NASA Student Launch

2017-2018 Preliminary Design Review (PDR)

November 3rd, 2017

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1 General Information

1.1 School Information/Project Title

School Name: University of Louisville
Organization: River City Rocketry
Location: J.B. Speed School of Engineering
132 Eastern Parkway
Louisville, KY 40292
Project Title: River City Rocketry 2017-2018

1.2 Team Officials

Advisor Name: Dr. Yongsheng Lian
Contact Information: y0lian05@louisville.edu **or** (502) 852-0804

Dr. Lian serves as a faculty member at the Department of Mechanical Engineering at the University of Louisville. He worked at the Ohio Aerospace Institute as a Senior Researcher from 2003 to 2005 and as a Research Scientist at the Aerospace Engineering Department of the University of Michigan from 2005 to 2008. He joined the University of Louisville in 2008. He has 21 years of experience in computational fluid dynamics. He developed algorithms to study fluid/structure interaction, laminar-to-turbulent flow transition, low Reynolds number aerodynamics, and its application to micro air vehicle, two-phase flow, and design optimization.

Team Captain/Safety Officer
Name: Maria Exeler
Contact Information: msexel01@louisville.edu **or** (859) 912-3547

Maria is currently a senior mechanical engineering student at the University of Louisville's J.B. Speed School of Engineering. This is Maria's second year in NSL and her first year as co-captain of River City Rocketry. After contributing to last year's successful season, Maria is looking forward towards improving on the team's safety while continuing to lead the team through new challenges. Maria plans to bring her experiences from working at GE Aviation to her position as co-captain and as safety officer. Throughout last year Maria gained valuable knowledge in fabrication, integration, and problem solving, and she hopes to both pass this knowledge down and employ these skills at GE Aviation following graduation.

Team Captain/Outreach Lead
Name: Gabriel Collins
Contact Information: gdcoll01@gmail.com **or** (502) 457-8829

Gabriel is currently a senior mechanical engineering student at the University of Louisville's J.B. Speed School of Engineering. This is Gabriel's third year in NSL and his first year as co-captain of River City Rocketry. After contributing to last year's success, Gabriel is looking forward towards improving on the team's integration while continuing to take the team to new heights. Gabriel plans to bring his experiences from working at PIA to his position as co-captain and Outreach Lead. Throughout this experience Gabriel has gained valuable knowledge in design, communication, and project optimization, and he hopes to both pass this knowledge down along with employing these skills in the aerospace industry following graduation.

1.3 Tripoli Rocketry Association Mentor

Name: Darryl Hankes
Certification: Level 3 Tripoli Rocketry Association
Contact Information: nocturnalknightrocketry@yahoo.com **or** (270) 823-4225

Darryl Hankes engaged himself in rocketry in February of 2003. In 2004, he joined Tripoli Indiana and where he received his Level 1 TRA certification. In 2006 at Southern Thunder, Hankes received his Level 2 TRA certification. A year later, in 2007, Hankes successfully attempted his Level 3 TRA Certification at Mid-West Power. Over the years, Hankes has flown an R10,000 twice in a team project along with countless M-R projects with clusters, staging, and air starts. He is the former prefect for the Tripoli Rocketry Association, Bluegrass Rocket Society (TRA #130), which provides launch support during test launches. Hankes has mentored the team through all seasons that River City Rocketry has participated in NASA's student launch competitions. The team is pleased to see his return for this year's competition.

1.4 Team Members and Organization

The University of Louisville's team this year will consist of approximately 30 students coming from a variety of backgrounds. To support the technical efforts on the project, the team consists of students from the mechanical engineering, electrical and computer engineering, and computer engineering and computer science departments (CECS). Additionally, the team has recruited other

STEM disciplines from across the university to support the team, specifically with the intent of enhancing our educational outreach.

This project has been broken up by technical design and the following subteams are as follows:

- *Launch Vehicle* – responsible for design, testing, and construction of the launch vehicle. A key responsibility is to ensure the desired altitude is achieved by closely monitoring the mass properties of the vehicle throughout the season.
- *Variable Drag System* – responsible for the electrical design, prototyping, and manufacturing of all electrical vehicle systems. This includes the continued refinement of our variable drag system (VDS).
- *Recovery* – responsible for the analysis, design, testing, and manufacturing of all competition parachutes for the team. Main responsibility is to ensure a safe landing for the launch vehicle while maintaining the kinetic energy requirement.
- *Payload* – responsible for the development, design, construction, and integration of the payload into the launch vehicle.

Each of these subteam has a lead position which has been assigned based on that member's experience, knowledge in the field, and leadership abilities. River City Rocketry is confident that the personnel selected to uphold these leadership positions have the technical knowledge and dedication to have their sub-team produce an innovative system to be showcased at the end of the season.

Other leadership roles that must be upheld are outreach lead and safety officer which have also been selected based on their knowledge of the subjects. These members also have experience and the skills required to successfully execute the required tasks.

2 Changes Since Proposal

2.1 Vehicle Design

2.1.1 Airframe

- Changed from filament wound carbon fiber to woven carbon fiber fabric. Further discussed in 233.3.4.5.

2.1.2 Nose Cone Design

- Changed from LD Haack design to a Parabolic design which is further outlined in 3.4.6.
- Added a 6 in. transition section at the base of the nose cone.

2.1.3 Fin Design

- Changed fin dimensions and removed bend in mid-section of fin. Further outlined in 3.4.3.6.

2.1.4 Centering Ring Design

- Changed minor dimensions of centering rings, shown in Figure 15.

2.1.5 VDS

- New inertial measurement unit chosen
- Telemetry system added
- Upgraded Printed Circuit Board Configuration
- Upgraded Power supply
- External accessibility port proposed

2.1.6 Recovery

2.2 Payload

2.2.1 Rover Orientation Correction System

- Aluminum bridging sled

2.2.2 Rover Locking Mechanism

- Solenoid selected

2.2.3 Rover Body Structures

- Projected dimensions and weights of the payload
- Aluminum body chosen

2.2.4 Rover Drive System

- Reduced number of passive pulleys

2.2.5 Solar Array Structure

- Solar Array Structure now consists of a single base section
- Solar Array Structure will be held in place by locking motor
- Towing peg design for Solar Array Structure support arms
- Spring hinge actuation

2.3 Safety

- Restructured Safety Officer requirements and verifications
- Updated team Safety Manual, MSDS, Risk Assessments, and Motor Safety

2.4 Project Plan

- Project Plan
    - Addition of team derived requirements
    - Increased scope for project schedule
    - Slightly increased projected costs
    - Additional outreach events added

3 Launch Vehicle

3.1 Mission Statement

River City Rocketry's mission for the 2017-2018 NASA Student Launch competition is to design, build, and launch a launch vehicle capable of reaching an apogee altitude of 5,280 ft. and then deploy a rover with foldable solar panels upon landing. For the mission to be successful, upon reaching apogee, the launch vehicle shall safely descend under parachute and land without inflicting damage to the rover, spectators, or the surrounding environment, such that the launch vehicle is capable of being re-flown. Additionally, River City Rocketry aims to inspire young minds in our community by teaching them about science and engineering.

3.2 Mission Success Criteria

1. The launch vehicle shall ascend upon motor ignition, exiting the launch rail at a velocity greater than 75 ft./s.
2. The launch vehicle's motor shall burn out without incident and the Variable Drag System (VDS) shall become active.
3. The launch vehicle shall reach an apogee altitude of 5,280ft. +/- 23ft. AGL.
4. All recovery events shall occur at their programmed altitudes.
5. All sections of the launch vehicle shall have a stable descent.
6. All sections of the launch vehicle shall land safely under kinetic energy requirements and be fully reusable.

3.3 System Level Trade Studies

To properly evaluate several different options for each subsystem of the launch vehicle, Multiple Criteria Decision Analysis (MCDA) was performed on the nose cone profile design, fin material selection, fin mounting system design, and airframe material selections. The Kepner Tregoe trade study method was used to evaluate each option against the alternatives.

3.3.1 Nose Cone Profile Trade Study

Four nose cone profiles were analyzed to determine the optimal design for the launch vehicle. A ½ Power Series, Conical, LD Haack, and Parabolic nose cone profile were modeled in SolidWorks using a curve generated in MATLAB. To determine the optimal nose cone for the launch vehicle, each design was evaluated based upon several performance criteria including mass, coefficient of drag, manufacturability, and internal volume. In the following section, a brief overview of each design option is presented, followed by the results of the trade study.

3.3.1.1 Conic Nose Cone

A conic nose cone design is a variation of the power series and parabolic nose cone designs. The conic nose cone was designed using

$$y = \frac{xR}{L} \qquad \textbf{(1)}$$

where y is the radius corresponding to each point x along L, L is the total length of the nose cone, and R is the radius of the nose cone's base. A plot of the resulting pressure distribution and wall shear for the conic nose cone profile is shown below in Figure 1.

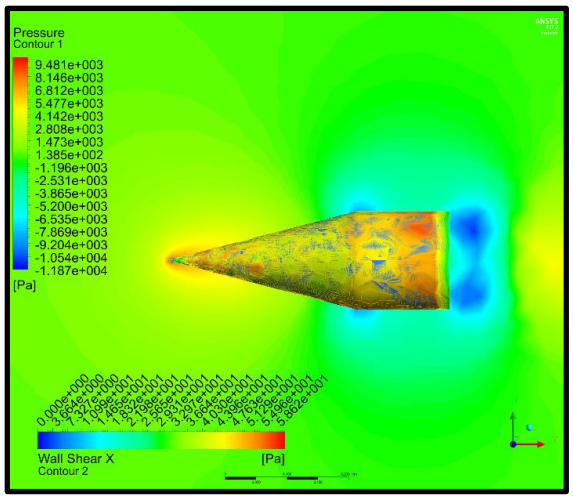


Figure 1: Conic nose cone pressure and wall shear plot.

3.3.1.2 ½ Power Series Nose Cone

The ½ power series nose cone shape is characterized by a blunt tip and a base that is not tangent to the launch vehicle's airframe. The ½ power series nose cone was designed using

$$y = R\left(\frac{x}{L}\right)^n \qquad \textbf{(2)}$$

where y, x, L, and R are as previously defined in (1), and n is a variable corresponding to the bluntness of the nose cone. A plot of the resulting pressure distribution and wall shear for the ½ power series nose cone profile is shown below in Figure 2.

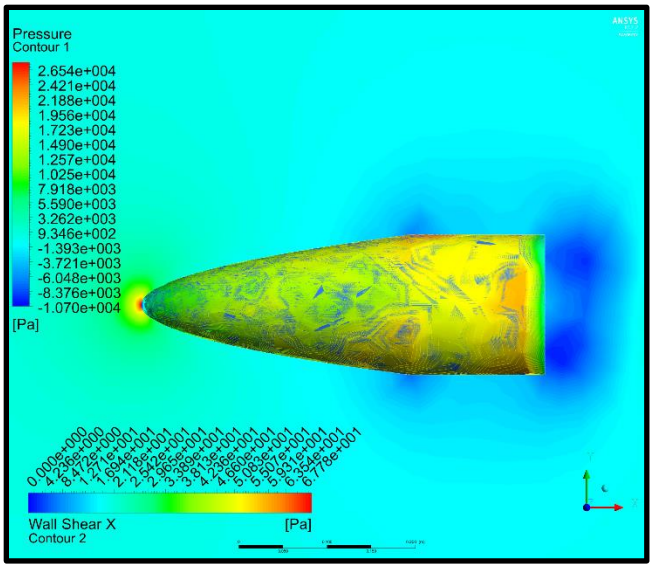


Figure 2: 1/2 power series nose cone pressure and wall shear plot.

3.3.1.3 LD Haack Nose Cone

The LD Haack nose cone shape is mathematically designed to minimize drag for a given length and diameter. The LD Haack nose cone design was generated using

$$y = \frac{R\sqrt{\theta - \frac{\sin(2\theta)}{2} + C sin^3\theta}}{\sqrt{\pi}} \qquad \textbf{(3)}$$

where y and R are as previously defined in (1), C, which is equal to zero for the LD Haack profile, is the parameter that determines the profile of the nose cone, and θ is calculated using

$$\theta = cos^{-1}\left(1 - \frac{2x}{L}\right) \qquad \textbf{(4)}$$

where x and L are as previously defined in (1). A plot of the resulting pressure distribution and wall shear for the LD Haack nose cone profile is shown below in Figure 3.

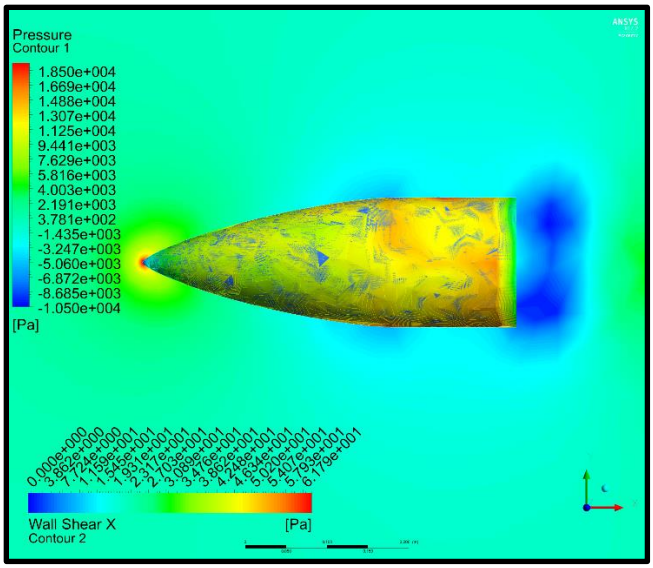


Figure 3: LD Haack nose cone pressure and wall shear plot.

3.3.1.4 Parabolic Nose Cone

The parabolic nose cone shape is generated by rotating a parabola around a line parallel to the central axis and is tangent to the airframe at its base. The parabolic nose cone design was generated using

$$y = R\left(\frac{2\left(\frac{x}{L}\right) - K'^{\left(\frac{x}{L}\right)^2}}{2 - K'}\right) \qquad \textbf{(5)}$$

where x and L are as previously defined in (1), K' determines the shape of the parabola and ranges from a cone at 0 to a full parabola at 1. K' was set as 1 for this trade study. A plot of the resulting pressure distribution and wall shear for the parabolic nose cone profile is shown below in Figure 4.

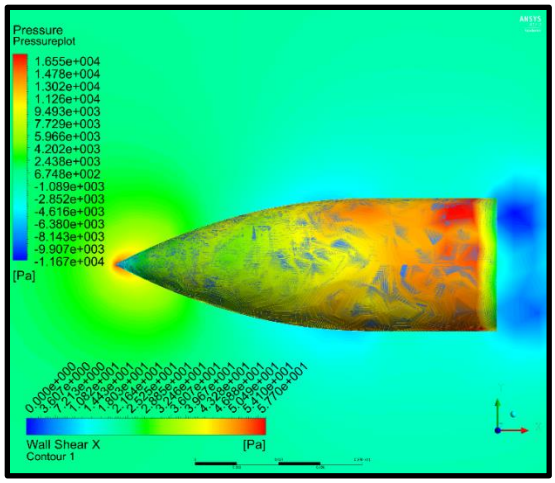


Figure 4: Parabolic nose cone pressure and wall shear plot.

3.3.1.5 Nose Cone Properties

The mass properties of each nose cone design were determined using SolidWorks, and the coefficient of drag for each design was determined using ANSYS Fluent version 17.2. The CFD simulation was configured with a symmetric constraint about the X, Y plane, and the projected frontal area of each nose cone profile was set as $0.1065 ft^2$. The fluid velocity was set to 700ft/s, which is the approximate max velocity of the launch vehicle. The total mass and coefficient of drag of each design alternative are shown below in Table 1.

Characteristic	**Nose Cone Profile**			
	Conic	**1/2 Power Series**	**LD Haack**	**Parabolic**
Mass (lbs.)	1.69	1.94	1.94	1.97
Drag Coefficient	0.3048	0.2861	0.2971	0.2773

Table 1: Nose cone design properties.

3.3.1.6 Nose Cone Design Trade Study Results

The results of the Kepner Tregoe trade study conducted between the conic, ½ power series, LD Haack, and parabolic nose cone designs are shown below in Table 2.

Nose Cone				
Options	12in. LD Haack	12in. 1/2 Power Series	12in. Conical	12in. Parabolic
Mandatory Requirements				
Overall length does not exceed 12 inches.	YES	YES	YES	YES

Coefficient of Drag less than 0.5.		YES		YES		YES		YES	
Wants	Weights	Value	Score	Value	Score	Value	Score	Value	Score
Coefficient of Drag (0-10)	35.00%	8	2.8	7	2.45	5	1.75	9	3.15
Mass (0-10)	30.00%	6	1.8	5	1.5	7	2.1	5	1.5
Manufacturability (0-10)	20.00%	6	1.2	5	1	7	1.4	6	1.2
Internal Volume	10.00%	8	0.8	8	0.8	6	0.6	9	0.9
Total Score		6.6		5.75		5.85		6.75	

Table 2: Kepner Tregoe trade study table for nose cone design.

Based on the results of the trade study, the parabolic nose cone profile is the current leading nose cone design. The conic and ½ power series designs were each eliminated from consideration due to a high coefficient of drag relative to other design options and low internal volume. The LD Haack design provides several desirable characteristics, including a low drag coefficient compared to both the conic and power series nose cone, and the lowest overall mass of all design alternatives. The parabolic nose cone profile provides a comparable mass to the LD Haack design, the lowest coefficient of drag of all alternatives, and a greater internal volume than the LD Haack. Therefore, the parabolic nose cone profile outperformed all other design alternatives and will be used moving forward in the design process. A rendering of the parabolic nose cone design is shown below in Figure 5.

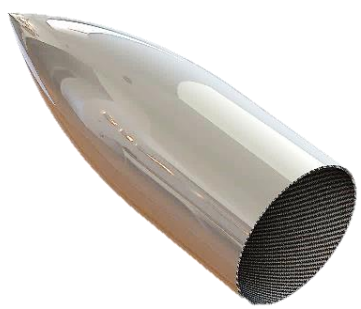


Figure 5. Parabolic nose cone profile.

3.3.2 Fin Mounting System Trade Study

To determine the optimal system for mounting the fins to the booster section of the launch vehicle, a Kepner Tregoe trade study was performed. The three designs considered were an epoxied through the wall mounting system, a removable fin system, and a fin can. The designs were evaluated based on fin rigidity, weight, cost, and durability.

3.3.2.1 Epoxied Through the Wall Fin Mounting

Mounting the launch vehicle's fins via an epoxied through the wall method is the simplest and lightest method considered for the launch vehicle. The system is appealing as it requires no additional parts such as a fin retainer or modified centering rings. The main disadvantages with epoxied fins are that the fin mounting is permanent, a damaged fin cannot be easily removed, and epoxy joints are vulnerable to fracturing upon landing as shown in Figure 6.

Figure 6: Cracked epoxy joint of through the wall fin mounting system.

If a fin is severely damaged during a test flight of the launch vehicle, the entire booster would have to be remanufactured. For this reason, the use of an epoxied through the wall fin mounting system has been deemed unacceptable for the launch vehicle.

3.3.2.2 Removable Fin System (RFS)

A removable fin system (RFS) is a system consisting of four parts that must be designed and manufactured in house. The key advantage to using an RFS is that the fins can be removed quickly and easily from the launch vehicle. This allows for a damaged fin to be replaced and eliminates the possibility of damaging the fins during transportation of the launch vehicle to and from the launch site. An RFS also allows different fin designs to be utilized during test launches to account for mass changes throughout the year. However, an RFS requires tight tolerances which, if exceeded, could drastically affect the fin rigidity and adversely affect the performance of the launch vehicle.

3.3.2.3 Fin Can

A fin can is an apparatus that externally mounts the fins to the booster section of the launch vehicle using an assembly of brackets and airframe sleeves. The fin can method of fin mounting is the most rigid method considered, as each fin is connected to each other and the airframe. A commercially available fin can is shown below in Figure 7.

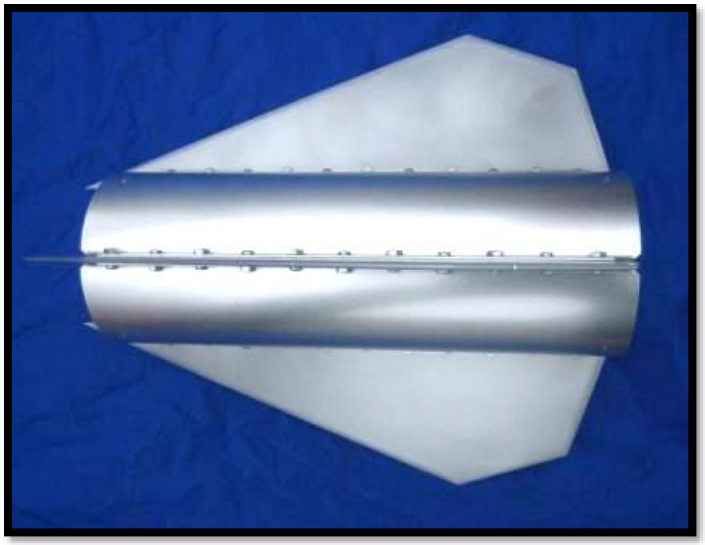


Figure 7: Commercially available fin can.

Utilizing the fin can design requires either purchasing a finished aluminum assembly from Max Q Aerospace or manufacturing the entire assembly in-house. A fin can meets the requirement of being able to replace a broken fin, however the extra fins may have to be machined or molded again which is a timely process. Another negative aspect to the fin can is that without developing several CNC machined molds, the fin can would have to be constructed from aluminum. Aluminum is an undesirable material for fins as it is heavy and the NAR High Power Rocketry safety code states that "A high power rocket may be constructed of paper, wood, fiberglass or plastic with a minimum amount of metallic parts" per NFPA 1127.

3.3.2.4 Fin Mounting System Trade Study Results

The results of the fin mounting system trade study are shown below in the Kepner Tregoe table shown in Table 3.

Fin Mounting System							
Options		Epoxied Through the Wall		Removable Fin System		Fin Can	
Mandatory Requirements							
Ability to replace broken fins		NO		YES		YES	
Wants (0-10)	Weights	Value	Score	Value	Score	Value	Score
Fin rigidity	40.00%	7	2.8	7	2.8	8	3.2
Weight	25.00%	9	2.25	7	1.75	5	1.25
Cost	5.00%	8	0.4	5	0.25	3	0.15
Durability	30.00%	6	1.8	8	2.4	7	2.1
Total Score		7.25		7.2		6.7	

Table 3: Kepner Tregoe trade study table for fin mounting system selection.

The trade study results show that the epoxied through the wall fin mounting system is the best choice of the three options, but due to the requirement that the fins must be able to be replaced if broken, it is unsuitable for use. The next best option, a Removable Fin System (RFS), shall be used going forward in the design phase. An RFS design is further outlined in section 3.4.3.4.

3.3.3 Fin Material Trade Study

To determine the optimal fin material for the launch vehicle, a Kepner Tregoe trade study was performed. The three materials considered were plywood, fiberglass, and carbon fiber. The materials were evaluated by comparing their performance affecting properties such as stiffness, durability, and mass. Each material's advantages and disadvantages are described in the following sections.

3.3.3.1 Plywood

The material of plywood is by far the cheapest, and most available material under consideration for the fins. Plywood is however the weakest of the materials, thus resulting in a higher thickness of 0.25 in. required for use. To use plywood for the fins, the RFS design outlined in 3.4.3.5 would have to be drastically altered to accommodate the increase in thickness. Due to the plywood being incompatible with the RFS, it has been eliminated from consideration as the fin material.

3.3.3.2 Fiberglass

Fiberglass's high flexural and compressive strengths make it an appealing material for the fins of the launch vehicle. A thickness of 0.125 in. has been found to be adequate, making fiberglass compatible with the current RFS design outlined in 3.4.3.5. Fiberglass is not the strongest material under consideration and has the highest density of the three options.

3.3.3.3 Carbon Fiber

Carbon fiber is an extremely stiff material making it very attractive for use on a launch vehicle traveling at close to transonic speeds. Carbon fiber has a lower density than that of fiberglass and can be used with a material thickness of 0.125 in., making it compatible with the current RFS design. Carbon fiber is however the most expensive material under consideration.

3.3.3.4 Fin Material Trade Study Results

The Kepner Tregoe trade study table for the fin material is shown below in Table 4

Fin Material							
Options		Plywood		Fiberglass		Carbon Fiber	
Mandatory Requirements							
Impact resistant		YES		YES		YES	
Compatible with RFS		NO		YES		YES	
Wants (0-10)	Weights	Value	Score	Value	Score	Value	Score

Stiffness	40.00%	4	1.6	8	3.2	9	3.6
Durability	40.00%	4	1.6	8	3.2	9	3.6
Cost	5.00%	10	0.5	5	0.25	1	0.05
Weight	15.00%	6	0.9	5	0.75	8	1.2
Total Score		4.6		7.4		8.45	

Table 4: Kepner Tregoe trade study table for fin material selection.

Carbon fiber has the best overall stiffness, durability, and mass of the materials considered for the fins. Fiberglass has a comparable stiffness and durability, a better cost, but a higher mass than carbon fiber. As the Kepner Tregoe study shows, carbon fiber has been determined to be the optimal material for the fins. Moving forward in the design phase, the launch vehicle will utilize 0.125 in. thick carbon fiber as the fin material. The fin design is further outlined in section 3.4.3.6.

3.3.4 Airframe Material Trade Study

To determine the optimal material to use for the launch vehicle's airframe, several materials were researched and a Kepner Tregoe trade study was performed on the leading candidates. The materials under consideration include G12 fiberglass, filament wound carbon fiber, A&P Technology QISO carbon fiber fabric, and the commercially available material BlueTube. Each material was evaluated based upon its mass, strength, availability, and cost.

3.3.4.1 G12 Fiberglass

G12 fiberglass, shown in Figure 8, is a commonly used material for large high-powered rockets. Many rocketry suppliers offer fiberglass airframe tubes in several different sizes that would be suitable for the launch vehicle. Fiberglass is a very strong material and is offered with a very smooth surface finish, both of which benefit the performance of the launch vehicle. Some negative aspects of using fiberglass airframe is the high cost and mass.

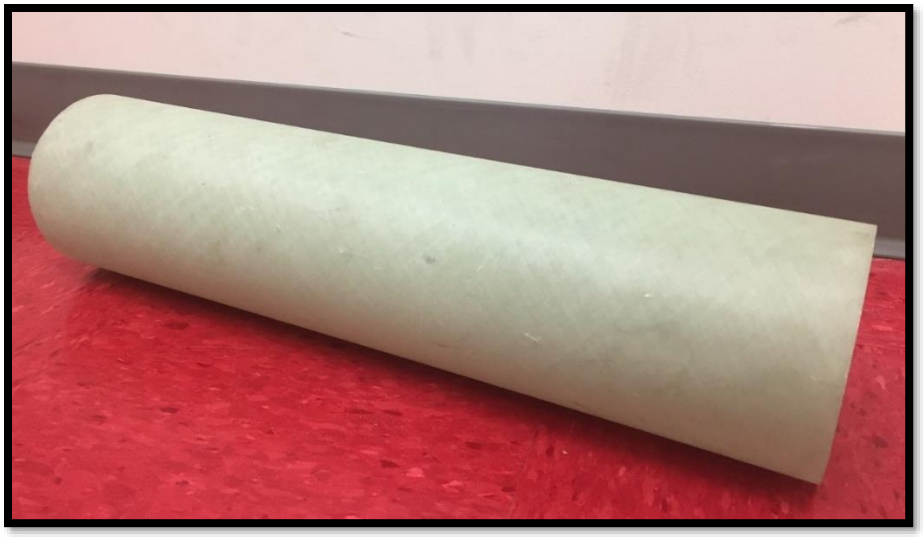


Figure 8: G12 Fiberglass airframe material.

3.3.4.2 Filament Wound Carbon Fiber

In the past, the team has used the X-Winder filament winder shown in Figure 9 to manufacture carbon fiber airframe in house. The ability to manufacture airframe in house quickly is a significant advantage that has proven to be an important asset to the team in the past. Some negative aspects to filament wound carbon fiber include that the carbon fiber filament and epoxy that must be purchased is very expensive, the surface finish is variable, and the winding process is time consuming, messy, and volatile.

