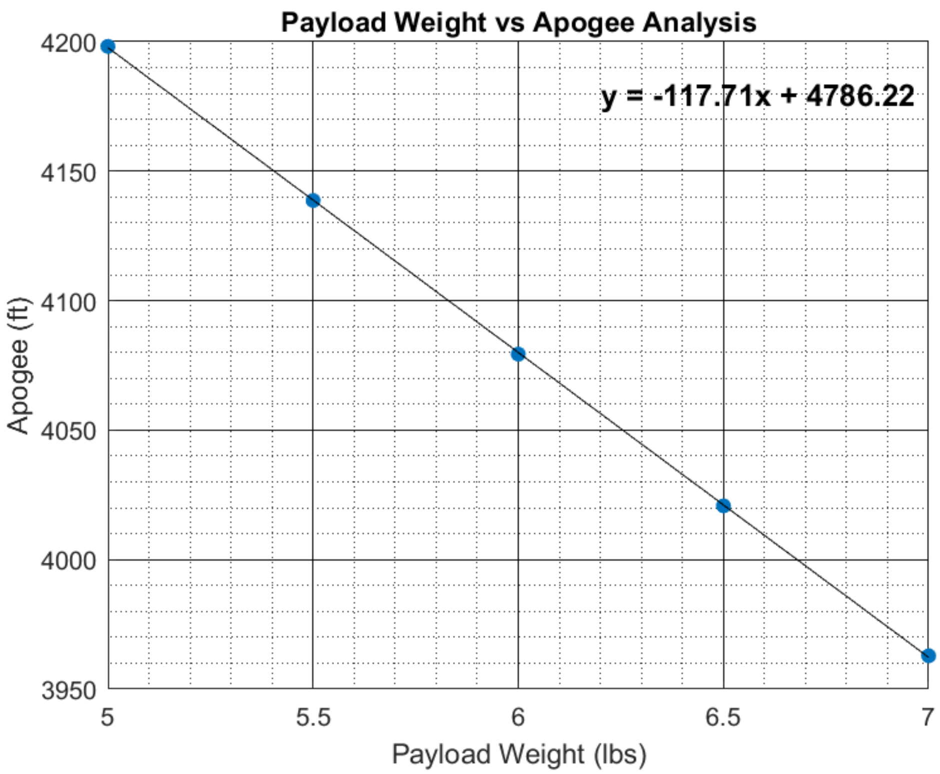


Figure 3.57: Apogee Analysis for a variable payload weight using RocketPy.

From this analysis, a linear trend line can be deduced with the first coefficient of -117.71. This leads to the assumption that for every pound of payload mass increase above the 5 lb. minimum payload requirement, the apogee loss will be equivalent to 117.71 feet. This analysis used real-time wind speed data for the launch site, so specific apogee analysis for the day of launch will be required to determine the day of launch ballasts for the launch vehicle. Given the current mass estimates of the payload, the desired apogee will be obtainable with the introduction of ballasts.

3.7.4 Wind Affect on Apogee Prediction

To determine the variation in apogee with variation in wind speeds, a similar analysis may be conducted. The environment class used to specify the atmospheric conditions may be used to define a standard wind speed throughout the atmosphere. Understanding the control that wind conditions have on the apogee will allow for the determination of the maximum ballasts weight required based on simulation results.

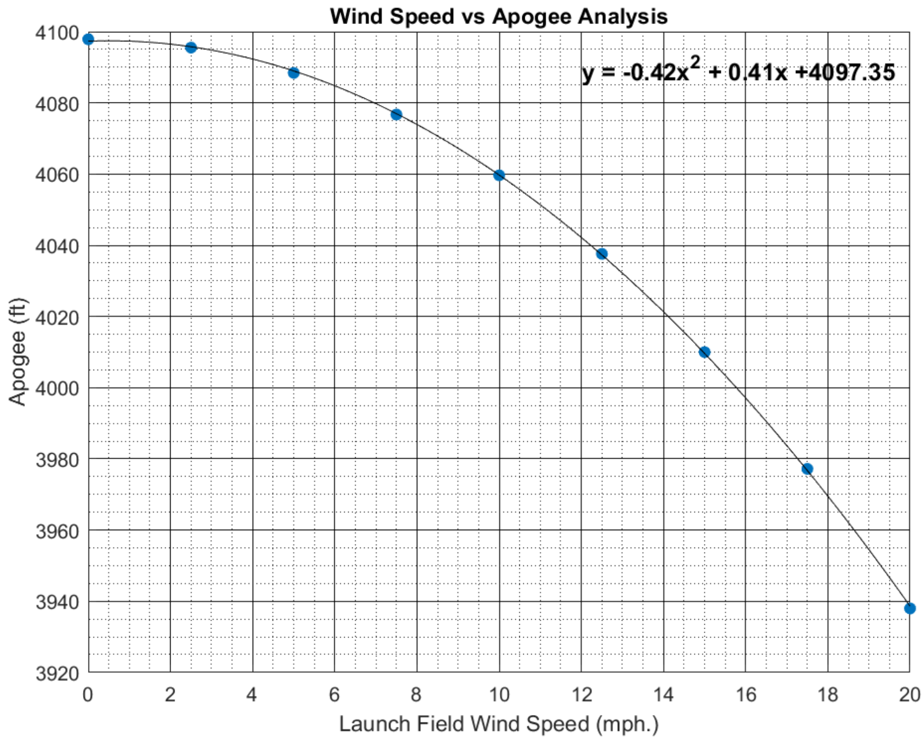


Figure 3.58: Apogee Analysis for a variable wind speed using RocketPy.

From the analysis, the apogee of the launch vehicle follows a parabolic nature to the wind speed on the launch field. This is due to fact that the dynamic pressure experienced by the launch vehicle is related to the velocity squared. It can be observed that the optimal wind speed for apogee is approximately 11 mph. Beyond this point, ballasts removal will be required to achieve the desired apogee. If the launch is conducted with a calm atmosphere, ballasts located on the AV bulkhead will need to be added to lower the apogee of the launch vehicle to an acceptable value.

3.7.5 Stability Margin

Each of the software suites used in the determination of launch day apogee uses its own method for analytical determination of the launch vehicle's center of pressure. This location of the center of pressure is important to know with certainty since it predicts the overall stability of the launch vehicle during flight. Provided below is the calculated stability margin of the launch vehicle at liftoff from each of the software suites.

Table 3.15: Stability Margin Determination Across Software Suites

Software	Stability Margin
OpenRocket	2.10 Calibers
RocketPy	2.04 Calibers
RasAero II	2.22 Calibers

From these tabulated values, there is a variation in the stability margin determined between the analysis software suites. However, the stability margins predicted all fall within NASA SL Requirement 2.14.

Verification Calculation

To bind the software-predicted stability margin values with analytical results, Barrowman equations can be used. These equations are widely used within the field of rocketry due to their simplicity and accuracy. The two subsections of the Barrowman equations that are relevant to the launch vehicle are the nose cone terms and fin terms. The equations used in this analysis have been provided below.

$$(C_{N_f}) = 1 + \frac{R}{S+R}\left[\frac{4N(\frac{S}{d})^2}{1+\sqrt{1+(\frac{2L_F}{C_R+C_T})^2}}\right] \tag{13}$$

$$X_f = X_B + \frac{X_R(C_R+2C_T)}{3(C_R+C_T)} + \frac{1}{6}\left[(C_R+C_T - \frac{C_RC_T}{C_R+C_T}\right] \tag{14}$$

$$X_{CP} = \frac{C_NX_N + C_FX_F}{C_N+C_F} \tag{15}$$

$$SM = \frac{X_{CP}-X_{CG}}{2R} \tag{16}$$

The constants used in this equation, along with the results of each equation, have been tabulated.

Table 3.16: Stability Margin Constants and Results

Constant	Variable Name	Value	Units
$(C_N)_N$	Nose Cone Coefficient	2	NA
X_N	Nose Cone Length Factor	11.184	in.
R	Body Radius	3.085	in.
S	Fin Span	5.25	in.
N	Number of Fins	4	NA
d	Base of Nose Diameter	6.17	in.
L_F	Fin Midchord Line Length	7.60	in.
C_R	Fin Root Chord Length	8	in.
C_T	Fin Tip Chord Length	4	in.
X_B	Nose to Root Chord LE length	93.5	in.
X_R	Nose to Root Chord LE length	7.5	in.
Equation	Result		Units
$(C_N)_f$	6.0722		NA
X_f	98.388		in.
X_{CP}	78.782		in.
SM	2.071		Calibers

From the Barrowman equations, the center of pressure can be estimated at 78.78 in. from the tip of the nose cone leading to a stability margin prediction of 2.07. This prediction falls between the predictions of RocketPy and OpenRocket and satisfies NASA SL Requirement 2.14. Overall this analysis method has a high-fidelity result that is powerful in its ability to deliver high accuracy estimation of the launch vehicle center of pressure given four equations.

3.7.6 Addition of Ballasts

The design of the launch vehicle incorporates specific locations by which ballasts may be used to tune the stability of the launch vehicle for launch. Given that the exact final mass of the payload is unknown at this stage of the payload maturity, an estimated payload mass has been used for preliminary sizing of ballasts. For the forward end of the launch vehicle, threaded rods used to secure the nose cone AV sled may also function as a way to secure ballasts. Ballasts in the form of high-density metal weights may be drilled to the specific diameter of the threaded rods and secured to the rods by using a set of jam nuts on either side of the weights. This method will allow for modification of the amount and location of ballasts along the threaded rod during launch vehicle tuning. For the aft end of the launch vehicle, the design of the removable fin system contains threaded rods that are used to secure the thrust plate and bulkheads to the launch vehicle. The area along these threaded rods may be used to add additional mass to the aft end of the launch vehicle with a similar jam nut configuration to the forward threaded rods.

To tune the apogee of the launch vehicle to the specific wind conditions present at the launch field for the day of launch, a method to secure additional weights to the AV bulkheads has been considered. This design features two locations by which threaded rods may be mounted to the bulkhead, and 50 g. weights may be stacked until enough ballasts have been introduced. Since the center of gravity of the launch vehicle is predicted to exist within the AV bay, and size constraints within the bay make the addition of ballasts infeasible, the AV bay is a good location to add additional ballasts. The total mass that can be added to the launch vehicle for the day of launch apogee adjustment has been determined to be 0.881 lbs. A table displaying the day of launch ballasts control authority is shown below.

Table 3.17: Day of Launch Ballasts Apogee Control Authority.

Ballasts Mass	Apogee
0.22 lb.	4085.10 ft.
0.44 lb.	4059.18 ft.
0.66 lb.	4007.85 ft.
0.88 lb.	3982.4 ft.

3.7.7 Kinetic Energy at Landing

Through the use of Newtonian Mechanics, the kinetic energy for the launch vehicle upon landing can be calculated using the equation below.

$$E = \frac{1}{2}mV^2 \tag{17}$$

In accordance with NASA requirement 3.3, the maximum impact energy allowed is 75 ft-lbf. Additional points can be attained for being below 65 ft-lbf. Using the equation listed above, the maximum impact velocity of each section of the launch vehicle under main parachute descent is shown in the table below. The descent velocity used is calculated for the current leading main parachute option stated in Section 3.6.6.

Table 3.18: Required Velocity for Kinetic Engergy Requirement

Section	Section of Mass	Descent Velocity Necessary to be Awarded Points	Descent Velocity Necessary to be Awarded Bonus Points
Nose Cone	.228 slugs	25.65 ft/s	23.88 ft/s
Main Parachute/ Payload Bay and Avionics Bay	.333	21.22 ft/s	19.76 ft/s
Drogue Bay/ Fin Can	.434 slugs	18.59 ft/s	17.31 ft/s

Mentioned in Section 3.6.6, the main parachute selected is the 84" Iris Ultra Compact by Fruity Chutes. Found in Section 3.5.6, the descent velocity calculated for this parachute is 15.85 ft/s. This is well under the descent velocity required to obtain bonus points for kinetic energy upon landing. From there, the descent velocity of each coupled component can be used to calculate their kinetic energy upon landing. These values are presented in Table 3.19 below.

Table 3.19: Landing Kinetic Energy for Each Section

Section	Section of Mass	Velocity Under Main Parachute	Impact Energy
Nose Cone	.228 slugs	18.49 ft/s	38.97 ft-lb
Main Parachute/ Payload Bay and Avionics Bay	.333 slugs	15.85 ft/s	41.83 ft-lb
Drogue Bay/ Fin Can	.434 slugs	15.85 ft/s	54.56 ft-lb

It can be seen from the table that the kinetic energy requirement for each section hitting the ground is satisfied. The velocities provided were calculated in the parachute alternative's section, where the AV bay, main parachute/payload bay, and fin can have the same descent rate since they are tethered together under the main parachute.

3.7.8 Expected Descent Time

The total descent time for the launch vehicle is broken up into two sections, the descent time under drogue, and the descent time under main parachute. Calculated in Sections 3.5.5, and 3.5.6, the descent velocity of the launch vehicle under drogue and main parachutes is used to find the descent time. The descent time is also a factor of the apogee height and the height at which main is deployed. Presented in the equation below is how the total descent time is found.

$$t = \frac{h_a - h_m}{v_d} + \frac{h_m}{v_m} \tag{18}$$

Let t represent the total descent time for the launch vehicle, h_a is the apogee altitude, h_m is the main deployment altitude, v_d is the descent velocity of the vehicle under drogue parachute, and v_m is the descent velocity of the launch vehicle under main parachute. As stated in Section 3.7.8, the total descent time for the launch vehicle was calculated to be 81.98 seconds, which meets NASA SL Requirement 3.12.

Additionally, the nose cone/ payload parachute descent time can be found using the descent velocity of the nose cone when the payload is and isn't attached and the height of main parachute deployment. These parameters are stated in Section 3.2.2.

$$t_n = \frac{h_m - h_p}{v_p} + \frac{h_p}{v_n c} \tag{19}$$

Where t_n is the total descent time of the nose cone, h_m is the height the main parachute is deployed (800 ft), h_p is the height the payload is deployed (450), v_p is the descent velocity of the nose cone and payload under the payload parachute, and $v_n c$ is the descent velocity of the nose cone without the payload attached. As stated in Section 3.2.2, the total descent time of the nose cone from its ejection at the main parachute event is 38.134 seconds, which meets NASA SL Requirement 3.12.

3.7.9 Expected Drift Distance

When determining the expected drift distance, a large overestimation is used in order to ensure the launch vehicle will not drift 2500 ft from the launch pad. This consists of assuming that the drift velocity of the launch vehicle is equal to the wind speed. In fact, the actual drift speed is a function of the drag from the parachute, and the descent velocity as well, though it will get us lower magnitudes. The team likes to overshoot to eliminate the risk of even getting close to the maximum 2500 drift distance. Using the following equation, the drift distance of the launch vehicle can be calculated for different wind speed conditions. Let v_w be the wind speed, t be the estimated descent time, and D is the expected drift. As mentioned in Section 3.7.8, the total descent time for the launch vehicle is approximately 81.98 seconds, and the descent time for the nose cone from the main deployment altitude is 31.134 seconds.

$$D = v_w t \tag{20}$$

Using this equation, the total wind drift can be calculated using the wind drift under the drogue parachute and adding the wind drift from the main parachute. Presented below in the table are the wind drift distances calculated for various wind speeds.

Table 3.20: Wind Drift Distances for Varying Wind Speeds

Wind Velocity	Launch Vehicle Drift Distance	Nose Cone Drift Distance from Main Deployment
0 mph	0 ft	0 ft
5 mph	601.19 ft	279.66 ft
10 mph	1202.38 ft	559.33 ft
15 mph	1803.57 ft	838.99
20 mph	2404.76 ft	1118.65 ft

From looking at the table, the overestimated maximum drift distance under a 20 mph wind will be 2404.76 ft, which is under the 2500 ft requirement 3.11 set by NASA.

3.7.10 Parachute Opening Shock Calculations

One of the largest loads the launch vehicle experiences is when the main parachute deploys. The shock force is a function of the time it takes the parachute to open, the change in velocity of the launch vehicle from drogue descent to main descent, and the mass of the launch vehicle. In order to calculate the shock force though, the time it takes the parachute to open needs to be calculated first. This is found using the equation below, where r is the radius of the parachute opening, v is the drogue descent velocity, and t is the time it takes the parachute to open.

$$t = \frac{8r}{v} \tag{21}$$

Found in a study by W. Ludtke on how to calculate opening shock forces for a parachute, the coefficient of 8 is necessary to find the time it takes the parachute to open. For the Fruity Chutes 84" parachute with a drogue descent rate of 113.10 ft/s, the time it takes to open is approximately 0.246 seconds. From there the shock force can be found using the equation below where F is the shock force, m is the mass of the launch vehicle, Δv is the change is descent velocity from drogue to main, and t is the time it takes the main parachute to open.

$$F = \frac{m\Delta v}{t} \tag{22}$$

Using this equation, the maximum shock force the launch vehicle will experience is approximately 299.47 lbf.

4 Payload Criteria

4.1 Payload Objective and Success Criteria

The payload mission objective is to safely jettison four non-living crew members, or STEMnauts, from the launch vehicle during descent and transport them to the ground. The payload lander, named the STEMnauts Atmosphere Independent Lander (SAIL), is intended to be functional in various atmospheres that coincide with different celestial bodies. Due to this requirement, the use of parachutes or streamers cannot be considered in the descent of the SAIL. The STEMnauts should be recovered under predefined survivability metrics, with the intention of representing human passengers.

The SAIL should be able to jettison from the launch vehicle with commands given during flight. Once given RSO permission, ground control will operate an electric latch on the launch vehicle using an RF command to release the SAIL. Upon release, contra-rotating rotor blades and landing legs will unfold, with the rotors being powered from an on-board motor. The descent velocity of the SAIL will then be controlled with an Arduino microcontroller, decreasing the speed through lift generated by the contra-rotating rotor configuration. Once the SAIL reaches the ground, the motor will disengage, ceasing function of the rotors.

Table 4.1: Payload Success Criteria

Success Level	Payload Aspect	Safety Aspect
Complete Success	The SAIL lands in the pre-defined orientation and with a landing velocity of under 5 mph. Additionally, the SAIL does not experience any sustained forces greater than 3g.	No personnel are harmed or at risk during payload recovery
Partial Success	The SAIL lands in the pre-defined orientation but with a velocity between 5 mph and 15 mph OR the SAIL lands with a velocity under 5 mph but does not come to rest in the pre-defined orientation.	No personnel are harmed during payload recovery but there is at least one close call.
Partial Failure	The SAIL impacts the ground with a velocity greater than 15 mph but receives no major damage.	Personnel receive minor injuries during payload recovery.
Total Failure	The SAIL impacts the ground with a velocity greater than 15 mph AND sustains catastrophic damage.	Personnel receive major injuries during payload recovery.

4.2 Potential Payload Designs

The following section breaks down the SAIL into its major components and discusses possible designs. Each design is then analyzed based on mission requirements, part reliability, cost, ease of manufacturing/ease of purchasing and ease of integration with the rest of the payload and launch vehicle.

4.2.1 Descent Method

The SAIL will be recovered using a rotor system to decrease the vertical descent velocity to safe levels (below 15 mph impact velocity). Four possible methods for the rotor blade configuration/operation are discussed below.

Auto-rotation

As the SAIL descends, the airflow going past the rotor blades will start a process called auto-rotation. With the increase of the rotor blade RPMs, lift increases and slows the vehicle down. This decrease in descent velocity then decreases the rate of spin of the rotor blades, decreasing lift. This process repeats cyclically until a steady-state condition is reached and the rotor blade is spinning at a constant RPM generating constant lift. In this state, the rotor blade is acting like a parachute and slowing the SAIL down to safe speeds. By considering the lift from the blades as drag and inputting them into the terminal velocity equation, rotor blade sizes are roughly estimated below in Figure 4.1

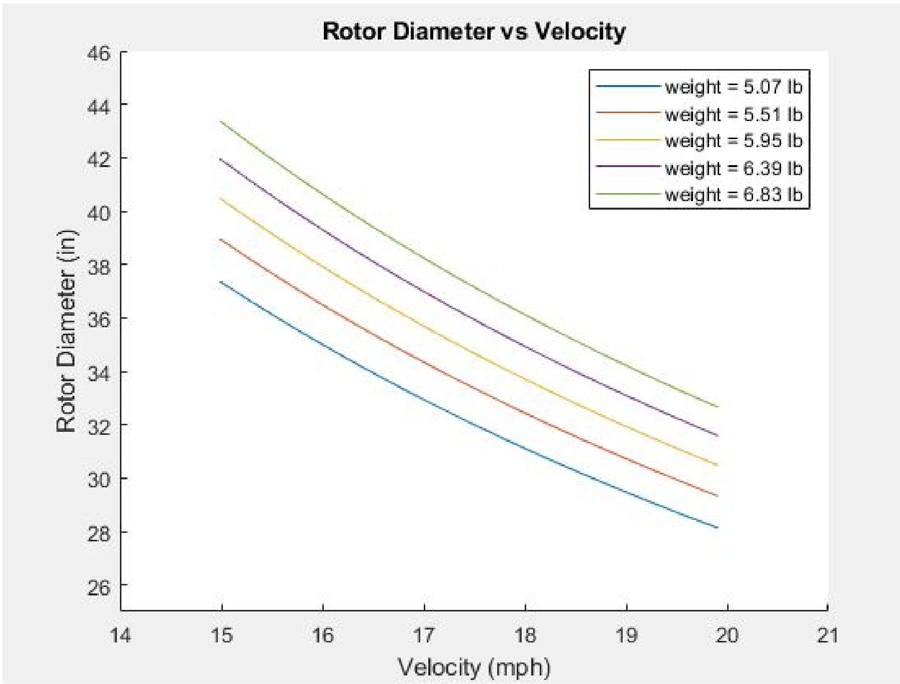


Figure 4.1: Rotor Diameter vs Descent Velocity

The rough estimations show that auto-rotation is within the realm of possibility given the size restraints for the payload bay. However, one major problem with auto-rotation is refining the simulation models and accurately calculating the descent velocity. Because there is no motor powering the rotor, there is not an analytical solution to find the RPM of the rotor blades. In order to estimate the lift generated by the blades, a 6 degree of freedom transient CFD analysis would be required. This form of analysis potentially requires more time and resources than are available for the project.

Another issue with auto-rotation is the difficulty in ground testing. Because the SAIL must be falling in order to generate lift, it would not be possible to test without dropping the SAIL from hundreds of feet above the ground. If the recovery method does not work the first time, the SAIL would most likely be catastrophically damaged. This puts even more emphasis on the accuracy of the complex CFD analysis.

Counter-rotating rotors

Counter-rotating rotor blades are co-axially mounted rotors that are powered independently of each other and that rotate in opposite directions. An advantage of this method is that there is no need for a complex gearing system to reverse the direction of one of the rotors.

Using Blade Element Momentum Theory (BEMT), rough estimations of thrust were calculated for 2-4 blade configurations using a NACA 0015 symmetric airfoil.

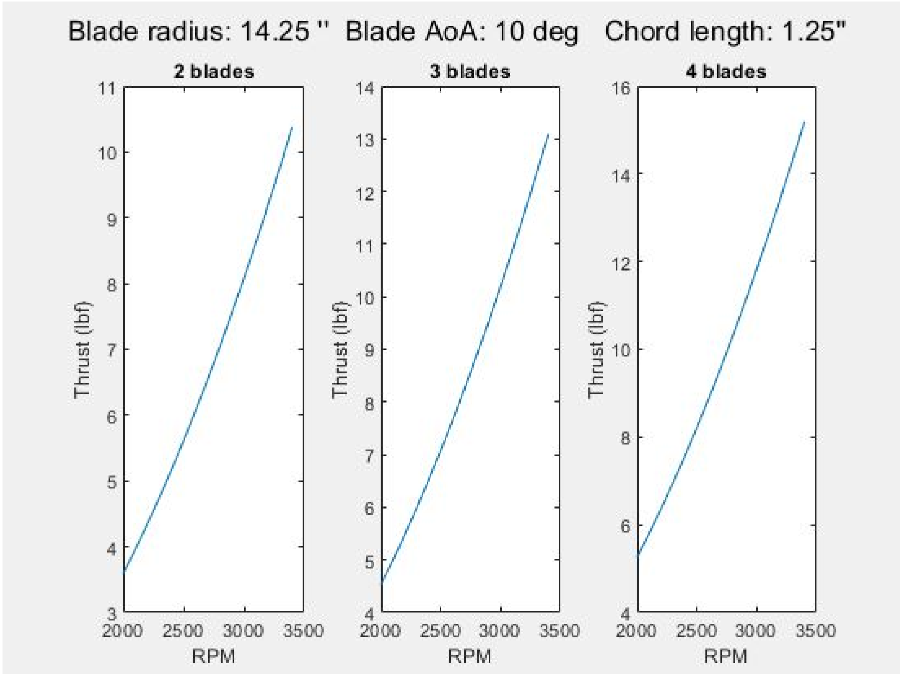


Figure 4.2: Thrust estimations for coaxial rotors.

One major advantage of using a motor to power the rotor blades is the ability to ground test the system. By securing the SAIL to a scale, the actual thrust of the system can easily be determined prior to flight. This greatly reduces the risk of a catastrophic failure during testing. Additionally, since the RPM is determined by the motor configuration, CFD for a powered system is much easier than the CFD required for auto-rotation.

One major difficulty with counter-rotating rotors is ensuring that both rotors are spinning at exactly the same RPM. Any slight difference in rotor speed will cause both the rotation of the vehicle body and a lateral thrust force. While differential RPM can be used to effectively steer the vehicle, this adds a high degree of complexity to the overall design.

Contra-rotating Rotors

Contra-rotating rotor blades are very similar to counter-rotating in that there are two rotors co-axially mounted. The main difference is that contra-rotating rotors are both powered by the same motor rather than being powered individually. This ensures that both blades are spinning at the same RPM, eliminating the differential torque that would cause the SAIL to rotate without requiring precise control over individual motor speeds. Figure 4.3 below illustrates one possible method of rotating the rotors in different directions utilizing a system with spur gears, a belt, and concentric axles.

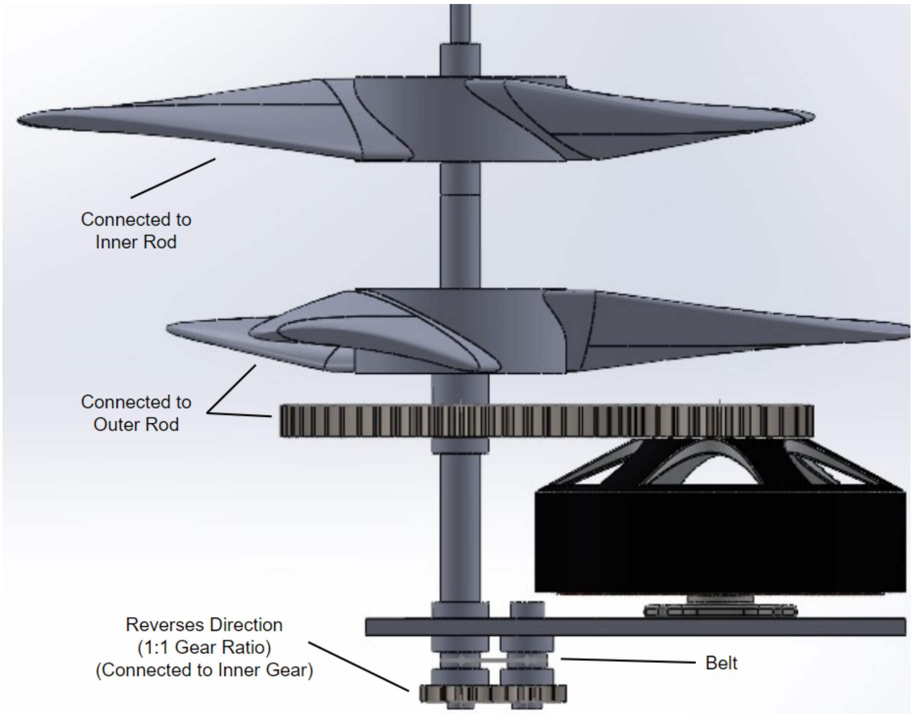


Figure 4.3: Spur gear design for contra-rotation.

One of the main drawbacks of contra-rotation is the requirement of a set of gears/belts to reverse the direction of one of the rotor blades. This adds mechanical complexity as well as weight that is avoided in other design options. The second main drawback is the requirement of a large brushless motor. One large motor is more expensive than two smaller motors, thus increasing the total cost of the SAIL.

Quadcopter

This method of recovery uses multiple small powered rotor systems rather than one single system. Some advantages of having a ”quad-copter” style propulsion system include full control authority via differential thrust and access to published specification sheets for propeller/motor combinations. However, having multiple powered rotors also introduces a large amount of complexity in terms of programming a flight computer capable of accurately controlling the independent motors to maintain a stable flight profile. Additionally, folding and stowing multiple motor/blade assemblies would be challenging with the size restraints of the payload bay.

4.2.2 Rotor Blade Shape

Every method of arresting the vertical descent rate discussed in the prior section implements a rotor system. Symmetric airfoils have been employed in many rotor systems to date and are the main airfoil shape in consideration. However, two variations between symmetric airfoil blades are discussed below.

Zero Twist, Zero Taper, Symmetric Airfoil

The simplest version of a rotor blade is a symmetric airfoil with zero twist and zero taper. This form of blade is easy to manufacture and has well known aerodynamic properties. While this design is not optimized for peak performance, it is capable of producing enough thrust for this project application.

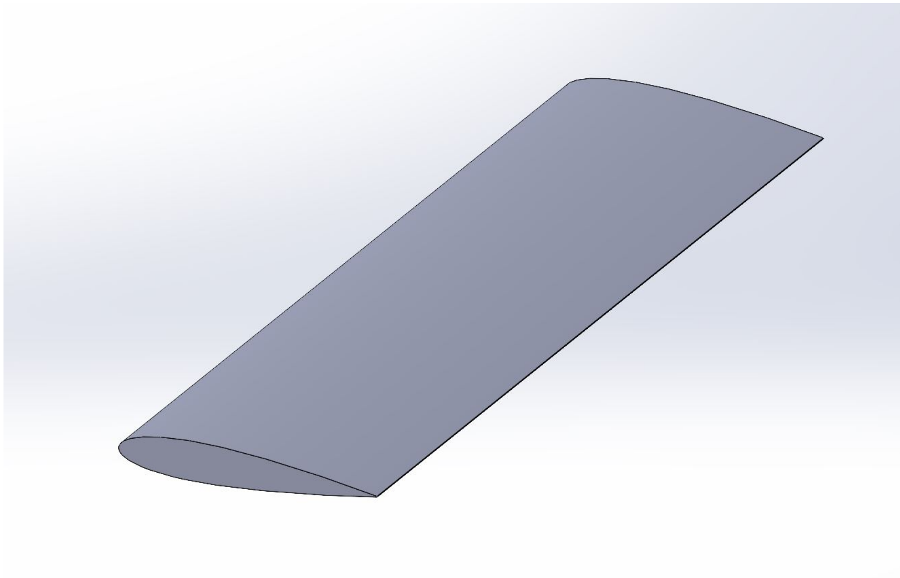


Figure 4.4: Zero twist, zero taper rotor blade

Twisted, Tapered, Symmetric Airfoil

The two main ways of increasing blade performance are tapering and twisting the blade profile. Creating a blade taper is relatively simple but only has a small affect on performance. Designing the proper blade twist is a complicated process that is a subject unto itself and would introduce a high level of complexity. Additionally, since the purpose of the rotor system is to slow the SAIL down and does not need a thrust to weight ratio of greater than 1, blade optimization is not a critical design aspect. A straight symmetric airfoil blade will provide enough performance to meet the project requirements.

4.2.3 Landing Legs

Due to the lack of translational control and high center of gravity, a stable landing platform is critical to maintain vertical orientation upon landing. Three possible mounting locations for the landing legs are discussed below.

Bottom Mounted

The first possible mounting location is placing the hinge towards the bottom of the SAIL and having the legs fold upwards. Upon deployment, the legs would rotate into the deployed position via a torsion spring in the hinge. Figure 4.5 below illustrates the configuration. There are two main disadvantages with this method: leg length and stability. Because the legs are extending upwards when stowed, the length of the legs is restricted by the rotor hub. Additionally, by placing the leg mount lower on the SAIL the center of gravity will be much higher, increasing the chance of a tip over. This mounting option does not provide as stable of a landing platform as a top mounted design.

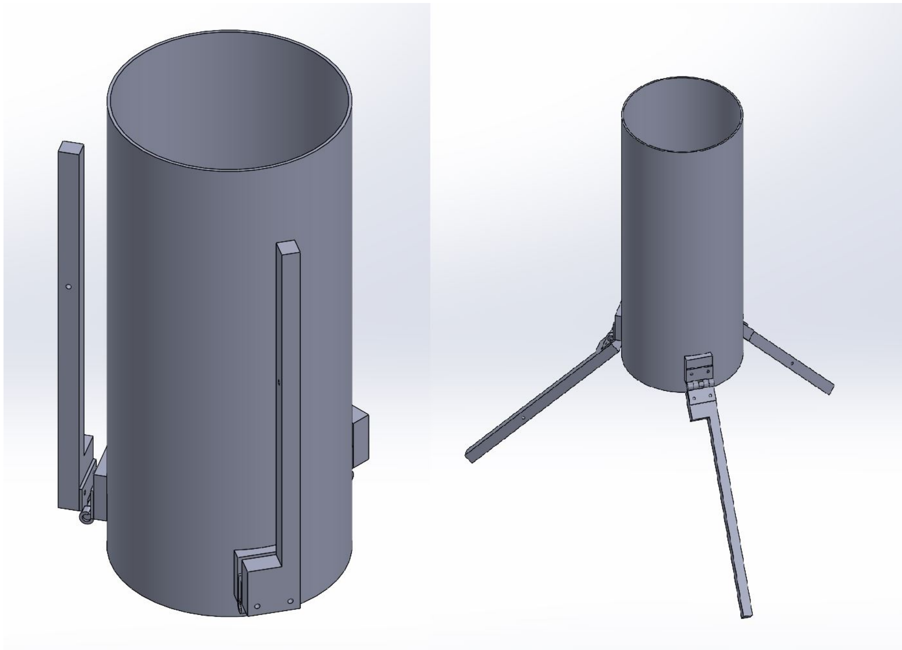


Figure 4.5: Stowed and deployed views of bottom mounted landing legs.

Top Mounted

Mounting the legs higher on the SAIL will provide much greater stability than legs mounted on the bottom by providing a larger footprint. Additionally, mounting the legs around the SAIL in combination with the folding rotors will simplify the deployment of the legs.

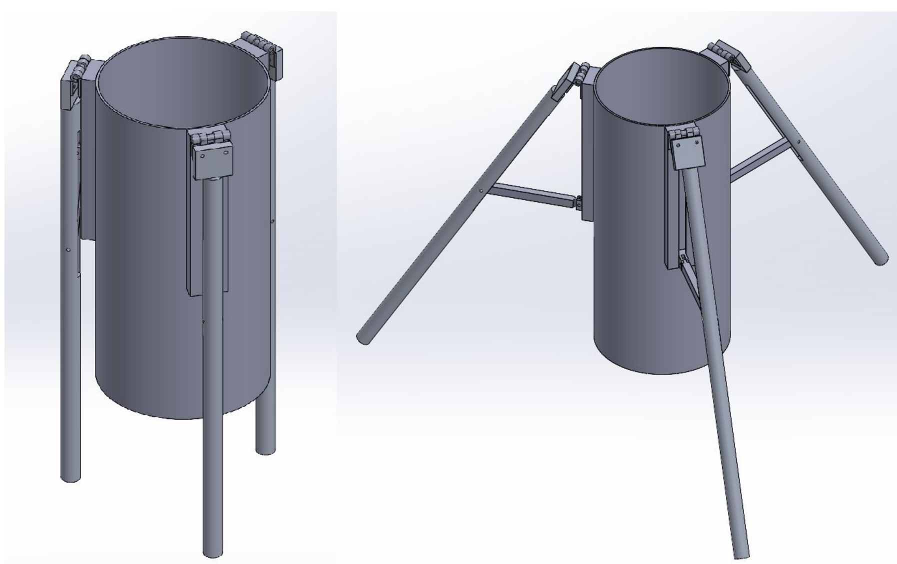


Figure 4.6: Stowed and deployed views of landing legs with rail system.

Figure 4.6 shows the landing legs with a rail/follower configuration. The legs would be mounted at the top with a hinge and have a rod connected towards the middle. This rod would be pinned on both ends to allow rotation and connected to a rail system. A spring located inside the rail system will push downwards on the follower, forcing the legs to deploy once outside of the payload bay. One disadvantage of this method are the tight tolerances required in the rail system to ensure smooth operation.

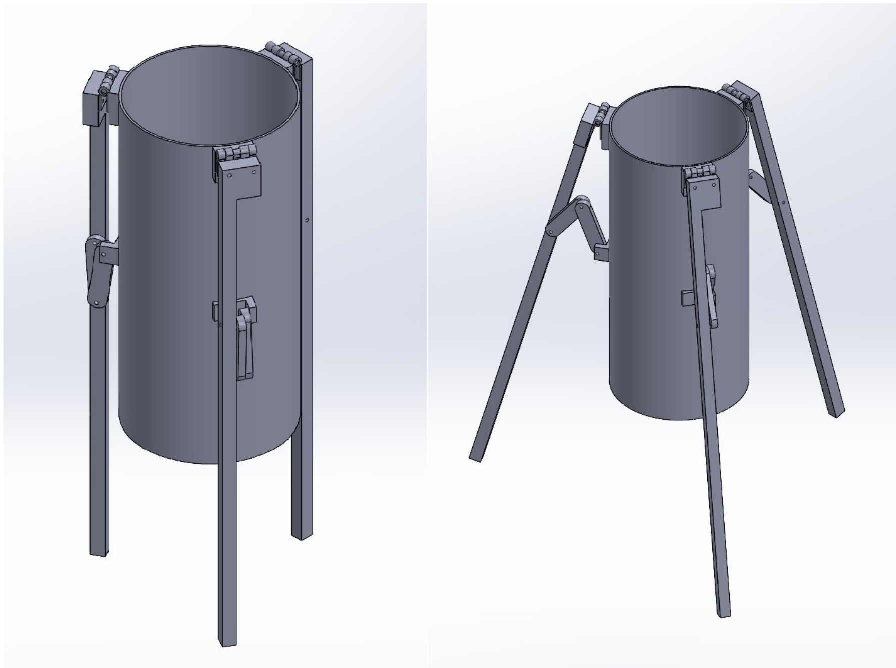


Figure 4.7: Stowed and deployed views of landing legs without rail.

Figure 4.7 shows another variation of the landing leg configuration that eliminates the rail system and instead has a folding bar pinned onto the vehicle body. An extension spring would be connected between the top hinge mount and the folding bar. Upon exit from the payload bay, the spring would force the landing legs into the deployed state. The advantage of this method is that the spring system is much easier to fabricate compared to the rail system.

4.2.4 SAIL Retention

The SAIL will be stowed in a configuration that allows for a simple vertical drop to support the stability of the descent. This will be accomplished through separation of the nose cone from the launch vehicle, which will be attached to the SAIL with a shock cord. With this setup, a remote-controlled electronic retention system is desired to release the shock cord, thus dropping the SAIL. There are a few commercially available electronic rotary latches, which are listed below.

Southco Electronic Rotary Latch

The first latch available is a Southco Electronic Rotary Push-to-Close latch. This latch can be controlled with a single voltage high, opening the latch in a simple manner. The latch closing is rated to withstand a tensile force of up to 180 lb., which may be a concern for initial shocks upon separation.

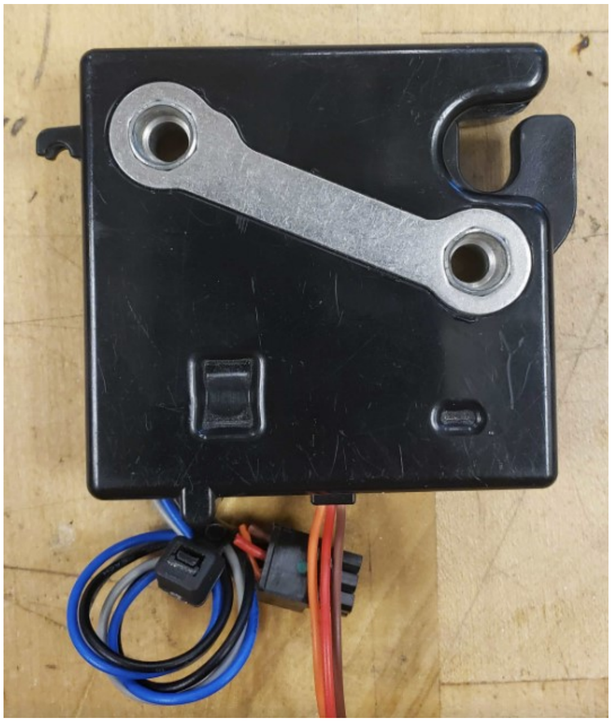


Figure 4.8: Southco electronic rotary latch.

However, the biggest concern is the structure of the latch arm. When under tension, the arm opens slightly, leaving a gap that the shock cord can easily slip through. This is a major concern on an otherwise acceptable latch.

Camlock Rotary Latch

The Camlock rotary latch functions very similarly to the Southco latch, however it is much more durable and can withstand a max tensile force of 7000 N, or about 1570 lb. This makes it more likely to survive shocks during nose cone separation. A concern for this latch is the size and weight. With the latch being made of metal, it would add quite a bit more weight to the nose cone. Although, the closing arms of the latch give confidence that the shock cord connection cannot become undone.

Figure 4.9: Camlock electronic rotary latch.

However, a major issue with this latch was found during testing. While under tension, the electronics inside the latch failed to actuate the release arm. If this were to occur during flight, then the SAIL would fail to deploy when commanded. Furthermore, the latch electronics were permanently damaged after attempting to unlatch under tension.

Custom 3D Printed Latch

A custom latch is the most viable option in terms of creating a custom release assembly. With this method, materials can be selected based on expected shock forces during release. This also allows for simple testing and integration when compared to an already manufactured product.

The challenge with this approach is keeping the mechanism simple enough to minimize errors during operation. The leading idea is to use a servo, which would rotate to open the latch. A strong enough servo would be required to overcome the tension of the shock cord. The leading servo choice is shown in Figure 4.10. Commercially available servo horns would not likely be viable in a custom latch design, meaning a custom horn would have to be designed.

Figure 4.10: 35 KG servo for latch opening.

In terms of material, carbon fiber PETG is the most viable candidate due to it's favorable material properties and low weight. Other materials, such as PLA and non-carbon fiber PETG, were considered for their ease of printing and lower cost, but ultimately these polymers do not have the strength needed for the extreme conditions experienced during flight.

4.2.5 SAIL Deployment Method

Due to the need of real-time RSO permission in order to release the SAIL, an RF controlled release mechanism seems to be the best option for real-time deployment. This method allows for full control of the release which is very important in the event that it is deemed unsafe to deploy the SAIL. We estimate that a communication distance of at least 1 mile is needed to ensure a high likelihood of the vehicle being in range when it is time to deploy the SAIL.

Module Name	Operating Frequency	Line-of-Sight Range	Current (Transmit | Receive | Idle)
Long Range RF Link Kit	433 MHz	2 km	'working current' = 2.5 mA
LORA Reyax RYLR896	915 MHz	4.5 km	43 | 16.5 | 0.5 mA
XBee Pro S3B	902-928 MHz	45 km	215 | 26 | 2.5 mA

Table 4.2: Comparison of Different Radio Modules

Long Range RF Link Kit with Arduino

For sending and receiving RF commands, there are a few options for hardware. A 2KM Long Range RF link kit, shown in Figure 4.11, interfaces well with an Arduino micro-controller, allowing for a simple code snippet to operate the release of the SAIL. A concern with such a simple communication system is that other radio signals may interfere with the release command sent by the team. To ensure that the SAIL is deployed only when commanded by the team, a unique pin would be required when transmitting the release command.

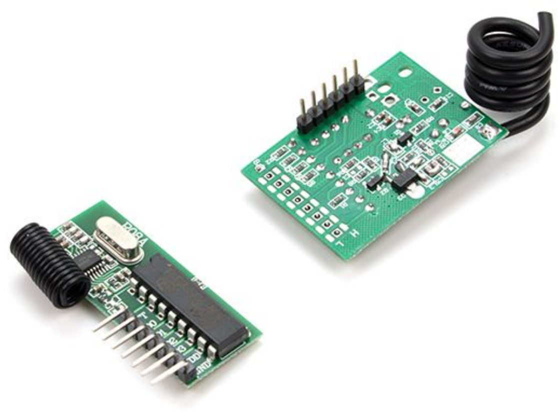


Figure 4.11: 2KM long range link kit.

With this kit, the receiver will be on the launch vehicle listening for a command from the transmitter, which will be with a designated personnel on the ground. The ground transmitter will be powered by a battery, with a switch to connect the transmitter to the power source. Once RSO permission is given to release the SAIL, the switch will be closed, giving power to the transmitter to output an RF signal. This is a fairly simple design and decreases the likelihood for errors.

LORA Reyax RYLR896 Transceiver Module

A similar RF communication setup involves using a transceiver pair for both transmitting and receiving commands. A good candidate for this is the Reyax RYLR896 Transceiver Module, shown in Figure 4.12. The range for this module is around 4.5 KM (about 2.8 miles) with a maximum operational distance of 15 KM (about 9.3 miles). With a planned SAIL deployment altitude of 800 ft, this module exceeds the necessary capabilities for an RF transmitter/receiver. The size of this module is also attractive, as it is only 25 x 17 mm (about 1.0 x 0.7 in). This allows for more available space in the nose cone as the operational size is limited inside.

Figure 4.12: RYLR896 transceiver module.

This module would work in a similar fashion as it would be placed in the nose cone and the ground to receive and send command, respectively. It is also fair to assume that this unit would work well with an Arduino, as the pin setup is streamlined for data input/output.

XBee Pro S3B

The final alternative is the XBee Pro S3B Transceiver. It boasts the highest range, but also the highest current draw and monetary cost. They are also slightly bigger than the LORA modules at 33 x 25 mm excluding an antenna. This module would require a breakout board of similar form factor to the transceiver in order to connect to an Arduino or micro-controller. For the software, it would be a similar form of communication as the other two modules.

Figure 4.13: XBee Transceiver

4.3 Feasibility Study

In order to finalize the design of the payload, a feasibility study was conducted using the Pugh method. Critical elements for each subsystem are identified and given a "Project Impact" rating on a scale of 1-5 (1 = negligible impact, 5 = extreme impact). Each option discussed previously was given a "Design Rating Impact". A weighted total was then calculated for each option by adding the products of the project impact and design rating impact values.

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

Table 4.3: Design Rating Impact on Critical Elements

Critical Element	Project Impact	Auto-rotation	Counter-rotation	Contra-rotation	Quad-copter
Mechanical Simplicity	3	1	-1	0	-1
Ease of Computer Modeling	4	-1	1	1	0
Ease of Manufacturing	3	1	0	0	-1
Ease of Ground Testing	5	-1	1	1	1
Weight	2	0	1	1	1
Cost	3	1	0	0	-1
RAW TOTALS		1	2	3	-1
WEIGHTED TOTALS		0	8	11	-3

Table 4.4: Pugh Matrix of Descent Methods

Critical Element	Project Impact	Straight	Tapered/Twisted
Design Complexity	4	1	-1
Ease of Manufacturing	3	1	0
Efficiency	2	-1	1
RAW TOTALS		1	0
WEIGHTED TOTALS		5	-2

Table 4.5: Pugh Matrix of Rotor Blade Shape Options

Critical Element	Project Impact	Bottom Mounted	Top Mounted w/ Rail	Top Mounted w/ Extension Spring
Stability	5	-1	1	1
Reliability	5	1	-1	1
Ease of Manufacturing	3	1	-1	1
Cost	3	1	0	1
RAW TOTALS		2	-1	4
WEIGHTED TOTALS		6	-3	16

Table 4.6: Pugh Matrix of Landing Leg Options

Critical Element	Project Impact	Southco Rotary Latch	Camlock Rotary Latch	Custom Latch
Strength	5	0	1	1
Adaptability	4	-1	-1	1
Size	3	0	-1	1
Cost	3	0	0	1
RAW TOTALS		-1	-1	4
WEIGHTED TOTALS		-4	-2	15

Table 4.7: Pugh Matrix of Latch Options

Critical Element	Project Impact	RF Link Kit	Reyax Transceiver	XBee Pro
Range	5	-1	0	1
Size	4	-1	1	1
Cost	3	1	0	-1
RAW TOTALS		-1	1	1
WEIGHTED TOTALS		-6	4	6

Table 4.8: Pugh Matrix of RF Hardware

4.4 Leading Payload Design

The most plausible payload design consists of the contra-rotating rotors where one motor is used to power two rotors in opposite directions. Using this method, the rotors will spin at the same rate, ensuring stability and preventing any rotation of the payload body. As for landing in the desired orientation, the top-mounted, folding design is most ideal so that the legs can hinge outwards with a spring and lock once fully deployed. To assemble the entire payload, there are five main components: hub-to-propeller assemblies, gearbox, motor assembly, payload body with electronics, and the landing legs. These are all secured to each other through various methods. The fully deployed payload with each of these components is shown in Figure 4.14.

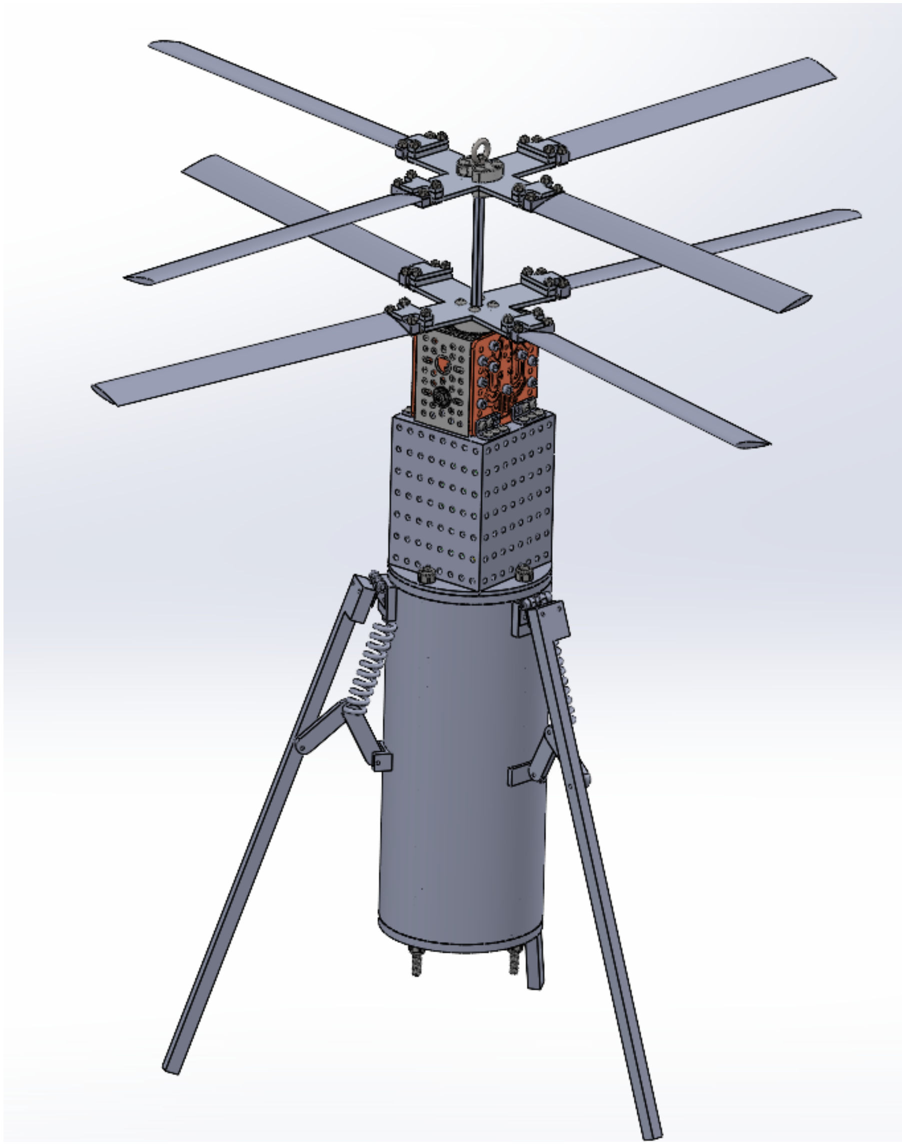


Figure 4.14: CAD model of the SAIL in a deployed configuration.

4.4.1 Major Components

Hub Assembly

A top-down approach will be used to dissect the payload's components. First, there are two hubs, upper and lower. These will be spinning opposite of each other and thus, will be secured to different parts of the assembly. Both hubs share the same shape and size but the central holes are different sizes to accommodate their fasteners.

Both hubs are set to have a 4.5 in. diameter with four prongs, each 1.25 in. wide. The thickness of each hub is 1/8 in. and will be made out of an aluminum sheet. At the end of each prong are mounting holes meant to secure spring hinges and hard stops. The only difference between the two hubs are the center holes as, later discussed, the "GoTube" and "sonic hub" have similar dimensions. The only exception is that the sonic hub has an extrusion that can fit within the 0.55 in. diameter hole, which is helpful for centering. Meanwhile, to increase strength, the lower hub only has the hex shaft running through it, making the center hole 0.35 in. in diameter. This difference is seen in

Figure 4.15. It is important to note that the GoTube is secured to the lower hub while the hex shaft is secured to the upper hub, allowing the two hubs to rotate opposite of each other. This will be further explained in the Gearbox section.

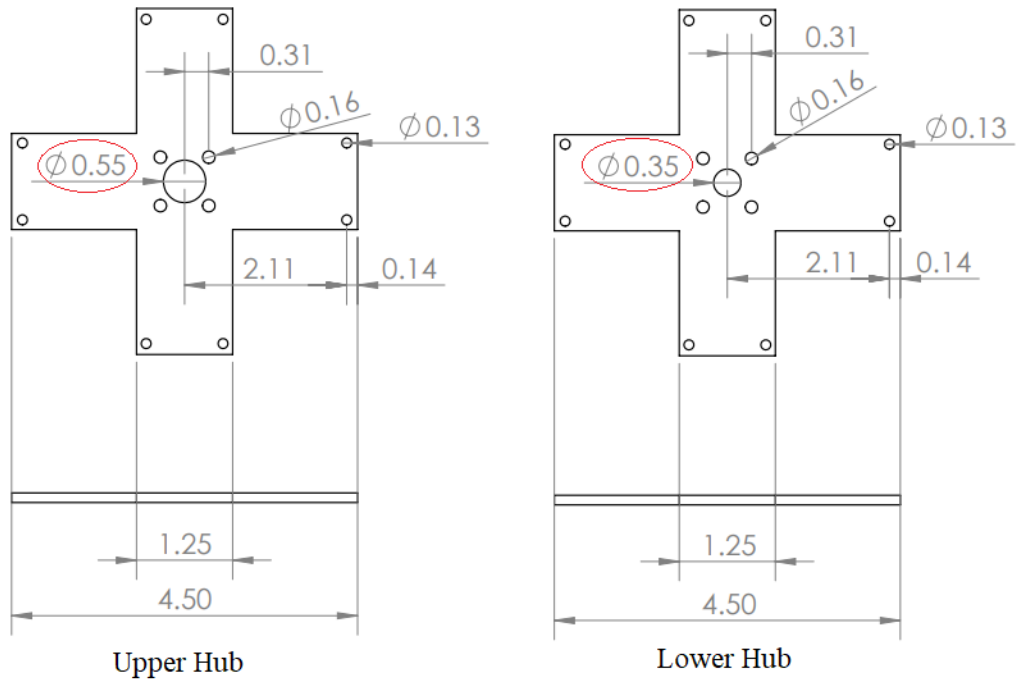


Figure 4.15: Upper Hub vs. Lower Hub (in.).

The upper hub (4.16) will be attached to a central hex shaft (0.32 in. diameter, 7.72 in. tall). This connection is made using two ”sonic hubs” that have two set screws each to fasten to the hex shaft. In each sonic hub are four M4 x 0.7 mm threads. Thus, four 4 mm holes will be cut out in the upper hub. With a sonic hub on each side, top and bottom, and using four 20 mm long, M4 x0.7 mm bolts, the hub will be easily fastened and compressed to it's ideal height on the shaft.

At the top of the hex shaft is a tapped M4 x 0.7 mm thread. This will be utilized for a fully sealed eye bolt that has an M4 x 0.7 mm extruded thread. The eye bolt is used to hold onto the shock cord for release. This will hold the entire payload inside the payload bay.