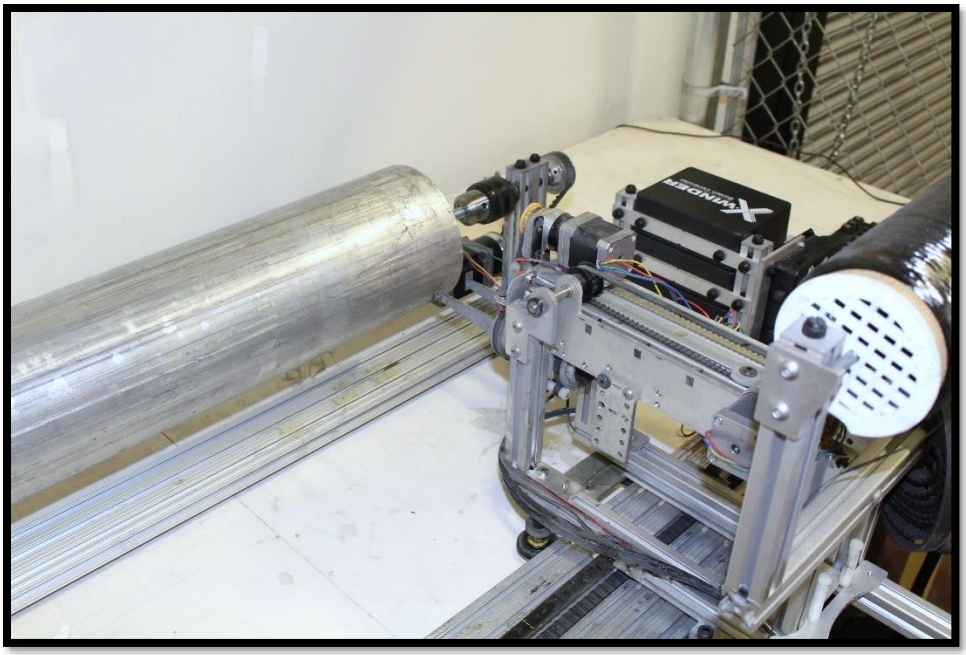


Figure 9: X-Winder filament winder.

3.3.4.3 A&P Technology QISO Carbon Fiber Fabric

One of River City Rocketry's sponsors, A&P Technology, manufactures braided carbon fiber fabric. A&P Technology has donated their product, shown in Figure 10, to the team for use during the season. The carbon fiber fabric is made from individual strands braided together at alternating angles to increase strength in varying directions. Another positive to using the carbon fiber fabric is that the epoxy levels can be easily controlled during manufacturing, thus allowing for optimized fiber-resin ratios and lower mass.

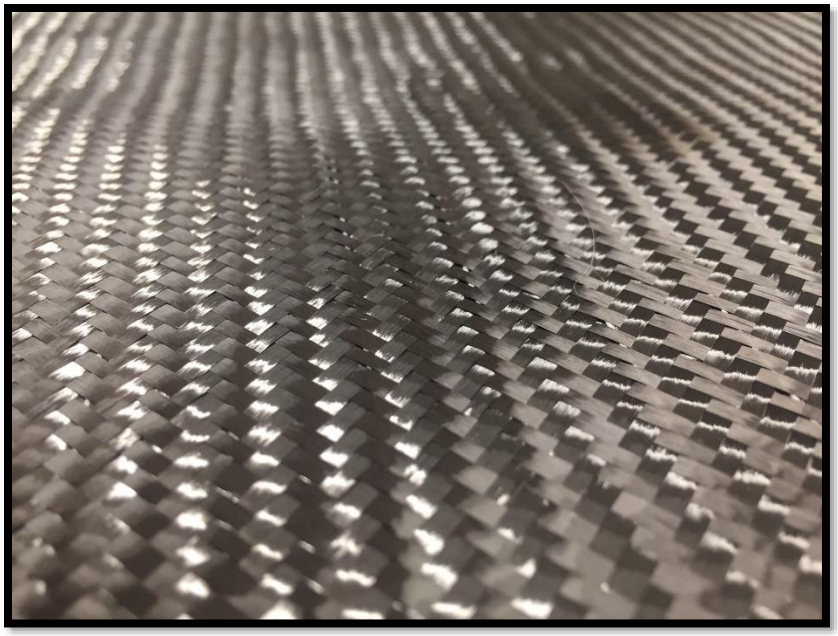


Figure 10: A&P Technology braided carbon fiber fabric.

3.3.4.4 BlueTube

BlueTube is a commercially available airframe material specifically made for rocketry. BlueTube is 36% lighter than G12 fiberglass, shock resistant, easy to cut, and available in sizes suitable for the launch vehicle. Some disadvantages to using BlueTube for the airframe material of the launch vehicle include that the team cannot manufacture it in house, it's low strength, and it is not waterproof. BlueTube airframe is shown below in Figure 11.

Figure 11: BlueTube airframe material.

3.3.4.5 Airframe Material Trade Study Results

The airframe material Kepner Tregoe trade study results are shown below in Table 5.

Airframe Material									
Options		Fiberglass		Filament Wound Carbon fiber		A&P Technology QISO Carbon Fiber Fabric		BlueTube	
Mandatory Requirements									
Support loads during lift off		YES		YES		YES		YES	
Impact resistant		YES		YES		YES		YES	
Wants (0-10)	Weights	Value	Score	Value	Score	Value	Score	Value	Score
Weight	35.00%	4	1.4	7	2.45	8	2.8	8	2.8
Strength	35.00%	8	2.8	9	3.15	9	3.15	5	1.75
Availability	20.00%	8	1.6	7	1.4	9	1.8	7	1.4
Cost	10.00%	7	0.7	3	0.3	9	0.9	8	0.8
Total Score		6.5		7.3		8.65		6.75	

Table 5: Kepner Tregoe trade study table for airframe material selection.

The Kepner Tregoe trade study results show that the A&P Technology QISO carbon fiber fabric is the most optimal airframe material considered for the launch vehicle. The fabric outscored every material in every category, thus, the launch vehicle will be designed with the intent of using the A&P Technology QISO carbon fiber fabric as the airframe material.

3.4 Leading Preliminary Design

3.4.1 Launch Vehicle Overview

The launch vehicle has been designed to safely deliver a rover payload to an apogee altitude of 5,280 ft. AGL. The launch vehicle consists of five sections: the booster, booster recovery bay, payload bay, payload recovery bay, and nose cone. Along with the airframe sections are the VDS coupler, payload coupler, and the payload recovery coupler. The launch vehicle will be constructed primarily of carbon fiber, wood, and aluminum. An overview of the launch vehicle is shown below in Figure 12.

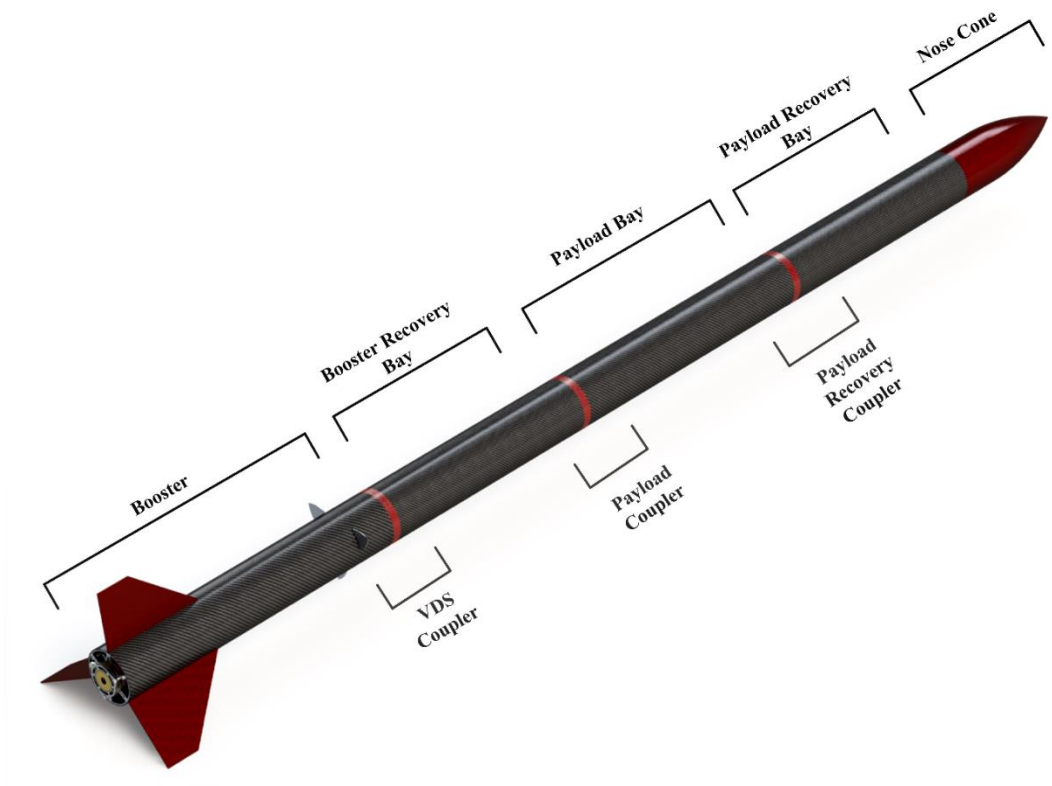


Figure 12: Launch Vehicle overview.

3.4.2 Launch Vehicle Dimensions

The launch vehicle is 6 inches in diameter and 145 inches in length. The launch vehicle's overall dimensions, and the dimensions of each section, are shown below in Table 6. The airframe section's lengths were determined by researching motor dimensions, estimating the rover payload size, and estimating the packed parachute sizes. Any witness rings' dimensions are included in the furthest aft section it corresponds to.

Section	Length
Booster	37 in.
Booster Recovery Bay	28 in.
Payload Bay	33 in.
Payload Recovery Bay	29 in.
Nose Cone	18 in.
Total Length	145 in.

Table 6: Launch vehicle dimensions.

3.4.3 Booster

The two primary goals for the booster section are to serve as the connection point for the RFS, and to effectively house the motor so that the launch vehicle is propelled upward upon motor ignition. Within the booster section is the motor mount tube and the RFS, further discussed in 3.4.3.1 and

3.4.3.5 respectively. In the upper half of the booster airframe section are three 0.1875 in. slots, cut 120° from each other, for the Variable Drag System (VDS) drag blades. The VDS is further discussed in 3.4.8. A rendering of the fully assembled booster section is shown below in Figure 13.

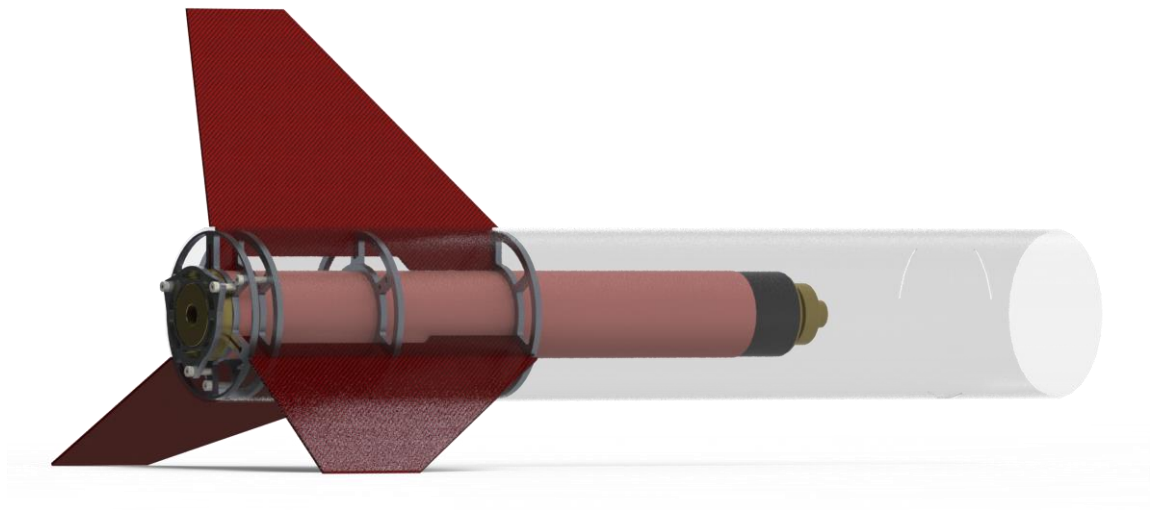


Figure 13: Assembled booster section with transparent airframe.

3.4.3.1 Motor Mount Tube

The motor mount tube is responsible for securing the motor in the center of the launch vehicle and serving as the connection point for the centering rings. The motor mount tube will be 22 inches in length and 3.25 inches in diameter and constructed of carbon fiber fabric.

3.4.3.2 Motor Alternatives

For a launch vehicle of this size and approximate weight, the motors shown in Table 7 were considered for use. The motor selection is further discussed in 3.6.3.

Motor	Total Impulse (Ns)	Average Thrust (N)
Aerotech L2200	5,104	2,200
Cesaroni L2375	4,905	2,375
Cesaroni L3150	4,806	3,150

Table 7: Different motor alternatives and their characteristics.

3.4.3.3 Motor Retainer

To secure the motor in the launch vehicle, a custom motor retainer has been designed. The motor retainer will be cut from 6061-T6 aluminum using a Maxiem 450 water jet. After the fins and motor are installed, the motor retainer is placed on the aft end of the motor casing, and then secured to the fin retainer via three stainless steel #10-32 UNF-3A shoulder bolts. A dimensional drawing of the motor retainer is shown below in Figure 14.

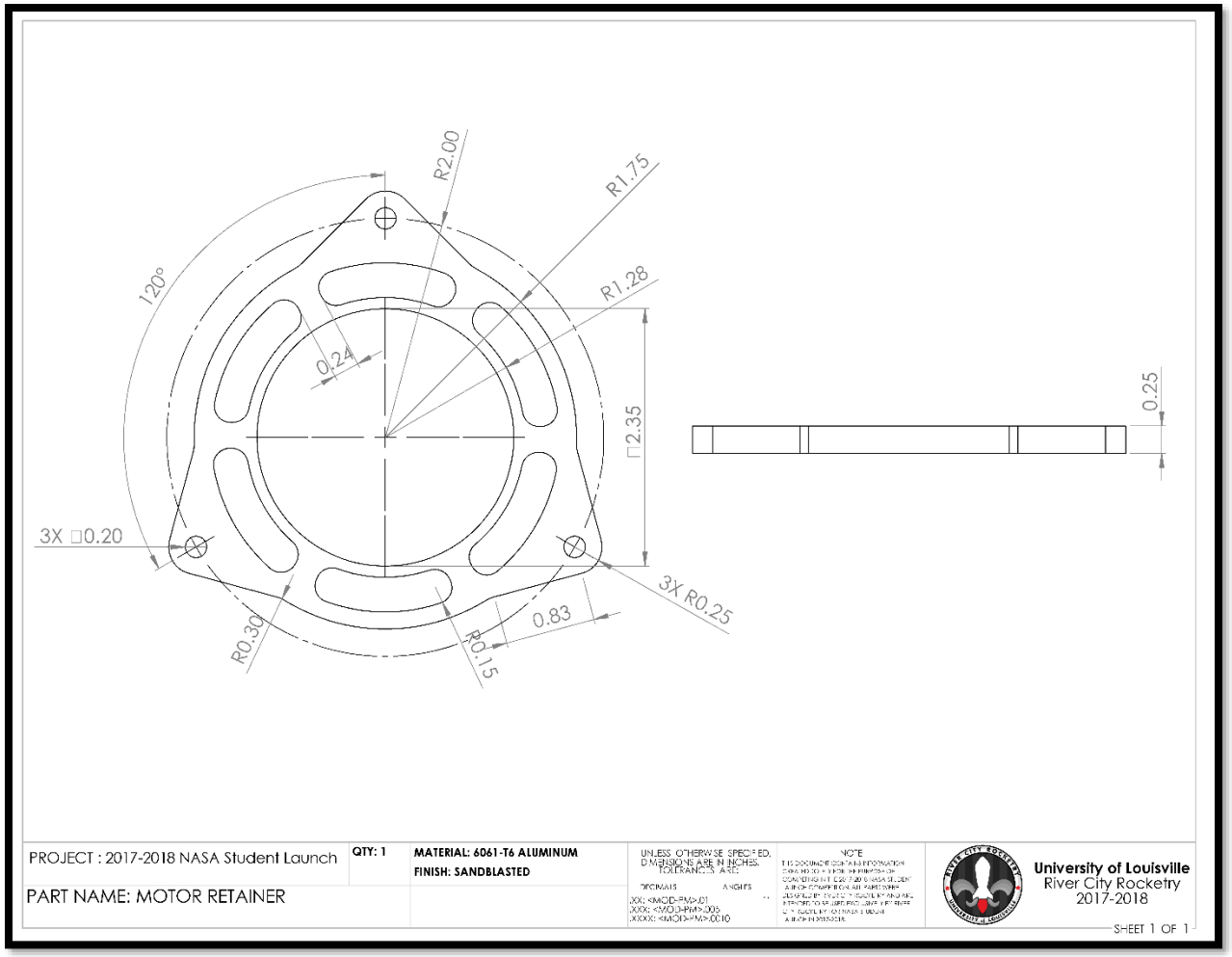


Figure 14: Dimensional drawing of the motor retainer.

3.4.3.4 Centering Ring Design

The launch vehicle will utilize three, 0.25 in. thick, 6061-T6 grade aluminum centering rings to transfer the thrust from the motor to the rest of the launch vehicle. The centering rings will be cut using a Maxiem 450 water jet. Dimensional drawings of the fore and mid centering rings are shown below in Figure 15 and Figure 16 respectively.

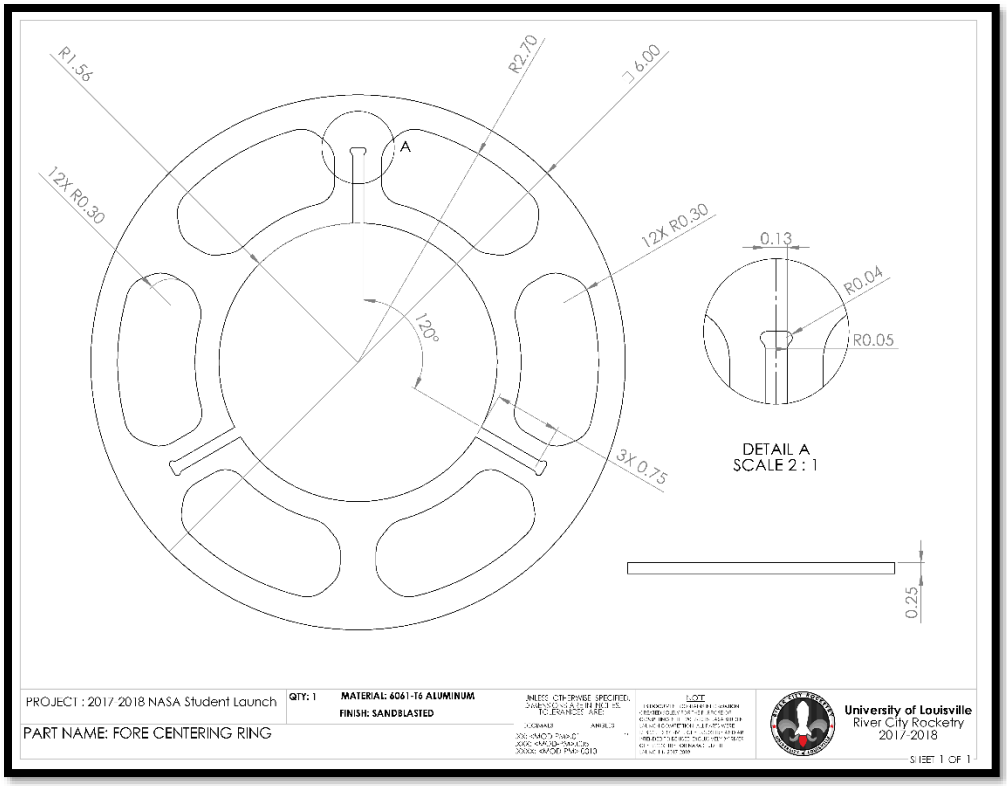


Figure 15: Dimensional drawing of the fore centering ring.

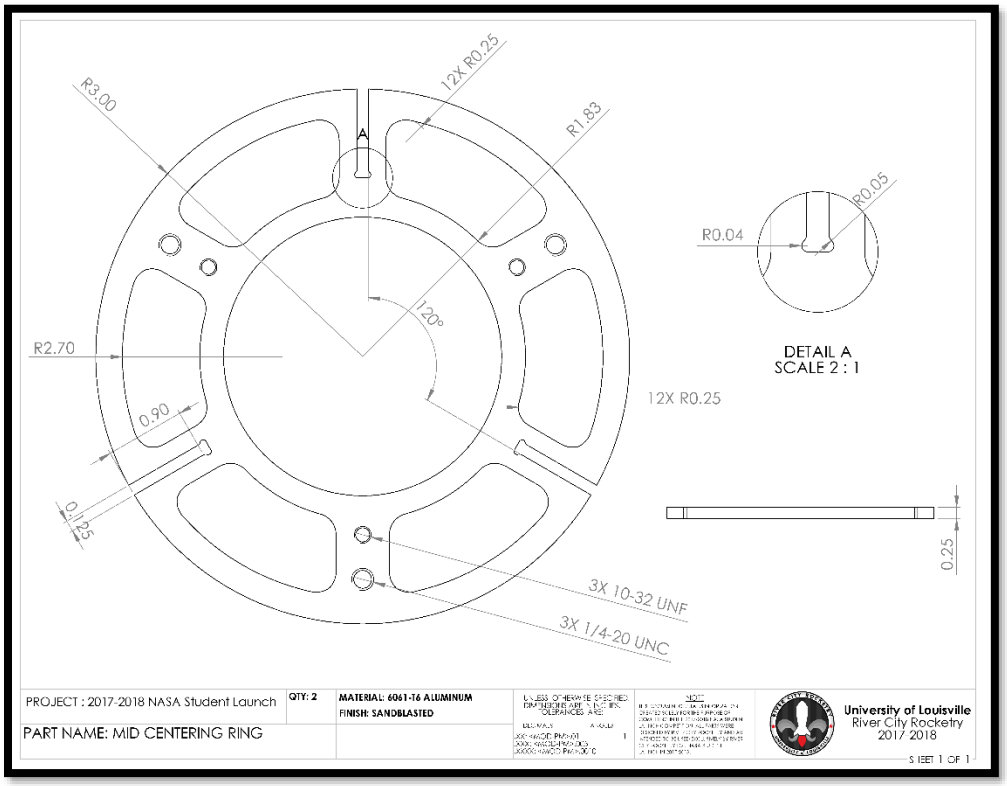


Figure 16: Dimensional drawing of the mid centering ring.

3.4.3.5 Removable Fin System (RFS)

The RFS consists of one fore centering ring, two mid centering rings, and a fin retainer. Each centering ring will be epoxied to the motor mount tube and booster airframe with high strength epoxy. The steps for inserting the fins into the RFS, and a schematic of the RFS, are shown below in Table 8, and Figure 17 respectively.

Steps	Instructions
A	Insert fore fin tab into fore centering ring slot.
B	Insert the mid-section of the fin into the fin slot in the mid centering ring.
C	Insert the aft-section of the fin into the fin slot in the aft centering ring.
D	Insert the aft fin tab into the fin retainer fin slot. Secure the fin retainer with three ¼''-20 18-8 stainless steel shoulder bolts threaded into the aft centering ring.

Table 8: Removable Fin System assembly steps.

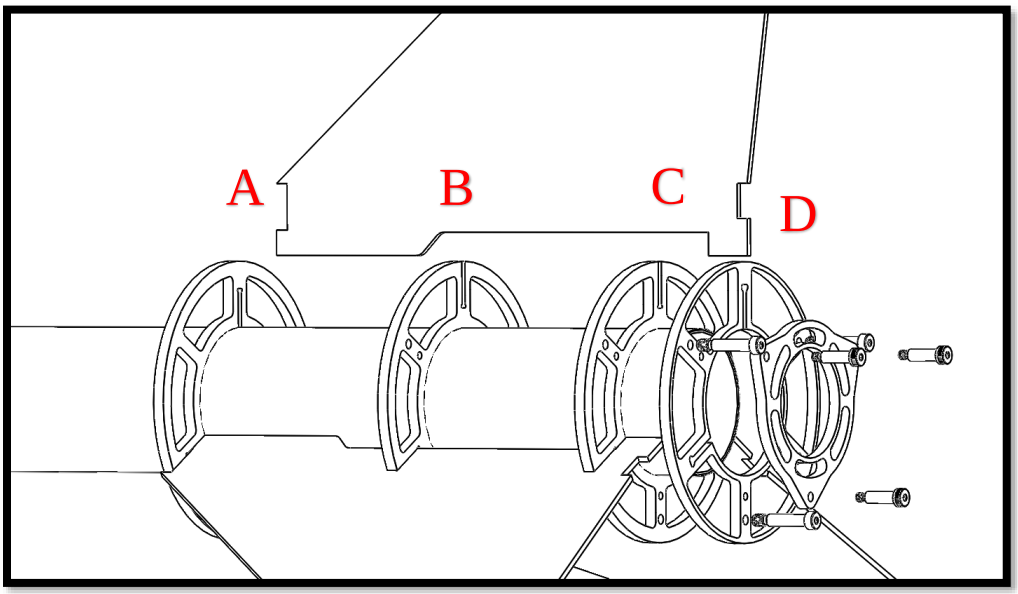


Figure 17: Schematic of the RFS.

3.4.3.6 Fin Design

To reduce drag and not be interfered with by the VDS, the launch vehicle will utilize three swept cropped delta fins. The fins will be cut from 0.125 in. thick carbon fiber sheet. Carbon fiber was chosen as the fin material due to the trade study results outlined in section 3.3.3.4. A dimensional drawing of the fin design is shown below in Figure 18.

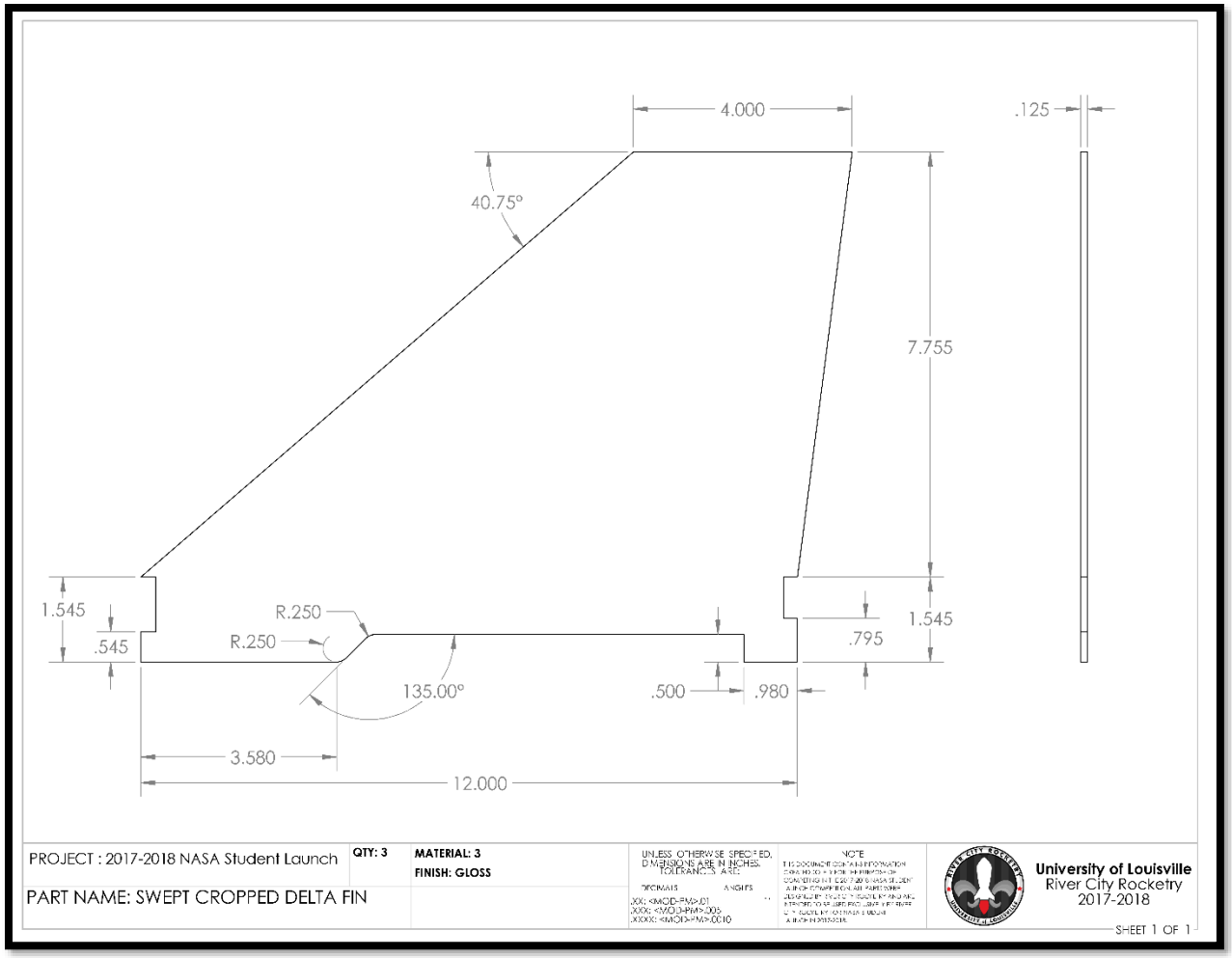
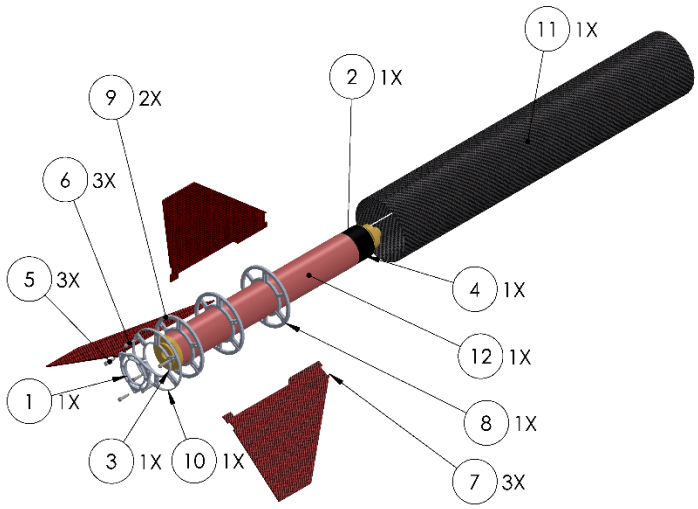


Figure 18: Dimensional drawing of the fin design.