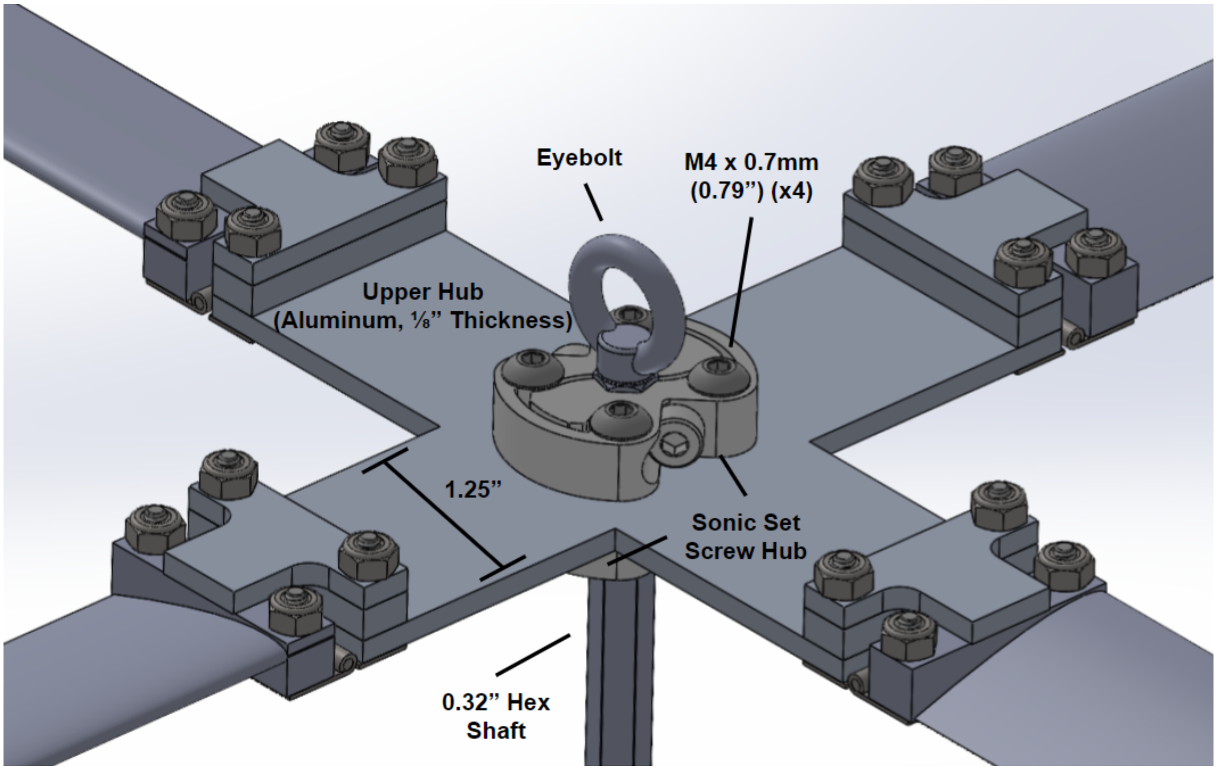


Figure 4.16: Upper hub assembly with four attached propeller blades.

The lower hub (seen in Figure 4.17)is placed 4 in. below the upper hub to reduce any potential turbulent air that may affect airflow on the upper hub's propeller blades. Furthermore, four 20 mm long, M4 x 0.7 mm bolts are used to secure the hub to the GoTube (seen in Figure 4.18).

The GoTube and hex shaft are connected to their respective bevel gears that rotate in opposite directions and thus, they secure the hubs in place to lock the contra-rotation. The GoTube is especially helpful as it opens up 12 x 12 mm of central space for the hex shaft to rotate freely while having M4 x 0.7 mm tapped holes to easily mount the lower hub. An area of concern was keeping the GoTube and hex shaft concentric, especially since the hex shaft extends over 4 in. beyond the GoTube. To aid this, an 8 mm hex flange bearing can fit on top of the GoTube to constrict the hex shaft's movement, making it purely rotational. With the flanged bearing, however, a spacer is needed to fill the gap between the hub and GoTube.

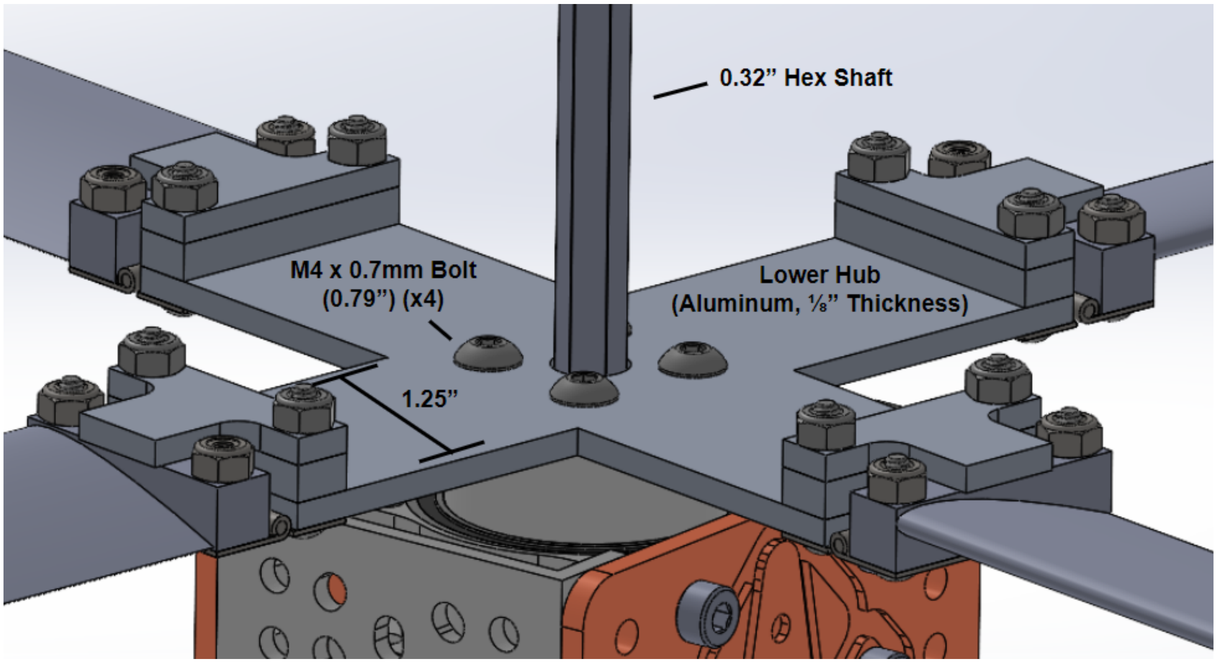


Figure 4.17: Lower hub assembly with four attached propeller blades.

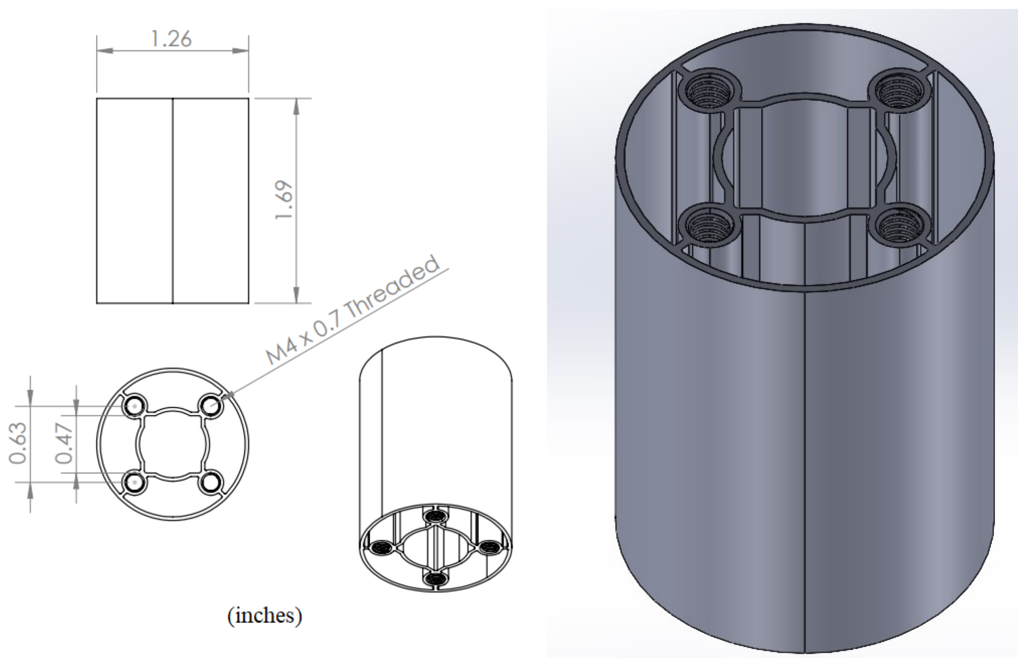


Figure 4.18: Drawing and 3D model of GoTube component (in.).

Moreover, spring hinges are used to connect the hubs to the propeller blades. The reason behind having an extruded, four-prong design is to allow for our propeller blades to be long enough such that they can generate a sufficient amount of lift. They will fold down to the sides of the payload bay and once released, the hinges will spring upwards to a horizontal position, maximizing propeller length. The folded position within the payload bay can be seen in Figure 4.19. However, it is important to note that the spring hinges will want to push upwards at all times, meaning they will be pushing against the payload bay, coincidentally keeping the payload centered within the bay.

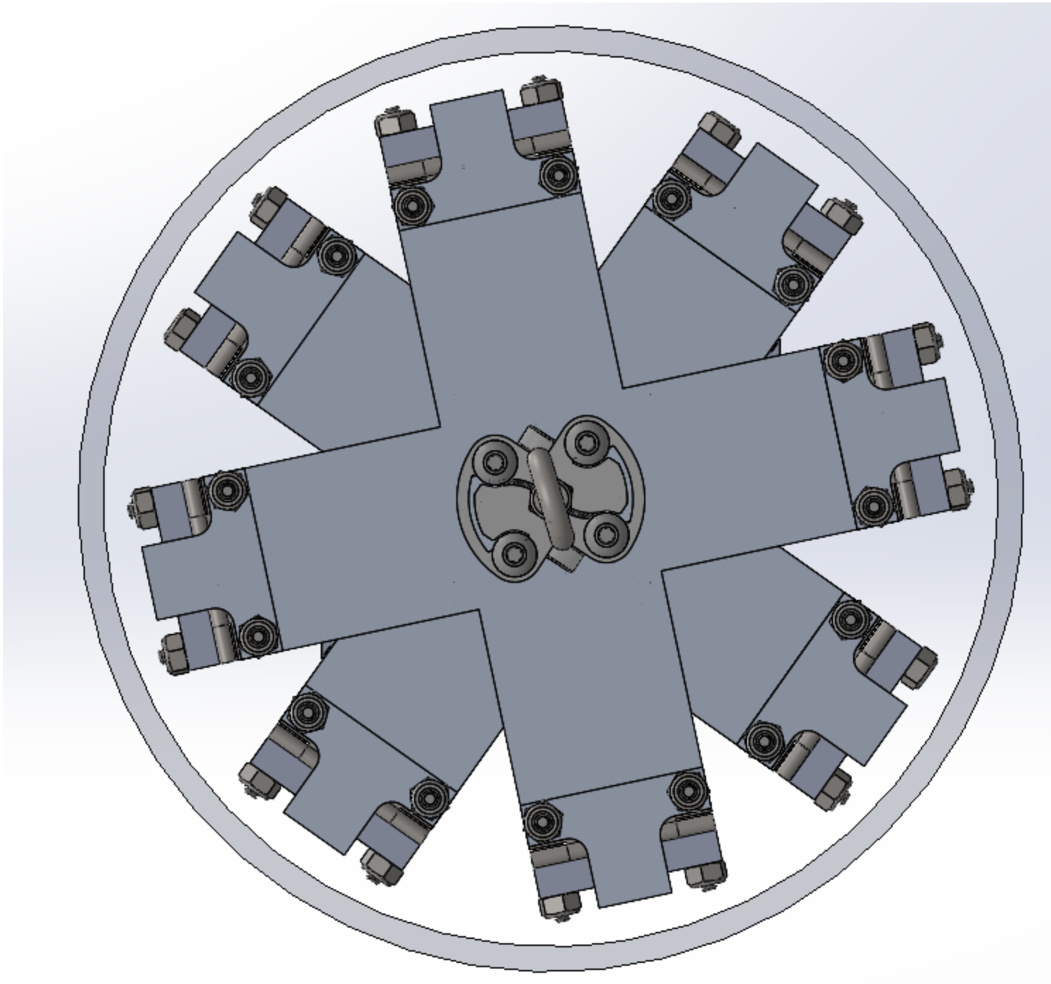


Figure 4.19: Top view of folded propeller blades.

In the hub-to-propeller assembly, there is the spring hinge, a hard stop, a spacer, and the respective bolts and lock nuts. The springing motion of the hinge, or centripetal forces once the rotors are spinning, could potentially extend the blades beyond the horizontal position. Thus, hard stops are put in to prevent any unwanted movement. The spacer is in place to prop the hard stop up to where the propeller blade's connection block lies. This layout can be seen in Figure 4.20.

The spring hinges have hole sizes meant for 4-40 threads. Hence, 4-40 bolts and lock nuts were used for the entire hub-to-propeller assembly. The bolts have button heads to streamline the flow and reduce any unnecessary drag. At the top side are 4-40 lock nuts to prevent any disassembly under high centripetal forces and vibration. The hard stop will be made out of 1/8 in. aluminum sheet metal. The spacer will also be made out of aluminum but using a different size sheet as it cannot accommodate the 1/8 in. thickness. Meanwhile, the propeller blade will be printed using the SLA resin printer, Formlabs 3L, at NCSU.

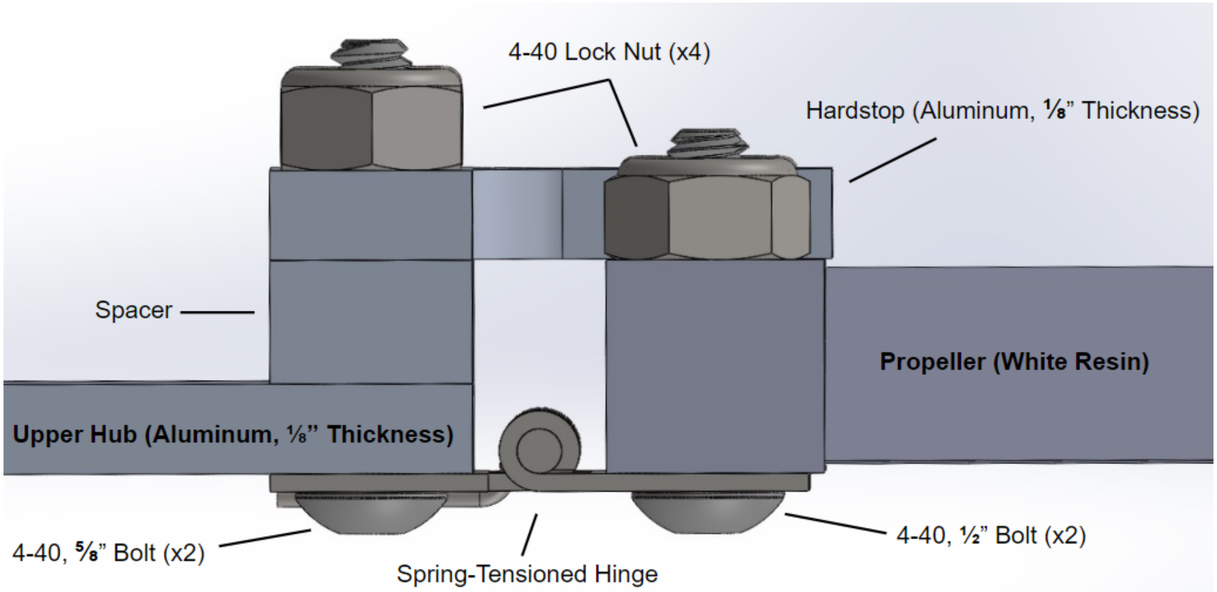


Figure 4.20: Side view of the extended, horizontal propeller blade position.

In order to demonstrate the top view's orientation, Figure 4.21 is helpful to see the folded propeller blade's orientation. The spring will allow it to rotate further, but this is the maximum angle that will be required in order to fit within the payload bay. This also shows an added benefit of having button head bolts as other types would collide. It is also why the lock nuts have to be on top.

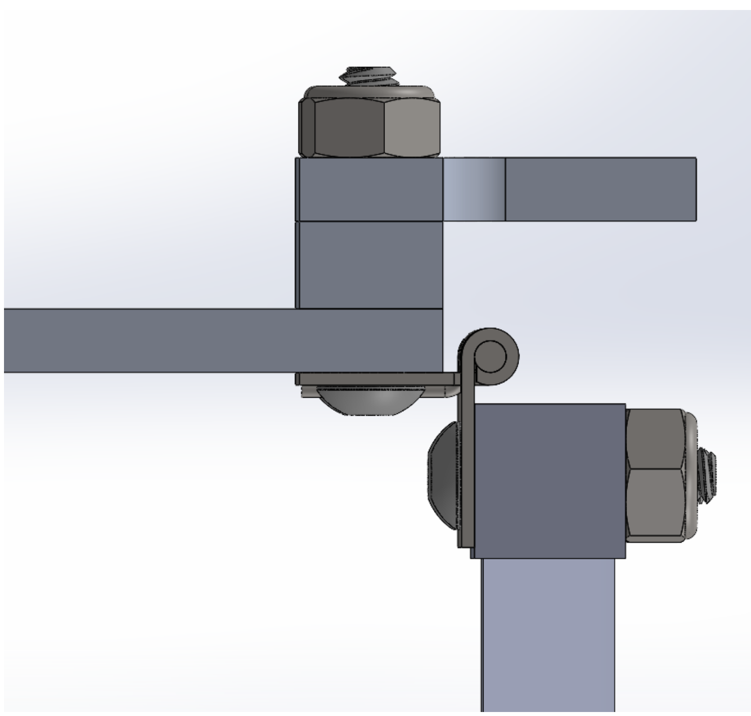


Figure 4.21: Side view of the folded, vertical propeller blade position.

Rotor Blades

The NACA-0015 airfoil was chosen as the airfoil shape for the rotor blades due to its relatively high stall angle compared to other symmetric airfoils. Using a calculated Reynolds number of approximately 200,000 for a 14” radius rotor blade with a 1.25” chord rotating at 2500 RPM, the stall angle of the NACA-0015 is roughly 15 degrees.

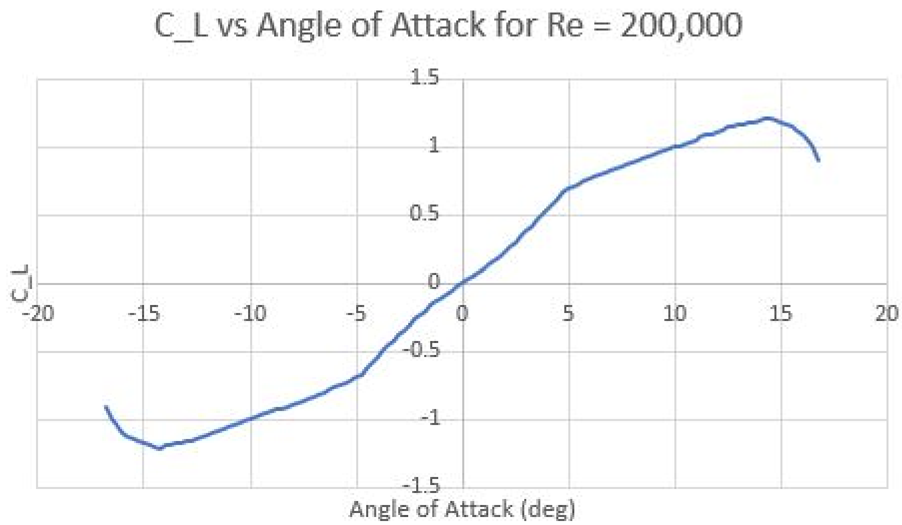


Figure 4.22: Coefficient of Lift vs Angle of Attack for NACA-0015

For early design of the system, an angle of attack (AOA) of 10 degrees was chosen. This AOA provides a high coefficient of lift while leaving roughly 5 degrees of headroom in order to avoid stalling the rotor blades. Further analysis using CFD will be conducted in order to better approximate the lift generated at various operating speeds as well as determining how many rotor blades are necessary for the mission and the space required between rotor hubs.

Gearbox

The leading design uses bevel gears to centralize the entire gearing system and make the payload as symmetric as possible. This helps to avoid any unnecessary body rotation and drifting.

To digest what is going on in the system, a condensed view (Figure 4.23) and an exploded view (Figure 4.24) were made. Looking at the condensed view, five bolts are shown to see what is constricting horizontal movement. These are screwed into three bearings, two for the GoTube and one for the hex shaft. These will help with stability and preventing the rotors from tilting. As seen in the exploded view, there is a fourth bearing. This flange bearing below the Lower Hub fits into the GoTube while having the 0.32 in. hex diameter for the hex shaft. This will help keep the two hub attachment pieces concentric to each other, avoiding any tilt of the shaft relative to the GoTube. To alleviate pressure on the hub and flange bearing, a spacer (labeled in Figure 4.24) will be cut out to fill in the gap where the bolts screw into the GoTube.

Furthermore, there is a tight space between the bevel gears where a 2-sided, 1-post hex bearing can fit. It is attached to the sides of the gearbox with a 0.47 in. long, M4 x 0.7 mm bolt on each side. The two GoTube bearings are also secured to each side of the gearbox with the same bolts. Additionally, there are spacers, upper and lower, to constrict any vertical movement of the GoTube-to-lower hub assembly. They utilize the stationary bearings to hold this assembly in place. These can be seen better in Figure 4.25.

The other two flange bearings on the left and right of the side view serve to hold the vertical bevel gears in place. There is a hole cut-out in the U-channel where the bearing will be placed within. A 0.95 in. long, 0.32 in. diameter hex shaft will sit inside the bearing and bevel gear. Additionally, the hex shaft has a clip to make it flush against the flange bearing. The bevel gear has a set screw to fasten to the hex shaft. However, there are slight gaps between the flange bearings and gears. Hence, thin spacers are put in between these two components to prevent any horizontal motion.

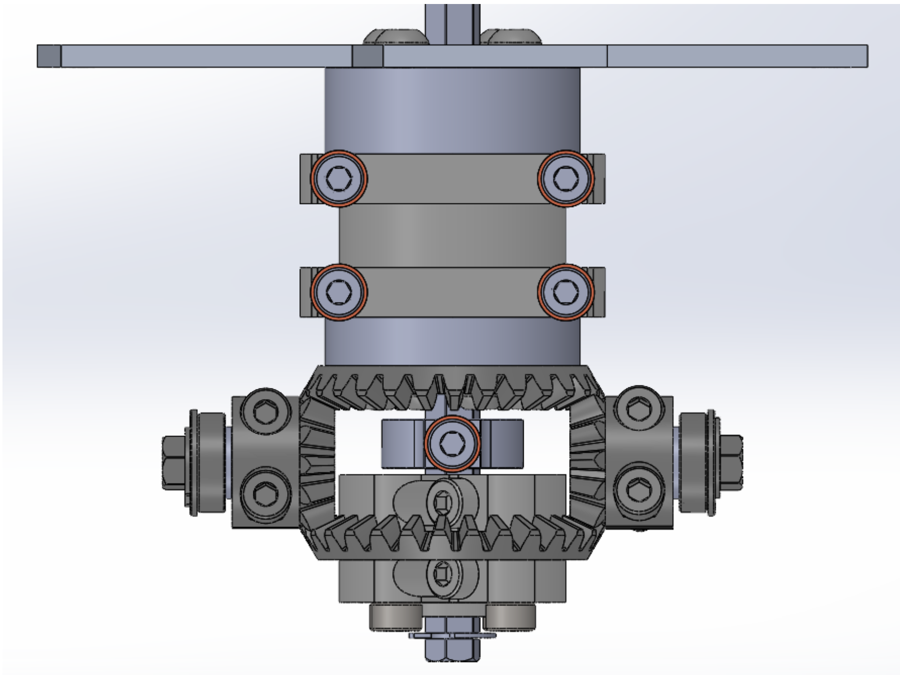


Figure 4.23: Bevel gear design for contra-rotation (condensed, side view).

Beyond the bearings, the upper, horizontal bevel gear is bolted to the GoTube with four 0.55 in. long, M4 x 0.7 mm bolts. The bevel gear has pre-cut holes to do this with ease. To keep the upper bevel gear in place, a spacer is fitted around the GoTube and situated between the lower hub and the upper GoTube flange bearing. As a result, the upper spacer is bolted down by the 0.79 in. bolts holding the lower hub and GoTube together. This spacer will prevent the assembly from falling and instead rests on the fixed GoTube bearing. A second spacer is placed between the lower GoTube bearing and the bevel gear. This will prevent the bevel gear from moving upwards. The lower, horizontal bevel gear will be bolted to two sonic hubs on each side with 0.79 in. long, M4 x 0.7 mm bolts. This will keep the lower bevel gear in place on the hex shaft. Keeping the bevel gears at their respective heights is important to reduce any potential friction and wear.

Once all together, the upper and lower bevel gears will be rotating in opposite directions. The hex shaft will attach to the lower bevel gear, which will feed through the GoTube and lower hub, and attach to the upper hub. Meanwhile, the GoTube will be attached to the upper bevel gear, which is connected to the lower hub. Thus, the upper and lower hub will rotate in opposite directions around the same axis using the same motor.

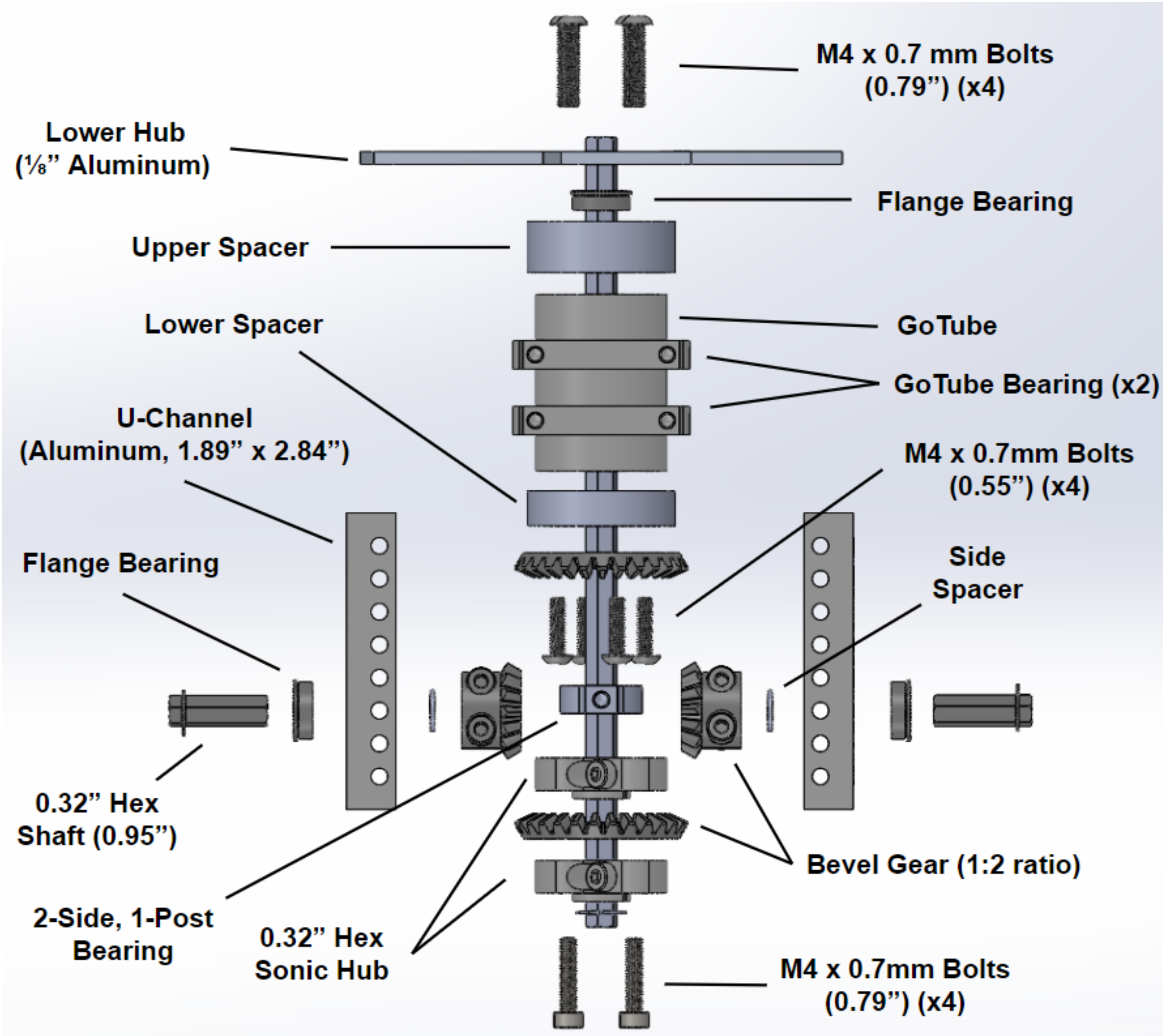


Figure 4.24: Bevel gear design for contra-rotation (exploded, side view).

To finalize the gearbox, the outer structure must be put together. The U-channel and custom-made orange plate are both used to hold the bearings in and are assembled together. Each are 1/8 in. of aluminum. They are bolted together at the lips of the U-channels and ends of the orange plate with 0.47 in. long, M4 x 0.7 mm bolts and lock nuts, seen in Figure 4.25. There will be four sets of the bolts and lock nuts at each corner of the gearbox to maximize strength while reducing any unnecessary weight. In total, there will be 22 bolts, 10 for the bearings and 12 for the U-channel and orange plate assembly. This does not include the fourth bolt and lock nut at the bottom of the components.

This fourth bolt is a button head, 0.55 in., M4 x 0.7 mm bolt but goes through a threaded L-bracket. There are two bolts and lock nuts being used in the vertical direction but the innermost bolt and lock nut only hold together the L-bracket and orange plate, disregarding the U-Channel. This is because the U-Channel has width for only one bolt. The third threaded hole in the L-bracket is not usable as the lock nut would interfere with the lower bevel gear. In total, there are eight M4 x 0.7 mm button head bolts and lock nuts for securing the L-bracket to the gearbox. This sums up to 20 M4 x 0.7 mm lock nuts for the gearbox.

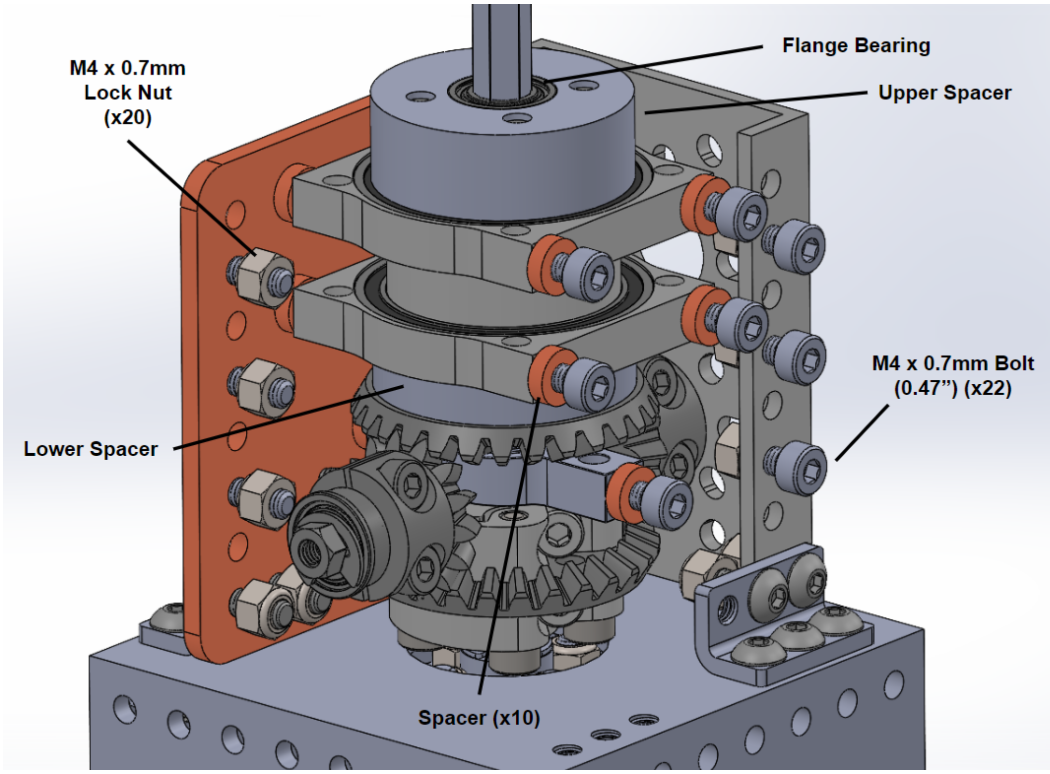


Figure 4.25: Gearbox (inside view).

Looking from an outside view of the gearbox in Figure 4.26, it is more clearly defined how it will be assembled. To connect to the motor housing, the L-bracket will be bolted down. The holes in the motor housing will be tapped as there is not enough space for a nut to fit. The bolt will be close to the wall and thus, tapping the motor housing is required, allowing the bolt to assemble both components together rigidly. These bolts are button headed, 0.55 in. long, and M4 x 0.7 mm threaded as are the other button head bolts because space does not allow for the standard, flat head type. That will total to 20 bolts of this kind.

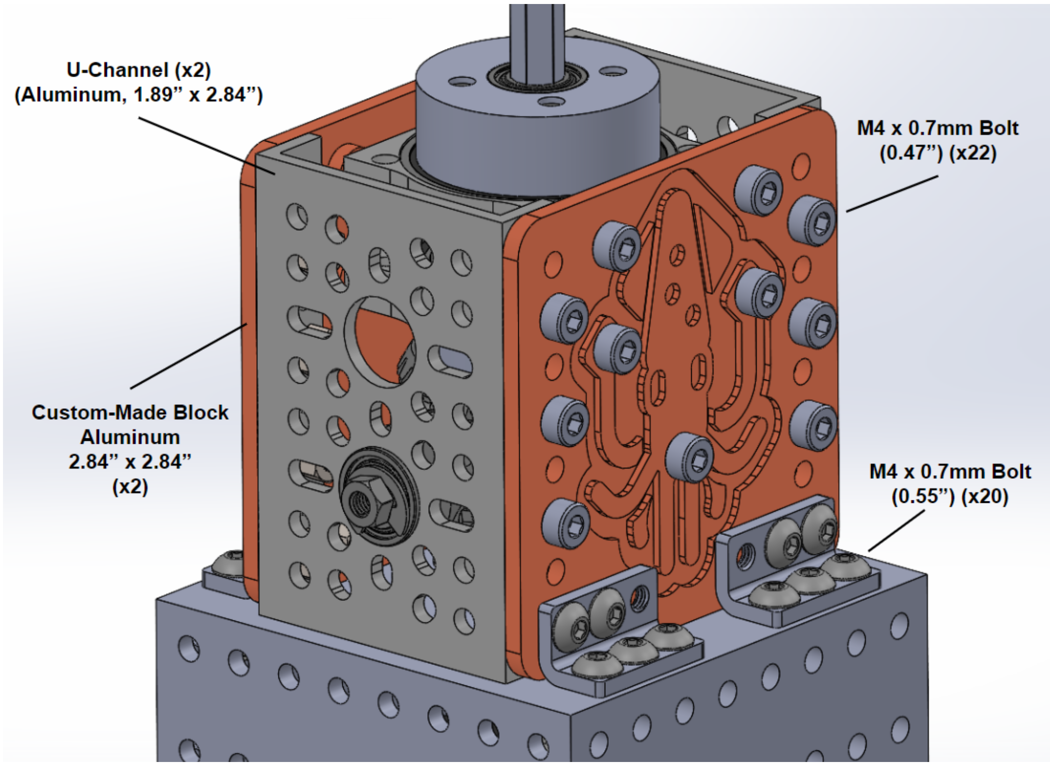
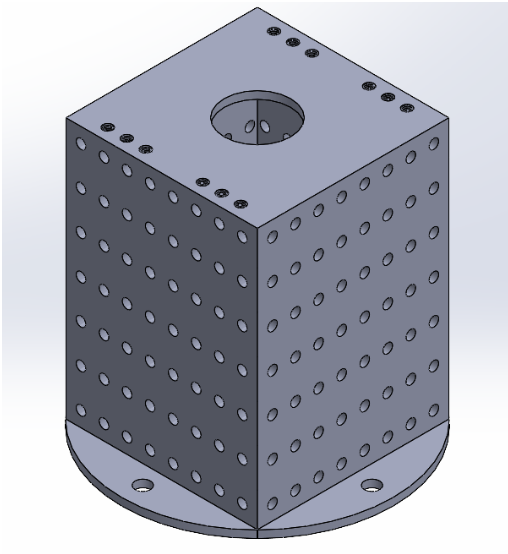


Figure 4.26: Gearbox to motor housing assembly (outside view).

Motor Housing

First, a motor housing must be established. This will be made out of 1/8 in. aluminum and custom-made. It will be made by cutting out the component seen in Figure 4.27. Once cut out, this component will be bent to the necessary dimensions. The 1.1 in. central hole is for the motor to hex shaft assembly. Near the edges of the top surface are the threaded holes for the M4 x 0.7 mm bolts that thread through the L-bracket and into this motor housing. On the sides of the motor housing are holes to let the motor exhaust heat as needed while ensuring there is still structural strength. The holes at the end of the flat plate and on the bottom-most surface (once bent) are 1/4 in. diameter for 1/4"-20 rods that will help assemble the payload body together (seen in Figure 4.31).



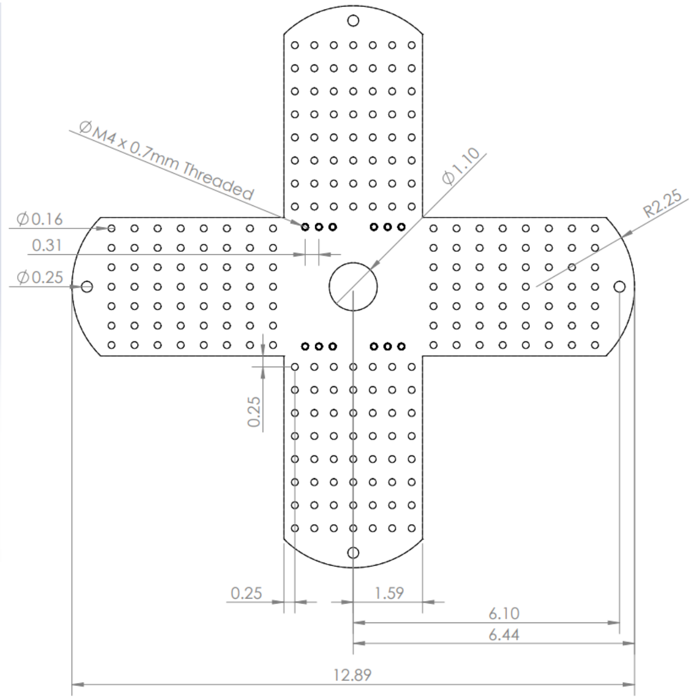


Figure 4.27: Motor housing and the flat plate for manufacturing (in.).

Within the motor housing is the motor, placed and secured centrally. To connect the motor to the shaft, flanges are used to fasten them together (seen in Figure 4.28). It is not pictured but these will have set screws to fasten each flange to their respective shaft. They are bolted together with four 0.47 in. long, M4 x 0.7 mm bolts and lock nuts. This should be enough for the motor to rotate the shaft and subsequent weight.

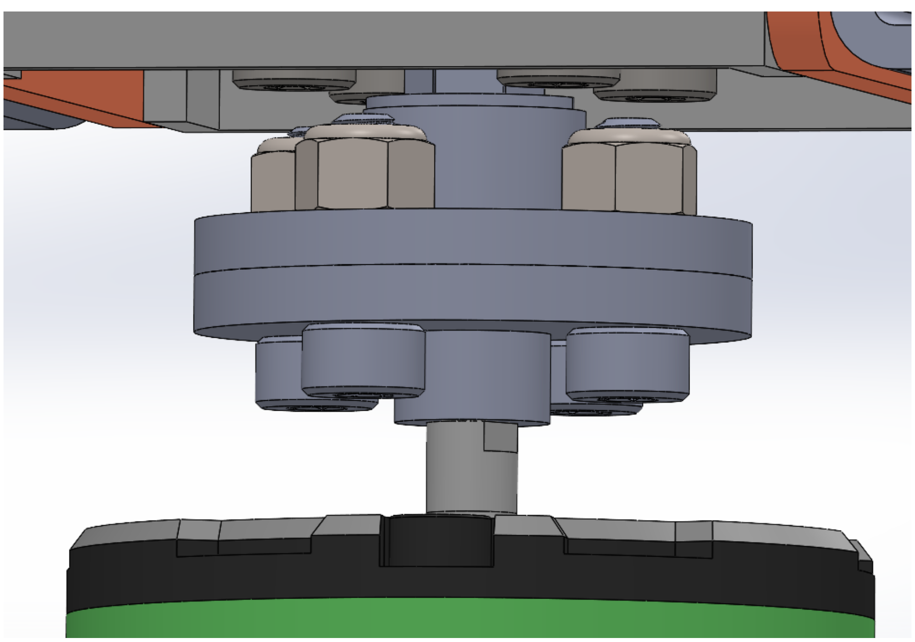


Figure 4.28: Motor shaft to hex shaft assembly.

Landing Legs

Among the landing leg options, the top-mounted, folding linkage system was chosen as the leading design (seen in Figure 4.7). This was chosen because of it's simple fabrication and strength. It would consist of two linkages, a 1/8 in. thick aluminum leg, a hinge, and a spring. A mounting piece will act as an interface between the cylindrical body and hinge. Seen in Figure 4.29, the spring will be attached to the connection point of the two hinges and fixed to a point on the mounting piece. Initially, the spring will be extended, forcing the leg outwards. However, since it will be in the payload bay the legs will only extend up until the walls. They will fully deploy once exiting the bay as seen in the second part of Figure 4.29. This is a simple method using an entirely mechanical and automatic spring mechanism.

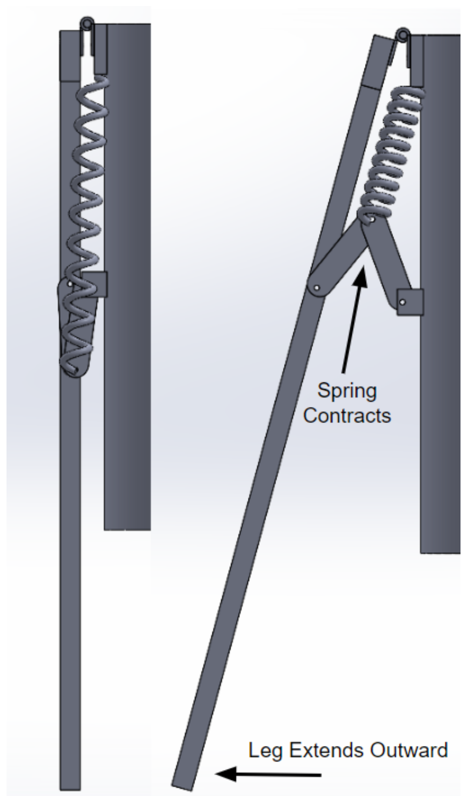


Figure 4.29: Springing motion of the landing legs.

Electronics

The electronics will be controlled by a Raspberry Pi 4b which is capable of simultaneously logging data from sensors and controlling the speed of the motor. Inertial Measurement Units (IMUs) will be used to measure the force of impact upon landing as well as the deceleration event after payload deployment. A gyroscope will be used to measure the angular velocity of the SAIL in order to calculate sustained centripetal g-forces during the descent. Lastly, an RF receiver will be used to receive a command to start rotating the rotor blades shortly after release from the deployment system in order to ensure that the blades do not spin while still attached to the shock cord. Figure 4.30 below shows the current electronic layout.

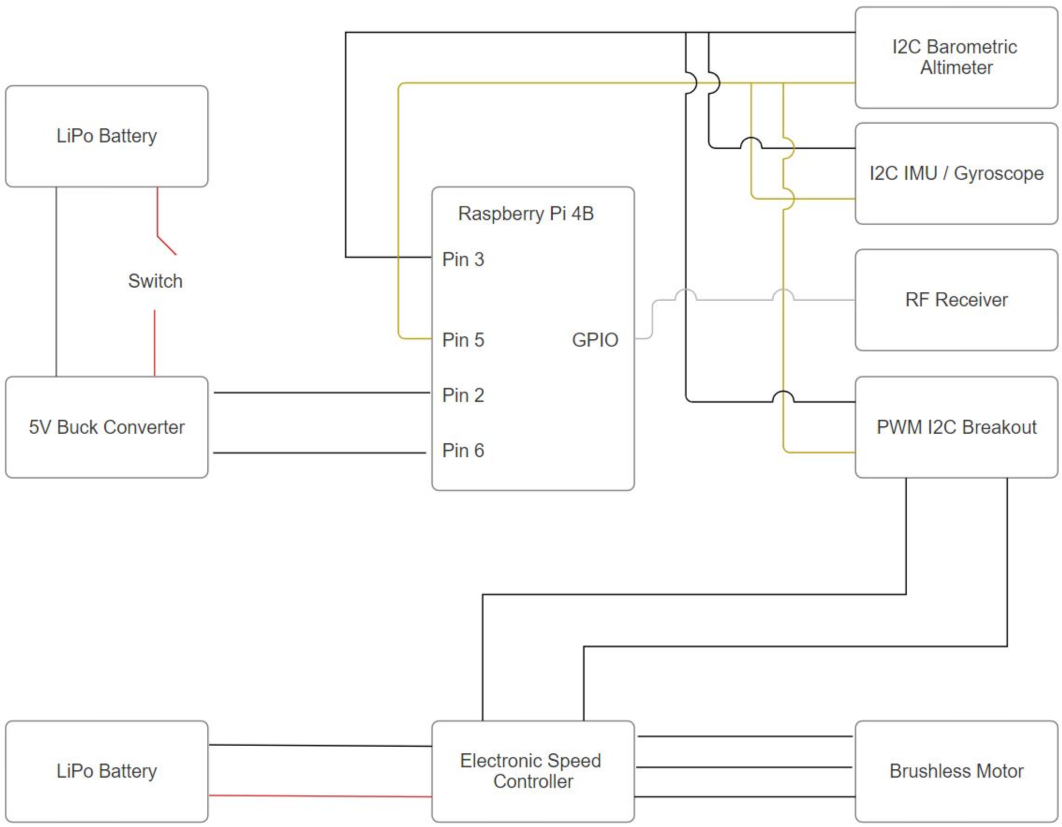


Figure 4.30: Electronics block diagram.

The electronics will be located on 3D printed sleds that will be secured inside the vehicle using threaded rods and nuts. This method of securing electronics is also used in the avionics bay and is a proven way of mounting electronics. These threaded rods will also secure the bulkheads and motor/hub assembly to the vehicle (Figure 4.31).

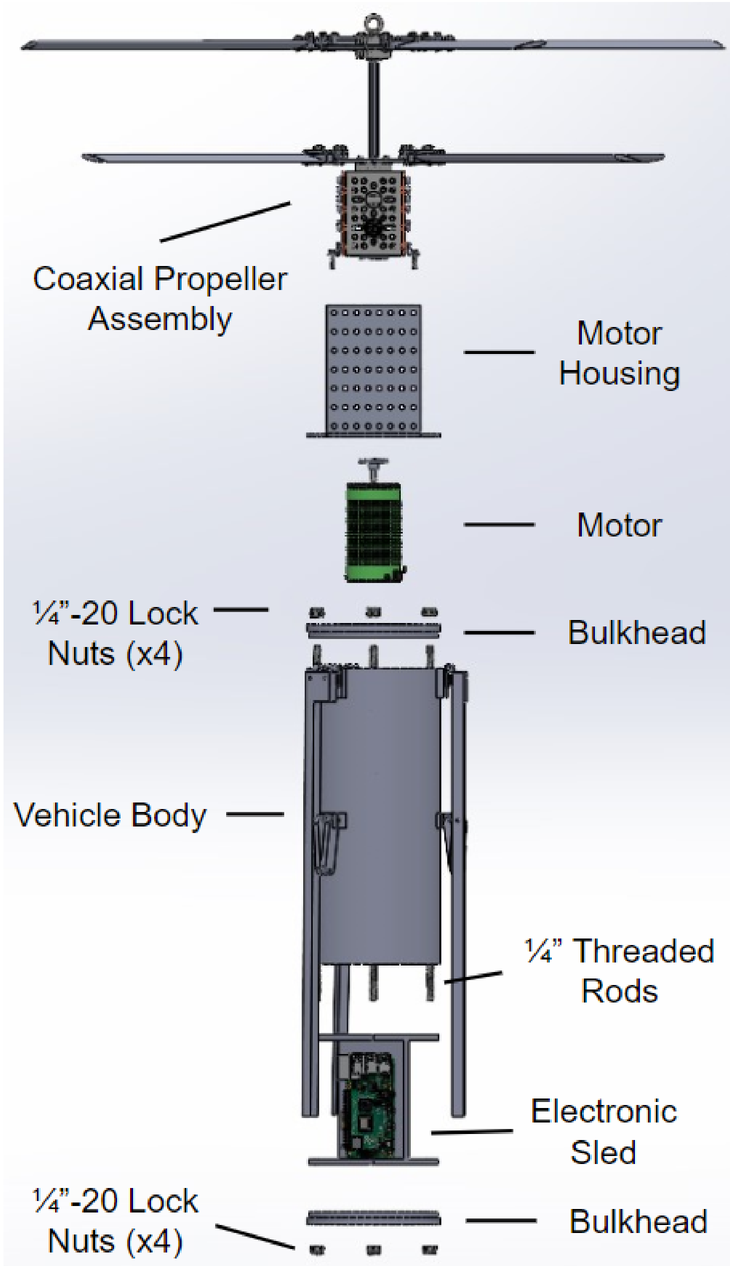


Figure 4.31: Exploded view of the SAIL with labeled components.

Table 4.9 below lists the major components and their estimated weights.

Component	Unit Weight (lb)	Quantity	Component Weight (lb)
Rotor blade	0.073	8	0.584
Rotor hub	0.113	2	0.226
Rotor hinges	0.009	8	0.072
1:2 bevel gear	0.106	2	0.212
GoTube bearing	0.042	2	0.084
Flanged bearing	0.008	3	0.024
Sonic hub	0.031	4	0.124
Rotor shaft	0.137	1	0.137
Rotor assembly hardware	Varied	1	0.426
Bulkhead	0.168	2	0.336
Brushless motor	1.317	1	1.317
Electronic Speed Controller	0.209	1	0.209
Motor housing	0.768	1	0.768
1/4” Threaded rods	0.044	4	0.176
1/4-20 Nuts	0.01	8	0.08
Fiberglass tube	0.562	1	0.562
Landing leg	0.23	3	0.69
Landing leg linkage	0.018	6	0.108
Raspberry Pi 4B	0.13	1	0.13
LiPo Battery (motor)	0.985	1	0.985
LiPo Battery (pi)	0.225	1	0.225
STEMnauts	0.003	4	0.012
		Total Weight	7.49

Table 4.9: Payload Weight Estimates

4.4.2 Manufacturing Methods

NCSU provides numerous manufacturing capabilities including hand fabrication (e.g. mills, lathes, etc.), 3D printers, water jets and laser cutters. The following section discusses potential manufacturing methods for the major components.

Rotor Blades

At NCSU, a Formlabs 3L SLA resin printer is available for student use (seen in Figure 4.32). The build dimensions are 13.2 x 7.9 x 11.8 in.. To achieve the desired propeller blade length, this can be used to print a custom, smooth airfoil that can be attached to the payload with ease. The ability to print and easily re-print altered airfoil designs makes for quick and easy prototyping.

An alternative solution to creating the propeller blades is 3D-printing. The downside with this is that the build dimensions are not big enough for the required propeller blade length. Thus, a compartmental design must be made. This can cause a weak point and potentially hurt the effectiveness of the propeller blades. Another alternative solution is cutting foam and laying it up with carbon fiber. The drawbacks of this are the expensive nature of carbon fiber and the extensive production process. Meanwhile, the resin printer can produce the propeller blades, currently at $8.40 per blade, cheaply and quickly.

There are various resins that NCSU supplies: Draft, White, Clear, and Flexible. Flexible resin was disregarded as the propeller blades should be mostly rigid. The other resins have an elongation of 4-6%, which is plenty for the payload’s purposes. White and Clear resin both have comparable properties. Draft resin has a tensile modulus of 334 ksi while the Clear/White resin has a tensile modulus of 402 ksi. The Clear/White resin also has a 6% elongation compared to the Draft resin’s

4%. Thus, White resin will be used for the propeller blades for the higher strength. Additionally, the color is preferred over clear for its improved clarity to see any breaks.

Figure 4.32: Large resin printer located at NCSU (Formlabs 3L).

Aluminum Parts

The rotor blade hubs, landing legs and landing leg connecting rods will all be made of aluminum 6061. Aluminum was chosen due to its high material strength and low weight in addition to ease of manufacturing. The parts will be cut out of a sheet of stock aluminum alloy using an Omax 1515 Water Jet (shown below in Figure 4.33). This water jet is capable of cutting sheet stock up to 60 x 48 in. and up to 2 in. thick to an accuracy of +- .0003 in..

Figure 4.33: "Big" Waterjet located at NCSU.

4.5 SAIL Release System

4.5.1 Leading Release Configuration

The SAIL will be stowed in the payload bay with the top of the rotor shaft looped through a shock cord. This shock cord will be attached to an eye bolt on one end and the radio-controlled latch on the other. Both of these attachment points will be located on the nose cone bulkhead. Once the latch is released, the shock cord will feed through the eye bolt on top of the payload, releasing to SAIL into free fall.

Stowing and Release

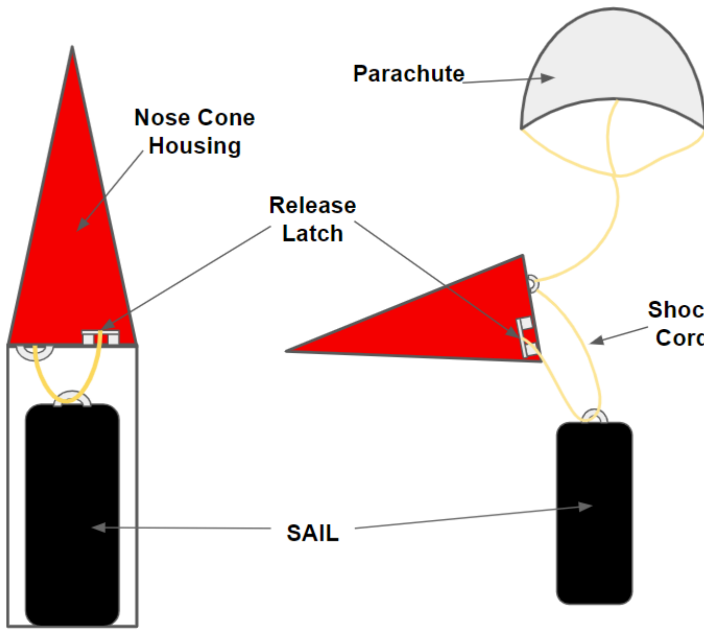


Figure 4.34: Stowing configuration of SAIL pre-release.

For retaining and releasing the SAIL on the sub-scale launch vehicle, a custom latch will be used, shown in Figure 4.35. The latch release will be powered by a 35 KG servo controlled by an Arduino. Once the Arduino receives a command sent over RF from ground control, the servo will rotate, providing clearance for the shock cord to drop through. The bulkhead will be cut to feed the shock cord through an opening in order to guide the cord upon release.

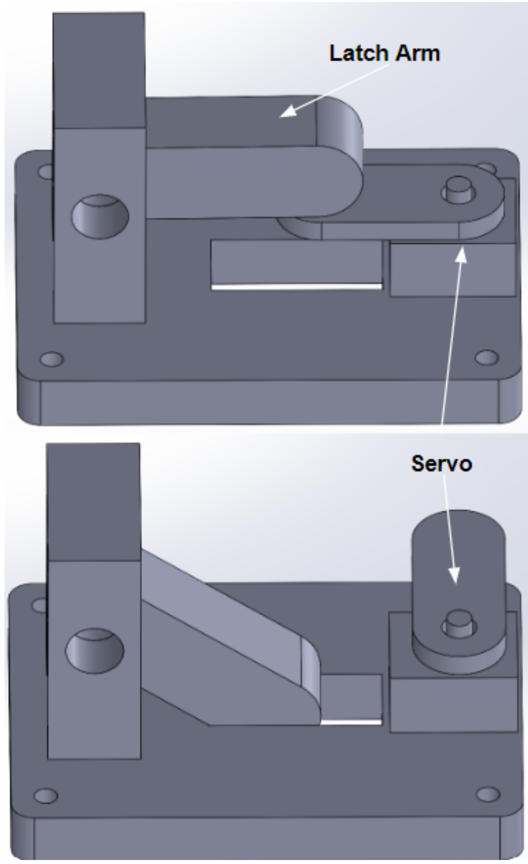


Figure 4.35: CAD model of custom servo latch.