3.4.3.7 Booster Overview

An exploded view and bill of materials for the booster section is shown below in Figure 19.



ITEM NO.	PART NUMBER	QTY.
1	Motor Retainer	1
2	Motor Casing	1
3	Aft Closure	1
4	Forward Closure	1
5	#10-32 Shoulder Bolt	3
6	1/4" - 20 Shoulder Bolt	3
7	Swept Cropped Delta Fin	3
8	Fore Centering Ring	1
9	Mid Centering Ring	2
10	Fin Retainer	1
11	Booster Tube	1
12	Motor Mount Tube	1

PROJECT : 2017-2018 NASA Student Launch
QTY: 1
PART NAME: BOOSTER BOM
UNLESS OTHERWISE SPECIFIED, DIMENSIONS ARE IN INCHES. TOLERANCES ARE:
DECIMALS ANGLES
.XX: <MOD-PM>.01
.XXX: <MOD-PM>.005
.XXXX: <MOD-PM>.0010
NOTE
University of Louisville
River City Rocketry
2017-2018
SHEET 1 OF 1

Figure 19: Exploded view of the booster section and the bill of materials.

3.4.4 Payload Bay

The payload bay is 33 inches in length and is responsible for securing the rover payload during flight. The payload bay consists of the Rover Orientation Correction System and the deployable rover, further outlined in . A rendering of the rover payload secured in the payload bay is shown below in Figure 20.

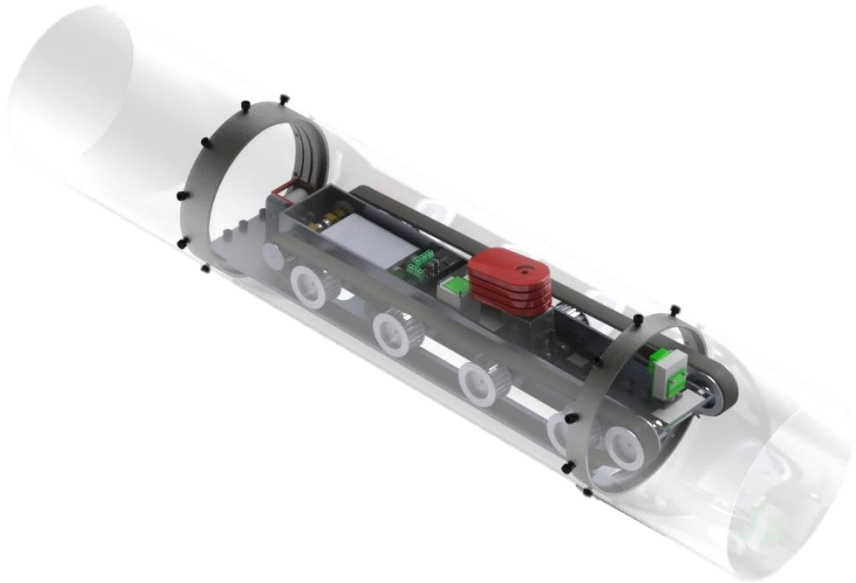


Figure 20: Rover payload secured inside payload bay.

3.4.5 Avionics

All avionics will be secured inside their respective section's coupler. The electronics will be secured inside the coupler via two threaded rods and a custom designed, 3D printed sled, as shown in Figure 21.

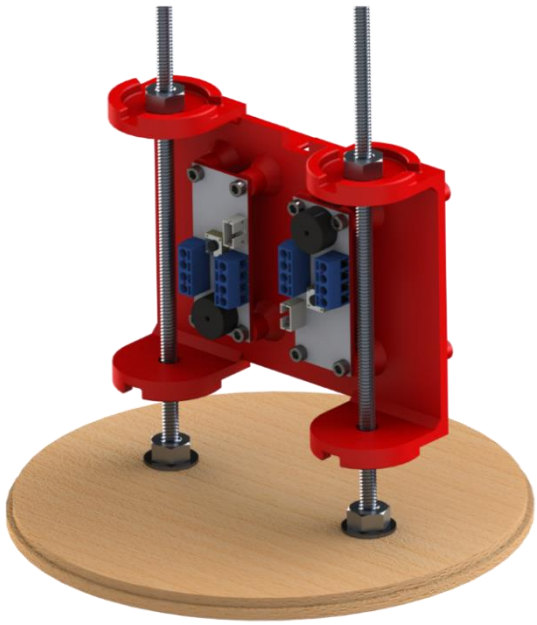


Figure 21: Secured recovery electronics on 3D printed sled.

The electronics will be secured by four 4-40 nylon screws threaded into extruded standoffs on the sled. All StratoLogger altimeters will each be powered by a Duracell 9-volt battery secured to the opposite side of the sled by a cover and four 4-40 nylon screws.

3.4.6 Nose Cone

A 12-in. parabolic nose cone profile with a 6-in. transition section will be used on the launch vehicle. This design provides minimal drag, low mass, and adequate internal volume for electronics. The nose cone design is further discussed in 3.3.1.4. A rendering of the nose cone is shown below in Figure 22.

Figure 22: Parabolic nose cone design.

3.4.7 Subscale Launch Vehicle Design

To test the recovery subsystem design and aerodynamic properties of the launch vehicle, a one-half scale model was designed in OpenRocket. The OpenRocket model is shown below in Figure 23.

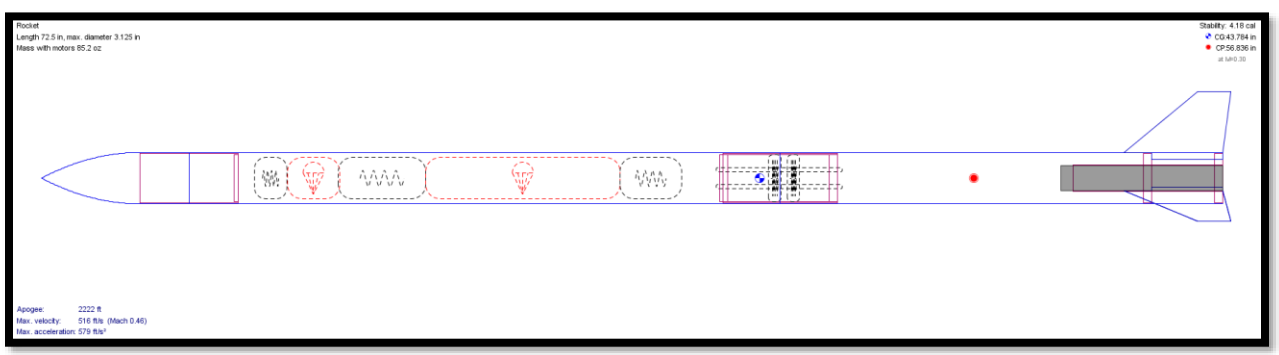


Figure 23: OpenRocket subscale launch vehicle model.

To reduce mass and manufacturing time, the subscale launch vehicle will utilize BlueTube as the airframe material. To allow for a standard dual deployment recovery configuration, the payload and payload recovery bays were replaced with a single recovery bay. The booster and booster recovery bay were replaced with a single booster section. This allows for a similar recovery configuration to the full-scale launch vehicle, while still maintaining the same external proportions. After analyzing several simulations' results, further discussed in 3.6.8, the subscale

vehicle will utilize an Aerotech I-300 motor. The subscale launch vehicle dimensions are shown below in Table 9.

Section	Length (in.)
Booster	27
Coupler	8
Recovery	36
Nose Cone	9

Table 9: Subscale launch vehicle dimensions.

3.4.8 Variable Drag System

To reach the target altitude of 5,280 ± 23 ft., the team has developed and implemented a dynamic target apogee air braking system called the "Variable Drag System" (VDS). Through ample test data and simulations, the VDS V2 proved to be an accurate system in slowing the vehicle with a precision of ± 31 feet of 5,280 ft. The goals of the VDS V3 are to improve the precision of the apogee to ± 23 ft. from target apogee, and to develop a system that will deliver data telemetrically to a custom ground station.

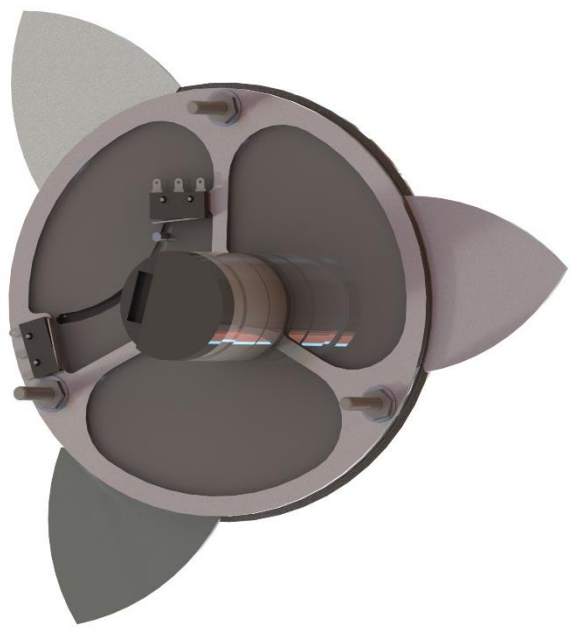


Figure 24: Variable Drag System.

The VDS functions by reducing the projected apogee of the vehicle to the specified target altitude by autonomously altering the drag force acting on the launch vehicle. Based on real time sensor data, three drag blades are actuated after motor burn. With the flat faces of the blades perpendicular to the airstream, the VDS increases the projected area of the vehicle by a factor of 1.28 and the coefficient of drag by an estimated factor of 1.35. Additionally, the VDS V3 will contain a radio

frequency telemetry system that will deliver data regarding the current state of the VDS to a ground station.

3.4.8.1 Design Optimization and System Level Trade Studies

The VDS V3 configuration will consist of two printed circuit boards, which utilize one shared Teensy 3.6 microcontroller, the data acquisition system (DAQ) is made up by a VN-100 IMU, a BTN7960 H-bridge motor driver circuit, as well as DC motor encoder and two limit switches. The software will consist of a program written in C/C++ which controls the sensors data acquisition and prediction to motor control, to the physical actuation of the drag blades. This system will also contain a telemetry system consisting of a Teensy 3.6, an XBee SX pro radio frequency (RF) transmitter and a power source.

These components are upgraded from the VDS V2 which shared a similar configuration, differing in its sensory unit and otherwise preliminary design. The upgrades to this system have undergone much consideration for the overall benefit of the vehicle, and are outlined below. Included in Table 10 is a reference as to what is being referred to as different versions of the system are mentioned.

VDS	VDS V2	VDS V3
Refers to the overall system and information that is otherwise unchanging/applicable as a whole.	Refers to the previous version of the VDS used in the 2016-2017 competition season.	Will be used to refer to upgrades that are currently in progress to improve the system in the 2017-2018 competition season.

Table 10: VDS versioning reference.

3.4.8.1.1 Drag Control Configuration systems

To explore the most ideal configuration for the VDS, various designs were considered for each aspect of the system, including the physical drag configuration, sensory units, and electronic components.

Research was conducted on the VDS V2 configuration to determine whether the blades were the most optimal way to increase the drag of the vehicle. Additionally, the sensory capabilities of the VDS V2 had relatively high error tolerances of ± ~10m, and recorded a large amount of noise that caused difficulty throughout the data analysis process. For these reasons, higher grade inertial measurement units were researched. These findings will be discussed in section 3.4.8.1.3. Below is a brief introduction into each system that was considered for implementation.

3.4.8.1.1.1 *Cold Gas Thrusters*

The Cold Gas Thruster (CGT) system was proposed as an alternative to drag blades. The CGT was designed to make use of thrust as a braking force rather than increasing the drag force on the launch vehicle. Compressed air was routed from a single air tank housed within the launch vehicle to two external nozzles directed into the oncoming airflow surrounding the vehicle. The airflow was designed to be controlled by a single solenoid valve fitted to the outlet of the compressed air tank. The advantage of this system is that the applied braking force is constant. The system is also able to apply this braking force with very little lag time at low speeds. This design was manufactured and tested extensively during offseason. The tests resulted in the elimination of this design as it failed to achieve the expected thrust output. This was due to inadequate airflow through the solenoid valve. A rendering of the design configuration is shown below in Figure 25.

Figure 25: Cold Gas Thruster braking system.

3.4.8.1.1.2 *Tri-Aileron System*

An aileron system was considered in the past for the VDS prototype design, as it maximizes projected area against airflow but was ultimately ruled out due to constraints on its actuating ability and lack of control capability. This system was reconsidered with small design alterations to its opening mechanism, this time utilizing a gear system with an opening arm as opposed to a linear actuator connected via three struts. This would allow for a more stability and control upon opening. Each aileron would have hinged about a clevis and pin assembly that would have fastened to the enclosure, which would be epoxied into a coupler enclosure within the vehicle. A rendering of the proposed design is shown below in Figure 26.

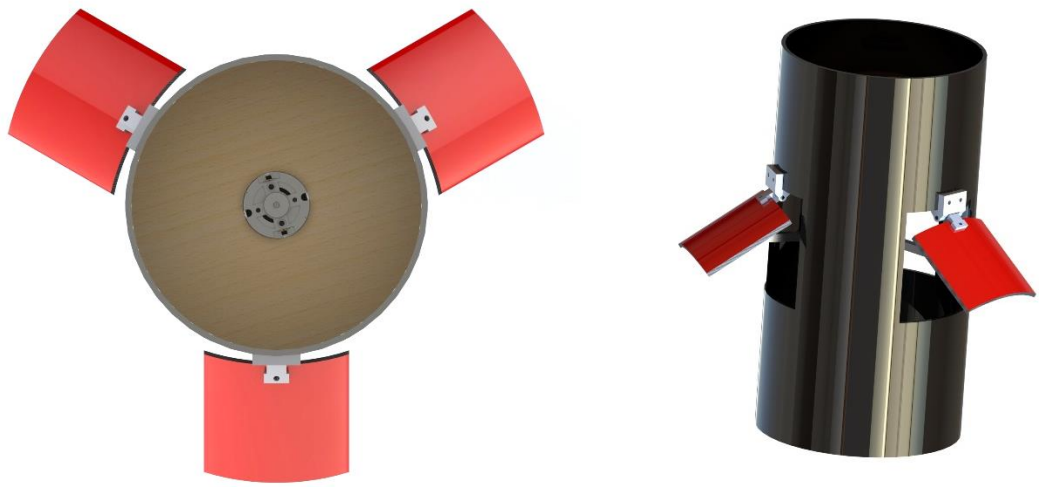


Figure 26: Tri-aileron drag control system.

3.4.8.1.1.3 *Three Blade Variable Drag System*

The original standing design of the VDS is the three-drag blade configuration. The design allows for blade actuation perpendicular to the airflow instead of against the airflow. Previous air braking designs have had actuating joints that work against the incident air current. Through the perpendicular actuation of this design, the overall volume is minimized, and the actuating device does not have to directly counteract the drag force. The entire VDS can fit inside a single 6 in. by 12 in. carbon fiber coupler. This design is the most compact, allowing for the overall launch vehicle length to be reduced. A rendering of the three bladed VDS is shown below in Figure 27.

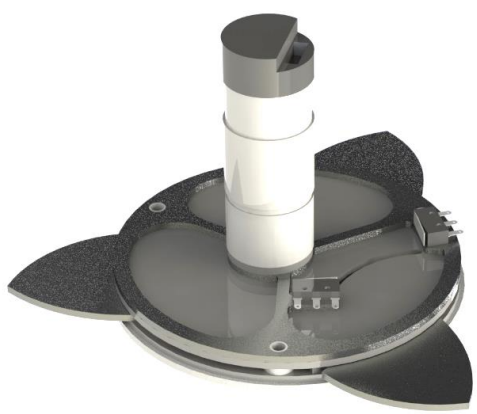


Figure 27: Three-bladed Variable Drag System.

3.4.8.1.2 Drag Control System Trade Study Results

The results of the system level trade study on the drag control system are shown in Table 11.

Drag Control System							
Options		Cold Gas Thrusters		3 Blade VDS		Aileron System	
Mandatory Requirements							
Located aft of the center of gravity of the launch vehicle.		YES		YES		YES	
Wants (0-10)	Weights	Value	Score	Value	Score	Value	Score
Braking Force	35.00%	3	1.05	6	2.1	7	2.45
Continuous Actuation	20.00%	1	0.2	8	1.6	6	1.2
Mass	20.00%	3	0.6	8	1.6	7	1.4
Volume	15.00%	3	0.45	7	1.05	7	1.05
Manufacturability	5.00%	2	0.1	7	0.35	7	0.35
Simplicity	5.00%	2	0.1	8	0.4	5	0.25
Total Score		2.3		6.35		6.1	

Table 11: Drag control system Kepner Tregoe trade study table.

After considering braking power, coupled with ease of integration, and strength against drag forces, it has been determined that the current tri-blade configuration is still the most optimal air braking system.

3.4.8.1.3 Inertial Measurement Units

Through analysis of the data collection system implemented in the VDS V2, it was apparent that the quality of the sensors used had room for improvement. The VDS V2 used a combination of the Bosch BMP 280 pressure altimeter, and BNO055 nine degrees of freedom (9DOF) sensor. Cumulatively, these units received barometric, acceleration and derived velocity, gyroscopic, and magnetometer data. These units are not rated for aerospace applications and the error tolerance was relatively high in practice (roughly ± ~10m off from actual values). Therefore, it was concluded that a more precise IMU should be implemented in the VDS V3.

3.4.8.1.3.1 *Kalman Filter/Noise in Data*

One of the largest obstacles in the data acquisition process through V2 was the issue with noise from the sensors. It was hypothesized that, because the electrical harness is not in a vacuum, nor airtight from the blade configuration, the actuation of the blades from the electronics bay was causing a pressure differential to form within the rocket as air was scooped from the outside of the vehicle into the bay. The data taken from the altimeter displayed some questionable ‘blips’ in the coast phase, which indicated a flight pattern that is not probable or even possible.

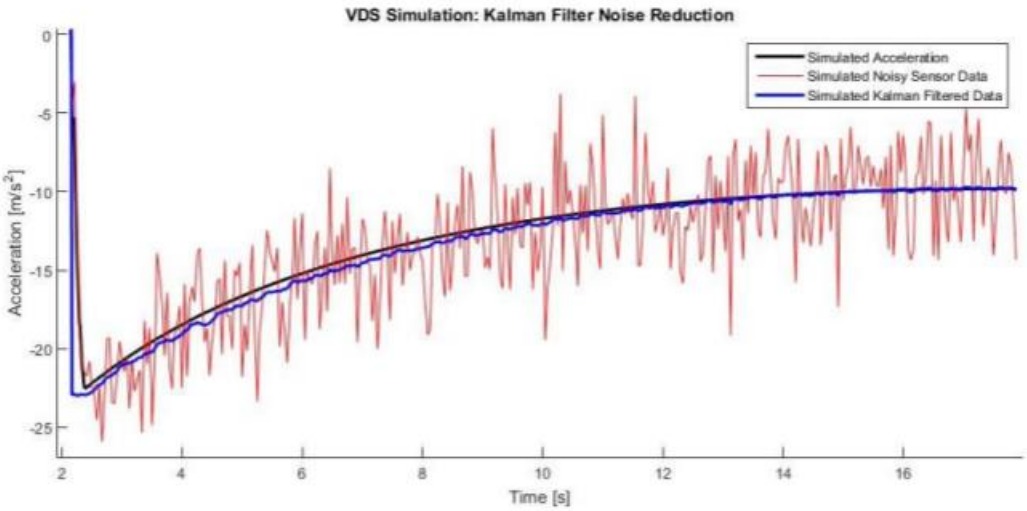


Figure 28: A representation of noisy data formerly mitigated by Kalman filtering

To account for these variances in data readings from the sensors utilized in V2, a Kalman filter was written with the expectation that more accurate data would be able to be derived from what was taken from test flight if there was excess noise in said data. The application of a Kalman algorithm proved to be challenging however partly successful in deriving the data required – the issue encountered was that this Kalman filter was not applied in real time, but on the data during after processing. Through further examination of this issue, it was determined that a sensor with a pre-programmed Kalman filter would be necessary to acquire the accuracy of data that is required to actuate the drag blades to achieve the target apogee.

3.4.8.1.3.2 *VN-100 IMU*

The VN-100 is an aerospace grade inertial measurement unit manufactured by Vectornav solutions. This sensor contains a gyroscopic range of ±2000 °/s, with an accelerometer limit of ±16g, and a pressure derived altimeter with limits of 10 to 1200 mbar. These limits meet and exceed the specifications for the VDS, and have been utilized in other rocketry applications. Additionally, the sensor has a built in kalman filter which would be used to mitigate the issues discussed in section 3.4.8.1.3.1. The VN-100 is widely documented and includes full software libraries that would be used to construct the software for the VDS if based around this sensors. The cost of the VN-100 ranges from $500-$1200 which also fits within the price requirements for this component of the system.

3.4.8.1.3.3 *BMP280/BNO055 Combination*

This configuration of sensors was utilized in the VDS V2, and have been proven to be semi-adequate for the needs of the system. The noise limits stated in the datasheets of these sensors are within the range of the needs of the VDS. Though the data processed while using these sensors is able to be filtered, when used in actual applications, this noise is above the threshold for optimal use of the VDS. The cost of this combination of sensors ranges from $25-$40 depending on the configuration and versioning, and software resources are widely available for both of the units.

3.4.8.1.3.4 *MSIMU3020*

The MSIMU is a navigational inertial measurement unit manufactured by Memsense. The most upgraded unit available of this sensor contains a gyroscopic range of ±1920 °/s, with an accelerometer limit of ±15g, and a magnetometer of ±1.5 gauss. This sensor does not contain an altimeter of any kind so if this sensor was to be chosen it would have to be supplemented with an independent altimeter unit. While the MUIMU lacks an altimeter, it has extremely low noise ranges and is the most precise of all other options considered. The cost of this sensor ranges from $1428-$1632.

3.4.8.1.3.5 *G-Wiz Flight Computer*

The G-wiz flight computer is not only an IMU but a system that has the potential to be the control center of the whole rocket; it is designed specifically for aerospace applications. The specific tolerances of the Accelerometer, magnetometer, and gyroscopes are not widely available however the tolerances for noise are comparatively low. The price range of this unit is from $150-$300 with dependence on the model.

3.4.8.1.3.6 *IMU Trade Study Results*

The options considered regarding the sensors are to deliver the most accurate data possible to the altitude correcting firmware so that the VDS can achieve peak accuracy and precision. The additional features considered were the quality and limits of the built-in magnetometers, gyroscopes, and accelerometers. Factors such as price of the unit and availability of documentation were also considered to aid integration into the system. The results of the trade study conducted to evaluate which IMU would best suit the precise needs of the VDS are shown in Table 12.

Inertial measurement Units									
Options:		VN-100		BMP 280/BNO055 combination		MS-IMU3020		G-wiz flight computer	
Mandatory requirements									
Contains an altitude derived pressure sensor with a tolerance of ±3mbar		Yes		Yes		No		Yes	
Categories	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Accelerometer	20.0%	9	1.8	8	1.6	7	1.4	5	1
Gyroscope	20.0%	10	2	9	1.8	9	1.8	7	1.4
Magnetometer	20.0%	8	1.6	10	2	6	1.2	4	0.8
Cost	15.0%	8	1.2	10	1.5	6	0.9	8	1.2
Software availability	10.0%	9	0.9	9	0.9	4	0.4	8	0.8
Kalman filter	10.0%	10	1	0	0	0	0	0	0
Total Score	**100%**		8.5		7.8		**5.7**		**5.2**

Table 12: IMU Kepner-Tregoe Trade Study.

As shown above, the Vectornav VN-100 was chosen, as it is manufactured for aerospace applications and is advertised to have a smaller margin of error than the current system during real life applications. Further experiments and tests are being designed in order to validate that upgrading to this IMU is in the best interest of the system.

The additional systems mentioned did not hold up to the standards set for this system, as they were both lacking in availability of documentation and in the range of data taking that the team anticipates will be necessary for the needs of the VDS. These units also did not contain a kalman filtering software.

3.4.8.2 Design Overview

The Variable Drag System has broken its design up into multiple subsystems. They are structured as follows;

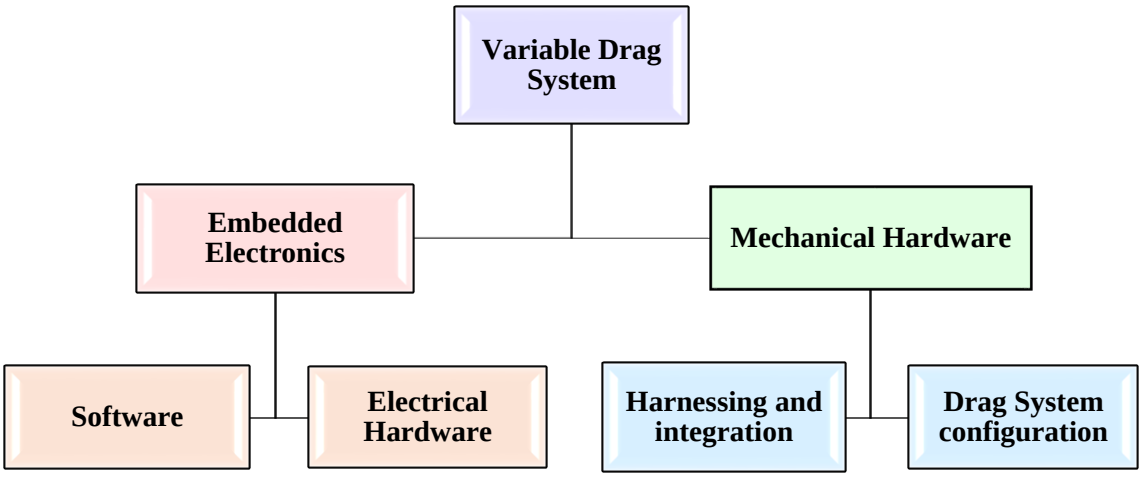


Figure 29: VDS design structure

3.4.8.2.1 Electrical Design

Several alternative electrical configurations were proposed as part of the redesign of the VDS V3. Trade-studies and test results have led to several reconsiderations such as the choice of main IMU and printed circuit board (PCB) design. The design decisions in the following sections have resulted in a new iteration of VDS electronics with the following main components:

- Teensy 3.6 Microcontroller
- LM7805 Linear Regulator
- BTN7960B Half Bridge Motor Drive (x2)
- VN-100 Inertial Measurement Unit
- Circular design printed Circuit board

These components are expected to produce accurate sensor readings coupled with regulated circuit power. The electronics will consist of two stacked PCBs, custom designed by the team. The top board will consist of a Teensy 3.6, the power conversion circuit, accessible system signals, power switches, power indication, and battery terminal connectors. The bottom board will contain system sensors, a motor-controlling H-bridge circuit, and the signals of the Teensy.

3.4.8.2.1.1 *Main Controller*

The Teensy 3.6 microcontroller provides the VDS with sufficient data processing and storage capabilities while minimizing power draw. The Teensy uses a 32-bit 180 MHz ARM Cortex-M4 processor and floating-point unit that can perform data computations from the sensors of the system. The Teensy has both non-volatile and volatile memory installed to take sensor values at a high enough rate. The Teensy contains 62 input and output pins that can provide Pulse Width Modulation (PWM), I2C communication, and UART serial communication. The input and output will be used to control motor actuation and data collection systems, as well as the telemetry system.

3.4.8.2.1.2 *Power Design*

The Power source is responsible for supporting the sensors, control circuits, microcontroller and the DC motor. The limitations when selecting the right power source include system operation time and component consumption. The run time of the electronics will ensure continuous operation during the full launch process. The current consumption of the system determines the battery to use. Table 13 below shows the resulting current consumption for the main components of the VDS:

Device	Current Consumption [mA]
Electronics power consumption	
VN-100 IMU	45
Half Bridge (x2)	0.3
Teensy 3.6 microprocessor	99.85
XBEE Telemetry system	1100
Total Electronics current	1245.15
with factor of safety = 2	**2490.3**
Motor Control Circuit Power Consumption	
Motor	**3000**

Table 13: Power Consumption of VDS electronics

The runtime of the system is directly dependent on the total current drawn by the components of the electronics. A factor of safety of 2 is added to account for potential jumps in current or environmental conditions that might vary the current draw on the system. This current draw will be adequately provided by the Teensy 3.6 microcontroller which draws 3.6 to 6 volts. To account for this power consumption while considering an appropriate amount of run time, the VDS will utilize two lithium polymer (Li-Po) batteries; one to support the power draw by the controls system, and one to support the power draw by the motor control circuit. Table 14 displays the options that could fit the VDS's power needs.

Current [mA with FOS of 2]	Battery rating (mAh)	Operation Time (hrs)
2490.3	180	0.072
	500	0.201
	1000	0.402
	1300	0.522
	2500	1.004
	4000	1.606
	5000	2.008

Table 14: Li-Po battery options for electronic support

This comparison of battery options also shows that a factor of safety of 2 can be achieved with a regulated supply. This regulation will be achieved using a 5V linear power regulator with an output current rated greater than .25 A. The decision to use this type of power regulator stems from the VDS V2 electronics configurations, where it was shown through multiple stages of testing to be an adequate solution. A large consideration of these power requirements are the size and weights of the batteries, as they could significantly alter the weight of the vehicle. Thorough tests will be conducted to decide which battery will be the most optimal regarding power consumption as well as sizing factors.

Wiring and Harnessing

The VDS avionics consist of two round PCBs, the bottom of which contains the control electronics and the top contains the regulation power circuit. The bottom section will house the sensor and drive the motor while the top PCB will regulate and supply the battery power, house the Teensy 3.6 microcontroller, and provide other external current signals. The structure of this design is laid out using the block diagram in Figure 30.

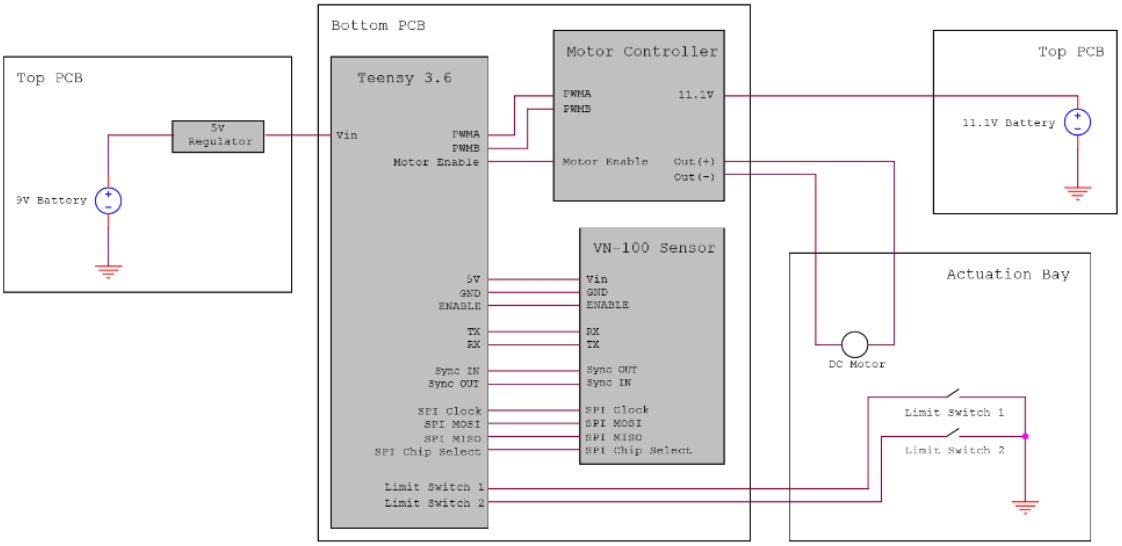


Figure 30: High Level Overview of VDS Electronics.

The avionics will be coupled in an adjacent bay to the motor blade configuration, separated by a bulk plate and connected through a D-sub style outlet within the plate. This bulk plate will be used in concurrence with the upgraded sensors to alleviate noise within the data collection system. The bulk plate will have an airtight fitting to prevent the pressure differential in the electronics bay caused by scooping of air into the vehicle by the blades.

3.4.8.2.2 Telemetry System

The team has proposed a newly designed telemetry system to allow the VDS to send state information to a custom ground station. This would allow for real time monitoring of vehicle status prior to launch and during flight. In VDS V2, data was stored on an SD card, then retrieved and analyzed post launch. The telemetry system would be able to detect potential issues pre-flight, deliver real time vehicle status, and would preserve data in the event of a failure.

3.4.8.2.2.1 *Radio Frequency Transmitter Trade Study*

Several RF transmitters were considered upon the design of the telemetry system. The factors outlined in FIGUREX will be tested during the subscale launch in order to determine if the chosen module meets these requirements set by the team.

The features which the team outlined as being crucial to this system were its transmit power, ease of integration and use, and data transmission rate. The transmit power is directly proportional to the range of transmission; a more powerful transmitter creates a larger distance from which it can receive and send signals. Additionally, the rate of data transmission affects the amount of data that is able to be sent, as well as the speed at which the data is sent to the ground station. Ease of integration refers to the ability for the transmitter to be programmed to receive signals directly from the VDS and to be able to be interfaced with the custom ground software that is being built

Telemetric Long Distance Radio (TLDR)									
Options:		P900		XBEE SX PRO		XBEE SX		RN2903A-I	
Mandatory requirements									
Range > 1 mile		Yes		Yes		Yes		Yes	
ISM Band		Yes		Yes		Yes		Yes	
Categories	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Transmit Power (0-10)	25.0%	10	2.5	10	2.5	6	1.5	7	1.75
Ease of integration (0-10)	25.0%	5	1.25	8	2	8	2	5	1.25
Data Rate (0 - 10)	20.0%	8	1.6	7	1.4	7	1.4	9	1.8
Sensitivity (0 - 10)	15.0%	6	0.9	5	0.75	5	0.75	9	1.35
Cost (0-10)	10.0%	4	0.4	2	0.2	7	0.7	10	1
Current Draw (0-10)	5.0%	2	0.1	4	0.2	8	0.4	6	0.3
Total Score			**6.75**		**7.05**		**3.25**		**4.45**

Table 7: Radio Transmitter Kepner-Tregoe trade study table.

Through trade study analysis, it was determined that the XBEE SX PRO would be the most optimal choice, as it's features align with the telemetry system's power, data rate, and usability needs. The XBEE SX Pro has a data transmission rate of 120 kb/s, with a line of site range of up to 65 miles in a rural area. The XBEE pro was also selected based on its receiver sensitivity of -103 dBm which is optimal in the case that the antenna experiences carbon fiber interference.

3.4.8.2.2.2 *Microcontroller*

The choice of microcontroller depends on the decision to integrate the system with the VDS. If the Telemetry system is decided to be a standalone system, then it could run off of its own Teensy 3.6. The Teensy 3.6 is powerful microcontroller with many features, as outlined in section 1.2.1.1.

To integrate the telemetry into the VDS electronics, considerations would have to be made to make sure that both system's power and data signal requirements are met. This could be done with a multicore microcontroller or microprocessor; however, the sensory data still must be communicated between systems. Further testing will be done to determine whether it is more efficient to integrate the telemetry system into the VDS or to leave it as a standalone system. This issue is referred to in more detail in section 3.4.8.2.1.1.

3.4.8.2.2.3 *Antenna*

Two distinct factors were taken into account when considering the integration of the antenna into the vehicle. Carbon fiber is opaque to RF transmission, so the antenna would have to be placed somewhere that the signal is able to propagate. Additionally, the antenna that is chosen must be omnidirectional so the RF receiver on the ground is able to maintain signal regardless of the direction the antenna is mounted. The frequency range of the RF module is SM 902 to 928 MHz, however the team will operate on a frequency which does not require an ARRL ham radio license to be acquired.

There are several proposed methods of mitigating this propagation during integration into the vehicle. Previous vehicle designs have included a transparent window to allow for access into the

VDS coupler, as well as to allow for an in-flight camera mounted to video record the ascent. This compartment could be utilized to mount an antenna.

Another potential mounting point are the fins of the rocket. This point would provide optimal line of sight between the antenna and the ground station as the vehicle ascends, as well as alleviating any connectivity issues that might incur from carbon fiber interference. This method could alter the aerodynamics of the vehicle and cause additional drag that would need to be taken into consideration. Additional testing will be done to determine the safest and most effective mounting point of the vehicle.

3.4.8.2.2.4 *Design Decision: Radio Operating Modes*

For some radio interface implementations, manufacturers provide a "transparent mode," in addition to a standard frame-based packet transmission operating mode, where the radio effectively acts as a direct serial connection between nodes, hiding the implementation of the underlying transmission protocol. Using radios in transparent mode eases integration with existing software. However, using frame-based packets for transmission offers device addressing and a simple way to communicate diagnostic information between nodes. It is the current decision of the team not to use radios in transparent mode. Instead, in favor of auditability and future extensibility, the team intends to use the standard frame-based transmission protocol.

3.4.8.2.2.5 *Stand Alone vs. Integrated with VDS*

The proposed telemetry system has the potential to be integrated into the VDS electronics or a standalone system. For this stand-alone system, the sensor data would be fed to a separate Teensy 3.6 from the main controller for the blades, where the microcontroller would parse and encode the data to prepare it for transmission, and sent the data to the ground station. This method of integration would alleviate some design concerns if sharing a microcontroller with another system, the main concern when integrating with the VDS would be degradation of the VDS software running on the microcontroller. The drawbacks for a standalone system are these additional requirements for powering a separate system as well as the integration involved in communication with the sensory data to an additional microcontroller.

A single microcontroller system would be ideal if the VDS maintained its level of precision. Powerful microcontrollers or microcomputers with multithreading capabilities are candidates for this. A multithreaded microcontroller or microcomputer would allow for each system to have its own dedicated thread and avoid depriving the VDS of resources.

3.4.8.2.2.6 *Testing*

A preliminary telemetry system will be tested during the required subscale launch. This test will include a teensy 3.6, the XBEE RF transmitter and a pressure sensor. The purpose of this test will be to evaluate the actual range of transmission between the rocket and the ground station, as well as to give insight into the data transmission rate that will be required for future full-scale launches.

3.4.8.3 Control Theory

The VDS performs an autonomous decision-making process through the control scheme implemented within its software. These controls are active during flight to achieve the overall goal

of an exact apogee altitude. It does this by continually comparing its real time vertical velocity to a predetermined ideal flight path and correcting for any deviations. This process is based on several key equations.

3.4.8.3.1 Applicable equations

The equations utilized in the VDS controls calculations have been derived from the coast phase deceleration equation. These applicable equations are used to model the coast phase of a launch to create the control scheme used to actuate the system; they have been verified experimentally and through using simulations.

$$\boldsymbol{a = -g - cv^2} \tag{6}$$

Where a is representative of the vertical component of acceleration, v represents the vertical component of velocity, and the constant c represents the vehicle's unique drag characteristics. The unique drag characteristic is calculated using

$$\boldsymbol{c = \frac{C_d \rho A}{2m}} \tag{7}$$

where A represents the cross-sectional area of the vehicle, C_d is the coefficient of drag of the vehicle, the density of air represented by ρ is taken to be a constant 1.225 [kg/m^3] (despite that it changes with altitude) and m is the mass of the vehicle after burn. The changes in air density were taken to be negligible and ignored for computational efficiency.

An alternative form of the coast phase deceleration equation known as the velocity WRT height form is shown below.

$$\boldsymbol{v(h) = -e^{-hc}\sqrt{\frac{g}{c}e^{2K_2C}} - e^{-2hc}} \tag{8}$$

3.4.8.3.2 Setpoint Path

The setpoint path (SPP) equation is used to determine the characteristics of actuation with a dependence on the altitude of the vehicle - It is an ideal predetermined scenario of the trajectory of the vehicle that sets the apogee at the desired value of 5,280 ft. The VDS control scheme references the SPP in order to correct the vehicle and slow it to the closest possible path to optimal.

The SPP is an equation of velocity as a function of altitude, $V_{spp}(h)$. It is derived from the coast phase deceleration equation and has an altitude axis (h) intercept equal to 5,280 ft. The SPP is given below in Equation (4).

$$V_{spp} = \begin{cases} -e^{h\bar{c}}\sqrt{\frac{g}{\bar{c}}e^{2K_2\bar{c}}} - e^{-2h\bar{c}}\left[\frac{m}{s}\right] & , v > 125\left[\frac{m}{s}\right] \\ -e^{-hc_{min}}\sqrt{\frac{g}{c_{min}}e^{2K_2C_{min}}} - e^{-2hc_{min}}\left[\frac{m}{s}\right] & , v < 125\left[\frac{m}{s}\right] \\ 0\left[\frac{m}{s}\right] & , h > 1609\ [m] \end{cases} \quad \textbf{(9)}$$

where $\bar{c}$ is the average drag characteristics constant given by

$$\bar{c} = \frac{\rho(A_r + A_{r+b})(C_r + C_{r+b)}}{8m} \quad \textbf{(10)}$$

where Ar represents the cross-sectional area of the vehicle, Ar+b represents the cross-sectional area of the rocket and brakes, Cr represents the coefficient of drag of the vehicle, and Cr+b is the coefficient of drag of the rocket and brakes. The minimum drag characteristics constant, c_{min}, is given by

$$c_{min} = \frac{\rho A_r C_r}{2m} \quad \textbf{(11)}$$

The equation is represented as a piecewise function, displayed above in three parts. The first increment, where h >125 [m], is calculated using an average drag characteristic constant to facilitate a smooth transition to the minimum drag characteristic path. The second part follows the minimum drag characteristic path to the target altitude. The third part where V_{spp}= 0 for h >1609 [m] (5,280 ft.) ensures that the VDS will fully deploy the brakes if it surpasses its target altitude.

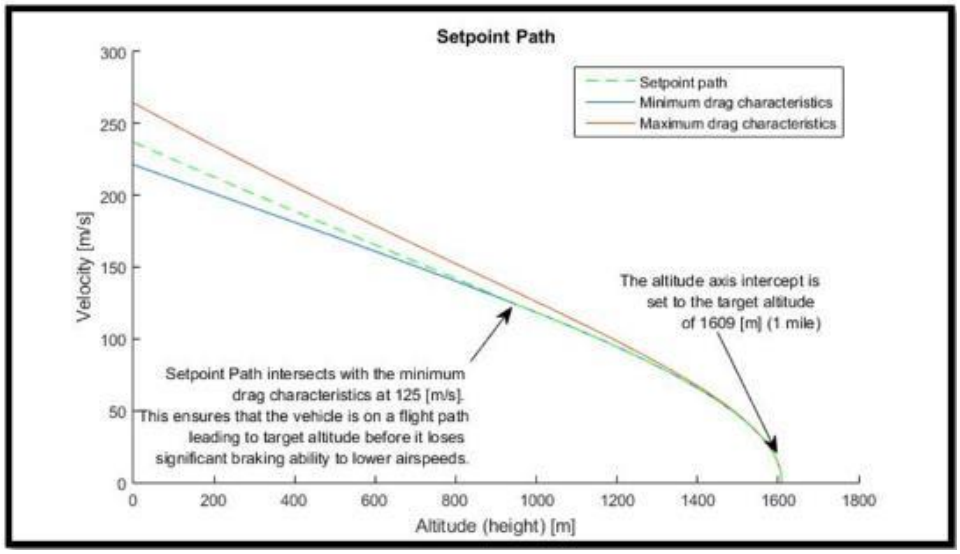


Figure 31: Setpoint path optimal trajectory

This piecewise SPP has been developed as an alternative to an originally not piecewise SPP that simply had average drag characteristics throughout. This SPP was found in both simulation and test launches to lead to an achieved apogee that was consistently higher than the target. This is likely due to the fact that the VDS loses braking ability as it slows down. This is remedied in the new SPP that puts the vehicle on a path to the target altitude at 125 m/s, before it loses significant braking ability

3.4.8.4 Mechanical Design

The goal for the VDS V3 mechanical design is to integrate the system into the launch vehicle efficiently while maintaining the functionality provided by the VDS V2. The system was has been configured with a refined gear design, and opportunities for improvements to the electronics integration have been identified.

3.4.8.4.1 Variable Drag System Flight Configuration

The VDS is designed to radially actuate three drag inducing blades into the airflow surrounding the launch vehicle with the purpose of ensuring that the launch vehicle reaches an apogee of 5280 ft. +/- 23ft. The drag blades will be radially extended into the airflow using a single central spur gear. All mechanical and electrical components of the VDS are designed to fit within a single coupler and be inserted into the launch vehicle as a single entity. A rendering of the current configuration of the VDS is shown below in Figure 1.

Figure 32: Variable Drag System.

3.4.8.4.1.1 *Actuation*

The VDS will radially actuate three drag-inducing blades from the launch vehicle's airframe using a single central spur gear press fit to the shaft of a motor. Each drag blade has a set of radial gear teeth designed to mesh with the central spur gear. The blades will rotate about a single 0.125 in. Dowel pin projecting approximately half of the surface area of each blade perpendicularly into the oncoming airflow. A rendering of the gear meshing is shown in Figure 33.

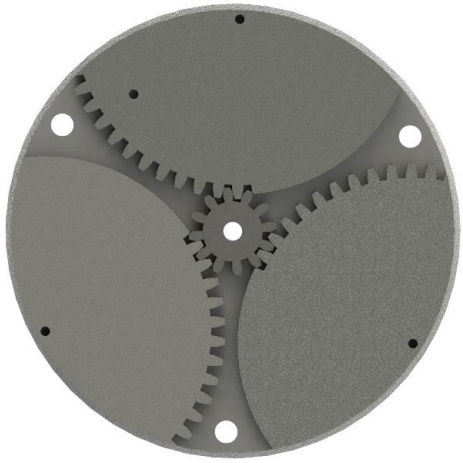


Figure 33: Rendering of VDS gear meshing.

Using an actuation method that extends the drag blades perpendicularly to the airflow reduces the motor torque requirements, thus decreasing the overall mass of the system. The final actuation design is optimized with respect to mass and the total projected area of the blades, and allows for

continuous and simultaneous control of all drag blades using a single motor. A rendering of the VDS before and after actuation is shown in Figure 34.

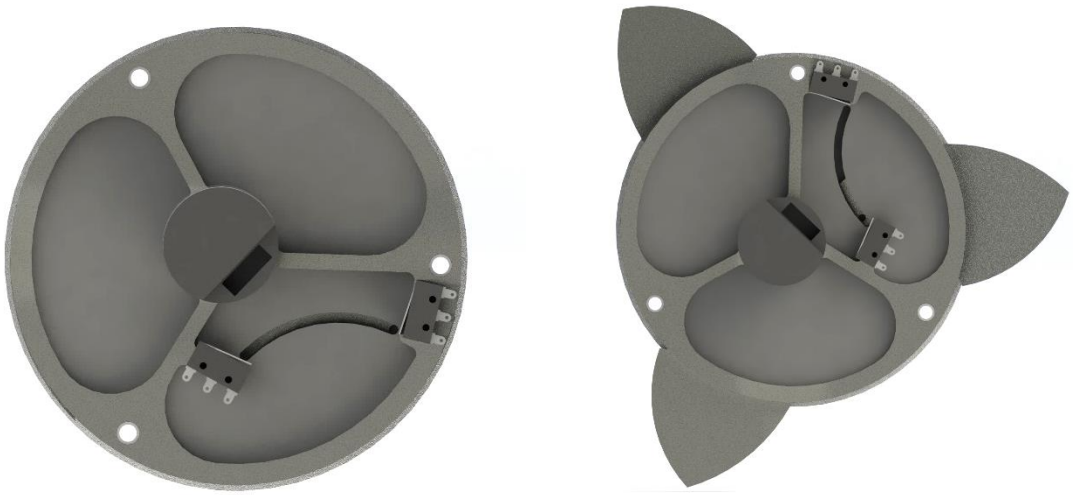


Figure 34: VDS blade actuation.

3.4.8.4.1.2 *Motor Selection Trade Study*

Three motor types were considered as part of a Kepner-Tregoe trade study to determine the optimal actuating device for the VDS. The motors considered were a stepper motor, a DC motor, and a servo motor. Each motor designation is described in the following sections.

3.4.8.4.1.3 *Stepper Motor*

A stepper motor was considered for the VDS V3 as it would provide increased actuation speed, stall torque, and precision of the drag blade actuation. A disadvantage to using a stepper motor as the actuating device for the VDS V3, is that it would substantially increase the overall mass and volume of the system.

Servo Motor

A servo motor was considered for the VDS V3 as it would decrease the overall mass and volume of the system. The disadvantages of using a servo motor in this application include it's low actuation speed and stall torque.

DC Motor

A DC Motor was used as the actuating device for the VDS V2. A DC motor can provide the VDS with an adequate actuation speed and stall torque along with an acceptable mass and volume.

Results

The results of the motor selection Kepner Tregoe trade study are shown below in Table 15.

Actuation Device			
Options	Stepper Motor	DC Motor	Servo Motor
Mandatory Requirements			

Able to provide 358 oz-in. of torque		YES		YES		NO	
Wants	Weights	Value	Score	Value	Score	Value	Score
Stall Torque	30.00%	10	3	7	2.1	1	0.3
RPM	25.00%	10	2.5	5	1.25	2	0.5
Mass	20.00%	1	0.2	8	1.6	10	2
Price	10.00%	8	0.8	7	0.7	5	0.5
Volume	15.00%	3	0.45	8	1.2	10	1.5
Total Score		6.5		6.85		3.3	

Table 15: Actuation device Kepner Tregoe Trade Study table.

The DC motor was determined to be the optimal actuation device for the VDS V3. The DC motor can provide adequate torque, while maintaining a relatively low mass and volume. Research is currently being conducted into a particular motor that would be ideal for the VDS V3.

3.4.8.4.1.4 *Variable Drag System Mechanical Components*

The VDS's three drag blades will be cut from 0.125 in. thick 6061-T6 Aluminum using a Maxiem 450 Water Jet. Each blade will include a set of radial gear teeth designed to mesh with a 0.125 in. mild carbon steel central spur gear. The design of the drag blades optimizes the projected area for a radially actuated braking system, and the flat plate shape of the blades provide a high coefficient of drag

A 0.125 in. Delrin plate will be placed on both the top and bottom side of the blade configuration. The Delrin plates provide a low coefficient of friction with the aluminum drag blades. This reduces the friction force that the motor must overcome to actuate the blades, allowing for faster actuation speed, lower overall mass, and less power consumption by the system. Three custom machined aluminum spacers will be placed between the Delrin plates to ensure proper alignment of the system and prevent overtightening on the drag blades.

The drag blade configuration and Delrin bearing plates will sit between two 0.125 in. 6061-T6 aluminum support plates. The support plates take much of the drag force acting on the VDS. Aluminum was chosen as the material for both the drag blades and the support plates because of its light weight, machinability, and rigidity. A rendering of the support plates is shown in Figure 35.

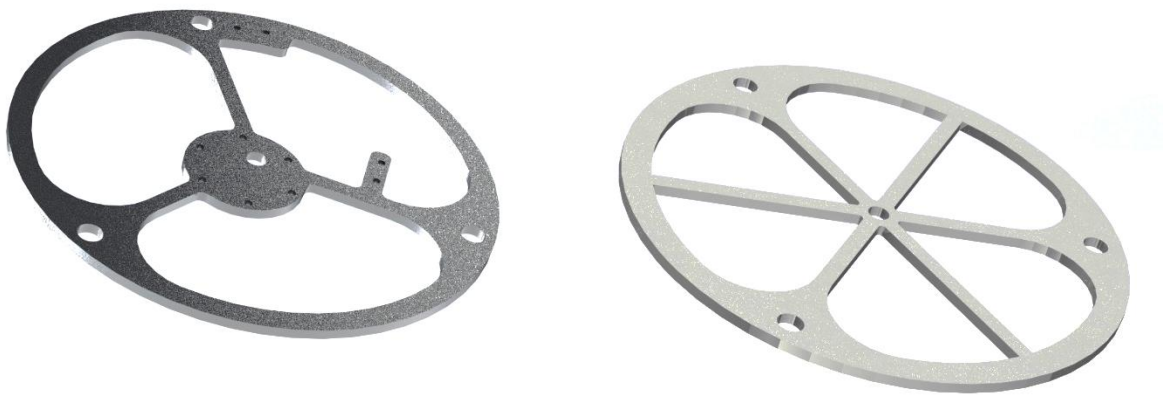


Figure 35: VDS top and bottom 6061-T6 Aluminum support plates.

3.4.8.4.1.5 *Analysis*

To ensure that the design will be robust enough to withstand the maximum in flight forces with a minimum acceptable factor of safety of 2.0, the mechanical components of the VDS V3 was verified using ANSYS Workbench 17. The maximum drag force exerted on the blades during test flights with the VDS V2 was approximately 20 lbs. As the same blade design will be implemented in the VDS V3, the same forces were used in the Finite Element Analysis simulations. Due to uncertainties in the drag force calculation and possible changes in the maximum velocity of the launch vehicle, each drag blade of the VDS was tested to withstand the full drag force multiplied by a factor of two. A minimum factor of safety of 6.51 was determined for each drag blade. The results of the analysis conducted on the drag blades are shown below in Figure 36 and Figure 37.

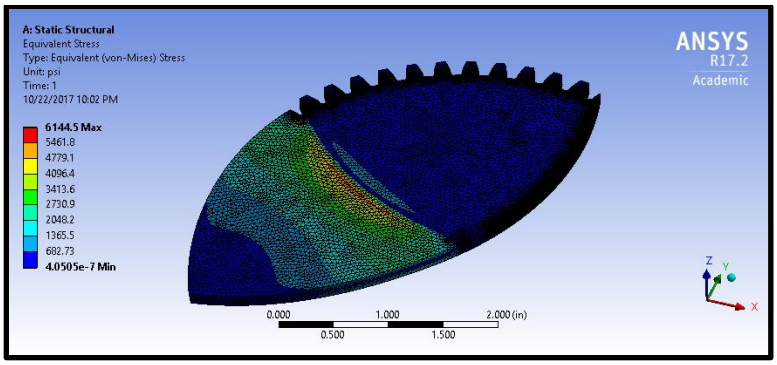


Figure 36: Drag blade stress plot at maximum braking force.

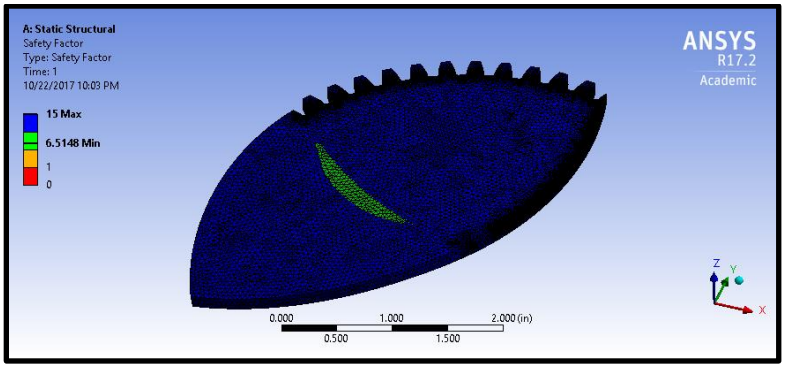


Figure 37: Drag blade factor of safety plot at maximum braking force.

FEA was done on the gear teeth of both the drag blades and the central spur gear to ensure they would perform safely under loading equivalent to the maximum stall torque of the NeveRest 40 DC motor applied between the central gear and a single drag blade. The results of the study showed a minimum factor of safety of 1.68 for the gear design on the drag blade and 1.73 for the gear design on the central spur gear. The results of the FEA are shown below in Figure 38, Figure 39, Figure 40, and Figure 41.

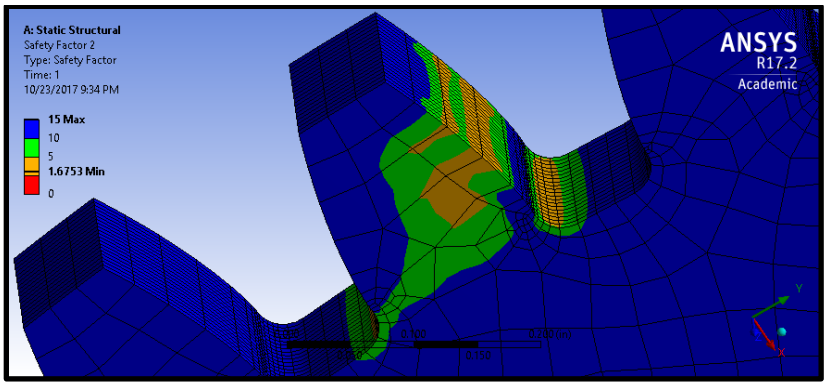


Figure 38: Drag blade gear tooth factor of safety plot.

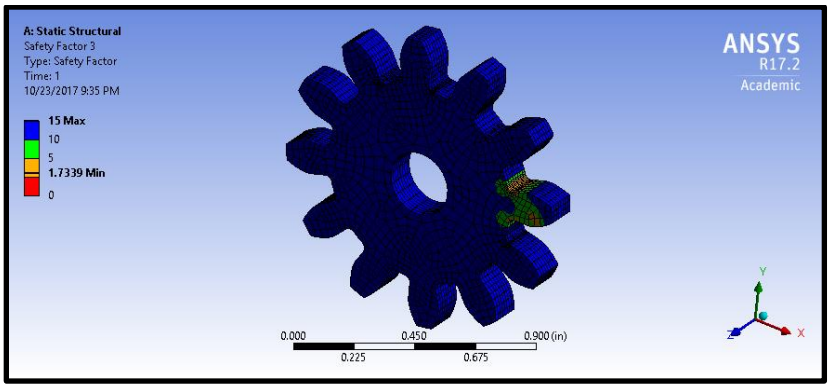


Figure 39: Central spur gear tooth factor of safety plot.