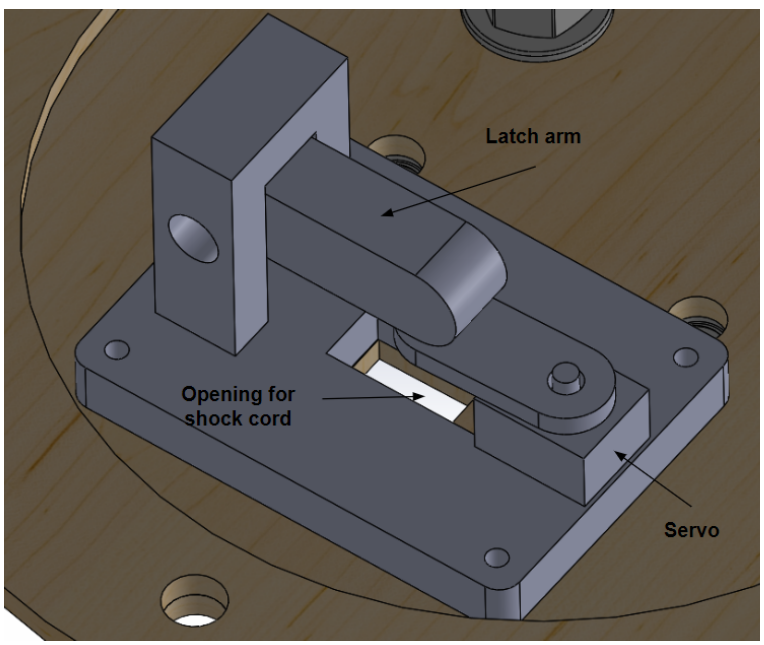


Figure 4.36: Mounting configuration for latch on nose cone bulkhead.

Using predicted descent velocities of around 90 ft/s decreased to 20 ft/s upon separation and with an estimated impulse time of 0.01 seconds, the shock cord will transfer a force of around 1100 lb. to the latch. This will require both the latch and servo arms to be able to withstand this shear force, which comes down to material selection. For a 3D printed application, carbon fiber PETG can be used, which can withstand a max stress of around 6100 psi. With the current dimensions of the latch arm, 0.35 x 0.95 in, the latch would be able to withstand up to a force of 2025 lb. This supports the idea that carbon fiber PETG would be a viable candidate. The leading idea is to have carbon fiber PETG components reinforced with metal, in order to ensure the latch does not sustain damage during separation. A similar configuration will be used for the servo arm.

The latch will be controlled by an Arduino connected to an RF receiver. Once RSO permission is given to release the SAIL, an RF command will be sent from the ground using an Arduino connected to an RF transmitter. When the RF command is heard by the receiver onboard the launch vehicle, the launch vehicle Arduino will send a signal to open the latch. At this point, the shock cord will be released from the latch and follow through the loop on top of rotor blades, releasing the SAIL.

An XBee Pro S3B transceiver will the be the operational RF hardware for the nose cone release mechanism. With a line-of-sight range of up to 45 KM, or around 28 miles, there should be no issue with the release mechanism being out of range of ground control. While this module has the highest current draw of the units that were considered, the XBee Pro has proven to have been effective in previous NCSU launch teams. Considerations will have to be made for the current draw, such as keeping the receiver off until the transmitter is ready to send the command to open the latch, as well as battery selection. While there is a higher current draw during transmitting, this would not affect the nose cone assembly as transmitting will only be done on the ground.

Sub-scale Nose Cone Release Configuration

The working configuration for the release system in the nose cone is shown in Figure 4.37. The transceiver will be soldered onto a solderable breadboard, along with an Arduino nano to control the input command to the servo. The breadboard will be secured on a 3D printed electronics sled. The sled will be placed onto a set of threaded rods, which will be bolted onto the nose cone bulkhead. A 7.4 V LiPo battery will be used to power the breadboard.

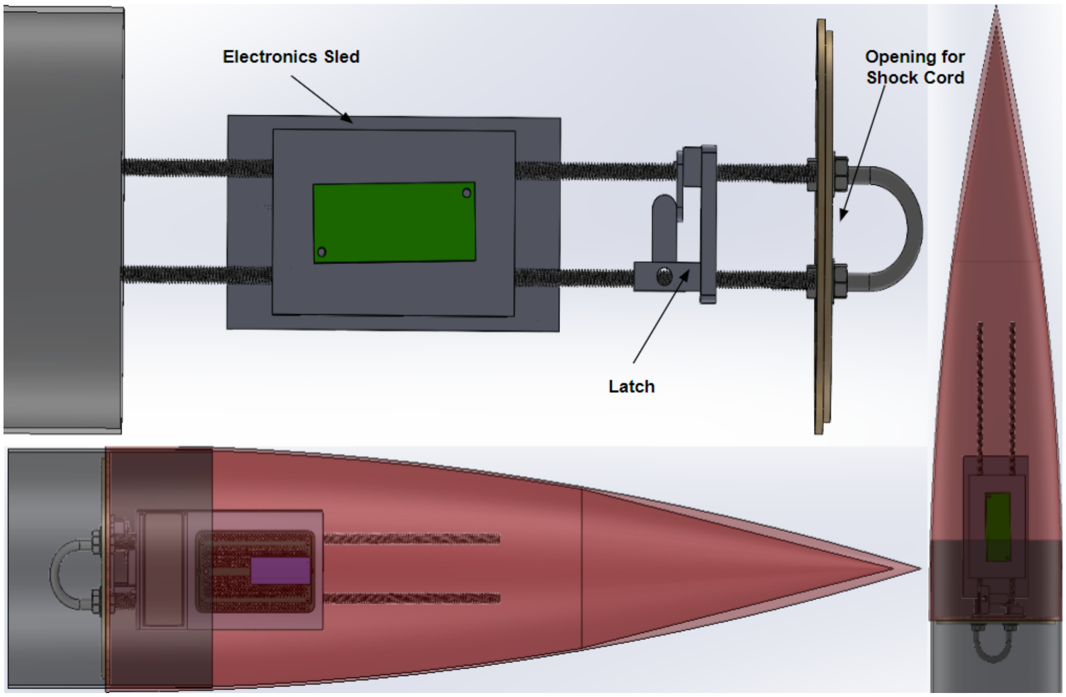


Figure 4.37: Exploded view of latch release in nose cone.

Figure 4.38 shows the contents of the nose cone sled. An Arduino nano will be the controller of the assembly. A Big Red Bee GPS tracker along with a pull pin switch will be in the nose cone as the it will be its own separate section once the payload is released.

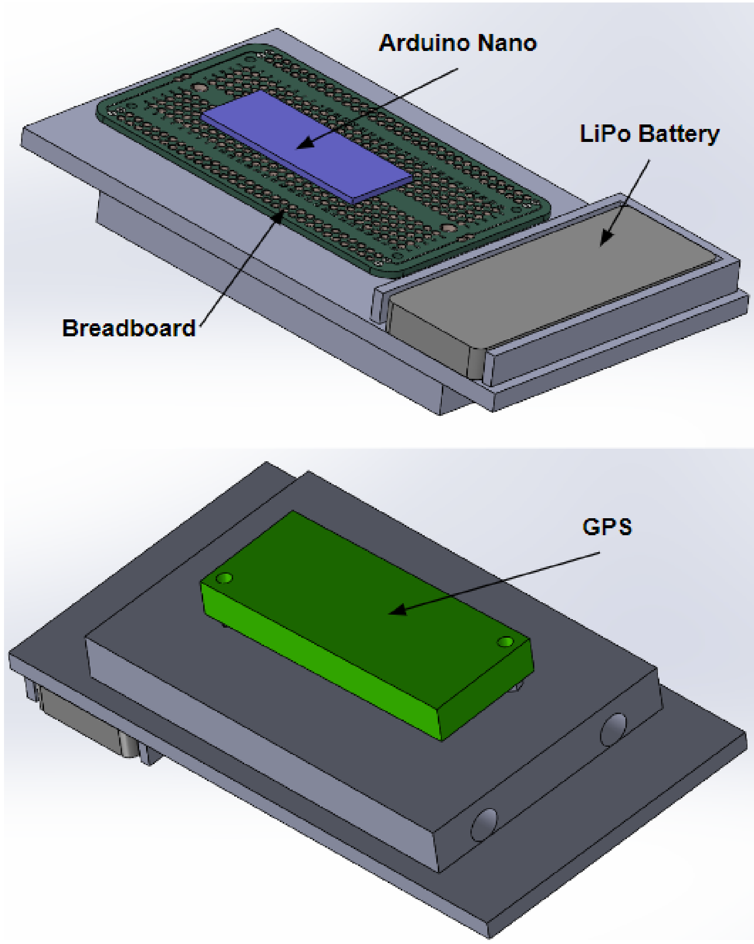


Figure 4.38: CAD model of nose cone electronics sled.

The core of the wiring will be contained on a solderable breadboard. This breadboard will support an Arduino nano, the XBee transceiver module, and a buck converter. Structure for this wiring is represented by the block diagram shown in Figure 4.39.

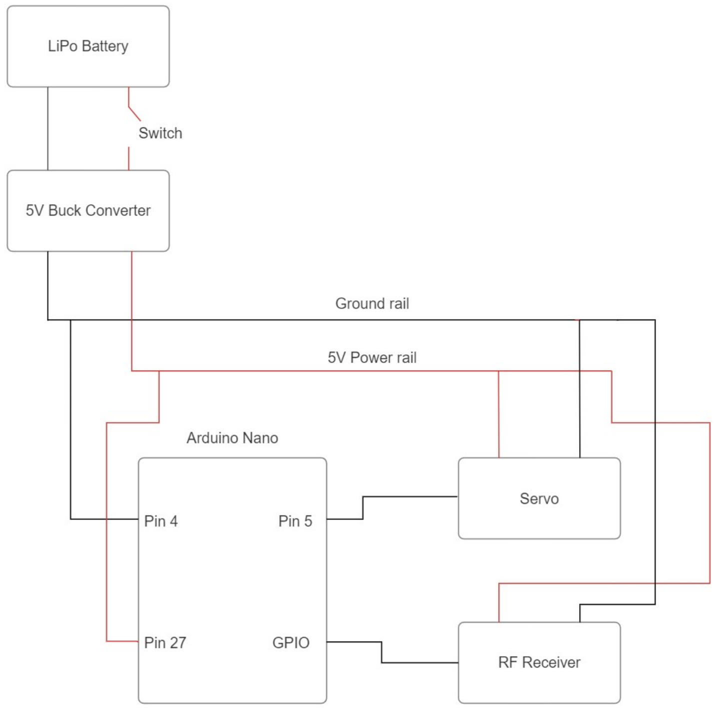


Figure 4.39: Electrical block diagram for nose cone sled.

4.6 STEMnaut Selection

Small resin ducks were chosen to be the STEMnauts for the launch. The commander of the launch will be a STEMnaut named Jeffrey pictured below in Figure 4.40. Jeffrey has previous flight experience flying on the NCSU's 2022-2023 SLI competition launch vehicle. Jeffrey will be accompanied by 3 fellow STEMnauts that will be named at a later date.

Figure 4.40: Commander Jeffrey

The STEMnauts will be contained in a bowl type housing (Figure 4.41) 3D printed into a sled. The STEMnauts will then be restrained using a strap running across their backs. This sled will be mounted near the top of the payload and will be easily removable for exhibiting the STEMnauts before their flight.

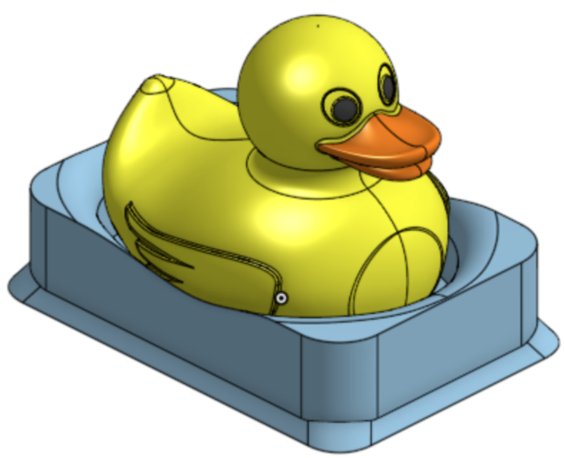


Figure 4.41: STEMnaut Chair

5 Safety

Megan Rink is the 2023-24 Safety Officer. Megan is responsible for ensuring the safe operation of lab tools and materials, including, but not limited to, drill presses, hand tools, band saws, power tools, flammable items, and hazardous materials. Megan is required to attend all launches and must always be present during the construction of the launch vehicle, payload, and associated components. Additionally, she is responsible for maintaining all lab space and equipment up to and exceeding NASA, MAE, and Environmental Health and Safety standards. This includes, but is not limited to, displaying proper safety information and documentation, maintaining the safe operation of a flame and hazardous materials cabinet, keeping lab inventory, and stocking an appropriate first aid kit. She can be reached via email at mdrink@ncsu.edu.

5.1 Safety Documentation Methods

Safety documentation will continue to be performed through FMEA analysis and Likelihood-Severity (LS) matrices. These matrices detail each hazard and the corresponding causes, effects, and LS, as determined by the matrix. Additionally, mitigation methods for each hazard have been analyzed and the LS after mitigation has been determined.

Verification of safety procedures is checked through various sources, including but not limited to, inspection, Launch Day checklists, NAR Safety Code, TRA Safety Code, and HPRC standards.

Below is the Likelihood-Severity matrix upon which all of the FMEA tables are based. Failure modes are defined as any hazard that is color-coded as orange or red. LS ratings both before and after mitigation are analyzed systematically in order to determine the percent likelihood and percent severity of failure for each launch vehicle system. There are additional matrices to better visualize the LS percentages both before and after mitigation for each subsection.

Table 5.1: FMEA LS Matrix

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

5.2 FMEAs

Table 5.2: Launch Vehicle Hazards

Label	Hazard	Cause	Effect	LS Before	Mitigation	LS After	Verification
Hazards to and from Bulkheads							
S.B.1	U-bolt failure	Excessive deployment force	Ballistic reentry	4A	Distribution of load during construction	4A	Tests and Verification Pending
S.B.2	Nosecone bulkhead bolt failure			4A	Load management during construction	4A	Tests and Verification Pending
S.B.3	Cracked bulkhead	Excessive stress on stress points	Bulkhead separates from airframe	3D		3B	Tests and Verification Pending
S.B.4	Bulkhead delamination	Excessive axial stress caused by shock cord connection points		3D		3B	Tests and Verification Pending
S.B.5	Separation of bulkhead from airframe	Epoxy is softened Latch connections cause excessive force	LV stabilization is changed	3D		3B	Tests and Verification Pending
S.B.6	Bulkhead exposure to hot ejection gases	Motor or ejection charges cause excessive heat		3B	Ensure LV is kept in optimal environmental conditions	3A	Tests and Verification Pending
Hazards to and from Removable Fin System							
S.F.1	Bolt failure	Excessive force caused by motor or landing	CATO, loss of stability, potentially repairable damage to LV components	3B	Bolts and rods selected have a high safety factor	3A	Tests and Verification Pending
S.F.2	Fin runners, threaded rods, or fin can buckle		CATO, loss of stability	3B		3A	Tests and Verification Pending
S.F.3	Thrust plate failure		CATO, airframe damage	3A	Material selected during design phase has a high safety factor	2A	Tests and Verification Pending
S.F.4	Fin breakage	Excessive force upon landing or fin flutter	Loss of stability in flight	3B	Reinforcement during construction	1C	Tests and Verification Pending
S.F.5	Delamination of or cracks to centering ring	Excessive force caused by motor	CATO, loss of stability, motor not securely held	3A	Proper construction techniques	1A	Tests and Verification Pending
S.F.6	Motor retainer-airframe connection failure	Epoxy weakened by heat or other factors	Motor descends prematurely and separate from LV	2B	Epoxy selected during design phase is rated for expected temperatures	1A	Tests and Verification Pending

Table 5.2 continued from previous page

Label	Hazard	Cause	Effect	LS Before	Mitigation	LS After	Verification
Hazards to and from Airframe							
S.A.1	Cracks in fin can body tube	Hoop stress caused by internal pressure	CATO, inability to relaunch LV	4A	Propellant grains are properly fastened in the appropriate motor tube, motor construction is overseen by mentors defined in Section 1.1.2	2A	Tests and Verification Pending
S.A.2	Cracks in AV bay body tube	Hoop stress caused by internal pressure	Inadequate force to separate LV sections	3B	Calculations performed to determine necessary amount of black powder, ejection tests performed prior to each flight	3A	Tests and Verification Pending
S.A.3	Zippering of body tube	Shock cord causes excessive forces; Excessively low altitude parachute ejection	Airframe rupture	2B	Fiberglass body tube and appropriately-sized couplers are used per SL Requirements 2.4.1 and 2.4.2	2A	Tests and Verification Pending
S.A.4	High-energy impact with ground	No or late parachute deployment	Airframe rupture	3B	Appropriate recovery system is used to decrease LV descent velocity	3A	Tests and Verification Pending
S.A.5	LV sections collide	Insufficient length of shock cord	Airframe rupture	3B	Appropriate recovery system is used to decrease LV descent velocity	3A	Tests and Verification Pending
S.A.6	Airframe exposed to water	Sudden inclement weather; LV lands in wet area of launch field	Airframe disintegration/ rupture; CATO	2C	Full scale LV airframe is constructed with fiberglass, sub-scale LV not constructed in inclement conditions	2B	Tests and Verification Pending
S.A.7	Airframe exposed to black powder	Uncontrolled ejection charges	Airframe disintegration/ rupture	1D	Airframes are constructed with heat-resistent materials	1C	Tests and Verification Pending
S.A.8	Body tube abrasion	High-energy impact with the ground; Body tube is dragged due to parachute re-inflation	Changes in LV center of pressure/stability, damage to LV	1C	Appropriate recovery system is used to decrease LV descent velocity; Launches will not occur in high winds	1B	Tests and Verification Pending

Table 5.3: Payload Hazards

Label	Hazard	Cause	Effect	LS Before	Mitigation	LS After	Verification
Hazards to/from Payload Structure							
PA.S.1	Cracking/breaking of payload	Impact between components of the payload and the inside of LV during launch/separation	Loss of power to payload electronics, loss of communication with payload, payload damage	1D	Payload is secured within the LV to prevent launch/separation forces from causing damage	1C	Verification Pending
PA.S.2	Payload rotor failure	Rotor system does not function properly, payload high-energy impact with ground	Payload is destroyed beyond repair	4C	Payload system is tested before it is dropped from the full required height	4A	Verification Pending
Hazards to/from Payload Electronics							
PA.E.1	Damage to LiPo battery connection/low power	LiPo battery is not fully charged, friction due to contact between cable and housing	Loss of power to payload electronics, loss of communication with payload	3D	Voltage of battery measured prior to flight, all wired connections secured	1A	Verification Pending
PA.E.2	Over-voltaging of electronic components	Voltage from LiPo battery is higher than components can withstand	Electronics are fried and no longer usable	2D	Use of buck converters to regulate voltage into components	1D	Verification Pending
PA.E.3	Wire shortage	Wires are loosely connected and contact each other	Incorrect voltages are passed through the circuit, excessive current flow, possible fire hazard	2D	Wires are properly soldered, all exposed wire is covered in shrink wrap and secured with electrical tape	1D	Verification Pending

Table 5.4: Hazards from Environmental Factors

Label	Hazard	Cause	Effect	LS Before	Mitigation	LS After	Verification
Hazards to LV Structure							
E.S.1	LV contact with water	LV lands in irrigation ditch, body of water	Structural damage to airframe	4C	LV is made of water-resistant materials	4B	Verification Pending
E.S.2	LV collides with birds	Birds fly in close proximity to LV flight path		2B	Flight path confirmed to be clear by RSO ahead of launch	2A	Inspection: Checklist Section Launch Pad, NAR Safety Code #9
E.S.3	LV lands in tree	Large gusts of wind, wind drift	Inability to recover LV	3D	Launches will not occur if wind speed at launch field exceed 20 mph	3C	
Hazards to Personnel							
E.PE.1	Personnel have excessive contact with sunlight and heat	Lack of appropriate PPE, hot launch conditions	Heatstroke, dehydration, sunburn	4B	Personnel are provided with sunscreen and are highly encouraged to bring sunglasses, a tent is set up at the launch field for personnel to take shelter	2B	Inspection: Checklist Section Launch Pad/Recovery
E.PE.2	Personnel slip, trip, or fall	Uneven ground, debris on the ground, working near/next to irrigation ditches	Bruising, broken bones, concussion	4C	Personnel are required to wear closed-toed shoes to all launch day activities, only specific personnel are allowed on the launch field itself	2B	Inspection: Checklist Section Launch Pad/Recovery
E.PE.3	Rain or hail	Inclement weather conditions	Damage to airframe	3C	No launches occur during periods of inclement weather, weather is monitored and launches may be postponed, personnel take shelter as appropriate	3A	Inspection: NAR Safety Code #9
E.PE.4	Lightning strike			1D		1A	Inspection: NAR Safety Code #9
E.PE.5	Wet and/or icy terrain		Personnel slip, trip, or fall	2C		1C	Inspection: Checklist Section Launch Pad/Recovery
E.PE.6	Pollen or other allergens present at launch site	Seasonal allergens, personnel allergic to crops grown at launch field	Potentially severe allergic reactions	3B	Personnel are asked to make the Safety Officer aware of any environmental allergies, antihistamines and other OTC allergy medications are kept in the Launch Day Safety Box	2B	Inspection: Checklist Section Launch Pad/Recovery

Table 5.4 continued from previous page

Label	Hazard	Cause	Effect	LS Before	Mitigation	LS After	Verification
Hazards to Payload System							
E.PA.1	Payload contact with water	LV lands in irrigation ditch, body of water	Damage to payload electronics	1C	Mitigation pending	1C	Verification Pending
E.PA.2	Lightning strike	Inclement weather conditions	Damage to payload electronics	3C	Launches will not occur in inclement weather, local Tripoli Prefect dictates if launch weekends are postponed	2A	Inspection: NAR Safety Code #9
Hazards to Mission Success							
E.M.1	Damp propellant grains	High humidity	No motor ignition, LV does not fly	1D	Launches will not occur in inclement weather	1B	Inspection: NAR Safety Code #9
E.M.2	Damp black powder grains	High humidity	No motor ignition, LV does not fly	2D	Launches will not occur in inclement weather	1B	Inspection: NAR Safety Code #9
E.M.3	LV flight path blocked by birds	Flight path not clear at launch	LV does not reach intended apogee	2B	RSO confirms LV flight path is clear before launch	2A	Inspection: NAR Safety Code #9
E.M.4	Unauthorized aircraft in designated airspace	Aircraft knowingly ignores restricted airspace designations	Any and all launches suspended until further notice	4A	RSO has contact with local air traffic controllers	1A	RSO is contacted directly by air traffic control

Table 5.5: Hazards to Environmental Safety

Label	Hazard	Cause	Effect	LS Before	Mitigation	LS After	Verification
Hazards to Wildlife							
E.W.1	Launch field catches fire	Motor ignition	Crop damage, harm to wildlife, personnel burns	3D	Areas surrounding launch pad are clear of flammable materials, blast plates are properly fitted to launch rails	3B	Inspection: NAR Safety Code #7
E.W.2		Black powder ignition		2C	Personnel are equipped with a functional fire extinguisher	1C	
E.W.3		Battery explosion		2D	Batteries are isolated from moisture, abrasion, and heat	2B	
E.W.4	Payload battery explosion	Battery is punctured, leading to contact with moisture Excessive heat surrounding battery	Hazmat leakage onto launch field, water contamination, fire on launch field	3D		3B	Inspection: NASA 2.22
E.W.5	LV comes into contact with flying birds	Birds fly in close proximity to LV	Wildlife injury or death, bird migration patterns are obstructed	1C	RSO confirms airways are clear ahead of launch	1A	Inspection: Checklist Section Launch Pad
E.W.6	Nomex permanently jettisons	Rips or tears in Nomex	Contamination of wildlife habitats, food supply, or water supply	2A	Nomex is rated to withstand flight forces, sheets are inspected before launch to ensure no rips or tears are present	1A	Inspection: Checklist Section Main/Drogue Recovery
E.W.7		Breakage in Nomex connection		1A	Nomex sheets are properly connected to shock cord with quick links, Safety Officer verifies proper connections	1A	
E.W.8	Parachute permanently jettisons	Quick link is not properly tightened and secured before parachute is inserted into LV	LV descends at an unsafe speed	1A	Parachutes are properly connected to shock cord with quick links, Safety Officer verifies proper connections	1A	
E.W.9	Hazmat deposit in irrigation ditch	Battery explosion Explosion byproducts	Toxins remain in food crops and could be consumed by humans/wildlife	2B	Any additional protective insulation is biodegradable	2A	Verification Pending
E.W.10	Wildlife consumes hazmats or other toxins	Littering of hazmats	Wildlife develop digestive issues or incur injury or death	3D		3C	

Table 5.5 continued from previous page

Label	Hazard	Cause	Effect	LS Before	Mitigation	LS After	Verification
E.W.11	CATO	Motor defects	Water supply is contaminated, wildlife incur injury or death	2D	AeroTech motors are selected for their low likelihood of catastrophic failure and personnel experience with the brand	2C	Verification Pending
E.W.12	LV lands in tree	Premature parachute deployment, wind drift	Destruction of wildlife habitats	4C	Recovery systems are tested and LV is flown away from trees	4B	Verification Pending
E.W.13	Emission of microplastics	Exceptionally high usage of single-use plastics	Wildlife infertility, bodily inflammation, choking/strangling/digestive hazard	4B	Personnel are encouraged to use reusable containers	4B	Inspection: Team Safety Briefing
Hazards to Land							
E.L.1	LV impacts with ground	Late or no deployment of parachute	Permanent ruts left in launch field, inability for soil to be used in future farming endeavors	3A	Recovery system utilizes altimeters to ensure accuracy in parachute deployment	2A	Verification Pending
E.L.2	Non-recoverable landing in tree	Premature parachute deployment, wind drift	Permanent tree damage	4C	Launch pads are placed far from trees or other hazards	4A	Verification Pending
E.L.3	Launch field catches fire	CATO, motor ignition, black powder detonation, battery explosion	Trees and crops destroyed, inability for land to be used in future farming endeavors	2D	Areas surrounding launch pad is clear of flammable materials, blast plates are properly fitted to launch rails	2B	Inspection: NAR Safety Code #3
Hazards to Air/Water							
E.A.1	Emission of greenhouse gases	Transportation to/from launch field, byproducts from motor and black powder ignition, use of power-drawing electronics	Air pollution, further contribution to climate change	4A	Personnel are encouraged to carpool, take public transportation, or walk to any club activities	4A	Inspection: Safety Briefing Slides
E.A.2	Emission of microplastics	Exceptionally high usage of single-use plastics	Pollution of air and water	4A	Use of single use plastics is limited in LV design	4A	Verification Pending

Table 5.5 continued from previous page

Label	Hazard	Cause	Effect	LS Before	Mitigation	LS After	Verification
E.A.3	Chemical off-gassing	Working with hazmats	Air pollution	1B	Hazmats that off-gas are used in well-ventilated areas with proper PPE	1A	Verification Pending
E.A.4	Emission of smoke	CATO		2B	AeroTech motors are selected for their low likelihood of catastrophic failure and personnel experience with the brand	2A	Verification Pending
E.A.5		Motor ignition		2B	LV operation produces few combustion byproducts in nominal conditions	2A	Verification Pending
E.A.6		Black powder detonation		2B		1A	Inspection: NAR Safety Code #3
E.A.7		Man-made wildfire		2D	Heat sources are not allowed within 25 feet of LV motors	2B	Inspection: Aerotech Motors Safety Data Sheet

Table 5.6: Hazards to Personnel Safety

Label	Hazard	Cause	Effect	LS Before	Mitigation	LS After	Verification
Hazards to Skin and Soft Tissue							
PE.S.1	Slips, trips, falls	Material spills	Skin abrasion or bruising	3B	Lab floors are inspected for spills after handling assembly materials	1B	Inspection: HPRC Safety Handbook, Checklist Section Field Recovery
		Wet or uneven launch field conditions			Only authorized personnel recover LV, recovery personnel are required to wear treaded closed-toe shoes		
PE.S.2	Appendage caught in bandsaw	Improper operation of bandsaw	Skin or muscle tear/abrasion	2D	Personnel are trained how to properly handle manufacturing tools, appropriate PPE is always used	2C	Inspection: HPRC Safety Handbook
		Jewelry or clothing caught in bandsaw blade					
PE.S.3	Skin comes into contact with hot soldering iron	Personnel negligence	Mild to severe burns	3C		3B	Inspection: Checklist Section Launch Pad
PE.S.4	LV collides with personnel	Launch rail tips with assembled LV	Skin or muscle tear/abrasion	2C	Launch rails are provided by TRA/NAR, launch rails have a locking mechanism that is engaged when LV is righted	2B	Inspection: Checklist Section Night Before Checklist
PE.S.5		Severe instability causes sideways propulsion		2B	The stability of LV is no less than 2.0	1B	Inspection: NASA 2.14
PE.S.6		LV lands within close proximity to personnel		1B	The LV is angled away from any personnel or spectators	1A	Inspection: NAR Safety Code
PE.S.7	Personnel muscles placed under high load	Heavy LV components	Muscle strain or tear	4C	At least two personnel carry the assembled LV, proper lifting techniques are always used	4A	Inspection: Checklist Section Launch Pad
PE.S.8	Insect sting/bite	Prolonged exposure to wildlife during launch day activities	Itchiness, rash, and/or anaphylaxis	4A	Bug spray is provided to personnel, personnel have knowledge on appropriate use of EpiPens	3A	Inspection: Checklist Section Launch Pad
PE.S.9	Personnel come into contact with black powder charges	Contact with unblown charges during recovery	Mild to severe burns or abrasions	3C	Personnel recovering the LV are provided with heavy duty gloves, LV sections are inspected for unblown charges before handling	3B	Inspection: Checklist Section Final Measurements