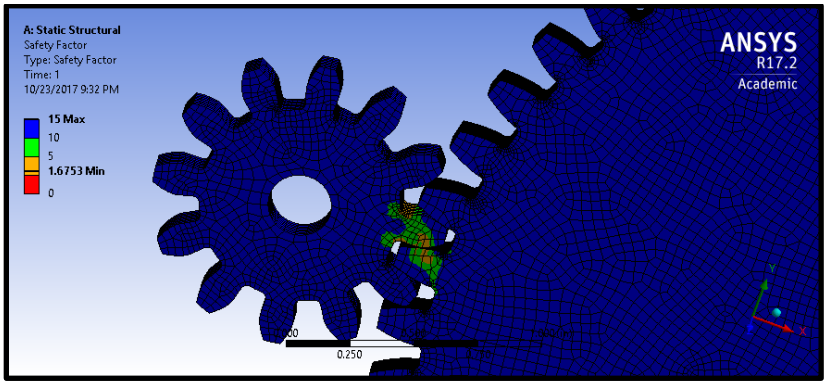


Figure 40: Overall factor of safety of VDS gear assembly.

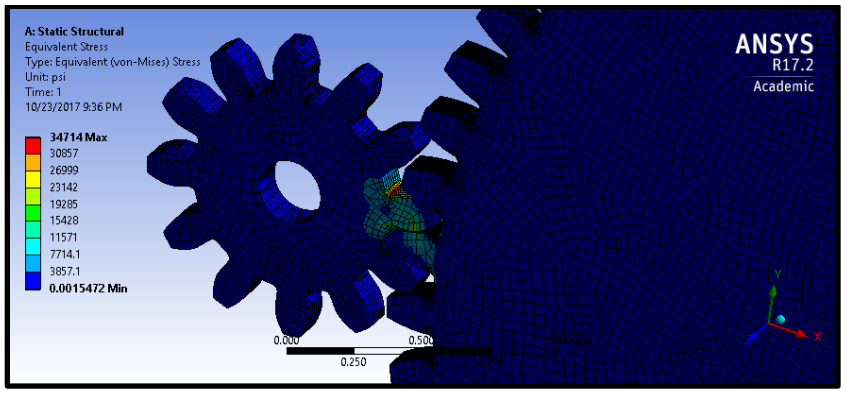


Figure 41: Maximum stress plot of VDS gear assembly.

The FEA results show that the critical point for yielding failure is at the root of the gear teeth. A fillet was included at the root of each set of gear teeth to alleviate the stress concentration. The results show that the gear design will perform with an acceptable factor of safety under three times the maximum torque.

3.4.8.4.2 Harnessing and Integration

One of the primary challenges identified with the VDS V2 was the ability to efficiently assemble the system and integrate it with the launch vehicle. A detailed description of how the flaws in the VDS V2 integration scheme will be mitigated in the VDS V3 design is shown in Table 16.

VDS V2 Integration Flaws	VDS V.3 Improvement Plan
The wiring to the VDS V2 limit switches on the top aluminum support plate was hot glued to the limit switch connectors. This was necessary because the limit switch connectors were altered to fit within the VDS coupler.	The limit switches will be repositioned on the VDS V3 top support plate to ensure there is enough room for all wiring to be securely soldered to the appropriate connectors.
The wiring from the VDS V2 limit switches and NeveRest 40 DC motor was not cut to an optimal length. This led to crowding and disorganization in the bottom half of the VDS coupler. The wiring was taped to the motor to limit the disorganization within the coupler.	The VDS V3 limit switch and motor wiring will be carefully measured and cut to proper length before being soldered to the limit switches and motor. The wiring will be routed through a removable wire strap rather than being taped to the motor casing.
All wiring from the VDS V2 limit switches and motor was routed to a single connector permanently mounted in the VDS Coupler mid bulkplate. On the top side of the bulk plate, wiring was routed around the outside of the VDS V2 custom 3D printed sled to a rectangular PCB harnessed within the sled.	The VDS V3 will route all wired connections through a similar connector permanently fixed to the mid bulkplate within the VDS coupler. All wires will be routed to circular PCB's harnessed within a custom 3D printed avionics sled. Circular PCB's will allow for a simpler avionics sled design.
To provide an external power supply to the VDS V2, a connector was wired out of the VDS avionics sled to a camera window in the vehicle's airframe. This design was not	A charging port will be mounted through the vehicle's airframe directly outside of the VDS avionics sled to provide constant access to the VDS electronics. This will ensure that the

adequate to ensure that the avionics could maintain a charge while the vehicle was on the launch pad.	VDS electronics will not lose power on the launch pad.

Table 16: VDS integration plan

The VDS mechanical and electrical components will all be secured within a single 12 in. carbon fiber coupler in the launch vehicle. The mechanical components will be securely fastened to #10-32 Aluminum all-thread rods on the bottom half of the coupler. The electronics will be housed within a custom designed 3D printed sled and secured to the launch vehicle using #10-32 Aluminum all-thread rods within the top half of the VDS coupler. A ½" wooden bulk plate will separate the electronics from the mechanical VDS.

3.5 Recovery Subsystem

3.5.1 Overview

A separation event at apogee will decouple the vehicle into two independent sections: the payload segment, and the booster segment. Both the nosecone and the coupler will separate from their respective airframe counterparts during main deployment and be recovered under the drogue parachutes as their own independent sections. These four independent segments are displayed below in Figure 42.

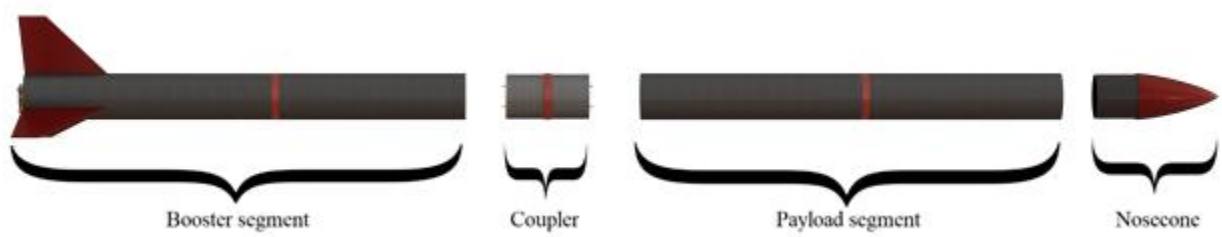


Figure 42: Independent sections of the launch vehicle upon descent

3.5.2 Release Device Trade Study

Due to the need to split the launch vehicle into two halves for the payload to deploy from the payload bay, the use of a dual deployment recovery method will be necessary. This will feature a recovery bay with a drogue and main parachute connected in series by a release device. A trade study was performed on two release devices, the Advanced Retention and Release Device (ARRD), and the Tender Descender. The release device functions as a connection point for the drogue parachute directly to the launch vehicle's bulkplate, as seen below in Figure 43. This retention passively tethers the drogue to the main deployment bag.

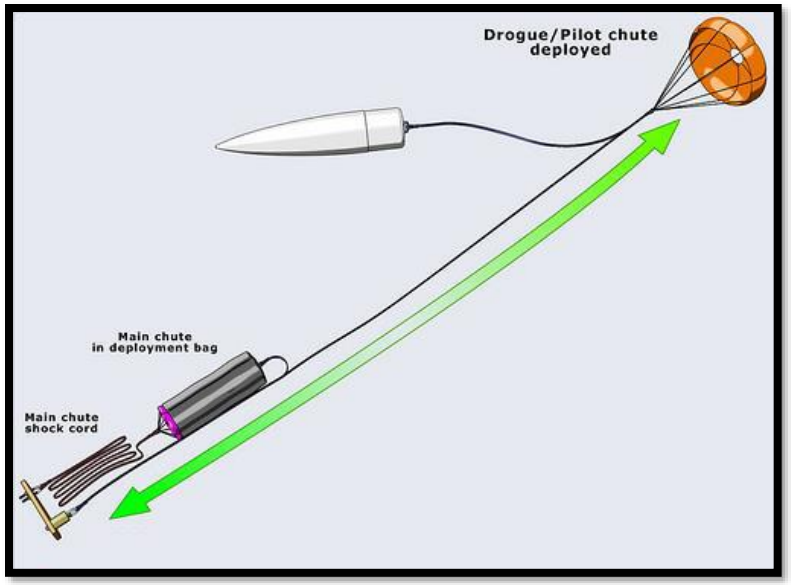


Figure 43: Payload drogue steady state

The release device lets the drogue free during the main event. This moves tension to the tether attached to the main deployment bag and allows the drogue to now act as a pilot parachute for the bag. The drogue pulls the deployment bag from the bay, and at line stretch, pulls the bag from the main parachute and frees the nosecone to descend under the retired drogue. This is detailed below in Figure 44.

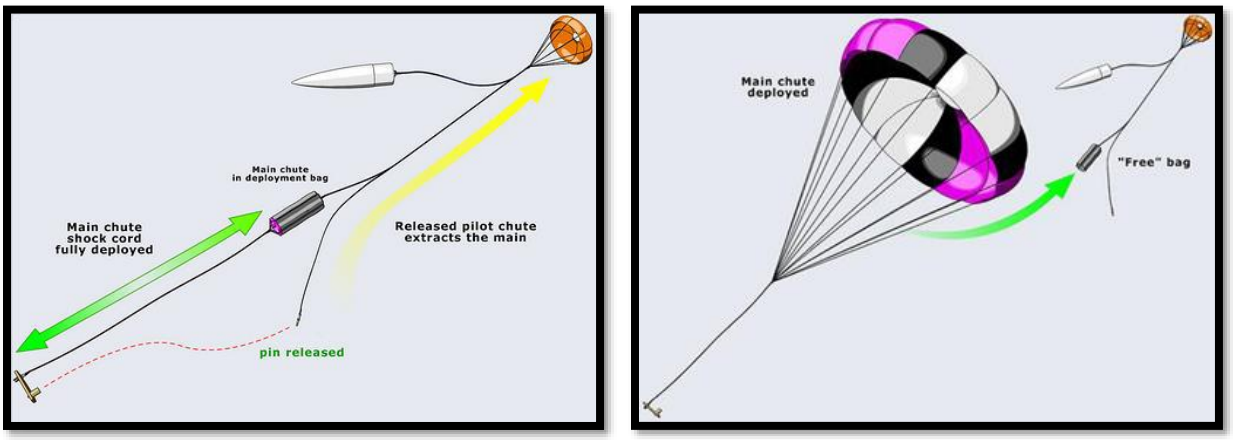


Figure 44: Release device engagement and main deployment

3.5.2.1 Advanced Retention and Release Device (ARRD)

The first release device considered by the team for use is the Advanced Retention and Release Device (ARRD). The ARRD, shown in Figure 45, is an assembly purchased from RATTworks that functions as a load bearing connection point for the drogue parachute directly to the launch vehicle's bulkplate. The ARRD is comprised of a base with attached threads for inserting into the bulkplate, a washer and hex screw, five ball bearings, anodized aluminum body, a piston, and toggle assembly.

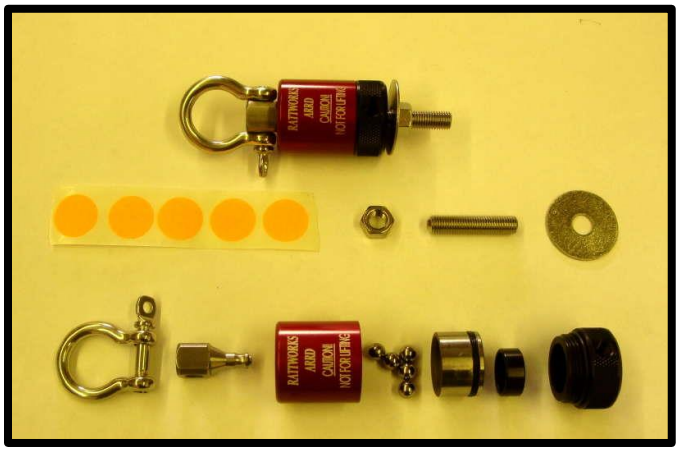


Figure 45: ARRD assembly

The ignition of a black powder charge forces the piston inside to release the attachment point out of the ARRD. This will ensure that the drogue does not act as the main pilot parachute until the main deployment event. The ARRD has been load tested to 2,000 lbs. The ARRD is easily reusable due to all components being recovered after flight. A disadvantage of the ARRD is that it has a possibility of releasing at apogee if the toggle is not properly secured into the assembly and tested before flight.

3.5.2.2 Tender Descender

The second release device considered is the L2 Tender Descender. The Tender Descender, shown in Figure 46, is a mechanism that separates two quick links at apogee. The Tender Descender is comprised of a housing with charge well, and a link retainer assembly

Figure 46: Tender Descender assembly

The Tender Descender is a simple mechanism and easy to assemble. After ignition of a black powder charge placed within the housing, the link retainer assembly will be ejected. This allows the quick links to separate from the mechanism. A cord prevents the link retainer assembly from ejecting from the launch vehicle. This allows for all parts of the Tender Descender to be reused. The Tender Descender has a maximum shock load of 2,000 lbs. Disadvantages to using the Tender Descender include that it has a possibility of impacting the airframe with the release pin during separations, and that the ejection charge could damage recovery components.

3.5.2.3 Release Device Trade Study Results

The two release devices were compared in the Kepner-Tregoe trade study shown below in Table 17.

Release device					
Options		ARRD		Tender Descender	
Mandatory requirements					
Provides retention until activated		Yes		Yes	
Wants (0-10)	**Weights**	**Value**	**Score**	**Value**	**Score**
Ease of Use	40%	7	2.8	6	2.4
Reliability	50%	8	4	8	4
Simplicity	10%	6	0.6	8	0.8
Total score		**7.4**		**7.2**	

Table 17: Release device Kepner Tregoe Trade Study

The ARRD is more easily integrated with the launch vehicle, but is more complex than the Tender Descender. The ARRD's parts are reusable, unlike the tender descender which can throw it's locking mechanism into the airframe or out of the vehicle if not tethered properly. For these

reasons, the ARRD has been chosen for use on the launch vehicle. The recovery subsystem will be further designed with the intent of using an ARRD.

3.5.3 Payload Bay Separation Method Trade Study

During the separation of the payload bay and booster recovery bay, it is imperative that the payload stays protected from the concussive properties of separation charges, as well as the black powder residue. To reduce the likelihood of damaging the payload sensors, two precautionary methods were chosen.

3.5.3.1 Black Powder Charge Wells

The black powder charge well method would encase black powder charges in canisters that direct the contents away from the payload equipment during ignition. This method will create a pressure change, but will keep the payload clear from residue. Multiple vent hole designs could be implemented, as shown below in Figure 47.

Figure 47: Charge well variations

The charge well will be attached to the coupler below the payload bay by a bolt through the lower cap which is held in place by a nut and washer on the other side of the bulkplate.

3.5.3.2 CO_2 Separation Method

The CO_2 separation method uses a pair of redundant cold gas canisters that expel compressed CO_2 into the bay for separation. The appeal for this separation method is that no black powder charge would be inside the recovery bay, thus eliminating any residue or fire. Disadvantages to using this system include the large amount of coupler space it would consume, and its high mass. A CO_2 ejection system is shown below in Figure 48.

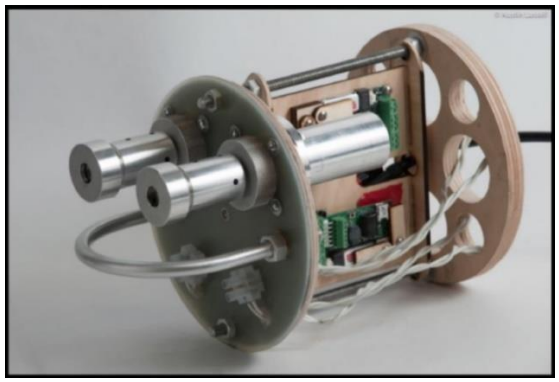


Figure 48: CO_2 separation method

3.5.3.3 Payload Bay Separation Method Trade Study Results

The payload bay separation method was determined using the Kepner-Tregoe trade study shown below in Table 18.

Payload Bay Separation Method					
Options		CO_2		Black Powder Charge Well	
Mandatory requirements					
Produces > 6 PSI		Yes		Yes	
Wants (0-10)	**Weights**	**Value**	**Score**	**Value**	**Score**
Cleanliness	40%	10	4	9	3.6
Reliability	30%	8	2.4	9	2.7
Simplicity	30%	5	1.5	9	2.7
Total score		**7.9**		**9**	

Table 18: Seperation device Kepner Tregoe trade study table.

The black powder charge wells were chosen for their effective simplicity, easy installation, and low mass. The recovery subsystem will be designed further with the intent of using black powder charge wells in the payload bay.

3.5.4 Parachute Design Trade Study

Six parachute designs were considered for the main and drogue parachute design for the launch vehicle. These parachute designs are detailed in the following sections.

3.5.4.1 Cruciform

The cruciform parachute consists of two rectangular layers laid in a cross shape. It can be optimized for efficiency or stability, at the cost of the other. It also tends to rotate. It the simplest to design, manufacture, and test. It is deployed using standard deployment techniques. It is shown below in Figure 49.

Figure 49: Cruciform parachute

3.5.4.2 Annular

The annular parachute is ring shaped. It is a conical parachute with a large inner vent with lines attached to outer suspensions. It has highest drag coefficient to canopy surface area, moderately high stability, and is moderately complex to manufacture. It is deployed using standard deployment techniques. It is shown below in Figure 50.

Figure 50: Annular parachute

3.5.4.3 Toroidal

The toroidal parachute is Torus shaped. It is a quarter spherical parachute with a large inner vent with lines attached to outer suspensions. It has a high drag coefficient to canopy surface area coupled with moderately high stability. It is of moderately high complexity to manufacture. It is deployed using standard deployment techniques. It is shown below in Figure 51.

Figure 51: Toroidal parachute

3.5.4.4 Flat Hexagonal

The flat hexagonal parachute is a conical parachute that approximates a flat circular parachute. It has moderate efficiency and stability. This parachute is simple to design, manufacture, and maintain. It is very reliable and easy to test. It is deployed using standard techniques. It is shown below in Figure 52.

Figure 52: Flat hexagonal parachute

3.5.4.5 Hemispherical

The hemispherical parachute is a half-sphere shaped parachute with a center vent. It has moderate efficiency and stability, and tends to rotate. It is difficult to manufacture, and is deployed using standard deployment techniques. It is shown below in Figure 53.

Figure 53: Hemisphereical parachute

3.5.4.6 Vortex Ring

The vortex ring parachute consists of four triangular gores with a leading and trailing edge to create a stabilizing rotary motion. It has a high drag coefficient coupled with high stability. It is highly complex to manufacture and is inconsistent in opening characteristics, making it somewhat difficult to test. It is shown below in Figure 54.

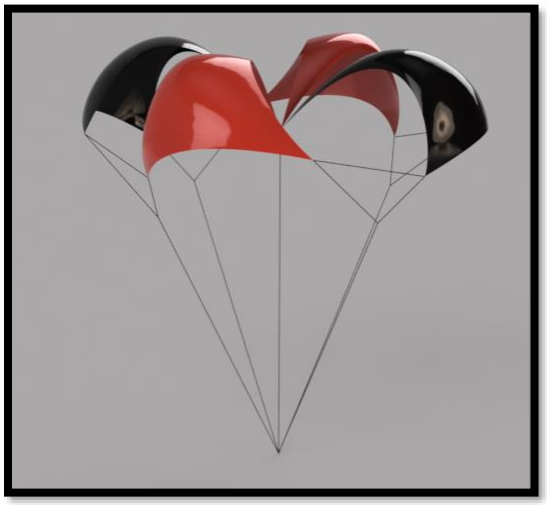


Figure 54: Votex ring parachute

3.5.4.7 Drogue Parachute Trade Study Results

The Drogue parachute for the launch vehicle was determined by Kepner-Tregoe trade studies comparing different parachute types. The drogue parachutes considered were weighted accordingly and evaluated in a trade study as seen in Table 19 below. The two lowest scoring parachutes were removed for simplicity. For the drag coefficient and stability criteria, a baseline was established by rating the cruciform and toroidal “10” in the respective categories, as it has the highest drag coefficient or lowest angle of oscillation. Values were then derived for the remaining

parachutes by calculating their characteristics as a percentage of the optimal values offered by the two.

Payload and Booster Drogue									
Options		Annular		Toroidal		Flat Hexagonal		Cruciform	
Mandatory requirements									
Oscillation < 10 degrees		Yes		Yes		No		Yes	
Wants (0-10)	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Efficiency (drag coefficient)	10%	5	0.5	8	0.8	0.4	0.04	3	0.3
Stability (angle of oscillation)	30%	3	0.9	3	0.9	1	0.3	7	2.1
Ease of Design	20%	7	1.4	6	1.2	10	2	9	1.8
Ease of Manufacturing	20%	7	1.4	6	1.2	10	2	9	1.8
Deployment Simplicity	15%	7	1.05	7	1.05	10	1.5	10	1.5
Testability	5%	7	0.35	7	0.35	10	0.5	10	0.5
Total score		**5.6**		**5.5**		**6.34**		**8**	

Table 19: Drogue parachute Kepner Tregoe trade study table.

The Cruciform design has been chosen for excelling in almost every category of the trade study that was conducted between the different designs. its simplicity in the manufacturing process provides more time to focus on manufacturing the larger more complex main parachutes. The low angle of oscillation will ensure a steady decent from apogee as well. The Simplicity of its inflation ensures that the drogue will deploy. This is especially important due to its secondary function as the main parachute for the nosecone and coupler once separated.

3.5.4.8 Main Parachute Trade Study Results

The Main parachute for the launch vehicle was determined by Kepner-Tregoe trade studies comparing different parachute types. The main parachutes considered were weighted accordingly and evaluated in a trade study as seen in Table 19 below. The two lowest scoring parachutes were removed for simplicity.

Payload and Booster Main									
Options		Annular		Toroidal		Vortex Ring		Cruciform	
Mandatory requirements									
Drag Coefficient > 0.8		Yes		Yes		Yes		No	
Wants (0-10)	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Efficiency (drag coefficient)	40%	5	2	8	3.2	10	4	3	1.2

Stability (angle of oscillation)	10%	3	0.3	3	0.3	10	1	7	0.7
Ease of Design	15%	7	1.05	6	0.9	2	0.3	9	1.35
Ease of Manufacturing	10%	9	0.9	7	0.7	2	0.2	8	0.8
Deployment Simplicity	20%	7	1.4	7	1.4	3	0.6	10	2
Testability	5%	7	0.35	7	0.35	2	0.1	9	0.45
Total score		**6**		**6.85**		**6.2**		**6.5**	

Table 20: Main parachute trade study

A toroidal design has been chosen for the main parachute of each the payload and booster segments for excelling in the trade study. The toroidal features a high drag coefficient for non-rotating parachutes and a simple yet weight and volume saving design when compared to the hemispherical design. This design is also packed and deployed simply, especially when compared to a vortex ring parachute, ensuring a reliable recovery system.

3.5.5 Staging Procedure

The launch vehicle separations will be staged in such a way that allows for each parachute to be deployed at safe altitudes from each other while also obeying the recovery distance requirements (RDR) outlined in SOW 3.9 of section 6.1.1. The recovery events are detailed in Table 21.

Event	Altitude	Phase	Description
1	Apogee	Booster Drogue Event	Launch vehicle separates into two independent sections at apogee. Booster drogue is pulled from the coupler where it is stored.
2	Apogee +2 sec. Delay	Nosecone Drogue Event	After a two second delay, the nosecone separates deploying the drogue for the payload section of the launch vehicle. The nosecone remains tethered to the payload recovery bay.
3	600 ft.	Payload Main Deployment	The main parachute is deployed from the payload recovery bay by a retention/release device, where the drogue acts as a pilot parachute for the main parachute's deployment bag.
3.1	600 ft.	Nosecone Untether	The activation of the retention/release device untethers the nosecone from the payload recovery bay and becomes an independent section under the drogue parachute.
4	500 ft.	Booster Main Deployment	The coupler above the booster recovery bay will be separated where the drogue acts as a pilot parachute and pulls the main parachute's deployment bag from the booster recovery bay.
4.1	500 ft.	Coupler Untether	The release of the coupler untethers it from the main body of the launch vehicle and it becomes an independent section under the drogue parachute.

Table 21: Recovery procedure

The sequence of events described in Table 21 are illustrated below in Figure 55, Figure 56, and Figure 57.

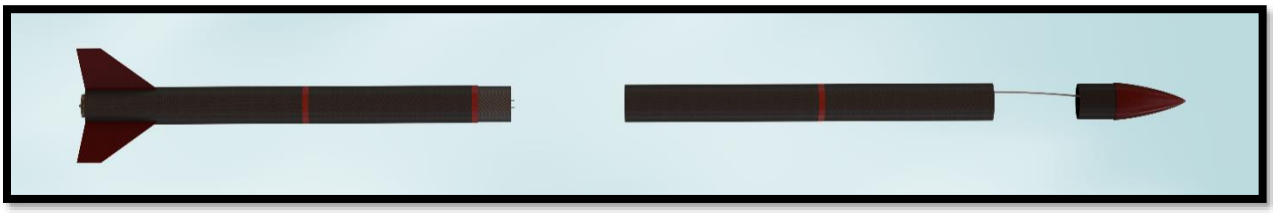


Figure 55: Launch vehicle separation at apogee.

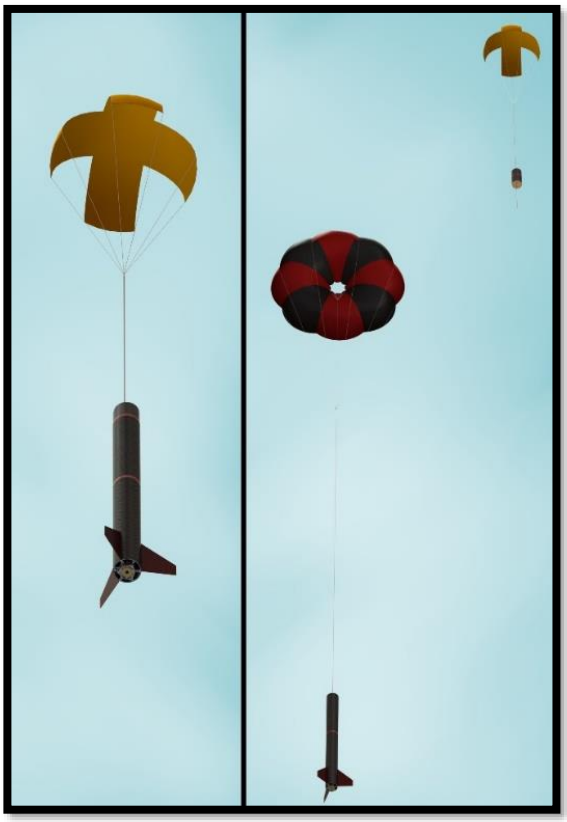


Figure 56: Booster drogue descent and coupler seperation.

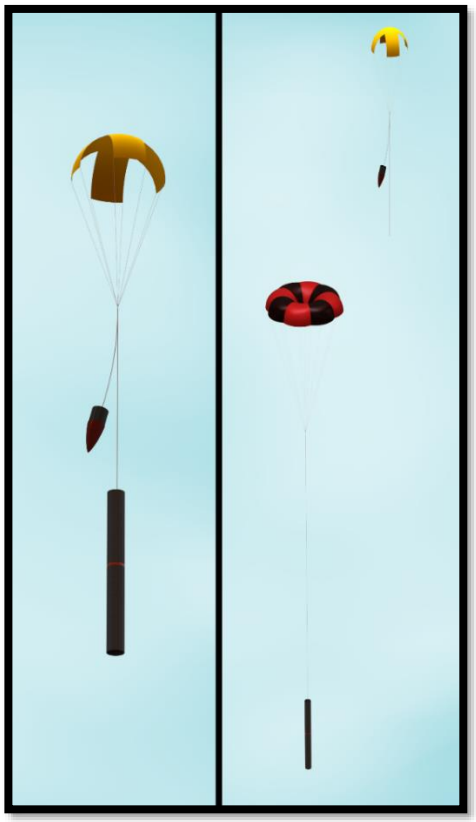


Figure 57: Payload drogue descent and release device seperation.

3.5.6 Coupler Deployment

The booster section of the launch vehicle will be recovered using a drogue and main separated into two bays. The drogue will be stowed in the coupler aft of the payload bay. During separation at apogee, the drogue will be loosely tethered to the payload section of the launch vehicle. As the payload bay separates from the coupler, it will pull the drogue away from its stowed position. Once the maximum length in the shock cord is reached, the tether will sever, and the drogue will be fully deployed. At main event, the coupler will be separated from the booster recovery bay via black

powder separation charges. The main parachute's deployment bag will be drawn out of the bay by the drogue as it is connected to the coupler. Once the maximum shock cord length is reached, the main will be pulled from the deployment bag and the coupler and drogue will become their own independent section.