Table 5.6 continued from previous page

Label	Hazard	Cause	Effect	LS Before	Mitigation	LS After	Verification
PE.S.10	Contact with large, airborne shrapnel	CATO	Severe skin abrasion or laceration	2D	Personnel are separated from the launch pad according to the minimum distance table, AeroTech motors are selected for their low likelihood of failure	2B	Inspection: NAR Safety Code
PE.S.11	Contact with small, airborne shrapnel	Sanding, cutting, drilling brittle or granular materials	Cuts or bruises	3C	Appropriate PPE is provided for personnel working with power tools	2C	Inspection: HPRC PPE Cabinet
PE.S.12	Exposure to uncured epoxy	Working with epoxy	Skin rash/irritation	3A	Appropriate PPE is provided for personnel working with hazardous materials	2A	
PE.S.13	Exposure to vaporous chemicals	Hazmat off-gassing		2A		2A	
PE.S.14	Excessive amount of walking	LV lands far from recovery personnel	Muscle strain, shin splints	3A	LV is equipped with a GPS, personnel wear shoes appropriate for walking large distances	2A	Inspection: Checklist Section Field Recovery
Hazards to Bones and Joints							
PE.B.1	Slips, trips, falls	Material spills, wet or uneven field conditions	Bone fracture/bruise, joint dislocation	1D	Lab floors are inspected for spills after handling assembly materials, only authorized personnel recover LV, recovery personnel are required to wear treaded closed-toe shoes	1C	Inspection: Checklist Section Field Recovery
PE.B.2	Excessive amount of walking	LV lands far from recovery personnel	Stress fracture	2D	LV is equipped with a GPS, personnel wear shoes appropriate for walking large distances	2C	Inspection: Checklist Section Field Recovery
PE.B.3	Appendage caught in bandsaw blade	Improper operation of bandsaw	Broken bone	2D	Personnel are trained how to properly handle manufacturing tools, appropriate PPE is always used	2C	Inspection: HPRC Safety Handbook
		Jewelry or clothing caught in bandsaw blade				2C	
PE.B.4	Contact with large, airborne shrapnel	CATO	Bone fracture/break/loss requiring immediate medical attention	2D	Personnel are separated from the launch pad according to the minimum distance table, AeroTech motors are selected for their low likelihood of failure	2C	Inspection: NAR Safety Code, RSO instruction
Hazards to Respiratory System							
PE.R.1	Inhalation of carcinogenic particles	Working with filet epoxy	Respiratory infection and/or irritation, cancer	4D	Personnel working with fillet epoxy are provided with appropriate PPE	3C	

Table 5.6 continued from previous page

Label	Hazard	Cause	Effect	LS Before	Mitigation	LS After	Verification
PE.R.2	Inhalation of epoxy fumes	Working with epoxy	Respiratory irritation, difficulty breathing	2C	Personnel working with epoxy are provided with appropriate PPE, an oxygen sensor is triggered if there is insufficient oxygen in the lab	2B	Inspection: HPRC PPE Cabinet
PE.R.3	Inhalation of aerosolized particles	Sanding, cutting, drilling brittle or granular materials		4B	Personnel are provided with appropriate PPE, including particle masks	4A	
PE.R.4	Inhalation of paint fumes	Use of spray paint for LV aesthetics		4B		4A	
PE.R.5	Inhalation of combustion reactants	Personnel are in close proximity to ejection charges		3B		3A	Inspection: NAR Safety Code, RSO instruction
Hazards to Head							
PE.H.1	High energy LV components come into contact with personnel	High energy LV components are in proximity to personnel during descent	Concussion, brain damage, memory loss, skull fracture	2D	Personnel are separated from the launch pad according to the minimum distance table, the LV recovery system is dual-redundant	2C	Inspection: NAR Safety Code
PE.H.2	LV tips during assembly	Launch rail is improperly assembled		3D	Launch rails are provided by TRA/NAR, launch rails have a locking mechanism that is engaged when LV is righted	3B	Inspection: Checklist Section Launch Pad
PE.H.3	Slips, trips, falls	Attempting to jump over irrigation ditches at launch field		3D	Personnel are made aware that jumping over ditches is forbidden	3D	Inspection: Checklist Section Field Recovery
PE.H.4	Contact with large, airborne shrapnel	CATO		2D	Personnel are separated from the launch pad according to the minimum distance table	2B	Inspection: NAR Safety Code
Hazards to Eyes							
PE.E.1	Exposure to epoxy fumes	Working with epoxy	Eye irritation, temporary blindness, permanent or semi-permanent blindness	3D	Personnel are provided with appropriate PPE	3B	Inspection: HPRC PPE Cabinet
PE.E.2	Exposure to aerosolized particles	Working with spray paint, sanding, cutting, drilling		2D		2B	
PE.E.3	Extended exposure to the sun	Maintained eye contact with descending LVs	Temporary or permanent blindness	1B	Personnel maintaining eyes with descending launch vehicles are encouraged to wear sunglasses or other forms of eye protection	1A	Inspection: Checklist Section Field Recovery

Table 5.6 continued from previous page

Label	Hazard	Cause	Effect	LS Before	Mitigation	LS After	Verification
Hazards from Payload							
PE.P.1	Personnel contact spinning rotor blades while payload is powered on	LV is released above personnel, recovery personnel approaches payload	Personnel injury to head, skin, bones, or soft tissue	4B	Payload is not released above or near crowds, payload is confirmed to be powered down before recovery personnel approach	2A	RSO permission to deploy payload required on launch day

6 Project Plan

6.1 Requirements Verification

6.1.1 Competition Requirements

Table 6.1: 2023-2024 NASA Requirements

NASA Req No.	Shall Statement	Success Criteria	Verification Method	Subsystem Allocation	Status	Status Description
General Requirements						
1.1	Students on the team SHALL do 100% of the project, including design, construction, written reports, presentations, and flight preparation, with the exception of assembling the motors and handling black powder or any variant of ejection charges or preparing and installing electric matches (to be done by the team's mentor). Teams SHALL submit new work. Excessive use of past work SHALL merit penalties.	Members of NC State's High-Powered Rocketry Club fabricate a solution to the criteria given in the Student Launch Handbook, implementing past ideas while developing new ones.	Inspection	Project Management	Not Verified	Students on the team use original work done by the team to complete the project.
1.2	The team SHALL provide and maintain a project plan to include, but not limited to, the following items: project milestones, budget and community support, checklists, personnel assignments, STEM engagement events, and risks and mitigations.	The Project Management Team, consisting of the Team Lead, Vice President, Integration Lead, Treasurer, Secretary, Safety Officer, Webmaster, and Social Media Lead manage the tasks related to this requirement.	Inspection	Project Management	Not Verified	See Section 6.4 for the project timeline.
1.3	Team members who will travel to the Huntsville Launch SHALL have fully completed registration in the NASA Gateway system before the roster deadline.	The Team Lead determines the team members attending Huntsville and ensures team members register and their application status is "submitted" in the NASA Gateway system no later than October 27th, 2023.	Inspection	Project Management	Not Verified	TBD
1.3.1	Team members attending competition SHALL include students actively engaged in the project throughout the entire year.	The Project Management Team determines the students that have been actively engaged to invite them to competition.	Inspection	Project Management	Not Verified	TBD
1.3.2	Team members SHALL include one mentor (see Requirement 1.13).	The Team Lead invites the mentor(s) identified in Section 1.1.2 to attend competition.	Inspection	Project Management	Not Verified	Team mentors are listed in Section 1.1.2.
1.3.3	Team members SHALL include no more than two adult educators.	The Team Lead invites the adult educator(s) shown in Section 1.1.2 to attend competition.	Inspection	Project Management	Not Verified	See Section 1.1.2 for team mentors and advisors.

1.4	Teams SHALL engage a minimum of 250 participants in Educational Direct Engagement STEM activities. These activities can be conducted in-person or virtually. To satisfy this requirement, all events SHALL occur between project acceptance and the FRR addendum due date. A template of the STEM Engagement Activity Report can be found on pages 86 – 89.	The Outreach Lead offers STEM engagement opportunities to K12 students for the duration of project development and submits STEM Engagement Activity Reports within two weeks of the event.	Inspection	Project Management	Not Verified	The Outreach Lead has begun to conduct and schedule STEM engagement activities.
1.5	The team SHALL establish and maintain a social media presence to inform the public about team activities.	The Webmaster and Social Media Officer collaborate to maintain our website and social media presence to educate the public about activities and events held by the team. Our social media platforms include, but are not limited to: our club website, TikTok, Facebook, and Instagram.	Inspection	Project Management	Verified	Any form of social media in relation to the team has been sent to the NASA project management team.
1.6	Teams SHALL upload all deliverables to the designated NASA SL Box submission portal by the deadline specified in the handbook for each milestone. No PDR, CDR, and FRR milestone documents SHALL be accepted after the due date and time. Teams that fail to submit the PDR, CDR, and FRR milestone documents SHALL be eliminated from the project.	The Team Lead uploads all documents to the designated NASA SL Box submission portal by the deadline specified.	Inspection	Project Management	Not Verified	Before the deadline the team lead upload all deliverables to the designated NASA SL Box submission portal.
1.7	Teams who do not satisfactorily complete each milestone review (PDR, CDR, FRR) SHALL be provided action items to be completed following their review and SHALL be required to address action items in a delta review session. After the delta session the NASA management panel SHALL meet to determine the team's status in the program and the team SHALL be notified shortly thereafter.	If a milestone review is not completed satisfactorily, the team completes any action items given and attends the delta review session to maintain their status in the program.	Inspection	Project Management	Not Verified	The team satisfactorily completes each milestone review and submits before the deadline.
1.8	All deliverables SHALL be in PDF format.	The Team Lead sends all deliverables in PDF format to the NASA Project Management Team.	Inspection	Project Management	Verified	All documents are changed to PDF format before submission.
1.9	In every report, teams SHALL provide a table of contents including major sections and their respective sub-sections.	The Team Lead creates and adjusts a table of contents in every report.	Inspection	Project Management	Verified	A table of contents is included in every report as seen in the Table of Contents above.

1.10	In every report, the team SHALL include the page number at the bottom of the page.	The team uses a template which displays the page number at the bottom of each page for every report.	Inspection	Project Management	Verified	The page number will be included at the bottom of the page in every report as seen in this document.
1.11	The team SHALL provide any computer equipment necessary to perform a video teleconference with the review panel. This includes, but is not limited to, a computer system, video camera, speaker telephone, and a sufficient Internet connection. Cellular phones should be used for speakerphone capability only as a last resort.	The team obtains the equipment needed to attend a video teleconference with the review panel.	Inspection	Project Management	Not Verified	The team plans to obtain and test any equipment needed to perform a video teleconference with the review panel.
1.12	All teams attending Launch Week SHALL be required to use the launch pads provided by Student Launch's launch services provider. No custom pads SHALL be permitted at the NASA Launch Complex. At launch, 8 ft. 1010 rails and 12 ft. 1515 rails SHALL be provided. The launch rails SHALL be canted 5 to 10 degrees away from the crowd on Launch Day. The exact cant SHALL depend on Launch Day wind conditions.	The Aerodynamics Lead designs a launch vehicle that utilizes 8 ft. 1010 rails or 12 ft. 1515 rails. The Structures Lead builds the launch vehicle according to these specifications.	Inspection	Aerodynamics, Structures	Not Verified	The team plans to use the launch pads provided for Launch Day.

1.13	Each team SHALL identify a "mentor." A mentor is defined as an adult who is included as a team member who SHALL be supporting the team (or multiple teams) throughout the project year, and may or may not be affiliated with the school, institution, or organization. The mentor SHALL maintain a current certification and be in good standing, through the National Association of Rocketry (NAR) or Tripoli Rocketry Association (TRA) for the motor impulse of the launch vehicle and must have flown and successfully recovered (using electronic, staged recovery) a minimum of 2 flights in this or a higher impulse class, prior to PDR. The mentor is designated as the individual owner of the launch vehicle for liability purposes and must travel with the team to Launch Week. One travel stipend SHALL be provided per mentor regardless of the number of teams he or she supports. The stipend SHALL only be provided if the team passes FRR and the team and mentor attend Launch Week in April.	The Team Leader determines a qualified adult to mentor the team throughout project development and attend Launch Week.	Inspection	Project Management	Verified	See Section 1.1.2 for team mentors.
1.14	Teams SHALL track and report the number of hours spent working on each milestone.	The team records the number of hours spent working on each milestone and documents this in the designated report.	Inspection	Project Management	Verified	See Section 1.1 pertaining to time spent on PDR.
Vehicle Requirements						
2.1	The vehicle SHALL deliver the payload to an apogee altitude between 4,000 and 6,000 ft. above ground level (AGL). Teams flying below 3,500 ft. or above 6,500 ft. on their competition launch will receive zero altitude points towards their overall project score and will not be eligible for the Altitude Award.	The Aerodynamics and Structures Leads design a launch vehicle to deliver the payload to an apogee between 4,000 and 6,000 ft. AGL. The team fabricates the launch vehicle as designed.	Analysis, Demonstration	Aerodynamics, Structures	Not Verified	See Section 3.7.1 for launch day target apogee.
2.2	Teams SHALL declare their target altitude goal at the PDR milestone. The declared target altitude SHALL be used to determine the team's altitude score.	The Aerodynamics Lead reports the target altitude goal by October 26, 2023 in the PDR milestone.	Inspection	Aerodynamics	Verified	See Section 1.2.1for official target apogee.

2.3	The launch vehicle SHALL be designed to be recoverable and reusable. Reusable is defined as being able to launch again on the same day without repairs or modifications.	The Recovery and Structures Lead design a recovery system that prevents the launch vehicle from being damaged upon ground impact.	Demonstration	Recovery, Structures	Not Verified	See Section 3.3 for leading launch vehicle design.
2.4	The launch vehicle SHALL have a maximum of four (4) independent sections. An independent section is defined as a section that is either tethered to the main vehicle or is recovered separately from the main vehicle using its own parachute.	The Aerodynamics and Recovery Leads design the launch vehicle to have no more than four independent sections.	Inspection	Aerodynamics, Recovery	Verified	See Section 3.3.1.
2.4.1	Coupler/airframe shoulders which are located at in-flight separation points SHALL be at least 2 airframe diameters in length (one body diameter of surface contact with each airframe section).	The Aerodynamics Lead designs the coupler/airframe shoulders at in-flight separation points at least 2 airframe diameters in length. The Structures Lead builds the couplers to the specified lengths.	Inspection	Aerodynamics, Structures	Not Verified	See Section 3.3 for the current launch vehicle design.
2.4.2	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.)	The Aerodynamics Lead designs the coupler/airframe shoulders at non-in-flight separation points at least 1.5 airframe diameters in length. The Structures Lead builds the couplers to the specified lengths.	Inspection	Aerodynamics, Structures	Not Verified	See Section 3.3 for the current launch vehicle design.
2.4.3	Nose cone shoulders which are located at in-flight separation points SHALL be at least 0.5 body diameters in length.	The Aerodynamics Lead designs the nose cone shoulders at in-flight separation points to be a minimum of 0.5 body diameter in length.	Inspection	Aerodynamics	Verified	See Section 3.3 for the current launch vehicle design.
2.5	The launch vehicle SHALL be capable of being prepared for flight at the launch site within 2 hours of the time the Federal Aviation Administration flight waiver opens.	The Project Management Team and Safety Officer creates a launch day checklist that can be completed within two hours.	Demonstration	Project Management, Safety	Not Verified	TBD
2.6	The launch vehicle and payload SHALL be capable of remaining in launch-ready configuration on the pad for a minimum of 3 hours without losing the functionality of any critical on-board components, although the capability to withstand longer delays is highly encouraged.	The Project Management Team and Safety Officer ensure functionality of electrical components for a minimum of three hours by monitoring power consumption.	Demonstration	Project Management, Safety	Not Verified	TBD
2.7	The launch vehicle SHALL be capable of being launched by a standard 12-volt direct current firing system. The firing system SHALL be provided by the NASA-designated launch services provider.	The Project Management Team and Safety Officer pick a motor from a designated launch services provider that can be ignited by a 12-volt direct current firing system.	Demonstration	Project Management, Safety	Not Verified	TBD

2.8	The launch vehicle SHALL require no external circuitry or special ground support equipment to initiate launch (other than what is provided by the launch services provider).	The Project Management Team and Safety Officer design the launch vehicle such that no external circuitry or special ground support equipment is needed for launch.	Demonstration	Safety	Not Verified	No use of external circuitry will be used as shown in Section 3.3.
2.9	Each team SHALL use commercially available e-matches or igniters. Hand-dipped igniters SHALL not be permitted.	The Project Management Team and Safety Officer utilize commercially available e-matches and igniters.	Inspection	Project Management, Safety	Not Verified	TBD
2.10	The launch vehicle SHALL use a commercially available solid motor propulsion system using ammonium perchlorate composite propellant (APCP) which is approved and certified by the National Association of Rocketry (NAR), Tripoli Rocketry Association (TRA), and/or the Canadian Association of Rocketry (CAR).	The Aerodynamics Lead selects a commercially purchased solid motor propulsion system with APCP certified by NAR, TRA, and/or CAR.	Inspection	Aerodynamics	Not Verified	The current motor choice can be seen in Section 1.2.2.
2.10.1	Final motor choice SHALL be declared by the Critical Design Review (CDR) milestone.	The Aerodynamics Lead states the finalized motor choice in the CDR milestone by January 8, 2024.	Inspection	Aerodynamics	Not Verified	TBD
2.10.2	Any motor change after CDR SHALL be approved by the NASA Range Safety Officer (RSO). Changes for the sole purpose of altitude adjustment SHALL not be approved. A penalty against the team's overall score SHALL be incurred when a motor change is made after the CDR milestone, regardless of the reason.	The Project Management Team requests approval from NASA RSO for a motor changed after the CDR milestone deadline.	Inspection	Project Management	Not Verified	TBD
2.11	The launch vehicle SHALL be limited to a single motor propulsion system.	The Aerodynamics Lead designs the launch vehicle to use a single motor propulsion system.	Inspection	Aerodynamics	Not Verified	See Section 3.3 for the current launch vehicle design.
2.12	The total impulse provided by a College or University launch vehicle SHALL not exceed 5,120 Ns (L-class).	The Aerodynamics Lead picks a motor that does not exceed a total impulse of 5,120 Ns.	Inspection	Aerodynamics	Not Verified	The current motor choice can be seen in Section 1.2.2.
2.13	Pressure vessels on the vehicle SHALL be approved by the RSO.	The Structures Lead gets RSO approval for any on-board pressure vessels.	Inspection	Structures	Not Verified	TBD
2.13.1	The minimum factor of safety (Burst or Ultimate pressure versus Max Expected Operating Pressure) SHALL be 4:1 with supporting design documentation included in all milestone reviews.	The Structures Lead provides design documentation in each milestone report supporting a minimum factor of safety of 4:1.	Analysis, Inspection	Structures	Not Verified	TBD

2.13.2	Each pressure vessel SHALL include a pressure relief valve that sees the full pressure of the tank and is capable of withstanding the maximum pressure and flow rate of the tank.	The Structures Lead picks pressure vessels which include a pressure relief valve system that sees the full pressure of the tank and can withstand the maximum pressure and flow rate of the tank.	Analysis, Inspection	Structures	Not Verified	TBD
2.13.3	The full pedigree of the tank SHALL be described, including the application for which the tank was designed and the history of the tank. This 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 Structures Lead describes the entire history of each pressure vessel, including the number of pressure cycles, the dates of pressurization/depressurization, and name of the person or entity administering each pressure event.	Inspection	Structures	Not Verified	TBD
2.14	The launch vehicle SHALL have a minimum static stability margin of 2.0 at the point of rail exit. Rail exit is defined at the point where the forward rail button loses contact with the rail.	The Aerodynamics Lead designs the launch vehicle to have a minimum static stability margin of 2.0 at the rail exit.	Analysis	Aerodynamics	Not Verified	See Section 3.7.3 for the projected stability margin.
2.15	The launch vehicle SHALL have a minimum thrust to weight ratio of 5:1.	The Aerodynamics Lead designs the launch vehicle to have a minimum thrust to weight ratio of 5:1.	Analysis, Inspection	Aerodynamics	Not Verified	The current motor choice can be seen in Section 1.2.2.
2.16	Any structural protuberance on the launch vehicle SHALL be located aft of the burnout center of gravity. Camera will be exempted, provided the team can show that the housing(s) causes minimal aerodynamic effect on the launch vehicle's stability.	The Aerodynamics Lead designs the launch vehicle to have any protuberances located aft of the burnout center of gravity. If camera's are included, the Aerodynamics Lead will prove the housings cause minimal aerodynamic effect on the launch vehicle's stability.	Analysis, Inspection	Aerodynamics	Not Verified	TBD
2.17	The launch vehicle SHALL accelerate to a minimum velocity of 52 ft/s at rail exit.	The Aerodynamics Lead designs the launch vehicle to reach a minimum velocity of 52 ft/s at the rail exit.	Analysis	Aerodynamics	Not Verified	See Section 1.2.2 for the projected velocity of the launch vehicle.

2.18	All teams SHALL successfully launch and recover a sub-scale model of their launch vehicle prior to CDR. Success of the sub-scale is at the sole discretion of the NASA review panel. The sub-scale flight may be conducted at any time between proposal award and the CDR submission deadline. sub-scale flight data SHALL be reported in the CDR report and presentation at the CDR milestone. sub-scales are required to use a minimum motor impulse class of E (Mid Power motor).	The Project Management Team launches a sub-scale model of the launch vehicle before CDR using an impulse motor of class E or higher. The Project Management Team and Safety Officer successfully recovers the sub-scale and reports flight data in the CDR milestone by January 8, 2024.	Demonstration	Project Management, Safety	Not Verified	See Section 6.4 for the project timeline and projected date of sub-scale flight.
2.18.1	The sub-scale model should resemble and perform as similarly as possible to the full scale model. However, the full scale SHALL not be used as the sub-scale model.	The Aerodynamics Lead designs a sub-scale model that performs similarly to the full scale model.	Inspection	Aerodynamics	Not Verified	TBD
2.18.2	The sub-scale model SHALL carry an altimeter capable of recording the model's apogee altitude.	The Recovery Lead attaches an altimeter to record the apogee altitude of the sub-scale model.	Inspection	Recovery	Not Verified	See Section 3.5 for potential altimeters.
2.18.3	The sub-scale launch vehicle SHALL be a newly constructed rocket, designed and built specifically for this year's project.	The Aerodynamics and Structures Leads design and fabricate a new sub-scale launch vehicle that meets the criteria for this year's project.	Inspection	Aerodynamics, Structures	Not Verified	TBD
2.18.4	Proof of a successful sub-scale flight SHALL be supplied in the CDR report.	The Project Management Team shows proof of successful sub-scale flight in the CDR report by January 8, 2024.	Inspection	Project Management	Not Verified	TBD
2.18.4.1	Altimeter flight profile graph(s) OR a quality video showing successful launch, recovery events, and landing as deemed by the NASA management panel are acceptable methods of proof. Altimeter flight profile graph(s) that are not complete (liftoff through landing) SHALL not be accepted.	The Recovery Lead makes an altimeter flight profile graph which displays all altitudes recorded from liftoff through landing.	Analysis	Recovery	Not Verified	TBD
2.18.4.2	Quality pictures of the "as-landed" configuration of all sections of the launch vehicle SHALL be included in the CDR report. This includes but is not limited to nose cone, recovery system, airframe, and booster.	The Project Management Team and Recovery Lead takes pictures of the landing configuration of all sections of the launch vehicle and includes them in the CDR milestone by January 8, 2024.	Analysis, Demonstration	Project Management, Recovery	Not Verified	TBD

2.18.5	The sub-scale launch vehicle SHALL not exceed 75% of the dimensions (length and diameter) of the designed full scale launch vehicle (if the full scale launch vehicle is a 4 in. diameter, 100 in. length rocket, your sub-scale SHALL not exceed 3 in. diameter and 75 in. in length).	The Aerodynamics and Structures Lead design the sub-scale launch vehicle to not exceed 75% of the dimensions used for the full scale launch vehicle.	Inspection	Aerodynamics, Structures	Not Verified	TBD
2.19.1	**Vehicle Demonstration Flight**. All teams SHALL successfully launch and recover their full scale launch vehicle prior to FRR in its final flight configuration. The launch vehicle flown SHALL be the same launch vehicle flown at competition launch. Requirements 2.19.1.1-9 SHALL be met during the Vehicle Demonstration Flight:	The Project Management Team launches and recovers the full scale vehicle, to be flown for competition, in its final flight configuration before the FRR milestone.	Demonstration	Project Management	Not Verified	See Section 6.4 for the project timeline and projected date of VDF.
2.19.1.1	The vehicle and recovery system SHALL function as designed.	The Project Management Team identifies no abnormalities in the performance of the vehicle and recovery system.	Demonstration	Project Management	Not Verified	TBD
2.19.1.2	The full scale launch vehicle SHALL be a newly constructed rocket, designed and built specifically for this year's project.	The Aerodynamics and Structures Leads design and build a new full scale launch vehicle, meeting the criteria for this year's project.	Inspection	Aerodynamics, Structures	Not Verified	TBD
2.19.1.3.1	If the payload is not flown during the Vehicle Demonstration Flight, mass simulators SHALL be used to simulate the payload mass.	The Structures Lead installs mass simulators to mimic payload mass if the payload is not flown during VDF.	Inspection	Structures	Not Verified	TBD
2.19.1.3.2	The mass simulators SHALL be located in the same approximate location on the launch vehicle as the missing payload mass.	The Structures Lead installs mass simulators at the approximate location on the launch vehicle as the missing payload if the payload is not flown during VDF.	Inspection	Structures	Not Verified	TBD
2.19.1.4	If the payload changes the external surfaces of the launch vehicle (such as camera housings or external probes) or manages the total energy of the vehicle, those systems SHALL be active during the full scale Vehicle Demonstration Flight.	The Payload Team activates systems during VDF if the payload changes the external surface or manages the total energy of the vehicle.	Inspection	Payload	Not Verified	TBD
2.19.1.5	Teams SHALL fly the competition launch motor for the Vehicle Demonstration Flight. The team may request a waiver for the use of an alternative motor in advance if the home launch field cannot support the full impulse of the competition launch motor or in other extenuating circumstances.	The Aerodynamics Lead selects the same motor for both competition launch and the VDF. If the selected motor cannot be flown for VDF due to extenuating circumstances, the Project Management Team requests a waiver for an alternative motor in advance.	Inspection	Aerodynamics, Project Management	Not Verified	TBD

2.19.1.6	The launch vehicle SHALL be flown in its fully ballastsed configuration during the full scale test flight. Fully ballastsed refers to the maximum amount of ballasts that SHALL be flown during the competition launch flight. Additional ballasts SHALL not be added without a re-flight of the full scale launch vehicle.	The Aerodynamics Lead determines the fully ballastsed configuration. The Structures Lead installs the needed ballasts for the full scale test.	Inspection	Aerodynamics, Structures	Not Verified	TBD
2.19.1.7	After successfully completing the full scale Vehicle Demonstration Flight, the launch vehicle or any of its components SHALL not be modified without the concurrence of the NASA Range Safety Officer (RSO).	The Project Management Team does not allow any further modifications of the launch vehicle or its components after VDF without NASA and RSO approval.	Inspection	Project Management	Not Verified	TBD
2.19.1.8	Proof of a successful Vehicle Demonstration Flight SHALL be supplied in the FRR report.	The Project Management Team provides proof of successful VDF in the FRR report.	Inspection	Project Management	Not Verified	TBD
2.19.1.8.1	Altimeter flight profile graph(s) that are not complete (liftoff through landing) SHALL not be accepted.	The Recovery Lead provides complete altimeter data acquired from the VDF in the FRR milestone.	Inspection	Recovery	Not Verified	TBD
2.19.1.8.2	Quality pictures of the ”as-landed” configuration of all sections of the launch vehicle SHALL be included in the FRR report. This includes but is not limited to nose cone, recovery system, airframe, and booster.	The Project Management Team and Recovery Lead takes pictures of the landing configuration of all sections of the launch vehicle and includes them in the FRR milestone.	Inspection	Project Management, Recovery	Not Verified	TBD
2.19.1.9	The Vehicle Demonstration Flight SHALL be completed by the FRR submission deadline. No exceptions SHALL be made. If the Student Launch office determines that a Vehicle Demonstration Re-flight is necessary, then an extension may be granted. Teams completing a required re-flight SHALL submit an FRR Addendum by the FRR Addendum deadline.	The Project Management Team completes the VDF by the FRR submission deadline. If re-flight is necessary, the team submits an FRR Addendum by the FRR Addendum deadline.	Inspection	Project Management	Not Verified	TBD
2.19.2	**Payload Demonstration Flight**. All teams SHALL successfully launch and recover their full scale launch vehicle containing the completed payload prior to the Payload Demonstration Flight deadline. The launch vehicle flown SHALL be the same launch vehicle to flown at competition launch. Requirements 2.19.2.1-4 SHALL be met during the Payload Demonstration Flight.	The Project Management Team launches and recovers the full scale launch vehicle containing the completed payload before the PDF deadline.	Inspection	Project Management	Not Verified	See Section 6.4 for the projected timeline and projected date of PDF.

2.19.2.1	The payload SHALL be fully retained until the intended point of deployment (if applicable). All retention mechanisms SHALL function as designed, and the retention mechanism SHALL not sustain damage requiring repair.	The Integration and Payload Leads ensure the payload is fully retained until the intended point of deployment, with each retention mechanism functioning as designed and not sustaining damage during flight.	Inspection	Integration, Payload	Not Verified	TBD
2.19.2.2	The payload flown SHALL be the final, active version of the payload.	The Project Management and Payload Teams ensures the payload flown during the PDF is the final active version of the payload.	Inspection	Project Management, Payload	Not Verified	TBD
2.19.2.3	If Requirements 2.19.2.1-2 are met during the original Vehicle Demonstration Flight, occurring prior to the FRR deadline and the information is included in the FRR package, the additional flight and FRR Addendum SHALL not be required.	The Project Management Team verifies all requirements are met for the VDF and are submitted prior to the FRR deadline. If all requirements are not met, the team performs an additional flight for PDF and submits the FRR Addendum.	Inspection	Project Management	Not Verified	TBD
2.19.2.4	Payload Demonstration Flights SHALL be completed by the FRR Addendum deadline.	The Project Management Team ensures the PDF is completed by the FRR Addendum deadline.	Inspection	Project Management	Not Verified	See Section 6.4 for the projected timeline and projected date of PDF.
2.20	An FRR Addendum SHALL be required for any team completing a Payload Demonstration Flight or NASA required Vehicle Demonstration Re-flight after the submission of the FRR.	The Project Management Team submits an FRR Addendum if the team completes the PDF or NASA required re-flight after the submission of the FRR.	Inspection	Project Management	Not Verified	TBD
2.20.1	Teams required to complete a Vehicle Demonstration Re-Flight and failing to submit the FRR Addendum by the deadline SHALL not be permitted to fly a final competition launch.	The Project Management Team ensures PDF and re-flight completion before the FRR Addendum deadline.	Inspection	Project Management	Not Verified	TBD
2.20.2	Teams who complete a Payload Demonstration Flight which is not successful may petition the NASA RSO for permission to fly the payload at the final competition launch. Permission SHALL not be granted if the RSO or the Review Panel have any safety concerns.	The Project Management Team petitions the NASA RSO for permission to fly the payload at the final competition launch if the PDF is not successful.	Demonstration	Project Management	Not Verified	TBD

2.21	The team's name and launch day contact information SHALL be in or on the launch vehicle airframe as well as in or on any section of the vehicle that separates during flight and is not tethered to the main airframe. This information SHALL be included in a manner that allows the information to be retrieved without the need to open or separate the vehicle.	The Project Management Team includes the team name and launch day contact information on the launch vehicle airframe, and any sections that separate during flight, such that it can be retrieved without the need to open or separate the vehicle.	Inspection	Project Management	Not Verified	TBD
2.22	All Lithium Polymer batteries SHALL be sufficiently protected from impact with the ground and SHALL be brightly colored, clearly marked as a fire hazard, and easily distinguishable from other payload hardware.	The Project Management Team and Safety Officer clearly mark all lithium polymer batteries as a fire hazard and sufficiently protects them from impact with the ground.	Analysis, Inspection	Project Management, Safety	Not Verified	TBD
2.23.1	The launch vehicle SHALL not utilize forward firing motors.	The Aerodynamics Lead selects a motor that is not forward firing.	Inspection	Aerodynamics	Not Verified	The current motor choice can be seen in Section 1.2.2.
2.23.2	The launch vehicle SHALL not utilize motors that expel titanium sponges (Sparky, Skidmark, MetalStorm, etc.)	The Aerodynamics Lead selects a motor that does not utilize motors that expel titanium sponges.	Inspection	Aerodynamics	Not Verified	The current motor choice can be seen in Section 1.2.2.
2.23.3	The launch vehicle SHALL not utilize hybrid motors.	The Aerodynamics Lead selects a motor that is not hybrid.	Inspection	Aerodynamics	Not Verified	The current motor choice can be seen in Section 1.2.2.
2.23.4	The launch vehicle SHALL not utilize a cluster of motors.	The Aerodynamics Lead designs the launch vehicle to be launched on a single motor.	Inspection	Aerodynamics	Not Verified	The team has no plans to use a cluster of motors as seen in Section 3.3 regarding launch vehicle design.
2.23.5	The launch vehicle SHALL not utilize friction fitting for motors.	The Structures Lead fabricates a motor retention system that does not use friction fitting to hold the motor.	Inspection	Structures	Not Verified	The current motor choice can be seen in Section 1.2.2.
2.23.6	The launch vehicle SHALL not exceed Mach 1 at any point during flight.	The Aerodynamics Lead designs the launch vehicle so that it does not reach Mach 1 at any point in flight.	Analysis	Aerodynamics	Not Verified	The current motor choice can be seen in Section 1.2.2.
2.23.7	Vehicle ballasts SHALL not exceed 10% of the total unballastsed weight of the launch vehicle as it would sit on the pad (i.e. a launch vehicle with an unballastsed weight of 40 lbs. on the pad may contain a maximum of 4 lbs. of ballasts).	The Aerodynamics Lead designs the launch vehicle such that vehicle ballasts does not exceed 10% of the total unballastsed weight of the launch vehicle.	Analysis, Inspection	Aerodynamics	Not Verified	See Section 3.7.4 pertaining to the addition of ballasts.
2.23.8	Transmissions from onboard transmitters, which are active at any point prior to landing, SHALL not exceed 250 mW of power (per transmitter).	The Recovery and Payload Leads choose onboard transmitters that do not exceed 250 mW of power (per transmitter).	Analysis	Recovery, Payload	Not Verified	TBD

2.23.9	Transmitters SHALL not create excessive interference. Teams SHALL utilize unique frequencies, handshake/passcode systems, or other means to mitigate interference caused to or received from other teams.	The Recovery and Payload Leads select transmitters that create minimal interference. The Safety Lead ensures the use of unique frequencies to mitigate interference with other teams.	Analysis, Demonstration	Recovery, Payload, Safety	Not Verified	TBD
2.23.10	Excessive and/or dense metal SHALL not be utilized in the construction of the launch vehicle. Use of lightweight metal SHALL be permitted but limited to the amount necessary to ensure structural integrity of the airframe under the expected operating stresses.	The Structures Lead fabricates the launch vehicle to have the minimal amount of metal used in the construction of the vehicle.	Inspection	Structures	Not Verified	TBD
Recovery Requirements						
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 Team ensures the launch vehicle is configured to fire a drogue parachute at apogee and a main parachute no later than 500 ft. AGL for both halves of the launch vehicle.	Demonstration	Recovery	Not Verified	See Section 3.4 pertaining to recovery design.
3.1.1	The main parachute SHALL be deployed no lower than 500 ft.	The Recovery Team ensures the main parachute deployment charge is programmed to fire prior to reaching 500 ft. for any and all independently descending launch vehicle segments.	Demonstration	Recovery	Not Verified	See Section 3.4 pertaining to recovery design.
3.1.2	The apogee event SHALL contain a delay of no more than 2 seconds.	The Recovery Team designs a recovery system that has an apogee event delay of no more than 2 seconds.	Demonstration	Recovery	Not Verified	See Section 3.4 pertaining to recovery design.
3.1.3	Motor ejection is not a permissible form of primary or secondary deployment.	The Recovery Team designs a recovery system that does not utilize motor ejection.	Inspection	Recovery	Not Verified	See Section 3.4 pertaining to recovery design.
3.2	Each team SHALL perform a successful ground ejection test for all electronically initiated recovery events prior to the initial flights of the sub-scale and full scale launch vehicles.	The Recovery Team performs ejection tests prior to each launch, confirming all recovery electronics are performing correctly.	Demonstration	Recovery	Not Verified	TBD

3.3	Each independent section of the launch vehicle SHALL have a maximum kinetic energy of 75 ft-lbf at landing. Teams whose heaviest section of their launch vehicle, as verified by Vehicle Demonstration Flight data, stays under 65 ft-lbf will be awarded bonus points.	The Recovery Team designs a recovery system such that the maximum kinetic energy experienced by the heaviest section of the launch vehicle does not exceed 65 ft-lbf.	Analysis	Recovery	Not Verified	See Section 3.7.5 for kinetic energy calculations.
3.4	The recovery system SHALL contain redundant, commercially available barometric altimeters that are specifically designed for initiation of launch vehicle recovery events. The term "altimeters" includes both simple altimeters and more sophisticated flight computers.	The Recovery Team designs a recovery system that uses primary and secondary altimeters for any and all AV bays.	Inspection	Recovery	Not Verified	See Section 3.6 for leading recovery design.
3.5	Each altimeter SHALL have a dedicated power supply, and all recovery electronics SHALL be powered by commercially available batteries.	The Recovery Team designs a recovery system that uses a separate, dedicated power supply, utilizing commercially available batteries, for any and all AV bays.	Inspection	Recovery	Not Verified	See Section 3.6 for leading recovery design.
3.6	Each altimeter SHALL be armed by a dedicated mechanical arming switch that is accessible from the exterior of the launch vehicle airframe when the launch vehicle is in the launch configuration on the launch pad.	The Recovery Team designs a recovery system that uses pin switches to activate any and all altimeters from the exterior of the launch vehicle.	Inspection	Recovery	Not Verified	See Section 3.6 for leading recovery design.
3.7	Each arming switch SHALL be capable of being locked in the ON position for launch (i.e. cannot be disarmed due to flight forces).	The Recovery Team designs a recovery system that uses arming switches that can be locked in the ON position for launch.	Inspection	Recovery	Not Verified	See Section 3.6 for leading recovery design.
3.8	The recovery system, including GPS and altimeters, electrical circuits SHALL be completely independent of any payload electrical circuits.	The Recovery Team designs a recovery system containing recovery electronics that are completely independent of the payload electronics.	Inspection	Recovery	Not Verified	See Section 3.6 for leading recovery design.
3.9	Removable shear pins SHALL be used for both the main parachute compartment and the drogue parachute compartment.	The Recovery Team designs a recovery system that uses removable shear pins such that separable sections of the launch vehicle are secured together on the pad and during launch.	Inspection	Recovery	Not Verified	See Section 3.6 for leading recovery design.
3.10	Bent eyebolts SHALL not be permitted in the recovery subsystem.	The Recovery Team designs a recovery system that does not use any bent eyebolts.	Inspection	Recovery	Not Verified	See Section 3.6 for leading recovery design.

3.11	The recovery area SHALL be limited to a 2,500 ft. radius from the launch pads.	The Recovery Team designs a recovery system containing parachutes that does not allow any separately descending segment of the launch vehicle to drift more than a 2,500 ft radius from the launch pad.	Analysis, Demonstration	Recovery	Not Verified	See Section 3.6 for leading recovery design.
3.12	Descent time of the launch vehicle SHALL be limited to 90 seconds (apogee to touch down). Teams whose launch vehicle descent, as verified by Vehicle Demonstration Flight data, stays under 80 seconds SHALL be awarded bonus points.	The Recovery Team designs a recovery system containing parachutes that allows any separately descending segments of the launch vehicle to safely land within 80 seconds of launch.	Analysis, Demonstration	Recovery	Not Verified	See Section 3.6 for leading recovery design.
3.13	An electronic GPS tracking device SHALL be installed in the launch vehicle and SHALL transmit the position of the tethered vehicle or any independent section to a ground receiver.	The Recovery Team designs a recovery system containing a GPS tracking device that transmits the position of each independent section of the launch vehicle.	Inspection, Demonstration	Recovery	Not Verified	See Section 3.6 for leading recovery design.
3.13.1	Any launch vehicle section or payload component, which lands untethered to the launch vehicle, SHALL contain an active electronic GPS tracking device.	The Recovery Team installs GPS tracking devices on any independent sections that land untethered to the launch vehicle.	Inspection	Recovery	Not Verified	See Section 3.6 for leading recovery design.
3.13.2	The electronic GPS tracking device(s) SHALL be fully functional during the official competition launch.	The Recovery Team tests GPS devices to ensure they remain completely functional during the official launch competition.	Inspection, Demonstration	Recovery	Not Verified	TBD
3.14	The recovery system electronics SHALL not be adversely affected by any other on-board electronic devices during flight (from launch until landing).	The Recovery Team designs a recovery system containing recovery electronics that are not affected by any other on-board electronic device.	Inspection, Demonstration	Recovery	Not Verified	See Section 3.6 for leading recovery design.
3.14.1	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 Team designs an AV bay containing altimeters in a compartment that is physically separate from any other radio frequency transmitting or magnetic wave-producing devices.	Inspection	Recovery	Not Verified	See Section 3.6 for leading recovery design.
3.14.2	The recovery system electronics SHALL be shielded from all onboard transmitting devices to avoid inadvertent excitation of the recovery system electronics.	The Recovery Team designs an AV bay containing recovery electronics that is shielded from all other onboard transmitting devices.	Inspection	Recovery	Not Verified	See Section 3.6 for leading recovery design.
3.14.3	The recovery system electronics SHALL be shielded from all onboard devices which may generate magnetic waves (such as generators, solenoid valves, and Tesla coils) to avoid inadvertent excitation of the recovery system.	The Recovery Team designs an avionics bay containing recovery electronics that is shielded from all other onboard magnetic wave generating devices.	Inspection	Recovery	Not Verified	See Section 3.6 for leading recovery design.

3.14.4	The recovery system electronics SHALL be shielded from any other onboard devices which may adversely affect the proper operation of the recovery system electronics.	The Recovery Team designs an AV bay containing recovery electronics that is shielded from all other onboard devices that may adversely affect the proper operation of the recovery system electronics.	Inspection	Recovery	Not Verified	See Section 3.6 for leading recovery design.
Payload Requirements						
4.1	**SL Payload Mission Objective** — College/University Division — Teams SHALL design a STEMnauts Atmosphere Independent Lander (SAIL). SAIL is an in-air deployable payload capable of safely retaining and recovering a group of 4 STEMnauts in a unique predetermined orientation without the use of a parachute or streamer. The landing SHALL occur under acceptable descent and landing parameters for the safe recovery of human beings. A STEMnaut SHALL be defined as a non-living crew member, to be physically represented as the team chooses, and is assumed to have human astronaut survivability metrics. The method(s)/design(s) utilized to complete the payload mission SHALL be at the team's discretion and will be permitted so long as the designs are deemed safe, obey FAA and legal requirements, and adhere to the intent of the challenge. NASA reserves the right to require modifications to a proposed payload.	The Payload and Integration Teams design a payload that is capable of safely returning the STEMnauts from the flight, follows all safety, FAA and NAR requirements, and is in accordance with the spirit of the competition.	Demonstration	Payload Electronics, Payload Structures, Payload Systems, Integration	Not Verified	See Section 4.4 for the leading payload design.
4.2.1	Teams SHALL not use parachutes or streamers that are commercially available or custom made. A parachute is defined as an open-faced canopy whose primary function is to reduce descent speed or increase drag. A streamer is defined as a long, narrow strip of material (typically affixed at one end) whose primary function is to reduce descent speed or increase drag.	The Payload Structures Team designs a SAIL that does not utilize any parachutes or streamers for recovery operations.	Inspection	Payload Structures	Not Verified	See Section 4.5 for the SAIL release system.
4.2.2	The SAIL SHALL be a minimum of 5 lbs inclusive of the jettisoned or separated landing capsule and the 4 STEMnauts.	The Payload Structures Team designs a SAIL that has a final weight of at least 5 lbs.	Inspection	Payload Structures	Not Verified	See Section 4.4 for the leading payload design.

4.2.3	Deployment of the SAIL SHALL occur between 400 and 800 ft. AGL. See Requirement 4.3.3 for deployment/jettison of payloads.	The Payload Structures, Recovery, and Integration Teams ensure SAIL ejection is designed to be within 400 and 800 ft. AGL.	Demonstration	Payload Structures, Recovery, Integration	Not Verified	See Section 4.2.5 for the SAIL deployment method.
4.2.4	The team SHALL pre-determine and land in a unique landing orientation to be verified by NASA personnel in Huntsville or by a non-affiliated NAR/TRA rep for at-home launches.	The Payload Teams design a SAIL that has a clear and defined landing orientation.	Demonstration	Payload Electronics, Payload Structures, Payload Systems	Not Verified	See Section 4.2.3 pertaining to landing orientation.
4.2.5	Teams SHALL design and implement a method of retention and ingress/egress for the STEMnauts.	The Payload Teams design a SAIL that retains the STEMnauts and allows easy access to the crew cabin for ingress/egress operations.	Inspection	Payload Electronics, Payload Structures, Payload Systems	Not Verified	See Section 4.4 for the leading payload design.
4.2.6	Teams SHALL determine acceptable descent and landing parameters, to be approved by NASA, and design their lander to meet those requirements.	The Payload Teams design a SAIL that has a final landing speed of 20 mph (according to criteria for NASA's Orion spacecraft) and limits angular velocity so that the STEMnauts experience a maximum of 3gs (according to NASA's Space Shuttle launch criteria).	Demonstration	Payload Electronics, Payload Structures, Payload Systems	Not Verified	See Section 4.4 for the leading payload design.
4.3.1	Black Powder and/or similar energetics are only permitted for deployment of in-flight recovery systems. Energetics will not be permitted for any surface operations.	The Payload, Recovery, and Integration Teams ensure that no energetics are used outside of in-flight recovery operations.	Inspection	Payload Structures, Recovery, Integration	Not Verified	See Section 4.4 for the leading payload design.
4.3.2	Teams SHALL abide by all FAA and NAR rules and regulations.	The Safety Team reviews the SAIL design throughout the design process to ensure compliance with all FAA and NAR rules and regulations.	Inspection	Safety	Not Verified	See Section 5 pertaining to all safety regulations.
4.3.3	Any payload experiment element that is jettisoned during the recovery phase SHALL receive real-time RSO permission prior to initiating the jettison event, unless exempted from the requirement by the RSO or NASA.	The Payload Systems and Safety Teams ensure that payload is not jettisoned without receiving RSO authorization.	Demonstration	Payload Systems, Safety	Not Verified	See Section 4.5 for the SAIL release system.
4.3.4	Unmanned aircraft system (UAS) payloads, if designed to be deployed during descent, SHALL be tethered to the vehicle with a remotely controlled release mechanism until the RSO has given permission to release the UAS.	The Payload Systems and Safety Teams ensures that any UAS that is deployed during the descent phase of flight is tethered to the vehicle and released on command after RSO permission is received.	Demonstration	Payload Systems, Safety	Not Verified	TBD

4.3.5	Teams flying UASs SHALL abide by all applicable FAA regulations, including the FAA's Special Rule for Model Aircraft (Public Law 112–95 Section 336; see https://www.faa.gov/uas/faqs).	The Payload and Safety Teams ensure that any UAS is flown in full compliance with FAA regulations.	Inspection	Payload Electronics, Payload Structures, Payload Systems, Safety	Not Verified	TBD
4.3.6	Any UAS weighing more than .55 lbs. SHALL be registered with the FAA and the registration number marked on the vehicle.	The Payload and Safety Teams ensure that any UAS weighing more than .55 lbs is registered with the FAA and the registration number is clearly marked on the vehicle.	Inspection	Payload Structures, Payload Systems, Payload Electronics, Safety	Not Verified	TBD
Safety Requirements						
5.1	Each team SHALL use a launch and safety checklist. The final checklists SHALL be included in the FRR report and used during the Launch Readiness Review (LRR) and any Launch Day operations.	Checklists are included in the FRR and are used during LRR and Launch Day activities.	Validation of Records	All	Not Verified	TBD
5.2	Each team SHALL identify a student Safety Officer who will be responsible for all requirements in Section 5.3.	The student Safety Officer, Megan Rink, is responsible for requirements listed in Section 5.3.	Validation of Records	Safety	Verified	The team has identified the student Safety Officer for the 2023-2024 year.
5.3.1.1	The designated Safety Officer SHALL monitor team activities with an emphasis on safety during design of vehicle and payload.	The student Safety Officer monitors team activities and ensures team members are practicing proper safety techniques.	Demonstration	Safety	Not Verified	TBD
5.3.1.2	The designated Safety Officer SHALL monitor team activities with an emphasis on safety during construction of vehicle and payload components.	The student Safety Officer monitors team activities and ensures team members are practicing proper safety techniques.	Demonstration	Safety	Not Verified	TBD
5.3.1.3	The designated Safety Officer SHALL monitor team activities with an emphasis on safety during assembly of vehicle and payload.	The student Safety Officer monitors team activities and ensures team members are practicing proper safety techniques.	Demonstration	Safety	Not Verified	TBD
5.3.1.4	The designated student Safety Officer SHALL monitor team activities with an emphasis on safety during ground testing of vehicle and payload.	The student Safety Officer monitors team activities and ensures team members are practicing proper safety techniques.	Demonstration	Safety	Not Verified	TBD
5.3.1.5	The designated student Safety Officer SHALL monitor team activities with an emphasis on safety during sub-scale launch test(s).	The student Safety Officer monitors team activities and ensures team members are practicing proper safety techniques.	Demonstration	Safety	Not Verified	TBD

5.3.1.6	The designated student Safety Officer SHALL monitor team activities with an emphasis on safety during full scale launch test(s).	The student Safety Officer monitors team activities and ensures team members are practicing proper safety techniques.	Demonstration	Safety	Not Verified	TBD
5.3.1.7	The designated student Safety Officer SHALL monitor team activities with an emphasis on safety during competition launch.	The student Safety Officer monitors team activities and ensures team members are practicing proper safety techniques.	Demonstration	Safety	Not Verified	TBD
5.3.1.8	The designated student Safety Officer SHALL monitor team activities with an emphasis on safety during recovery activities.	The student Safety Officer monitors team activities and ensures team members are practicing proper safety techniques.	Demonstration	Safety	Not Verified	TBD
5.3.1.9	The designated student Safety Officer SHALL monitor team activities with an emphasis on safety during STEM engagement activities.	The student Safety Officer monitors team activities and ensures team members are practicing proper safety techniques.	Demonstration	Safety	Not Verified	The Safety Officer monitored previous STEM engagement activities and will continue to do so for future events.
5.3.2	The designated student Safety Officer SHALL implement procedures developed by the team for construction, assembly, launch, and recovery activities.	The Safety Team writes and implements procedures and checklists for assembling, launching, and recovering the launch vehicle.	Demonstration	Safety	Not Verified	TBD
5.3.3	The designated student Safety Officer SHALL manage and maintain current revisions of the team's hazard analyses, failure modes analyses, procedures, and MSDS/chemical inventory data.	The student Safety Officer manages all safety documentation for the team.	Inspection	Safety	Verified	TBD
5.4	During test flights, teams SHALL abide by the rules and guidance of the local rocketry club's RSO. The allowance of certain vehicle configurations and/or payloads at the NASA Student Launch does not give explicit or implicit authority for teams to fly those vehicle configurations and/or payloads at other club launches. Teams SHALL communicate their intentions to the local club's President or Prefect and RSO before attending any NAR or TRA launch.	The Safety Team ensures all local rocketry club rules and regulations are followed by all team members.	Demonstration	Safety	Not Verified	TBD
5.5	The team SHALL abide by all rules set forth by the FAA.	The Safety Team ensures all rules from the FAA are followed.	Demonstration	Safety, Project Management	Verified	The Safety Team ensures team members follow FAA reglations at all times.
Final Flight Requirements						

6.1	Teams SHALL conduct the final flight in Huntsville during Launch Week, NASA Launch Complex, by the applicable deadlines as outlined in the Timeline for NASA Student Launch.	The team completes final flight at the NASA Launch Complex by the deadline given in the timeline for NASA Student Launch.	Demonstration	Project Management	Not Verified	See Section 6.4 for the project timeline.
6.1.1	Teams SHALL not show up at the NASA Launch Complex outside of launch day without permission from the NASA management team.	The team requests permission from the NASA management team if needing to show up at the NASA Launch Complex outside of launch day.	Demonstration	Project Management	Not Verified	TBD
6.1.2	Teams SHALL complete and pass the Launch Readiness Review conducted during Launch Week.	The team completes and passes the Launch Readiness Review.	Inspection; Demonstration	Project Management	Not Verified	TBD
6.1.3	The team mentor SHALL be present and oversee launch vehicle preparation and launch activities.	The team mentor oversees all launch activities.	Demonstration	Team Mentor	Not Verified	TBD
6.1.4	The scoring altimeter SHALL be presented to the NASA scoring official upon recovery.	The recovery lead presents the scoring altimeter to the NASA scoring official.	Demonstration	Recovery	Not Verified	TBD
6.1.5	Teams SHALL launch only once. Any launch attempt resulting in the launch vehicle exiting the launch pad, regardless of the success of the flight, SHALL be considered a launch. Additional flights beyond the initial launch, SHALL not be scored and SHALL not be considered for awards.	The team launches the launch vehicle only once.	Inspection; Demonstration	Project Management	Not Verified	TBD

6.1.2 Team Derived Requirements

Table 6.2: Launch Vehicle Team Derived Requirements

ID	Description	Justification	Success Criteria	Verification Method	Status	Status Description
Functional Requirements						
LVF 1	The launch vehicle SHALL be designed with fins that do not contain curved geometry.	Complex fin geometry reduces the manufacturability of the fins increasing the amount of labor and cost of producing flight and critical spares of the fins.	The fins contain a linear external profile.	Inspection	Verified	See Section 3.3.9 pertaining to fin design.
LVF 2	The launch vehicle SHALL be designed with removable ballasts.	Design changes made to the vehicle or payload after the PDR milestone may dictate a modification to the total ballasts of the vehicle.	ballasts is not permanently mounted to the vehicle allowing for removal or addition with hand tools.	Inspection	Not Verified	See Section 3.7.4 pertaining to ballasts.
LVF 3	The launch vehicle apogee verification SHALL be conducted by no less than 3 separate analysis programs.	Multimodal analysis of the apogee of the vehicle will increase the confidence in the apogee declared in the competition.	At least three different analysis programs are used in the development of a target apogee for the launch vehicle.	Inspection	Not Verified	See Section 3.7.1 regarding apogee calculations.
Design Requirements						
LVD 1	The launch vehicle SHALL utilize four fins.	Maximization of the aerodynamic surface area of the fins will increase the fin control authority of the launch vehicle's trajectory, reducing the risk of launch vehicle instability during flight.	The launch vehicle has four fins mounted to the airframe.	Inspection	Not verified	See Section 3.3.9 for current fin design.
LVD 2	The launch vehicle SHALL have symmetrical fins.	Ensures the CG is centered and aerodynamic forces are balanced.	The launch vehicle has fours fins around the airframe that are equally spaced.	Inspection	Not Verified	See Section 3.3.9 for current fin design.
LVD 3	The launch vehicle SHALL use at least two centering rings to support the motor tube.	Provides adequate support to the motor tube when the motor is experiencing forces during launch.	The launch vehicle has three centering rings supporting the motor tube.	Inspection	Not Verified	See Section 3.3 pertaining to launch vehicle design.
LVD 4	The launch vehicle SHALL have a stability margin between 2 and 2.7 upon rail exit.	A stability margin of 2.0 or greater is needed per NASA requirement 2.14. With a maximum stability of 2.7, undesired weather cocking can be avoided when launching in high winds.	The estimated launch vehicle stability will be between 2.0 and 2.7.	Analysis	Not Verified	See Section 3.7.3 for expected stability margin of the leading launch vehicle design.
LVD 5	The launch vehicle stability margin SHALL maintain a variability of no more than 0.3 calibers between the sub-scale and full scale vehicle.	Modification of the launch vehicle's center of pressure and center of gravity degrades the validity of sub-scale testing.	The stability of the sub-scale and full scale launch vehicle evaluate to a difference of no more than 0.1 calipers.	Analysis	Not Verified	See Section 3.7.3 for expected stability margin of the leading launch vehicle design.