3.5.7 Electronics

Each recovery event will be triggered by a redundant set of PerfectFlite StratologgerCFs. The PerfectFlite StratologgerCF altimeter records its altitude at a rate of 20Hz with a 0.1% accuracy. In previous testing, the altimeter was found to be accurate to ±1 foot. Each StratologgerCF will be powered by an individual Duracell 9V battery. Duracell batteries have been selected due to their reliability and the feature that their leads are internally soldered.

3.5.8 Design

3.5.8.1 Parachute Sizing Requirements

The main parachute diameters were derived to meet the Kinetic Energy Requirements (KER) in SOW 2.3 outlined in section 6.1.1. Nominal diameter for a parachute can be found using

$$\boldsymbol{D_o = \sqrt{\frac{4m_v m_s g}{\pi E C_D \rho}}} \tag{12}$$

where D_o is the nominal diameter, m_v is the mass of the vehicle, m_s is the mass of the subsection, g gravitational acceleration, E is the kinetic energy, C_D is the coefficient of drag and ρ is the air pressure. Each main parachute will have one tethered section, so this equation simplifies to

$$\boldsymbol{D_o = \sqrt{\frac{4m^2 g}{\pi E C_D \rho}}} \tag{13}$$

These sizes were calculated and are shown below in Table 22.

Section	**Weight (kg)**
Payload bay	7.650
Booster	7.550
Parachute	**Diameter (in)**
Payload main	81
Booster main	80

Table 22: Main parachute sizes.

The cruciform drogue for each bay was sized to firstly meet the KER stated in SOW 3.3 of section 6.1.4.1 where each drogue will become the main parachute for the coupler and nosecone after main deployment. The minimum diameter was calculated using

$$\boldsymbol{D_o = \sqrt{\frac{4m_v m_s g}{\pi E C_D \rho}}} \tag{14}$$

Second, the parachutes were sized to descend at a constant velocity where the RDR will not be exceeded. With the maximum drift distance given, we can solve for a terminal velocity needed to stay within the boundaries using the main drift distance as a set value, since it depends upon KER only. In 20 MPH (29.3 ft/s) winds, the drift during main decent phase is 800 ft. This means that the drogue must remain within 1,700 ft. of the launch rail from apogee. We use this data to solve for allotted time during drogue phase before the drift is too large and find that the drogue must descend 4,780 ft. in 88 seconds or at a rate of 54.3 ft/s. We can then solve for a surface area using

$$\boldsymbol{S_o = \frac{2mg}{C_D V_e^2 \rho}} \tag{15}$$

Where V_e is terminal velocity which in this case is 54.3 ft/s. The cruciform design is rectangular where the width of a panel is ¼ the size of the length. given the surface area, the length is solved for using

$$\boldsymbol{S_o = 2\left(L * \frac{L}{4}\right) - \left(\frac{L}{4}\right)^2} \tag{16}$$

Where L is length. When arranged to solve for panel length, it simplifies to

$$\boldsymbol{L = \frac{4}{7}\sqrt{7 * s_o}} \tag{17}$$

This shows nominal diameter is larger than the minimum diameter. It must also satisfy TDR4 in section 6.1.4.2 such that the drag of the drogue parachute will be greater than the fins of the booster segment of the rocket. This failure mode is illustrated below in Figure 58 .

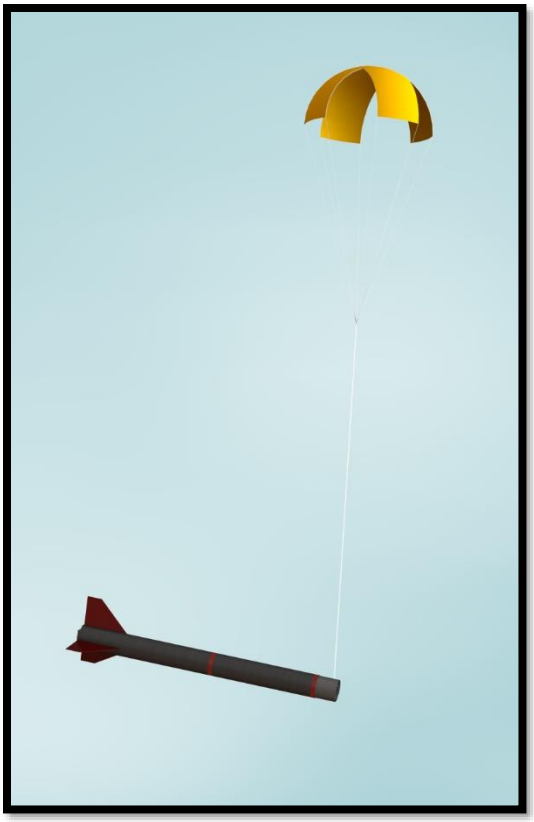


Figure 58: Drogue drag force failure mode

The drag force of the booster section was found using ANSYS fluent simulations and was compared to the drag force of the drogue parachute using

$$\sqrt{\frac{2mg}{C_{D_b}S_{o_b}\rho}} > \sqrt{\frac{2mg}{C_{D_d}S_{o_d}\rho}} \tag{18}$$

where the right side is the terminal velocity of the drogue and the left side is the terminal velocity of the booster section. After simplification of terms, this equation simplifies to

$$\frac{1}{C_{D_b}S_{o_b}} > \frac{1}{C_{D_d}S_{o_d}} \tag{19}$$

With these requirements in place the drogue size for the launch vehicle was calculated as seen below in Table 23.

Section	**Weight (kg)**
Nosecone	1.440
Payload bay	7.650
Coupler	0.929
Booster	7.550
Parachute	**Diameter (in)**

	KER (minimum)	RDR (maximum)
Payload Drogue	23	58
Booster Drogue	15	56

Table 23: Drogue parachute calculations

3.5.8.2 Drift Velocity Calculations

The drift distances of the launch vehicle were calculated using terminal velocity values, altitudes of parachute deployment, and decent times. Two drift values were calculated, one assuming the launch vehicle fly perfectly vertical, and another that factors in weather cocking of the vehicle due to wind. The drift values with no weather cocking are shown below in Table 24 for 0 mph, 5 mph, 10 mph, 15 mph, and 20 mph cases.

Wind speed	Distance from rail (Ft.)			
	Booster	Payload	Coupler	Nosecone
0 MPH	0.0	0.0	0.0	0.0
5 MPH	834.9	829.4	732.4	732.4
10 MPH	1669.9	1658.7	1464.8	1464.8
15 MPH	2504.8	2488.1	2197.2	2197.2
20 MPH	3339.7	3317.5	2929.6	2929.6

Table 24: Initial drift values

These drift calculations show that the launch vehicle will exceed the 2,500 ft. boundary put in place by R3.9. However, when the amount of weather-cocking the launch vehicle will experience under these wind conditions is considered, the drift values decrease significantly to within the acceptable ranges. The distance traversed laterally due to weather-cocking was calculated using the angle of attack in the different cases and the anticipated altitude to find the x component of the flight. The more accurate drift distances are shown below in Table 25.

Wind speed	Distance from rail (ft.)			
	Booster	Payload	Coupler	Nosecone
0 MPH	0.0	0.0	0.0	0.0
5 MPH	634.9	629.4	532.4	532.4
10 MPH	1269.9	1258.7	1064.8	1064.8
15 MPH	1904.8	1888.1	1597.2	1597.2
20 MPH	2539.7	2517.5	2129.6	2129.6

Table 25: Drift values including weather-cocking

These values were also compared to an OpenRocket simulation from each wind speed case and were found to be within 15% of the calculated values where the simulation found shorter distances. One of the simulations is shown below in Figure 59 as a graph of distance from the rail over time.

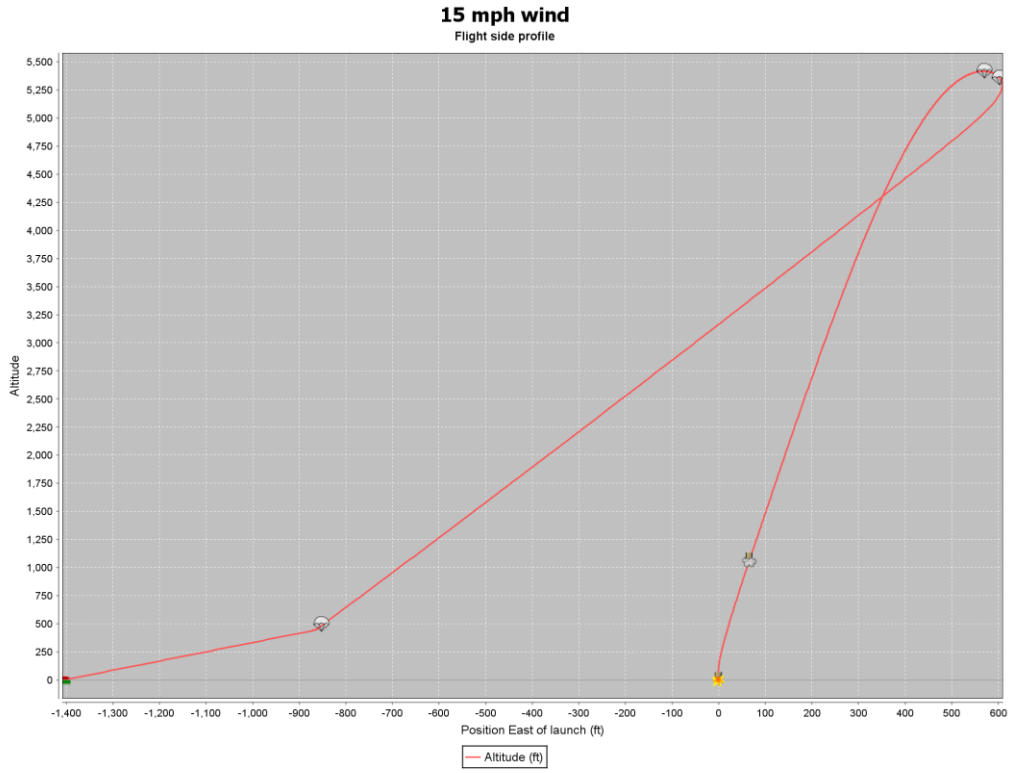


Figure 59: Side profile of simulation in Openrocket with 15 MPH cross wind

These simulation data and calculation values will be compared to, and verified by flight test data during the subscale and full-scale flights of the launch vehicle.

3.5.8.3 Opening Force Reduction

Opening force reduction rings will be used on all main parachutes to ensure the opening force does not exceed the acceleration seen during the burn of the motor as defined in TDR2 in section 6.1.4.2. An opening force reduction ring slides over the shroud lines to the mouth of the parachute. The parachute inflating will force the shroud lines to separate outwards but will be retained by the ring. This will push the ring further down the shroud lines until it reaches the quick link at the end connecting it to the shock cord. This is shown in Figure 60 below.

Figure 60: Opening force reduction ring

Tests have been conducted which show that the ring does qualitatively decrease opening forces. Subsequent quantitative data will be acquired by ground tests and subscale flights.

3.5.9 Subscale Launch Vehicle Recovery

Subscale testing will be performed to verify the TDR and the SOW requirements. The subscale launch vehicle will consist of one double-staging single recovery bay that replicates the full-scale vehicle at ½ scale. The subscale will utilize an ARRD to verify its effectiveness. The drogue and main parachutes will be deployed from the recovery bay of the launch vehicle. The parachute parameters were calculated using section 3.5.8 and are shown below in Table 26.

Parachute	Diameter (in.)
Main	30
Drogue	20

Table 26: Subscale recovery system

3.6 Mission Performance Predictions

3.6.1 Applicable Equations

While software exists that can accurately simulate the flight of the launch vehicle, hand calculations were performed to verify the simulation's accuracy. To assess the performance of the vehicle in flight, three main values are calculated: peak altitude, center of gravity, and the center of pressure of the vehicle. Calculating peak altitude requires a specific sequence of equations. First, average mass of the vehicle before burnout is calculated using

$$m_a = m_r + m_e - \frac{m_p}{2} \tag{20}$$

in which m_r is the mass of the rocket, m_e is the mass of the motor, and m_p is the propellant mass. Then the vehicle's aerodynamic drag coefficient (kg/m) is calculated using

$$k = \frac{1}{2}\rho C_D A \tag{21}$$

where ρ is air density (1.22kg/m^3), C_D is the drag coefficient, and A is the vehicle's cross-sectional area (m^2). Burnout velocity coefficient (m/s) is calculated using

$$q_1 = \sqrt{\frac{T - m_a g}{k}} \tag{22}$$

where T is the motor thrust and g is the gravitational constant (9.81 m/s^2). The vehicle's burnout velocity delay coefficient (1/s) is calculated using

$$x_1 = \frac{2kq_1}{m_a} \tag{23}$$

The burnout velocity (m/s) is calculated using

$$v_1 = q_1 \frac{1 - e^{-x_1 t}}{1 + e^{-x_1 t}} \tag{24}$$

where t is the motor burnout time (s). The vehicle's altitude at motor burnout can then be computed using

$$y_1 = \frac{-m_a}{2k} \ln\left(\frac{T - m_a g - kv_1^2}{T - m_a g}\right) \tag{25}$$

After the altitude at burnout is calculated, the vehicle's coasting distance must then be calculated. Comparable to burnout altitude, vehicle mass must be calculated first. The coasting mass is calculated using

$$m_c = m_r + m_e - m_p \tag{26}$$

Using coasting mass, the coasting velocity coefficient is calculated using

$$q_c = \sqrt{\frac{T - m_c g}{k}} \tag{27}$$

Also using coasting mass, the coasting velocity delay coefficient was calculated using

$$x_c = \frac{2kq_c}{m_c} \tag{28}$$

The vehicle's coasting velocity is then found using

$$v_c = q_c \frac{1 - e^{-x_c t}}{1 + e^{-x_c t}} \tag{29}$$

The coasting distance is found using

$$y_c = \frac{m_c}{2k} \ln\left(\frac{m_c g + k{v_c}^2}{T - m_c g}\right) \tag{30}$$

The peak altitude of the vehicle can then be found using

$$PA = y_1 + y_c \tag{31}$$

The vehicle's center of gravity location is calculated using

$$c_g = \frac{d_n w_n + d_r w_r + d_b w_b + d_e w_e + d_f w_f}{W} \tag{32}$$

where W is the total weight of the launch vehicle and d is the distance between the denoted section's center of gravity (nose, body, rocket, body, engine, and fins, respectively) and the aft end. The vehicle's center of pressure measured from the nose tip is computed using

$$X = \frac{(C_N)_N X_N + (C_N)_F X_F}{(C_N)_N + (C_N)_F} \tag{33}$$

where C_{NN} is the nose cone center of pressure coefficient (2 for conical nose cones). X_N is calculated using

$$X_N = \frac{1}{2} L_N \tag{34}$$

where L_N is the nose cone's length. Variable C_{NF} of (35) is defined by the fin center of pressure coefficient calculated using

$$(C_N)_F = \left[1 + \frac{R}{S + R}\right]\left[\frac{4N\left(\frac{S}{d}\right)^2}{1 + \sqrt{1 + \left(\frac{2L_f}{C_R + C_T}\right)^2}}\right] \tag{35}$$

where R is the cross-sectional radius of the vehicle body at the aft end, S is the fin semispan, N is the number of fins, L_F is the length of the fin mid-chord line, and C_T is the fin tip chord length. X_F is calculated using

$$X_F = X_B + \frac{X_R(C_R + 2C_T)}{3(C_R + C_T)} + \frac{1}{6}\left[(C_R + C_T) - \frac{(C_R C_T)}{(C_R + C_T)}\right] \tag{36}$$

where X_B is the distance from the nose tip to the leading edge of the fin root chord, X_R is the distance between the fin root leading edge and the fin tip leading edge measured parallel to the vehicle body. (30) through (36) are also known as the Barrowman Equations.

Note that (36) makes use of a simplified form because the vehicle makes no transition in the body diameter, thus the transitional terms have been omitted. These equations are used to verify the OpenRocket simulations conducted on the full-scale launch vehicle.

3.6.2 Simulations

The OpenRocket software application was used to simulate the launch vehicle's flight. In the simulation setup, it was assumed that the launch vehicle launches from a 141-in. tall rail. All simulations, unless stated otherwise, were run with a wind speed of 10mph, under international standard atmospheric conditions, launching from Bragg Farms in Toney, AL, at an elevation of 251 meters above sea level, and conducted under the assumption that the launch vehicle's surface finish consists of pits and valleys 150 µm in height. The OpenRocket model of the full-scale launch vehicle is shown below in Figure 61.

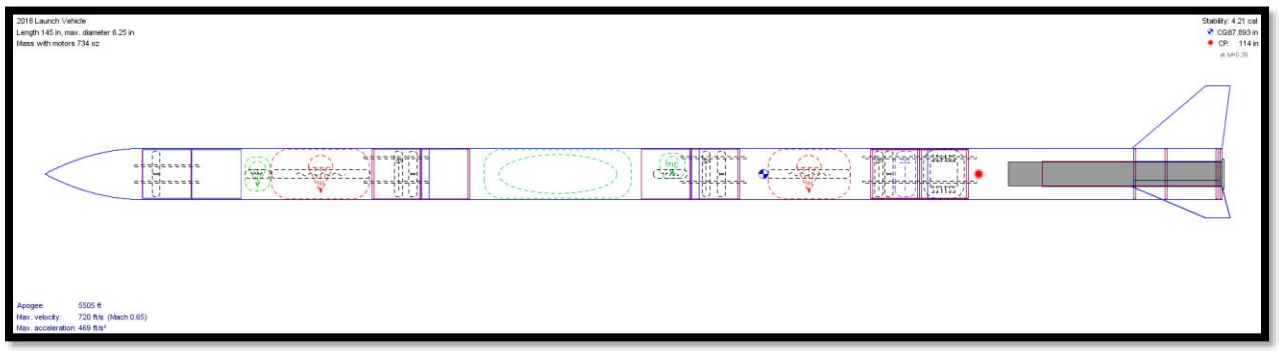


Figure 61: Full-scale OpenRocket model.

3.6.2.1 Component Masses

In the OpenRocket model, the mass of each component was estimated using material densities and measurements taken from similar components from previous year's launch vehicles. These estimates were used to calculate the total approximate mass of the launch vehicle. Each subsystem's estimated mass, including the masses of all components in each subsystem, is shown below in Table 27.

Launch Vehicle Section	Mass (lbs.)
Booster	17.35
VDS Coupler	4.29
Booster Recovery	2.52
Payload Coupler	2.31
Payload Bay	13.1
Payload Recovery Coupler	1.8
Payload Recovery Bay	2.01
Nose Cone	2.5
Total Launch Vehicle	45.88

Table 27: Launch vehicle component properties used in OpenRocket simulations.

3.6.3 Motor Selection

OpenRocket simulations were performed to determine which motor would be optimal to deliver the launch vehicle to an apogee of 5,500ft with an inactive VDS. The different motors considered, and their simulated apogee altitudes, are shown below in Table 28.

Motor	Total Impulse (Ns)	Average Thrust (N)	Predicted Apogee Altitude (ft.)
Aerotech L2200	5,104	2,200	5,510
Cesaroni L2375	4,905	2,375	5,369
Cesaroni L3150	4,806	3,150	5,211

Table 28: Different motors and their respective apogee altitudes with an inactive VDS.

Upon analyzing the OpenRocket simulation results, it has been determined that the launch vehicle will utilize an Aerotech L2200 rocket motor. If the launch vehicle's design leads to a lower mass than expected, the launch vehicle may utilize an alternative. The Aerotech L2200's specifications are shown below in Table 29.

Specification	Numerical Value
Diameter	75 mm
Length	68.1 cm
Total Weight	4,783 g
Propellant Weight	2,518 g
Average Thrust	2,200 N
Maximum Thrust	3,104 N
Total Impulse	5,104 Ns
Burn Time	2.3 sec

Table 29: Aerotech L2200 specifications.

3.6.3.1 Simulated Motor Thrust Curve

In the OpenRocket software's data sheets, it lists that the Aerotech L2200's thrust curve used in simulations was acquired from Aerotech Consumer Aerospace. The simulated motor thrust curve is shown below in Figure 62.

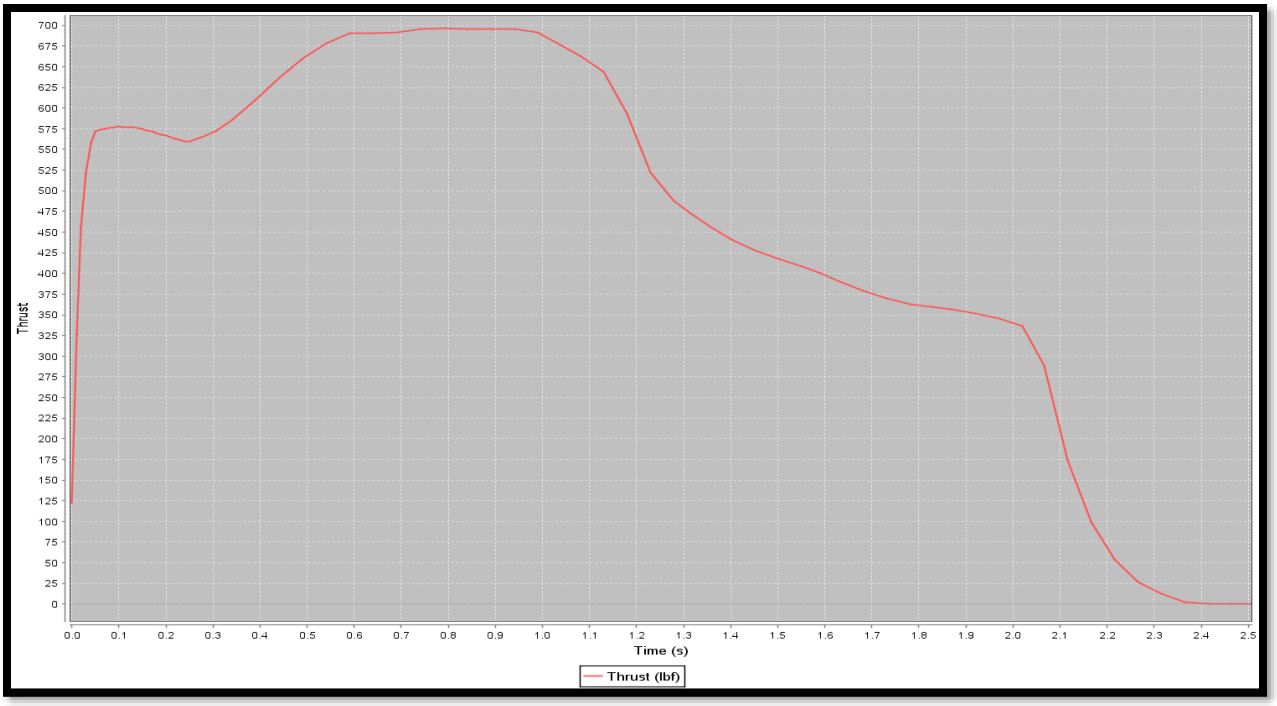


Figure 62: Simulated motor thrust curve.

The maximum thrust produced by the L2200 motor is approximately 700lbs. To account for the possibility of a centering ring failure, each centering ring was designed to withstand a minimum load of 350 lbs. while maintaining a minimum factor of safety greater than 2.0. Finite Element Analysis (FEA) was performed on each centering ring using ANSYS to verify that they could support the loads generated during motor burn. As shown in Figure 63 and Figure 64, the FEA showed that both centering rings met the requirement.

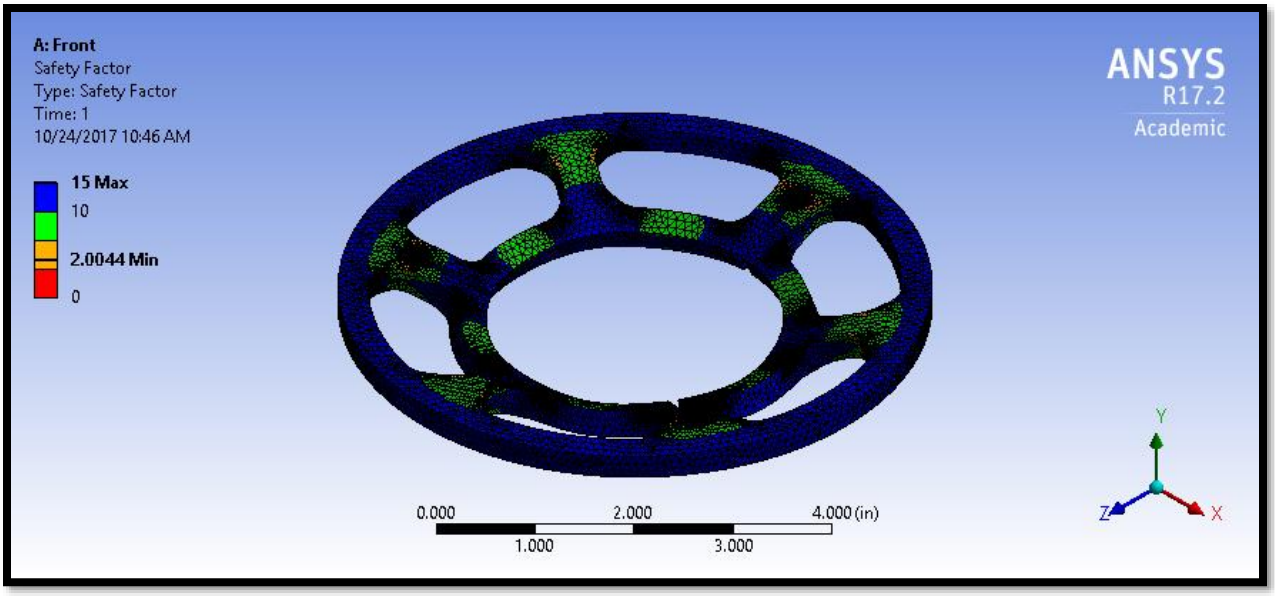


Figure 63: Fore centering ring FEA results.

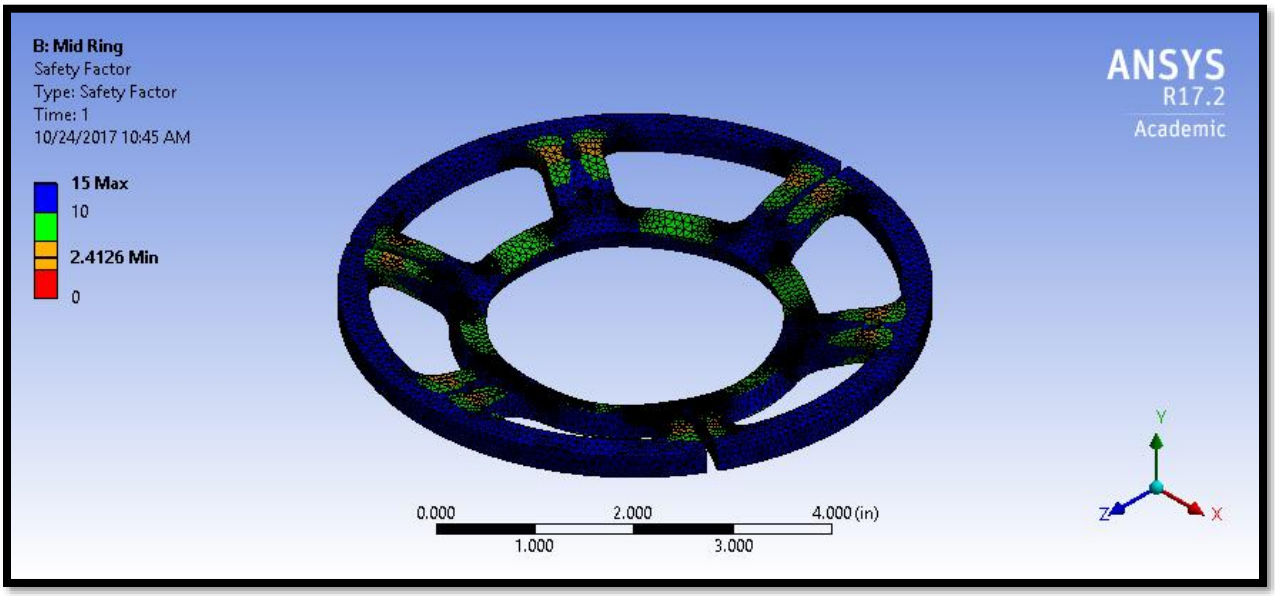


Figure 64: Mid centering ring FEA results.

3.6.4 Altitude Predictions

Several OpenRocket simulations were performed to confirm that the AeroTech L2200 could deliver the launch vehicle to an apogee of 5,500 ft. in varying weather conditions. The simulations were conducted with wind conditions varying from 0mph, up to the maximum safe wind speed of 20mph. The launch vehicle's predicted apogees with an inactive VDS are shown below in Table 30.

Wind Speed	Predicted Apogee (ft.)
0mph	5,536
5mph	5,529
10mph	5,510
15mph	5,486
20mph	5,453

Table 30: OpenRocket simulated apogee altitude at varying wind speeds.

A custom MATLAB simulation was created to verify the accuracy of the OpenRocket simulations and verify the capabilities of the VDS. The simulation was set up to launch from a 141-in. launch rail, with no wind. The simulated flight path of the launch vehicle, with an inactive VDS, is shown below in Figure 65.