LVD 6	The launch vehicle SHALL not exceed a maximum velocity of Mach 0.7.	High launch vehicle atmospheric loading increases the risk of structural component and payload hardware damage.	Simulations dictate a maximum launch vehicle velocity of no more than Mach 0.7.	Analysis	Verified	See Section 1.2.2 for the selected motor.
LVD 7	The Launch vehicle SHALL not exceed a maximum instantaneous acceleration of 14 G's during flight.	High acceleration of the vehicle during flight increases the risk of structural safety margin degradation along with the risk of payload hardware damage.	Simulations dictate a maximum launch vehicle acceleration of no more than 14 G's of acceleration during the entire flight profile.	Analysis	Verified	See Section 3.7 for performance predictions.
LVD 8	The launch vehicle SHALL use no more than 4 pounds of ballasts in the nose cone.	Volume constraints within the nose cone of the launch vehicle dictate a finite amount of ballasts that can reasonably be placed within the section.	The ballasts measurement of the full scale vehicle is at or below 4 pounds.	Inspection	Not Verified	See Section 3.7.4 pertaining to ballasts.
LVD 9	The launch vehicle SHALL be developed with a methodology for altering the final mass of the vehicle by at least 0.25 lb on the day of launch.	Variability of wind speeds on the day of launch may dictate the addition of additional ballasts to compensate for cosine losses of total apogee.	A system for the alteration of the final vehicle mass by at least 0.25 lb is incorporated into the vehicle.	Inspection	Not Verified	See Section 3.7.4 pertaining to ballasts.
LVD 10	The launch vehicle SHALL be designed to minimize cyclical angle of attack oscillations during liftoff.	Minimization of cyclical oscillations of the launch vehicle during flight will improve the probability of the vehicle achieving the target apogee.	Specific targeted analysis is presented in the CDR regarding design decisions made to minimize cyclical angle of attack oscillations	Inspection	Not Verified	See Section 3.3 pertaining to launch vehicle design.

Table 6.3: Recovery Team Derived Requirements

ID	Description	Justification	Success Criteria	Verification Method	Status	Status Description
Functional Requirements						
RF 1	Fully charged 9V batteries SHALL be used for the altimeters before every flight.	Without sufficient voltage to the blast cap, black powder charges may not properly ignite.	9V batteries will be determined as fully charged before being inserted into the AV sled.	Inspection, Analysis	Not Verified	TBD
RF 2	The secondary black powder charges SHALL be larger than the primary black powder charges.	The secondary charges are in place if the primary charges do not initially separate the sections of the launch vehicle. The secondary charges have to be larger than the primary charges to ensure complete separation during flight.	The black powder added to the secondary blast cap will be more than the amount in the primary blast cap.	Inspection	Not Verified	See Section 3.5.10 for ejection charge sizing.
RF 3	The descent velocity under drogue SHALL be less than 120ft/s.	Lower descent velocities under drogue decrease the loading on the main parachute when deployed.	The drogue parachute will slow the launch vehicle to a terminal velocity of less than 120ft/s.	Analysis	Not Verified	See Section 3.6.2 for parachute selection.
RF 4	A fully charged 2S/7.4 lipo battery SHALL be used for the Quasar dual-deploy altimeter and tracker for every flight.	The tracker may not sufficiently work or black powder might not be properly ignited if there is insufficient voltage.	The 2S/7.4 lipo battery will be verified as fully charged before inserted into the AV sled.	Inspection, Analysis	Not Verified	TBD
Design Requirements						
RD 1	A deployment bag SHALL be used to protect the main parachute from ejection gasses.	If exposed to ejection gasses, the main parachute may burn/melt causing the parachute to fail. Additionally, the deployment bag prevents the main parachute shock cords from tangling with other shock cords within the launch vehicle.	The main parachute will be fully inserted into a deployment bag before being put into the launch vehicle.	Inspection	Not Verified	TBD
RD 2	Nomex cloth SHALL be used to protect the drogue and payload parachutes from ejection gases.	If exposed to ejection gasses, the parachutes may burn/melt causing the parachutes to fail.	The drogue and payload parachutes will be fully wrapped inside a Nomex cloth before being attached to the respective shock cords and bays.	Inspection	Not Verified	See Section 3.4.1 pertaining to Nomex cloth.
RD 3	Only U-Bolts SHALL be used for all shock chord connections.	Using U-bolts disperses the shock to multiple points, increasing bulkhead stability.	U-bolts are used on every bulkhead as an anchor point for the recovery harness.	Inspection	Not Verified	U-bolts will be added to bulkheads for shock chord connections.
RD 4	Threaded quick-links SHALL be used to attach all recovery harnesses to the launch vehicle attachment points.	Threaded quick links are easy to install around U-bolts and are unlikely to detach during flight.	Quick links will be used to attach any recovery harness to its respective U-bolt.	Inspection	Not Verified	Threaded quick links will be used to attach recovery harnesses dueing launch vehicle assmebly.

Environmental Requirements						
RE 1	All protective insulation SHALL be biodegradable.	In the case insulation falls out of the launch vehicle, the insulation used will have no negative environmental consequences.	There will be verification of biodegradable insulation before inserting into the parachute bays.	Inspection	Not Verified	See Section 3.4.1 for insulation use.

Table 6.4: Payload Team Derived Requirements

ID	Description	Justification	Success Criteria	Verification Method	Status	Status Description
Functional Requirements						
PF 1	All electronic components in the launch vehicle SHALL be removable.	With removable electronics, easier adjustments can be made to the payload design.	No electronic components within the launch vehicle are fixed in place.	Inspection, Demonstration	Not Verified	Current electronic housing has not been completed.
PF 2	The RF Transmitter and Receiver for release SHALL have an operational range of at least 1 mile.	While the SAIL will be deployed at a maximum of 800 ft, the possibility of wind drift increases the chance of a large distance between the receiver and transmitter. This minimum specification will allow for the release latch to be operational over long distances.	Before inserting the transmitter and receiver in the SAIL, it will be tested and verified to have an operational range of at least 1 mile.	Inspection, Demonstration	Not Verified	TBD
Design Requirements						
PD 1	The SAIL SHALL land with an impact velocity of less than 15 mph.	Having a descent velocity greater than 15 mph increases risk to the STEMnauts by applying greater G forces and higher velocity upon landing.	The rotor blades produce enough thrust to maintain a descent velocity less than or equal to 15 mph.	Analysis, Demonstration	Not Verified	See Section 4.5 pertaining to the SAIL release system.
PD 2	The SAIL SHALL not experience more than 3 G's of sustained centripetal forces during the descent.	Having more than 3 G's of centripetal force increases risk to the STEMnauts during descent.	Contra-rotating blades spinning at a similar RPM will help minimize spinning of the SAIL.	Analysis, Demonstration	Not Verified	See Section 4.5 pertaining to the SAIL release system.
PD 3	The SAIL SHALL land in a vertical orientation resting on the extended landing legs.	Landing in a vertical orientation reduces risk to STEMnauts upon landing at a higher velocity. Resting upon the landing legs prevents the hub from tipping over.	The landing legs span wider than the base of the hub to provide sufficient support upon landing.	Inspection, Demonstration	Not Verified	See Section 4.4.1 pertaining to the landing legs.
PD 4	The contra-rotating rotor blades SHALL rotate at the same RPM.	This eliminates rotation caused by imbalanced aerodynamic torque on the rotors.	The contra-rotating rotors will be designed to work off one motor to ensure the rotors are operating at the same RPM.	Analysis, Demonstration	Not Verified	See Section 4.4.1 pertaining to motor assembly.
PD 5	The SAIL SHALL be a maximum of 8 lb. in total.	Keeping the weight at a reasonable value facilitates a lightweight launch vehicle and improves the performance of the rotor blades.	The SAIL will be a minimum of 5 lb., as per NASA requirement 4.2.2, and a maximum of 8 lb.	Inspection	Not Verified	TBD

Table 6.5: Safety Team Derived Requirements

ID	Description	Justification	Success Criteria	Verification Method	Status	Status Description
Functional Requirements						
SDR 1	All epoxy SHALL be left to cure for at least 24 hours before a load is applied.	The chances of structural failure increases when using uncured epoxy as it weakens the structural integrity of the launch vehicle.	Parts using epoxy are labeled and untouched until the time and date shown on the label.	Inspection	Not Verified	Current fabrication procedures require at least 24 hours of curing for all epoxied parts.
SDR 2	Safety glasses SHALL be provided to each personnel working with or around power tools.	Wearing PPE during power tool operation reduces the risk of skin and eye injury from debris.	Safety glasses provided for every working HPRC member are located in the rocketry lab's PPE closet.	Inspection	Verified	25 pairs of safety glasses are kept in the PPE closet which exceeds lab capacity.
SDR 3	Nitrile gloves, safety glasses, and particulate masks SHALL be provided to all personnel working with volatile liquid and/or powder chemicals.	Wearing PPE when working with hazardous liquids and/or powders reduces the risk of skin and eye injury from debris.	Gloves, safety glasses, and masks provided for every working HPRC member are located in the rocketry lab's PPE closet.	Inspection	Verified	7 boxes of nitrile gloves, 25 pairs of safety glasses, and 2 cases of masks are kept in the PPE closet which exceeds lab capacity.
SDR 4	All launch day attendees SHALL maintain a walking pace at all times on the launch field, including during assembly, launch, and recovery of the launch vehicle.	Walking at a steady pace decreases the risk of falling, tripping, or slipping.	Members attending launch day will maintain a steady walking place while on the launch field.	Inspection	Not Verified	Team members will be briefed before the launch on launch field safety and etiquette.
SDR 5	Hazards identified as orange or red in the risk assessment matrix SHALL be decreased to yellow or green in the CDR through mitigations.	Filtering frequent and/or potentially dangerous hazards allows for a more durable launch vehicle and payload system.	After mitigation, all hazards included in CDR will fall in the yellow or green zones.	Inspection	Not Verified	All potential hazards will fall in the green or yellow zones.
SDR 6	All hazardous/flammable liquids and/or powder chemicals SHALL be stored in a designated flame cabinet whenever it is not being used.	Storing hazardous powders and liquids in a fireproof cabinet reduces risk of injury to students and lab equipment.	Hazardous liquids and powders remain in the flame cabinet unless actively being used by a team member.	Inspection	Verified	All hardeners, resins, lubricants, cleaners, aerosol paints, black powder, oxidizers, and igniters used by the team are stored in a JUSTRITE Flammable Liquid Storage Cabinet.

6.2 Budget

Table 6.6 below details the year-long budget for the 2023-2024 Student Launch Competition.

Table 6.6: 2023-2024 NASA Student Launch Competition Budget

	Item	Quantity	Price Per Unit	Item Total
sub-scale Structure	Plastic 4 in. 4:1 Ogive Nosecone	1	$ 29.80	$ 29.80
	4 in. Blue Tube	2	$ 43.95	$ 87.90
	4 in. Blue Tube Pre-Slotted	1	$ 53.50	$ 53.50
	4 in. Blue Tube Coupler	4	$ 12.31	$ 49.24
	AeroTech I435T-14A Motor	2	$ 80.24	$ 160.48
	Aero Pack 38mm Retainer	2	$ 29.17	$ 29.17
	AeroTech RMS-38/600 Motor Casing	1	$ 98.86	$ 98.86
	Standard Rail Button - 1010	2	$ 4.25	$ 8.50
	U-Bolts	4	$ 1.00	$ 4.00
	Blast Caps	4	$ 1.80	$ 7.20
	Terminal Blocks	4	$ 3.00	$ 12.00
	Double Pull Pin Switch	2	$ 11.95	$ 23.90
			Subtotal:	**$ 564.55**
Full Scale Structure	6 in. Nosecone Fiberglass Ogive 4:1	1	$ 149.99	$ 149.99
	6 in. G12 Fiberglass Tube (60 in.)	1	$ 259.00	$ 259.00
	6 in. G12 Fiberglass Tube (48 in.)	1	$ 207.20	$ 207.20
	6 in. G12 Fiberglass Coupler	4	$ 77.50	$ 310.00
	AeroTech High-Power L1520T-PS Motor	2	$ 289.99	$ 579.98
	Aero Pack 75mm Retainer	1	$ 59.50	$ 59.50
	AeroTech RMS-75/3840 Motor Casing	1	$ 526.45	$ 526.45
	Large Rail Button -1515	2	$ 4.25	$ 8.50
	U-Bolts	8	$ 1.00	$ 8.00
	Blast Caps	4	$ 1.80	$ 7.20
	Terminal Blocks	4	$ 3.00	$ 12.00
	Double Pull Pin Switch	2	$ 11.95	$ 23.90
	Subtotal:			**$ 2,151.72**
Payload	Rotor Blades	2	$ 99.99	$ 199.98
	12” Axle	1	$ 19.99	$ 19.99
	Hinges	8	$ 6.99	$ 55.92
	Rotor Bearings	2	$ 4.99	$ 9.98
	Locking Collars	2	$ 9.99	$ 19.98
	Shock Absorbers	1	$ 13.99	$ 13.99
	Motor	1	$ 49.99	$ 49.99
	Carbon Fiber PETG Filament	1	$ 29.99	$ 29.99
	Arduino	1	$ 50.00	$ 50.00
	IMU	1	$ 15.30	$ 15.30
	RC Transmitter/Receiver	1	$ 19.99	$ 19.99
	Structural/Housing Materials	1	$ 300.00	$ 300.00
	Subtotal:			**$ 785.11**

Recovery and Avionics	Iris Ultra 72 in. Standard Parachute	1	$ 313.37	$ 313.37
	Iris Ultra 60 in. Standard Parachute	1	$ 266.07	$ 266.07
	12 in. Compact Elliptical Parachute	2	$ 67.41	$ 134.80
	Eggtimer Quasar	2	$ 99.99	$ 199.98
	Eggfinder TX Transmitter	1	$ 70.00	$ 70.00
	6 in. Deployment Bag	2	$ 54.40	$ 108.80
	4 in. Deployment Bag	2	$ 47.30	$ 94.60
	18 in. Nomex Cloth	2	$ 26.40	$ 52.80
	13 in. Nomex Cloth	2	$ 17.60	$ 35.20
	5/8 in. Kevlar Shock Cord (per yard)	25	$ 6.99	$ 174.75
	3/16 in. Stainless Steel Quick Links	14	$ 6.98	$ 97.72
	Firewire Electric Match	16	$ 2.00	$ 32.00
	AeroTech Ejection Charge - 1.4g	24	$ 1.25	$ 30.00
	Small Nylon Shear Pins	40	$ 0.18	$ 7.20
	Subtotal:			**$ 1,404.20**
Miscellaneous	Paint	12	$ 18.00	$ 216.00
	Domestic Birch Plywood 1/8 in.x2x2	12	$ 14.82	$ 177.84
	West Systems 105 Epoxy Resin	2	$ 109.99	$ 219.98
	West Systems 206 Slow Hardener	2	$ 62.99	$ 125.98
	PLA 3D Printer Filament Spool	1	$ 26.00	$ 26.00
	ClearWeld Quick Dry 2-Part Epoxy	1	$ 20.28	$ 20.28
	Wood Glue	1	$ 7.98	$ 7.98
	Misc. Bolts	1	$ 20.00	$ 20.00
	Misc. Nuts	1	$ 10.00	$ 10.00
	Misc. Washers	1	$ 8.00	$ 8.00
	Tinned Copper Wire Kit	1	$ 25.00	$ 12.00
	Zip Ties Pack	1	$ 6.59	$ 6.59
	Hook and Loop Strips Box	1	$ 10.00	$ 10.00
	9V Battery Pack	1	$ 12.00	$ 12.00
	Misc. Tape	1	$ 20.00	$ 20.00
	Estimated Shipping			$ 1,000.00
	Incidentals (replacement tools, hardware, safety equipment, etc.)			$ 1,500.00
	Subtotal:			**$ 3,413.63**
Travel	Student Hotel Rooms – 4 nights (# Rooms)	8	$ 898.45	$ 7,187.60
	Mentor Hotel Rooms – 4 nights (# Rooms)	2	$ 556.03	$ 1,112.06
	NCSU Van Rental (# Vans)	3	$ 798.00	$ 2,394.00
	Subtotal:			**$ 10,693.66**
Promotion	T-Shirts	40	$ 15.00	$ 600.00
	Polos	15	$ 25.00	$ 375.00
	Stickers	500	$ 0.43	$ 215.00
	Subtotal:			**$ 1,190.00**
Total Expenses:				**$ 20,202.87**

As highlighted in Figure 6.1, our expenses can be divided into different sub-sections with travel funds taking up the majority of our spending for this year.

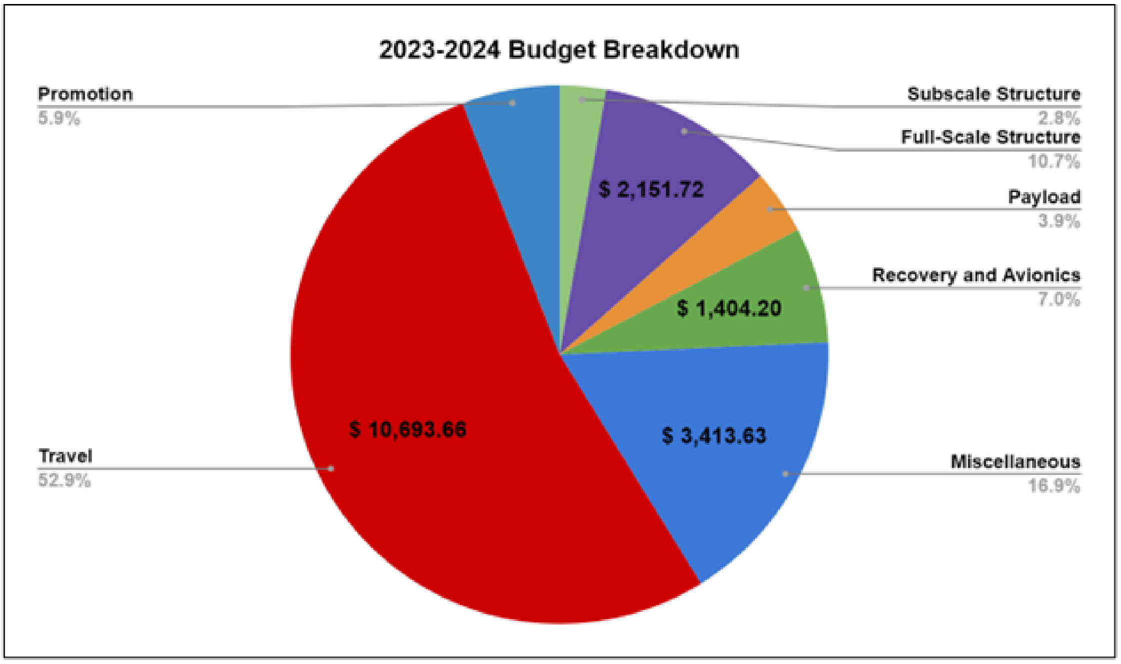


Figure 6.1: 2023 - 2024 Budget Breakdown

6.3 Funding Plan

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

NC State's Student Government Association's (SGA) Appropriations Committee is responsible for distributing university funding to nearly 600 different organizations on campus. Each semester the application process consists of a proposal where the club outline's what they are requesting from SGA, how much money they estimate to receive from other sources, and the anticipated club expenses for the academic year. The club then meets with representatives from SGA and give a presentation outlining club activities and the overall benefit the club provides the university. SGA then collectively allocates money to each organization on campus. In the 2022-2023 academic year, HPRC received $1,592.00 from SGA; $796.00 in the fall semester and $796.00 in the spring semester. For this academic year, a request of $2,000 was submitted for the fall semester and another $2,000 request will be submitted in the spring semester, assuming SGA regulations and budget remain the same.

The Educational and Technology Fee (ETF) is an NC State University fund that allocates funding for academic enhancement through student organizations. In the 2022-2023 academic year, HPRC received $3,000 from ETF and the club anticipates to receive $2,000 for this academic year. This funding will be used primarily to pay for the team's faculty advisors' travel costs.

Student travel costs will primarily be covered by NC State's College of Engineering Enhancement Funds. These funds come from a pool of money dedicated to supporting engineering extracurricular activities at NC State. Based on the 2022-2023 academic year, it is estimated HPRC will receive $7,500 this year.

In addition to funding through NC State organizations, North Carolina Space Grant is a large source of HPRC's funds. NCSG accepts funding proposals during the fall semester and teams can request up to $5,000 for participation in NASA competitions. NCSG will review the proposal and inform the club of the amount awarded. In previous academic years, this has been the maximum amount of $5,000, which will be available for use starting November 2023.

In the past, HPRC has held sponsorship's with Collins Aerospace, Jolly Logic, Fruity Chutes, and more. The team is currently seeking out new sponsorship's and reaching out to past sponsors. The team has found that companies are more likely to donate gifts in kind rather than provide monetary sponsorship. The team estimates to receive $1,000 in gifts of kind this academic year.

These totals are listed in Table 6.7 below, which outlines the projected costs and incoming revenue for the 2023-2024 academic year.

Table 6.7: Projected Funding Sources

Organization	Fall Semester	Spring Semester	Academic Year
Educational and Technology Fee	$0	$2,000	$2,000
Engineering Enhancement Fund	$0	$7,500	$7,500
NC State Student Government	$2,000	$2,000	$4,000
North Carolina Space Grant	$5,000	$0	$5,000
Sponsorship	$500	$500	$1,000
Total Funding:			**$20,500.00**
Total Expenses:			**$20,202.87**
Difference:			**$297.13**

6.4 Project Timelines

Table 6.8: NASA SL Competition Dates and Deadlines

Date/Deadline	Event/Task
14 August 2023	Request for Proposal released
11 September 2023 at 8am CST	Proposal due
4 October 2023	Awarded proposals announced
5 October 2023	PDR Q&A
26 October 2023	PDR packet due at 8am CST
9 November 2023	PDR video teleconference
7 December 2023	CDR Q&A
8 January 2023	sub-scale flight deadline
8 January 2023	CDR packet due at 8am CST
16 Januray - 6 February 2023	CDR video teleconferences
8 February 2023	FRR Q&A
4 March 2023	Vehicle Demonstration Flight deadline
4 March 2023	FRR packet due at 8am CST
11-19 March 2023	FRR video conferences
1 April 2023	Payload Demonstration Flight deadline
1 April 2023	Vehicle Demonstration Flight (reflights only)
1 April 2023	FRR Addendum due at 8am CDT
1 April 2023	Launch window opens for teams not traveling to Huntsville
4 April 2023	Launch Week Q&A
10 April 2023	Arrival in Huntsville
11-12 April 2023	Launch week events
13 April 2023	Launch day
14 April 2023	Backup launch day
23 April 2023	PLAR due at 8am CDT (Huntsville attendees)
30 April 2023	Launch window closes for teams not traveling to Huntsville

TACHO LYCOS

2023-24 Student Launch Competition Gantt Chart

				Aug				Sept				Oct				Nov				Dec				Jan				Feb				Mar				Apr			
Task Name	Task Number	Start Week	End Week	W01	W02	W03	W04	W05	W06	W07	W08	W09	W10	W11	W12	W13	W14	W15	W16	W17	W18	W19	W20	W21	W22	W23	W24	W25	W26	W27	W28	W29	W30	W31	W32	W33	W34	W35	W36
Proposal	1	W02	W06		1	1	1	1	1																														
PDR Q&A	2	W10	W10										2																										
PDR	3	W07	W12							3	3	3	3	3	3																								
PDR Presentation	4	W13	W13													4																							
Subscale Launch	5	W14	W14														5																						
CDR Q&A	6	W17	W17																	6																			
Backup Subscale Launch	7	W18	W18																		7																		
CDR	8	W13	W21													8	8	8	8	8	8	8	8	8															
CDR Presentation Window	9	W22	W25																						9	9	9	9											
FRR Q&A	10	W25	W25																									10											
Vehicle Demonstration Flight Window	11	W25	W28																									11	11	11	11								
FRR	12	W22	W29																						12	12	12	12	12	12	12	12							
FRR Presentation Window	13	W30	W32																														13	13	13				
Payload Demonstration Flight Window	14	W25	W32																									14	14	14	14	14	14	14	14				
FRR Addendum	15	W29	W33																													15	15	15	15	15			
Launch Week Q&A	16	W33	W33																																	16			
Huntsville Launch Week	17	W34	W34																																		17		
Competition Launch	18	W34	W34																																		18		
PLAR	19	W34	W36																																		19	19	19

Figure 6.2: SL Competition Gantt chart.

TACHO LYCOS

2023-24 Student Launch Competition Development Gantt Chart

				Aug				Sept				Oct				Nov				Dec				Jan				Feb				Mar				Apr			
Task Name	Task Number	Start Week	End Week	W01	W02	W03	W04	W05	W06	W07	W08	W09	W10	W11	W12	W13	W14	W15	W16	W17	W18	W19	W20	W21	W22	W23	W24	W25	W26	W27	W28	W29	W30	W31	W32	W33	W34	W35	W36
Brainstorming	1	W02	W05		1	1	1	1																															
Vehicle Design	2	W05	W08					2	2	2	2																												
Payload Design	3	W05	W16					3	3	3	3	3	3	3	3	3	3	3	3																				
Subscale Parts Ordering	4	W08	W16								4	4	4	4	4	4	4	4	4																				
Subscale Manufacturing	5	W10	W14										5	5	5	5	5																						
Subscale Launch	6	W14	W14														6																						
Payload Parts Ordering	7	W16	W17																7	7																			
Fullscale Parts Ordering	8	W18	W21																		8	8	8	8															
Fullscale Manufacturing	9	W21	W25																					9	9	9	9	9											
Payload Manufacturing	10	W21	W28																					10	10	10	10	10	10	10	10								
Fullscale Components Testing	11	W23	W26																							11	11	11	11										
Recovery System Testing	12	W23	W26																							12	12	12	12										
Vehicle Launch Window	13	W25	W28																									13	13	13	13								
Payload Testing	14	W25	W29																									14	14	14	14	14							
Payload Launch	15	W29	W32																													15	15	15	15				
Competition Launch	16	W34	W34																																		16		

Figure 6.3: SL Development Gantt chart.

Table 6.9: sub-scale Build Schedule

September						
Sunday	Monday	Tuesday	Wednesday	Thursday	Friday	Saturday
9/17	**9/18**	**9/19**	**9/20**	**9/21**	**9/22**	**9/23**
-	CAD bulkheads (fin can, AV bay)	Wellness Day	CAD RFS centering rings and runners	Laser cut bulkheads (fin can, AV bay)	-	-
9/24	**9/25**	**9/26**	**9/27**	**9/28**	**9/29**	**9/30**
-	Bulkhead layups (fin can, AV bay)	Laser cut RFS centering rings and runners	Sand bulkheads (FRS, AV bay, fin can); RFS bulkhead layups; Cut AV bay threaded rods	-	-	-
October						
Sunday	Monday	Tuesday	Wednesday	Thursday	Friday	Saturday
10/1	**10/2**	**10/3**	**10/4**	**10/5**	**10/6**	**10/7**
-	Cut body tubes; Cut nose cone shoulder; Epoxy AV bay coupler and body tube	CAD nose cone bulkhead; Laser cut a wood fin for reference; Laser cut new thrust bulkhead	Thrust bulkhead layups; Prep and assemble RFS; Cut threaded rods for RFS	-	-	-
10/8	**10/9**	**10/10**	**10/11**	**10/12**	**10/13**	**10/14**
-	Fall Break	Fall Break	Epoxy runners to RFS	Laser cut nose cone bulkhead and centering ring; Weld nuts to L-brackets	-	-
10/15	**10/16**	**10/17**	**10/18**	**10/19**	**10/20**	**10/21**
-	Cut fin slots into airframe; Assess fit of nose cone permanent and removable bulkheads; Assess placement of blast caps and termminal blocks; Cut and sand motor tube to attach retaining ring	-	Drill holes for blast caps and terminal blocks; Drill holes in airframe for RFS; Nose cone bulkhead layups	-	PDR soft deadline	-
10/22	**10/23**	**10/24**	**10/25**	**10/26**	**10/27**	**10/28**
-	-	Attach T-nuts to nose cone permanent ring; Cut threaded rods for nose cone sled; Epoxy nose cone permanent ring to nose cone; Epoxy motor tube to thrust plate	PDR due Epoxy motor retaining ring to thrust bulkhead; Trace fins geometry onto fiberglass; Start cutting fins out of fiberglass	-	-	-
10/29	**10/30**	**10/31**	-	-	-	-

-	Sand and bevel fiberglass fins; Fill ridges of airframe with spackle to prepare for paint; Drill shear pin and rivet holes into airframe	-	-	-	-	-
November						
Sunday	Monday	Tuesday	Wednesday	Thursday	Friday	Saturday
-	-	-	**11/1**	**11/2**	**11/3**	**11/4**
-	-	-	Sand airframe to prepare for paint; Drill pressure port holes; Drill switchband hole for pull pin; Prime airframe, fins, and nose cone	-	-	-
11/5	**11/6**	**11/7**	**11/8**	**11/9**	**11/10**	**11/11**
-	Paint airframe, fins, and nose cone	-	Clear coat airframe, fins, and nose cone	-	Dry run 1	-
11/12	**11/13**	**11/14**	**11/15**	**11/16**	**11/17**	**11/18**
-	-	-	-	Ejection testing	Dry run 2; Packing for launch day	Launch day

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