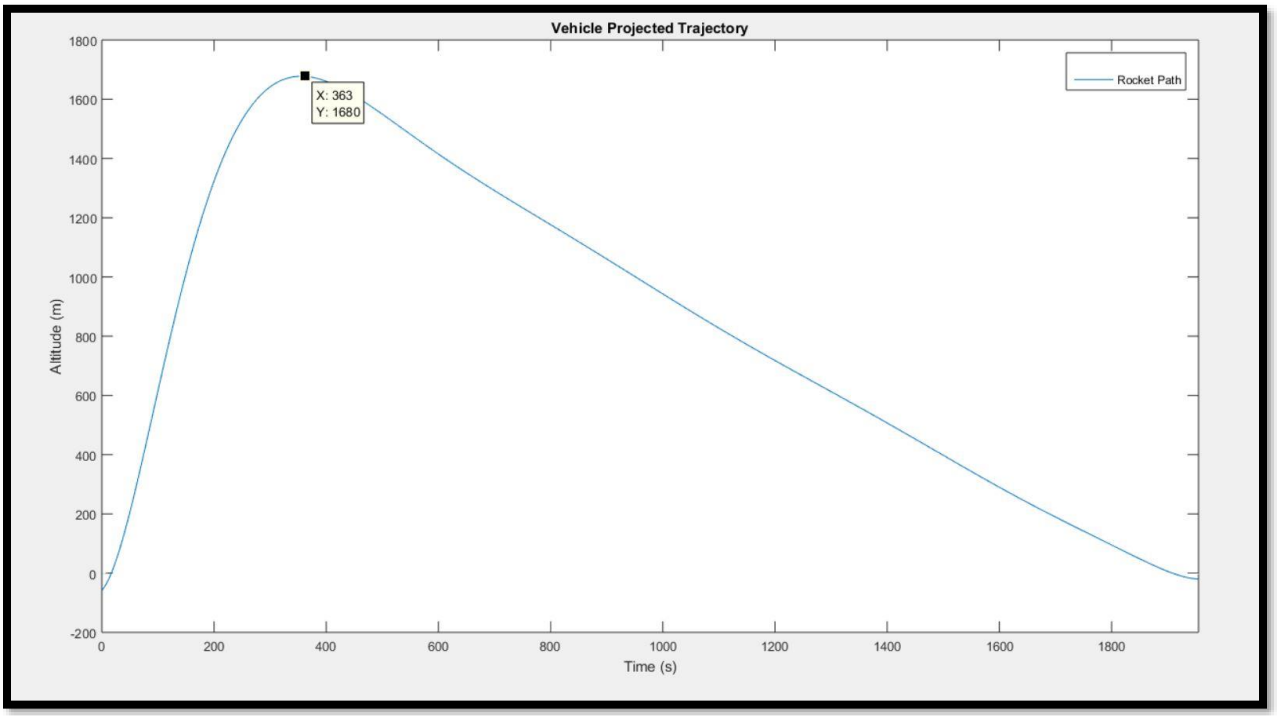


Figure 65: Simulink simulation of the launch vehicle flight with an inactive VDS.

The MATLAB simulation results show an apogee altitude of 1,609 m, or 5,511 ft. The apogee altitudes from the OpenRocket and MATLAB simulations differ by only 25 ft., or 0.45%. This minimal difference between the two simulation methods verify that the predicted apogee altitudes results are accurate. A MATLAB simulation, shown in Figure 66, was also conducted to simulate the launch vehicle's flight with an active VDS. The results of the simulation show an apogee of 1,615 m, or 5,298 ft.

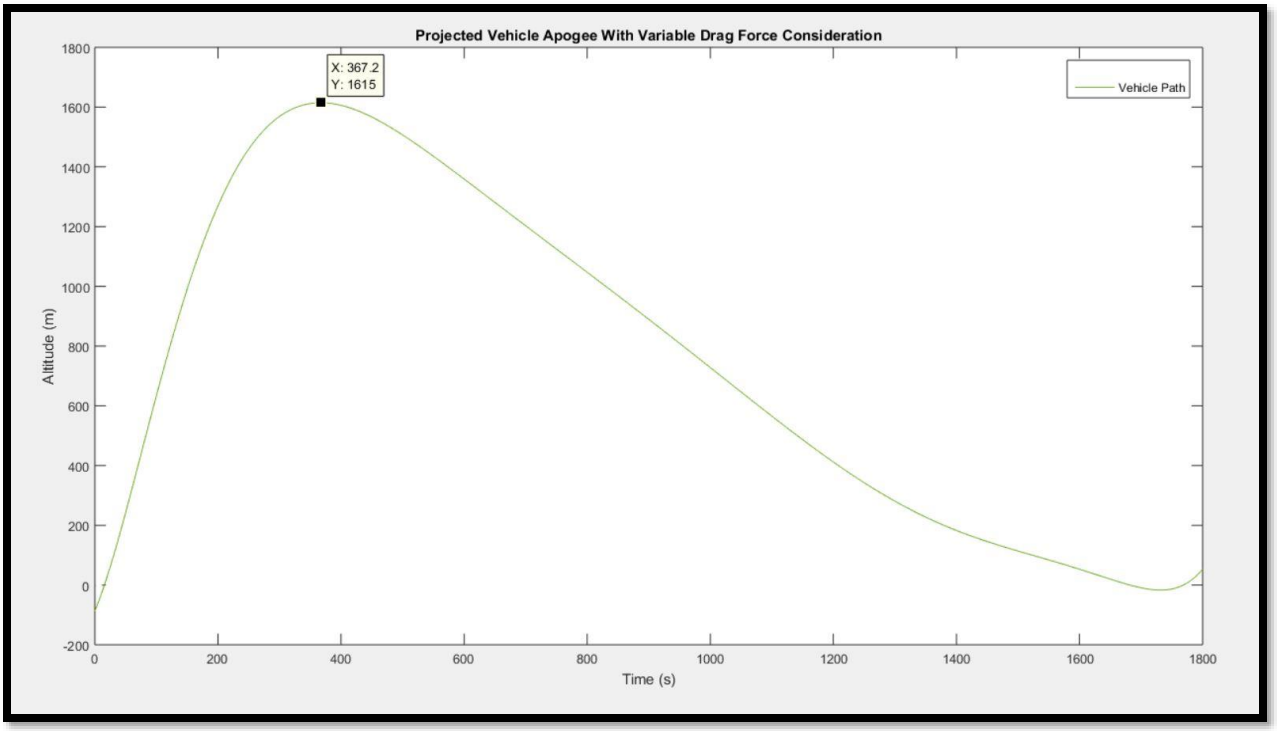


Figure 66: Simulink simulation of the launch vehicle's flight with an active VDS.

3.6.5 Flight Profile Simulations

A flight profile simulation was conducted in OpenRocket to simulate the ascent and descent of the launch vehicle. The launch vehicle's flight profile plot is shown below in Figure 67.

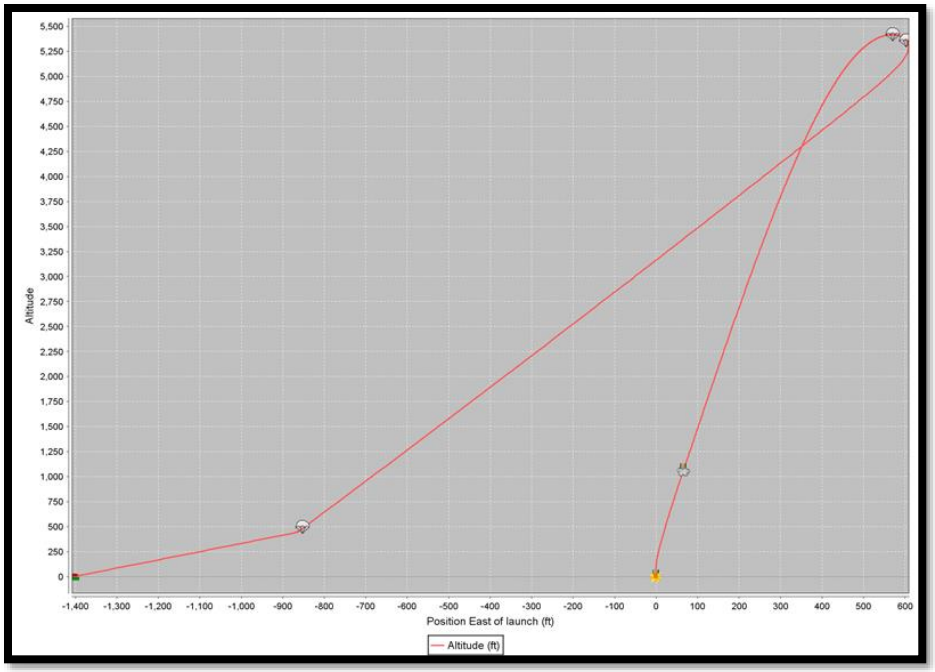


Figure 67: Flight profile simulation graph.

3.6.6 Angle of Attack

During high winds, one of the main concerns associated with launching safely is the angle of attack just after rail exit. With a high angle of attack, the launch vehicle could potentially fly off course and result in a lower altitude and a recovery radius greater than 2,500ft. A low altitude and wide recovery radius both affect the success of the mission; therefore, the launch vehicle has been designed to minimize the angle of attack at rail exit. Several OpenRocket simulations were run under varying weather conditions to predict the angle of attack at rail exit. The OpenRocket simulation results are shown below in Table 31.

Wind Speed	Angle of Attack at Rail Exit
0 mph	1.05
5 mph	4.39
10 mph	8.79
15 mph	12.98
20 mph	17.09

Table 31: Flight characteristics in varying weather conditions.

3.6.7 Flight Characteristics

OpenRocket simulations, and hand calculations using the equations listed in section 3.6.1, have been performed to determine the launch vehicle's flight characteristics. Key flight characteristics found are shown below in Table 32.

Flight Characteristic	Numerical Value
Stability Margin at Rail Exit (Cal.)	2.25
Simulated Center of Pressure (CP) Location from Nose Cone Tip (in.)	96.51
Center of Gravity (CG) Location from Nose Cone Tip (in.)	82.33
Exit Rail Velocity (ft./s)	95.4
Maximum Velocity (ft./s)	732
Maximum Acceleration (ft./s^2)	479
Thrust-to-Weight Ratio	15.26

Table 32: Key flight characteristics.

3.6.8 Subscale Launch Vehicle Flight Characteristics

Flight simulations were conducted in OpenRocket to verify the subscale represents similar flight characteristics of the full-scale launch vehicle. The flight characteristics of the subscale launch vehicle, compared to that of the full-scale launch vehicle, are shown below in Table 33.

Characteristic	Sub-Scale	Full-Scale
Stability Margin at Rail Exit (in.)	2.23	2.25
Simulated Center of Pressure (CP) Location from Nose Cone Tip (in.)	50.40	96.51
Center of Gravity (CG) Location from Nose Cone Tip (in.)	43.42	82.33
Exit Rail Velocity (ft./s)	94.9	95.4
Maximum Velocity (ft./s)	515	732
Maximum Acceleration (ft./s^2)	595	479
Simulated Apogee (ft.)	2,214	5,562

Table 33: Sub-scale launch vehicle flight characteristics compared to full-scale launch vehicle flight characteristics.

4 Safety

4.1 Safety Requirements

Maria Exeler is the Safety Officer for River City Rocketry during the 2017-2018 season. As Safety Officer, she is responsible for ensuring the overall safety of the team, students, and public throughout all team lead activities.

4.1.1 Statement of Work Requirements

The statement of work requirements for Safety were provided by NASA and are shown in Table 34.

Requirement Number	Requirement	Verification
5.1	Each team will use a launch and safety checklist. The final checklists will be included in the FRR report and used during the Launch Readiness Review (LRR) and any launch day operations.	**Demonstration** Thorough checklists will be created prior to test launches that will require two member signatures for each step. The checklists will be updated after each test launch and will be finalized prior to FRR.
5.2	Each team must identify a student safety officer who will be responsible for all items in section 5.3.	**Demonstration** Maria Exeler was identified as the Safety Officer.
5.3	The role and responsibilities of each safety officer will include, but not limited to: Safety 5.3.1.- Safety 5.3.4.	**Demonstration** Revision F of the team Safety Manual and has been created to address these requirements.
5.3.1.	Monitor team activities with an emphasis on Safety during: design, construction, assembly, and ground testing of vehicle and payload, sub-scale and full-scale launch tests, launch day, recovery activities, and educational engagement activities.	**Demonstration** Designs and construction progress will be reviewed weekly through subteam lead meetings. Mandatory safety briefings will be held prior to all construction and testing. Educational engagement activity safety will be reviewed prior to all team events.
5.3.2.	Implement procedures developed by the team for construction, assembly, launch, and recovery activities.	**Demonstration** Newly identified hazards will be reviewed prior to the construction, assembly, and testing of any components.

5.3.3.	Manage and maintain current revisions of the team's hazard analyses, failure modes analyses, procedures, and MSDS/chemical inventory data	**Demonstration** Maria has updated and reviewed these items with the team prior to Preliminary Design Review and reviews will continue as new materials are used and processes are conducted.
5.3.4.	Assist in the writing and development of the team's hazard analyses, failure modes analyses, and procedures.	**Demonstration** The Safety Officer will meet with the lead of each subsystem to review and update the hazard and failure analyses and procedures prior to each test flight and major review.
5.4	During test flights, teams will 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 Initiative does not give explicit or implicit authority for teams to fly those certain vehicle configurations and/or payloads at other club launches. Teams should communicate their intentions to the local club's President or Prefect and RSO before attending any NAR or TRA launch.	**Demonstration** By agreeing to the Safety Manual, all team members agreed to follow decisions made by the RSO during all launches.
5.5	Teams will abide by all rules set forth by the FAA.	**Demonstration:** By signing the Safety Manual, all team members agreed to follow the regulations defined in FAR 14 CFR.

Table 34: Statement of work safety requirements and verifications.

4.1.2 Team Derived Safety Requirements

Safety will be further ensured through the team derived responsibilities that are listed in Table 94.

4.2 Hazard Analysis

Project Risk Management
The project was review for risks including budget, integration, and equipment availability. Table 111 shows the result of the review.

Risk Assessment Matrix
By methodically examining each human interaction, environment, rocket system and component, hazards have been identified. However, hazards and risks will continue to be revised through the

competition as new components are designed and manufactured. Risk assessment and mitigation are vital to the success of our project and team safety.

Each currently identified hazard has been evaluated through a risk assessment process that recognizes the hazard's potential causes and results, the severity and probability of the hazard, and how the hazard can be mitigated.

A severity value between 1 and 4 has been assigned to each hazard with a value of 1 being the most severe. To determine the severity of each hazard, the outcome of the mishap was compared to an established set of criteria based on the severity of personal injury, environmental impact, and damage to the rocket and/or equipment. These criteria are outlined below in Table 35.

Severity		
Description	**Value**	**Criteria**
Catastrophic	1	Could result in death, significant irreversible environmental effects, complete mission failure, or monetary loss of $5k or more.
Critical	2	Could result in severe injuries, significant but reversible environmental effects, partial mission failure, or monetary loss of $500 or more but less than $5k.
Marginal	3	Could result in minor injuries, moderate environmental effects, complete failure of non-mission critical system, monetary loss of $100 or more but less than $500.
Negligible	4	Could result in insignificant injuries, minor environmental effects, partial failure of non-mission critical system, monetary loss of less than $100.

Table 35: Severity value criteria.

A probability value between 1 and 5 has been assigned to each identified hazard with a value of 1 being most likely. The probability value was determined for each hazard based on an estimated percentage chance that the mishap will occur with the following conditions considered:

- All personnel involved have undergone proper training on the equipment and material being used or processes being performed
- All personnel have read and acknowledged that they have a clear understanding of all rules and regulations set forth by the latest revision of the Safety Manual
- Personal Protective Equipment (PPE) is used properly as indicated by the Safety Manual and MSDS
- All procedures were correctly followed and verified during the construction of the rocket, testing, pre-launch preparations, assembly, and during launch
- All components were thoroughly inspected for damage and fatigue prior to any test or launch

The criteria for the selection of the probability value is outlined below Table 36.

Probability

Description	Value	Criteria
Almost Certain	1	Greater than a 90% chance that the mishap will occur
Likely	2	Between 50% and 90% chance that the mishap will occur
Moderate	3	Between 25% and 50% chance that the mishap will occur
Unlikely	4	Between 1% and 25% chance that the mishap will occur
Improbable	5	Less than a 1% chance that mishap will occur

Table 36: Probability value criteria.

Through the combination of the severity level and the probability value, an appropriate risk level has been assigned using the risk assessment matrix found in Table 37. The matrix identifies each combination of severity and probability values as either a high, moderate, or low risk. The team's goal is to have every hazard to a low risk level by the time of the competition launch. Hazards that are not currently at a low risk level will be readdressed with redesign, additional safety regulations, or other measures as required. Risk levels may also be reduced through verification systems.

Risk Assessment Matrix				
Probability Level	**Severity Level**			
	Catastrophic (1)	Critical (2)	Marginal (3)	Negligible (4)
Almost Certain (1)	2-High	3-High	4-High	5-Moderate
Likely (2)	3-High	4-High	5-Moderate	6-Moderate
Moderate (3)	4-High	5-Moderate	6-Moderate	7-Low
Unlikely (4)	5-Moderate	6-Moderate	7-Low	8-Low
Improbable (5)	6-Moderate	7-Low	8-Low	9-Low

Table 37: Risk Assessment Matrix.

Preliminary risk assessments have been completed for possible hazards that have been identified at this stage in the design. Identifying the hazards at the current design stage brings attention to the components as possible failure mechanisms. The team can improve the designs and create future refinements with these mechanisms in mind. The team will work to mitigate the hazards throughout the design phase.

Some identified risks are currently unacceptably high. This is because all risks have been identified and addressed through preliminary concept design work, hand calculations, and recommended practices. As components and design concepts are tested, designs will be verified as safe. Risk levels will only be lowered once physical testing has been performed and the safety of the design has been verified.

Workshop Risk Assessment

Construction and manufacturing of parts for the rocket will be performed in both on-campus and off-campus labs. The hazards assessed in Table 112 associated with machinery, tools, and chemicals in the lab. lab.

Stability and Propulsion Risk Assessment

The hazards outlined in Table 106 are risks associated with stability and propulsion. The team has one member with certifications supporting that he can safely handle motors and design stable

rockets the size of the competition rocket. Other members are currently working toward certifications as well. This assessment is considered a moderate risk for the team and lessons learned from the 2016-2017 season will keep the team vigilant about mitigating these risks.

Vehicle Assembly Risk Assessment
The hazards outlined in Table 107 are risks that could potentially be encountered throughout the assembly phase and during launch preparations.

VDS Actuation Risk Assessment
The hazards outlined in this section discuss the risks associated with testing and flight of the VDS. The VDS interfaces with the main structure of the vehicle with potential risk from manufacturing, assembly, and installment. The VDS hazards are outlined in Table 108.

Recovery Risk Assessment
The hazards outlined in Table 109 are risks associated with the recovery. Since there are two recovery systems onboard, many of the failure modes and results will apply to all the systems but will be stated only once for conciseness.

Payload Risk Assessment
The payload will not be permanently fixed in the launch vehicle and will require multiple components to ensure proper deployment. contains the testing, assembly, and flight hazards associated with the payload.

Environmental Hazards to Launch Vehicle Risk Assessment
The hazards outlined in Table 113 are risks from the environment that could affect the rocket or a component of the rocket. Several of these hazards resulted in a moderate risk level and will remain that way for the remainder of the season. These hazards are the exception for needing to achieve a low risk level. This is because several of these hazards are out of the team's control, such as the weather. In the case that environmental hazards present themselves on launch day, putting the team at a moderate risk, the launch will be delayed until a low risk level can be achieved. The hazards that the team can control will be mitigated to attain a low risk level.

Launch Vehicle Hazards to Environment Risk Assessment
The hazards outlined in Table 114 are risks that construction, testing or launching of the rocket can pose to the environment.

Launch Procedures
The Safety Officer is responsible for writing, maintaining, and ensuring the use of up to date launch procedures. These are critical to ensure the safety of personnel, spectators, equipment, and the environment. Checklists will be used for all test launches.

The checklists are broken up into checklists for each subsystem for pre-launch preparations and launch day. This allows the team to maintain organization and ensures a quick and efficient

preparation for launch day. Each subsystem checklist must be 100% complete and signed by a representative of that subsystem. Checklists will be verified and collected by the Safety Officer. Overall final assembly checklist can be started once all subsystems are prepared. After completion of the final assembly, all subteam leads, captains, and the Safety Officer must approve the rocket as being a go for launch. The "at the launch pad" checklist can be started and personnel are assigned tasks of tracking each section of the rocket during recovery. Maria maintains the right to call off a launch at any time if she determines anything to be unsafe or at too high of a risk level.

Each checklist will be thoroughly written to prepare the team up for a safe and successful launch. Each subsystem checklist includes the following features to ensure that assemblers are well equipped, safe, and recognize all existing hazards:

- Required hardware
- Required equipment list
- Required PPE

CAUTION – label to identify where PPE must be used.

WARNING – label to signify importance of procedure by clearly identifying a potential failure and the result if not completed correctly.

DANGER – label to signal the use of explosives and indicates specific steps that should be taken to ensure safety.

4.3 NAR/TRA Procedures

4.3.1 NAR Safety Code

Table 38 describes each component of the High Power Rocket Safety Code, as provided by NAR, and how the team will comply with each component. This table is included in the team safety manual that all team members are required to review and acknowledge compliance.

NAR Code	Team Compliance
1. **Certification.** I will only fly high power rockets or possess high power rocket motors that are within the scope of my user certification and required licensing.	Only Matthew Cosgrove, the Launch Vehicle Lead, Darryl Hankes, the team mentor, or certified team members are permitted to pack or handle the rocket motors.
2. **Materials.** I will use only lightweight materials such as paper, wood, rubber, plastic, fiberglass, or when necessary ductile metal, for the construction of my rocket.	The Vehicle and Payload sub-teams will select appropriate materials for the rocket while considering structure and weight.

3. **Motors.** I will use only certified, commercially made rocket motors, and will not tamper with these motors or use them for any purposes except those recommended by the manufacturer. I will not allow smoking, open flames, nor heat sources within 25 feet of these motors.	The motors will be purchased from Chris' Rocket Supplies and will be stored and handled only by certified members. The entire team will understand and agree to the motor safety portion of the safety manual.
4. **Ignition System.** I will launch my rockets with an electrical launch system, and with electrical motor igniters that are installed in the motor only after my rocket is at the launch pad or in a designated prepping area. My launch system will have a safety interlock that is in series with the launch switch that is not installed until my rocket is ready for launch, and will use a launch switch that returns to the "off" position when released. The function of onboard energetics and firing circuits will be inhibited except when my rocket is in the launching position.	All launches will be conducted at NAR/TRA certified events. The Range Safety Officer will have the final say over any safety issues. There will be arming switches for the altimeters that will inhibit premature activation of firing circuits that will not be armed before the rocket is on the launch pad. These arming switches may include screw switches, key switches, or pull pins.
5. **Misfires.** If my rocket does not launch when I press the button of my electrical launch system, I will remove the launcher's safety interlock or disconnect its battery, and will wait 60 seconds after the last launch attempt before allowing anyone to approach the rocket.	The Safety Officer will remind the team of this rule prior to the 5-second countdown. The Safety Officer will communicate any precautions given by the Range Safety Officer the day of the launch.
6. **Launch Safety.** I will use a 5-second countdown before launch. I will ensure that a means is available to warn participants and spectators in the event of a problem. I will ensure that no person is closer to the launch pad than allowed by the accompanying Minimum Distance Table. When arming onboard energetics and firing circuits I will ensure that no person is at the pad except safety personnel and those required for arming and disarming operations. I will check the stability of my rocket before flight and will not fly it if it cannot be determined to be stable. When conducting a simultaneous launch of more than one high power rocket I will observe the additional requirements of NFPA 1127.7.	The Safety Officer will sound an air horn prior to the 5-second countdown to ensure spectator awareness. In the event of a ballistic rocket, the airhorn will be used again to warn all spectators and personnel at the launch field. The team will comply with this rule and any other rules given by the Range Safety Officer the day of the launch.

7. **Launcher.** I will launch my rocket from a stable device that provides rigid guidance until the rocket has attained a speed that ensures a stable flight, and that is pointed to within 20 degrees of vertical. If the wind speed exceeds 5 miles per hour I will use a launcher length that permits the rocket to attain a safe velocity before separation from the launcher. I will use a blast deflector to prevent the motor's exhaust from hitting the ground. I will ensure that dry grass is cleared around each launch pad in accordance with the accompanying Minimum Distance table, and will increase this distance by a factor of 1.5 and clear that area of all combustible material if the rocket motor being launched uses titanium sponge in the propellant.	The team will comply with this rule by launching off a 12 ft. 1515 rail which is the same rail provided at competition. The team prefers to launch in surface winds less than 4 times the exit rail velocity or less than 20 miles per hour to ensure the stability of the rocket.
8. **Size.** My rocket will not contain any combination of motors that total more than 40,960 N-sec (9,208 pound-seconds) of total impulse. My rocket will not weigh more at liftoff than one-third of the certified average thrust of the high power rocket motor(s) intended to be ignited at launch.	The team will comply to this rule when designing the rocket and selecting an appropriate motor.
9. **Flight Safety.** I will not launch my rocket at targets, into clouds, near airplanes, nor on trajectories that take it directly over the heads of spectators or beyond the boundaries of the launch site, and will not put any flammable or explosive payload in my rocket. I will not launch my rockets if wind speeds exceed 20 miles per hour. I will comply with Federal Aviation Administration airspace regulations when flying, and will ensure that my rocket will not exceed any applicable altitude limit in effect at that launch site.	A wind gauge and weather predictions will be used to make a launch day weather assessment. Appropriate FAA waivers and adequate notice will be in place before the launch occurs. The team will comply with this and any determination made by the Range Safety Officer on the day of the launch.

10. **Launch Site.** I will launch my rocket outdoors, in an open area where trees, power lines, occupied buildings, and persons not involved in the launch do not present a hazard, and that is at least as large on its smallest dimension as one-half of the maximum altitude to which rockets are allowed to be flown at that site or 1,500 feet, whichever is greater, or 1,000 feet for rockets with a combined total impulse of less than 160 N-sec, a total liftoff weight of less than 1,500 grams, and a maximum expected altitude of less than 610 meters(2,000 feet).	All team launches will be at NAR/TRA certified events. The Range Safety Officer will have the final say over any rocketry safety issues.
11. **Launcher Location.** My launcher will be 1,500 feet from any occupied building or from any public highway on which traffic flow exceeds 10 vehicles per hour, not including traffic flow related to the launch. It will also be no closer than the appropriate Minimum Personnel Distance from the accompanying table from any boundary of the launch site.	The team will comply with this rule and any determination the Range Safety Officer makes on launch day.
12. **Recovery System.** I will use a recovery system such as a parachute in my rocket so that all parts of my rocket return safely and undamaged and can be flown again, and I will use only flame-resistant or fireproof recovery system wadding in my rocket.	The Recovery subteam will be responsible for designing, constructing, and testing a safe recovery system for the rocket. A clear recovery checklist will be followed for launch day to ensure that all critical steps in preparing and packing the recovery components are completed.
13. **Recovery Safety.** I will not attempt to recover my rocket from power lines, tall trees, or other dangerous places, fly it under conditions where it is likely to recover in spectator areas or outside the launch site, nor attempt to catch it as it approaches the ground.	The team will comply with this rule and any determination the Range Safety Officer makes on launch day. If necessary, professionals will be contacted for rocket retrieval.

Table 38: NAR High Power Rocket Safety Code compliance.

4.4 Team Safety Manual

The most current team Safety Manual will always be available for review on the team website at https://www.rivercityrocketry.com/documents.

4.4.1 Safety Manual Content

At the time of the Preliminary Design Review, the team safety manual covers the following topics:

- Workshop safety
- Machine Cage specific rules
- Material Safety
- Personal Protective Equipment regulations
- Engineering Garage emergency equipment locations
- Launch safety preparations
- Educational engagement safety
- MSDS location

4.4.2 Member Agreement

A team safety review was held prior to Preliminary Design Review in which all current team members agreed to the current Safety Manual, Revision F. Members who are absent on co-op or internship at the time of the Preliminary Design Review have been identified in the form and will be required to agree to the Safety Manual the next semester they are available to participate on the team.

By signing the Safety Manual Agreement Form, all team members agreed to the stipulations below:

- I agree to the rules and regulations detailed in the River City Rocketry Safety Manual
- I will report any concerns or safety violations to the Safety Officer
- I will follow these rules and maintain safety as my highest concern when completing any work for the team

The signed form is shown below in Figure 68.

Safety Manual Agreement Form			•I agree to the rules and regulations detailed in the River City Rocketry Safety Manual. •I will report any concerns or safety violations to the Safety Officer •I will follow these rules and maintain safety as my highest concern when completing any work for the team	
Last Name	First Name	Participation and Release Form Received	I agree to the above statements	Date
Basil	Alex	Yes	Absent on co-op	
Beville	Zach	Yes		10/26/17
Bloom	David	Yes		10/26/17
Brutscher	Austin	Yes		10/26/17
Cassady	Jake	Yes	Absent on co-op	
Cockerline	Kevin	yes		10/26/17
Collins	Gabriel	yes		10/26/17
Cosgrove	Matthew	yes		10/26/17
Coyle	Jarett	Yes		10-26-17
Culver	James	yes		10/27/17
Dannunzio	Blaise	Yes		10/28/17
Epps	Alex	Yes		10/27/17
Exeler	Maria	Yes		10/26/17
Fowler	James	Yes		10/30/17
Garcia	Erik	Yes		10/26/17
Holden	Nolan	Yes	Absent on co-op	
Hsieh	Taylor	Yes		10-26-17
Huddleston	Luke	Yes		10/26/17
Johnson	Justin	Yes	Absent on co-op	
Joy	Denny	Yes		10/26/17
Ketron	Joellyn	Yes		10/26/17
Kindred	Micah	yes		10/26/17
Lewis	Eric	yes		10/26/17
Malone	Brody	Yes		10/26/17
Malone	Luke	Yes		10/26/17
Marcum	Jacob	Yes		10/26/17
Mazarakis	Alora	YES		10/26/17
McClain	Gloria	yes		10/26/17
Meier	Michael	yes		10/26/17
Meyer	Kristian	yes		10/26/17
Norvell	Kaylee	Yes		10/26/17
Pyle	Ryan	yes		10/27/17
Stringer	Ben	Yes	Absent on co-op	
Tran	Justin	Yes		10/26/17
Williams	Robert	yes		10/26/17
Williams	Samuel	yes		10/26/17
Young	Jeff	yes		10/26/17

Figure 68. Safety Manual Agreement Form.

4.4.3 Manual Updates and Review

Safety update meetings will be held prior to the start of construction and testing of the launch vehicle and payload to address newly identified hazards and established protocol. New protocol will be added to the Safety Manual and revised versions will be released and agreed to during weekly team meetings. The Safety Officer will ensure that team members are up to date and compliant with any changes in regulations.

Prior to each launch, a briefing will be held to review potential hazards and accident avoidance strategies. Only team members that are present for the briefing will be allowed to attend the launch.

4.4.4 Manual Violation

Should a violation of the Safety Manual occur, the violator will be revoked of his or her eligibility to work on rocket components in the Garage and his or her ability to attend launches until having a meeting with the Safety Officer. The violator must review and reconfirm compliance with all safety rules prior to regaining eligibility.

4.5 Local/State/Federal Law Compliances

The team has reviewed and acknowledged regulations regarding unmanned rocket launches and motor handling. Federal Aviation Regulations 14 CFR, Subchapter F, Part 101, Subpart C, Code of Federal Regulation 27 Part 55: Commerce in Explosives; and fire prevention, and NFPA 1127 "Code for High Power Rocket Motors" documentation is available to all members of the team in the team safety manual.

The team's preferred launch fields are listed below in Table 39.

Field Location	Status	Team Objective
Elizabethtown, Kentucky	1) Pending on waiver approval up to 7,000 ft 2) Less than an hour of travel 3) Moderate field size	1) Ideal field for test flights 2) Ideal for travel 3) Ideal for 0-20 mph winds
Manchester, Tennessee	1) Operational to 10,000 ft 2) Only available part of the fall and spring semesters 3) Over 3 hours away 4) Large field size	1) Ideal field for test flights 2) Moderately inconvenient due to travel 3) Ideal for 0-20 mph winds
Memphis Tennessee	1) Operational to 5,000 ft 2) Available almost every weekend 3) Over 5 hours of travel 4) Small field size	1) To utilize this field as a backup field 2) Not ideal for launches due to travel 3) Ideal for 7 mph winds or lower

Table 39: Prefered launch sites for the 2017-2018 competition.

These fields are all in compliance with the requirements. The Elizabethtown Kentucky field will be the main launch field for the 2017-2018 season.

4.6 Motor Safety

Darryl Hankes, the team mentor, who has obtained his Level 3 TRA certification, will be responsible for acquiring, storing, and handling the teams rocket motors at all times. Team member Matthew Cosgrove, who obtained his Level 2 certification in October 2017, is also permitted to assist in this responsibility. If at any time, another member of the team acquires the appropriate certification, they will also be permitted to handle the team's motors. By having obtained at minimum a Level 2 certification, the individual has demonstrated that he or she understands the safety guidelines regarding motors. Any certified member of the team that handles or stores the

team's motors is responsible for following the appropriate measures. The motors for both test and competition launches will be transported by car to the launch site.

4.6.1 Malfunctioning Engine Statistical Survey (MESS)

The team will reference the Manufacture Notifications and Modification Announcements at http://www.motorcato.org/ to ensure that there have not been any warnings issued about the motors scheduled for use in the subscale and full-scale launches.

If a catastrophic event at take off (CATO) occurs during any season flight, the team will report it through the MESS to assist in tracking the reliability of rocket motors.

5 Technical Design: Payload

5.1 Selection, Design, and Rationale of Payload

The Deployable Rover with foldable solar cell panels has been selected as this year's experimental payload. The preliminary design for the selected rover payload is shown below in Figure 69.

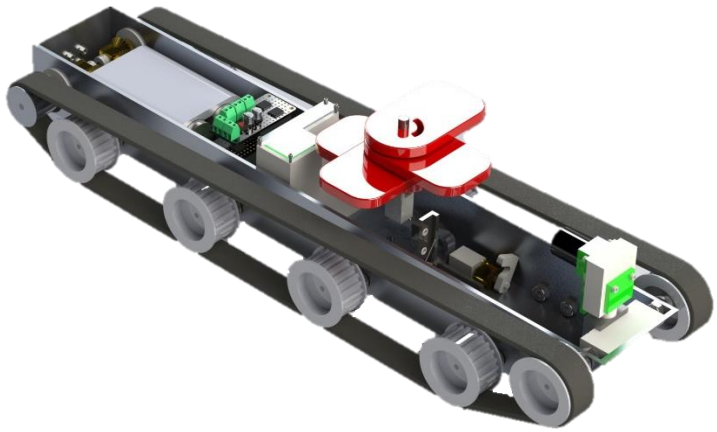


Figure 69: Rover fully deployed configuration assembly.

5.1.1 Statement of Work (SOW)

The statement of work requirements for the Deployable Rover payload has been provided by NASA and is displayed below in Table 40.

NASA Student Launch Handbook Requirement No.	Requirement
4.5.1	Teams will design a custom rover that will deploy from the internal structure of the launch vehicle.
4.5.2	At landing, the team will remotely activate a trigger to deploy the rover from the rocket.
4.5.3	After deployment, the rover will autonomously move at least 5 ft. (in any direction) from the launch vehicle.
4.5.4	Once the rover has reached its final destination, it will deploy a set of foldable solar cell panels.

Table 40: Statement of Work requirements.

5.2 Mission Success Criteria

The payload mission will be deemed a success if the following criteria are met in full.

1. The payload will remain secured inside the launch vehicle throughout the course of the flight.
2. The payload must land safely with no damage to the payload.
3. The deployable rover must exit the payload bay in its upright orientation.

4. The rover must halt at a distance of at least five feet from the launch vehicle.
5. The rover must deploy the onboard foldable solar array.

Mission success will be confirmed at the time of RSO permission to retrieve the payload.

5.3 System Level Trade Study

To ensure that the design will be capable of meeting all requirements in the statement of work, the payload has been broken into four main subsystems. These main subsystems were analyzed through trade studies to determine the optimal design for meeting the requirements. This section will give a brief explanation of the trade study, each of the designs that were determined to be viable candidates for the system, and the results of the study. The breakdown of the four main subsystems is shown below in Figure 70. An overview of each of the main subsystems is then provided in Table 41.

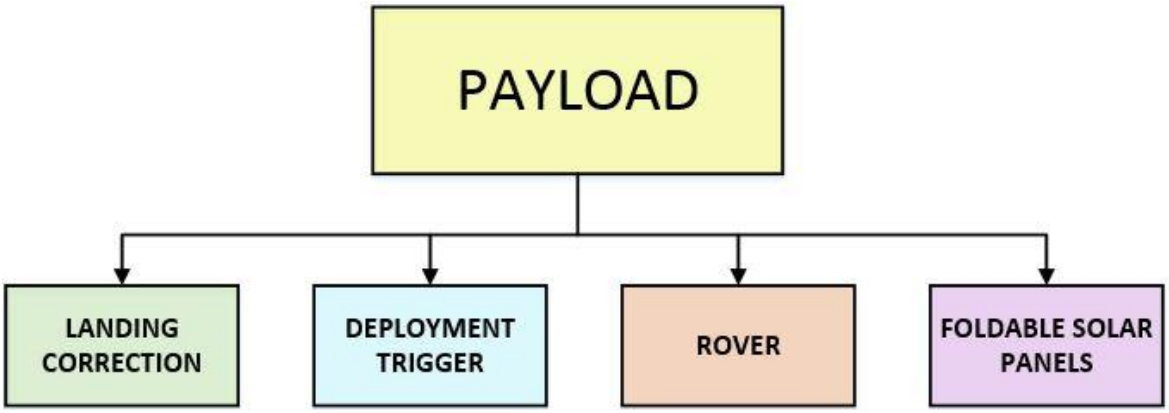


Figure 70: Main subsystems.

Main Subsystem	Subsystem Overview
Landing Correction	The Landing Correction System will support the rover while inside the airframe as well as ensure proper upright orientation before the rover deploys from the launch vehicle.
Deployment Trigger	The Deployment Trigger System will be responsible for satisfying requirement 4.5.2 of the SOW (section 5.1.1).
Rover	The Rover System will be the custom land vehicle designed to satisfy requirements 4.5.1 and 4.5.3 of the SOW (section 5.1.1).
Foldable Solar Panels	The Foldable Solar Panels System will be responsible for satisfying requirement 4.5.4 of the SOW (section 5.1.1).

Table 41: Main subsystem overviews.

5.3.1 Landing Correction Trade Study

The Landing Correction System will be a solution to the issue of unpredictable landing orientation of the payload bay. Three solutions were considered in the design of the Landing Correction trade

study. The three designs considered in this study were center bearings, perimeter bearings, and actuators.

5.3.1.1 Center Bearings

A landing correction system utilizing bearings mounted along the common center axis of the launch vehicle and rover was considered as a possible solution. A rigid rod would be mounted to the center of the recovery bulk plate forward of the payload bay. The center rings of the bearings would be slid onto the rod and locked in place. The outer rings of the bearings would be mounted to the body structure of the rover. This would allow the rover to rotate independently of the airframe of the launch vehicle during landing. The rover would then be pushed off of the rod and continue on its mission.

Advantages of the center bearings design include the ability to account for the payload bay rolling on the ground after landing and could be designed to use commercially available bearings. The main disadvantage to this design is that the rod would need to support the rover sufficiently close to both the front and rear to support the full weight of the rover without the slope in the rod allowing the rover to make contact with the interior wall of the payload bay. Having the rod pass through the entire body of the rover would take up space that could otherwise be utilized for electronics and the solar array.

5.3.1.2 Perimeter Bearings

This design is a landing correction system utilizing bearings mounted to the inner wall of the payload bay was considered as a possible solution. The outer rings of the bearings would be mounted to the interior wall of the payload bay. A single bearing would be mounted on both ends of the rover, the inner rings of which being connected by a sled that would hold the rover in place. The sled and rover would be free to rotate independently of the payload bay during landing. The rover would then be able to drive itself off the sled and continue its mission.

An advantage of the perimeter bearings design is the ability to account for the payload bay rolling on the ground after landing. This system also allows unobstructed space of the rover body which provides maximum space for electronics and the solar array. The main disadvantage to this design is that the large diameter bearings are heavier and would require a thin profile to allow maximum space for the rover to exit.

5.3.1.3 Actuators

This design is a landing correction system utilizing linear actuators to extend sections of the airframe radially outward after landing. This design would require sections of the airframe to be cut and hinged prior to flight that would then be pushed out radially after the payload bay had settled in its final landing position. The hinged sections would actuate one at a time in opposite directions and roll the payload bay into the proper orientation for the rover to drive itself out.

An advantage of this design is no interface is needed between the Landing Correction System and the rover. The rover would have maximum space inside the airframe with the only limiting factor being the inner diameter of the airframe. The main disadvantage of this design is the potential risk of a catastrophic failure during ascent of the launch vehicle. A failure during ascent would not only

render the payload mission a failure, but likely cause major damage to the rest of the vehicle and pose risk to bystanders.

5.3.1.4 Trade Study Categories

Two mandatory requirements were set for this study. The first requirement stated that the design be achievable to create, test, and implement within one season. The second requirement stated that the system reliably ensure that proper orientation of the rover is achieved. The categories considered for the study are listed and explained below in Table 42.

Category	Description
Integration	Ease of integration into the launch vehicle with the rover.
Simplicity of Design	The lack of complex mechanisms that could lead to project failure.
Manufacturability	The ability to manufacture the design with the resources at our disposal.
Affordability	The cost effectiveness of the design.
Possible Effect on Ascent Attitude	Possible effect on the flight of the launch vehicle during ascent.
Payload Weight	Effect of the design on the overall weight of the payload.
Impact on Size of Rover	Impact that the Landing Correction System design dimensions would have on the maximum rover dimensions.

Table 42: Landing Correction System trade study categories.

5.3.1.5 Results

The three designs explained above were compared using the Kepner Tregoe table shown below in Table 43.

Landing Correction Trade Study							
Options:		Center Bearings		Perimeter Bearings		Actuators	
Mandatory Requirements							
Achievable within 1 season		YES		YES		YES	
System will ensure correct orientation of rover		YES		YES		YES	
Categories	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Integration	25.00%	6	1.5	9	2.25	3	0.75
Simplicity of Design	20.00%	7	1.4	9	1.8	2	0.4
Manufacturability	15.00%	8	1.2	10	1.5	1	0.15
Affordability	10.00%	10	1	5	0.5	2	0.2
Possible Effect on Ascent Attitude	10.00%	10	1	10	1	3	0.3
Payload Weight	10.00%	8	0.8	6	0.6	2	0.2

Impact on Size of Rover	10.00%	7	0.7	4	0.4	10	1
Total Score	**100%**		76.00%		80.50%		30.00%

Table 43: Landing Correction System trade study.

The results of the trade study have concluded that the perimeter bearing design will be pursued for the Landing Correction System. This design allows maximum usage of the rover body's internal volume for necessary components and structures. The perimeter bearings are also expected to reliably ensure that proper orientation of the rover is achieved prior to deployment.

5.3.2 Rover Trade Study

The rover will be an autonomous land vehicle that will be capable of being housed inside the airframe of the launch vehicle and will be designed to accomplish all requirements in the statement of work. It has been determined that the design of the wheels for the rover will have a large impact on the design of the body structure of the rover. The rover trade study was performed to determine the design for the wheels, and thus the body. The following four styles were considered for this study: augers, standard wheels, tank treads, and standard wheel/tank tread combination.

5.3.2.1 Augers

This wheel style would use two auger style power screws to control the motion of the rover. The augers would be driven by a pair of motors mounted directly behind each. All other electronic and mechanical systems would then be mounted on top of a sled that would bridge the gap between the two augers. A similar design was created by the team in the 2013-2014 NASA Student Launch Competition as shown in Figure 71.

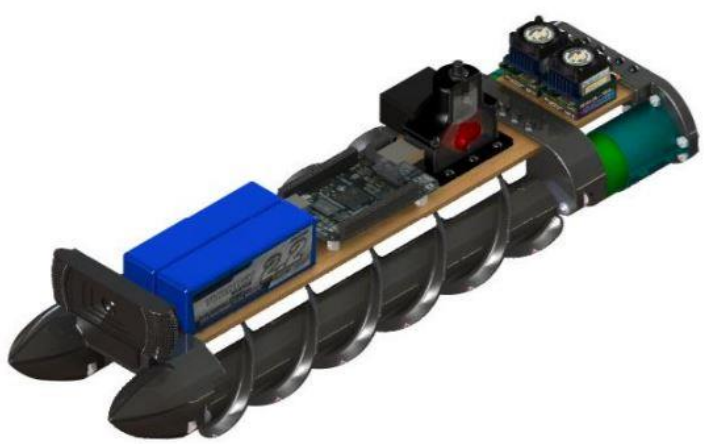


Figure 71: Auger style rover.

The advantages of this design include its ability to perform well in rough terrain, the simplicity of the drive mechanism given that the motor shafts can be directly mounted to the augers, and the potential ease of integration due to the compact, in-line design. Disadvantages to this design include the difficulty in manufacturing the augers and their impact on the weight of the rover.

5.3.2.2 Standard Wheels

This wheel style would utilize three or four standard circular wheels to control the motion of the rover. The wheels would be attached to a body structure that would allow one of the wheels (in a three-wheel design) or two of the wheels (in a four wheel design) to be turned left and right to maneuver the rover during its mission. The wheels would all have a tread design on the outer surface to increase traction. An image of a Lynx Motion 4WD1 Rover Kit is shown below as a reference of a four-wheel design with standard wheels in Figure 72.

Figure 72: Standard wheels style rover.

The advantages of this system include ease of manufacturing, simplicity of control, affordability, and reduction of the weight of the rover. Disadvantages of this style include maneuverability due to a larger turning radius and terrain handling. Standard wheels provide small contact area with the ground and are only able to surmount obstacles that are half the radius of wheel or less.

5.3.2.3 Tank Treads

This wheel style would utilize two continuous track belts mounted on either side of a central rover body structure to control the motion of the rover. All electronic and mechanical systems would be secured in the main body structure between the two tracks. Each track would have a tread design along the outer surface to increase traction. This style is typically used on military tank vehicles and heavy construction vehicles. An image of an M1 Abrams Main Battle Tank is shown below in Figure 73.

Figure 73: Tank tread style rover.

Advantages of this style of rover include ease of integration into the launch vehicle due to its in-line design, high maneuverability due to very tight turning radius, and ability to perform in any terrain. The ability to drive each track in opposite directions allows for high maneuverability. All terrain handling is achieved due to the large surface area of the treads that contacts the ground at any given time. Disadvantages to this style include overall weight of the rover and possible difficulties in manufacturing components for the design.

5.3.2.4 Standard Wheel/Tank Tread Combination

This wheel style would implement a tank tread style drive in the rear of the rover and standard wheels for front of the rover. An advantage to combining the two styles is good terrain handling ability due to the rear treads. Disadvantages of this design include decreased maneuverability and difficulty in manufacturing relative to the other designs, and the weight of the rover due to the parts and components needed for both styles.

5.3.2.5 Trade Study Categories

A mandatory requirement was set for this study that the design be achievable to create, test, and implement within one season. The categories considered for the study are listed and explained below in Table 44.

Category	Description
Integration	Ease of integration of the rover into the launch vehicle.
All Terrain Handling	Ability of the design to allow the rover to perform its mission most effectively in any terrain.
Drive Mechanism/Control Simplicity	Simplicity of the mechanical and electrical systems required to operate the design.
Maneuverability	Ability to turn the rover and continue driving to avoid insurmountable objects.
Payload Weight	Effect of the design on the overall weight of the payload.

Manufacturability	The ability to manufacture the design with the resources at our disposal.
Affordability	The cost effectiveness of the design.

Table 44: Rover trade study categories and descriptions.

5.3.2.6 Results

The four designs explained above were analyzed using the Kepner Tregoe table shown below in Table 45.

Rover Trade Study									
Options:		Augers		Standard Tires		Tank Treads		Treds/Tires Combo	
Mandatory Requirements									
Able to advance rover on multiple terrains		YES		YES		YES		YES	
Categories	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Integration	25.00%	8	2	8	2	8	2	7	1.75
All Terrain Handling	20.00%	8	1.6	4	0.8	10	2	8	1.6
Drive Mechanism/Control Simplicity	20.00%	9	1.8	9	1.8	8	1.6	4	0.8
Maneuverability	10.00%	5	0.5	6	0.6	9	0.9	5	0.5
Payload Weight	10.00%	5	0.5	8	0.8	6	0.6	5	0.5
Manufacturability	10.00%	6	0.6	9	0.9	6	0.6	5	0.5
Affordability	5.00%	6	0.3	9	0.45	7	0.35	6	0.3
Total Score	**100%**	73.00%		73.50%		80.50%		59.50%	

Table 45: Rover trade study.

The results of the rover trade study have determined that a tank tread style rover will be pursued as the rover design. This design will provide the best terrain handling and maneuverability of the designs considered while providing sufficient space for all other systems. This design is expected to ensure that the rover meets all requirements in the Statement of Work.

5.3.3 Deployment Trigger System Trade Study (DTS)

The Deployment Trigger System (DTS) will be responsible for allowing the rover to continue its mission after gaining approval to proceed from the RSO. The system will consist of a transmitter module held behind the flight line by a team member and a receiver module mounted with the payload inside the launch vehicle. The launch vehicle's current design of a carbon fiber airframe poses a challenge for integration of the receiver module due to the high conductivity of carbon fiber restricting a signal from passing through it. The following four designs were considered for

this study: detachment of the antenna, a tether, a forward protruding antenna, and fiberglass airframe.

5.3.3.1 Detachment

The detachment design would allow the rover's receiver module antenna to be securely fastened to the exterior of the airframe and the module mounted inside the airframe. The antenna would either be mounted along the length of the airframe or wrapped around the airframe. After receiving confirmation from the RSO to continue the mission, the deployment signal is sent from the transmitter, and the rover would begin to drive forward and detach itself from the antenna module.

Advantages of this design include the ability to securely mount the antenna, an increased probability of line-of-sight being achieved between the transmitter and receiver, and low potential for damage to the antenna due to rigid mounting. This design is also cost effective due to no extra components being needed to achieve the design besides the module and the antenna. The disadvantage of this design is complex radiation patterns due to the curvature and conductivity of the carbon fiber airframe.

5.3.3.2 Tether

The tether design utilizes a tether inside of which would be electrical wires connected at one end to the rover and at the other end to a home base in the payload bay. Not only the receiver module, but many of the rover's electronics would remain stationary in the home base while the rover performs its mission.

The advantages of this system include reducing the overall weight of the rover itself and the ability to rigidly mount the antenna to the airframe. The mounting of the antenna would be similar to the detachable antenna design explained above in section 1.3.3.1. This increases the probability of line-of sight being achieved between the transmitter and receiver. The disadvantages of this system include the possible effect that the tether would have on the forward motion of the rover, the complex radiation patterns due to the curvature and conductivity of the carbon fiber airframe, and the complexity of integrating both the home base and rover into the payload bay.

5.3.3.3 Forward Protruding Antenna

This design would utilize a rigid or semi-rigid antenna mounted to the front of the rover. After separation of the payload bay at apogee, the antenna would protrude from the now open end of the payload bay. The antenna would remain mounted to the rover as the rover performs its mission.

The advantages of this design include no need to detach from the receiver module after the deployment signal has been acquired and an increase in the probability of line-of-sight being achieved between the transmitter and receiver. The disadvantages of this design include the high probability of the antenna being damaged when the payload bay first makes contact with the ground during recovery and the complications that the design would impose on an obstacle avoidance system also mounted to the front of the rover.

5.3.3.4 Fiberglass Airframe

While carbon fiber restricts signals from passing through it, fiberglass would allow the signal to penetrate. The antenna for the receiver module could then be mounted anywhere inside of the payload bay.

The advantages of this system include the ability to ensure that risk of damage to the antenna is very low, the simplicity of the design, and the ease of integration considering no wires would need be plugged in after the payload has already been inserted into the payload bay. The disadvantage of fiberglass airframe is that it would be a major change to the design of the launch vehicle. In addition to this, it will block line-of-sight between the transmitter and receiver modules. While signals pass through fiberglass well, it still acts as a barrier.

5.3.3.5 Trade Study Categories

A mandatory requirement was set for this study that the design have little to no effect on the design of the launch vehicle. The categories considered for the study are listed and explained below in Table 46.

Category	Description
Integration	Ease of integration of the rover into the launch vehicle.
Barriers to signal	The effect of any barriers that would break line-of-sight between the transmitter and receiver.
Potential for damage to antenna	Probability and severity of damage done to the receiver antenna during nominal flight and recovery of the launch vehicle.
Simplicity of design	The lack of complex mechanisms that could lead to project failure.
Affordability	The cost effectiveness of the design.
Complexity of signal radiation pattern	The level of complexity of the radiation pattern from the receiver antenna.
Effect on motion of the rover	Any effects that may hinder the motion of the rover.

Table 46: Deployment Trigger System trade study categories and descriptions.

5.3.3.6 Results

The four designs explained above were analyzed using the Kepner Tregoe table shown below in Table 47.

Deployment Trigger Trade Study									
Options:		Detach Receiver		Tether		Protruding Antenna		Fiberglass Airframe	
Mandatory Requirements									
Little to no effect on the design of the launch vehicle		YES		YES		YES		NO	
Categories	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Integration	20.00%	7	1.4	7	1.4	6	1.2	10	2
Barriers to signal	20.00%	10	2	10	2	10	2	2	0.4

Potential for damage to antenna	20.00%	8	1.6	8	1.6	2	0.4	10	2
Simplicity of Design	10.00%	8	0.8	5	0.5	9	0.9	10	1
Affordability	10.00%	9	0.9	7	0.7	9	0.9	6	0.6
Complexity of signal radiation pattern	10.00%	5	0.5	5	0.5	9	0.9	7	0.7
Effect on motion of the rover	10.00%	7	0.7	5	0.5	10	1	10	1
Total Score	**100%**	79.00%		72.00%		73.00%		77.00%	

Table 47: Deployment Trigger System trade study.

The results of this study have determined that a detachable antenna design will be pursued as the design for the DTS receiver module. This design will provide the ability to perform consistent tests, high probability of achieving line-of-sight communication between the receiver and transmitter, and low probability of the antenna being damaged during nominal flight.

5.3.4 Foldable Solar Panels Trade Study

The Foldable Solar Panels System will deploy from the rover at the conclusion of its primary mission. The system will consist of a mechanism that will increase the surface area of exposed solar panels. The following four designs were considered: 180 degree flip deployment, tower rotate deployment, tent/origami style deployment, and zig zag deployment.

5.3.4.1 180 Degree Flip Deployment

In the stowed configuration of this design, the solar panels would be folded on top of each other such that the solar cells of each panel face inward toward each other. The panels would then be mounted on or in the body structure of the rover. During deployment of the panels, the solar panels that were on top would be rotated 180 degrees exposing both the panels. A picture of the Elfeland SP-19 Folding Module Monocrystalline Solar Panel is shown in Figure 74.

Figure 74: 180 degree flip foldable solar panels.

Advantages of this system include ease of integration, the simplicity of the mechanism required, and the affordability of the panels and materials required. The main disadvantage is the potentially low total surface area of exposed solar panels after deployment relative to other designs.

5.3.4.2 Tower Rotate Deployment

In the stowed configuration of this design, solar panels would be stacked on top of each other with their solar cells facing upwards. The panels would be mounted to a tower that would initially lay inside the rover's body structure. During deployment, this tower would be raised up along with the solar panels mounted to it providing the panels with adequate clearance from all other components of the rover. A motor would then begin to turn, and unstack the panels from each other. The final deployed configuration would resemble a circle of solar panels mounted on a tower similar to the Smartflower Pacific's Smartflower POP is shown below in Figure 75.

Figure 75: Tower rotate foldable solar panels.

An advantage of this design is its potential for a large increase in exposed solar panel surface area after deployment. Disadvantages of this design include the impact on the payload's overall weight and potential difficulties integrating the tower into the body structure of the rover.

5.3.4.3 Tent/Origami Deployment

Solar panels in this design would be mounted to a flexible canvas. In the stowed configuration, the canvas would be folded in such a way that a motor turning or actuator armature extending would entirely unfold the canvas. During deployment, a motor or actuator would unfold the canvas, and

a support structure under the canvas, exposing the solar panels mounted on it. A rendering of an experimental design for future space craft solar arrays is provided below as a reference in Figure 76.

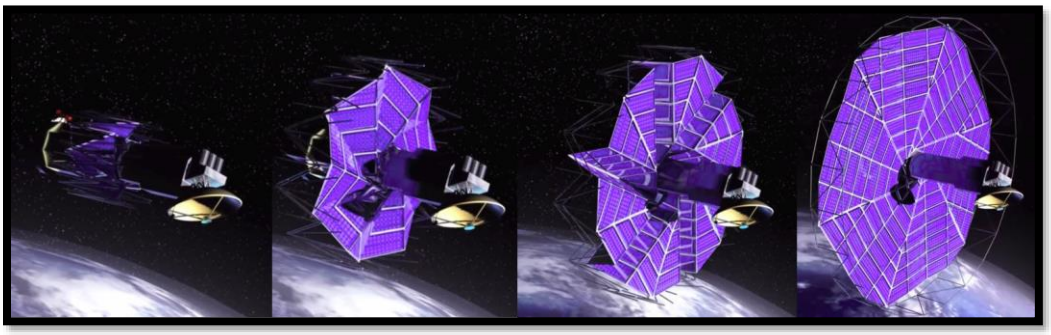


Figure 76: Origami foldable solar panels.

An advantage of this design is the potential for a very large increase in the exposed surface area of solar panels. Disadvantages include the complexity of the origami folds, support structure, consistent unfolding of the canvas, and potential difficulties integrating the design into the body structure of the rover.

5.3.4.4 Zig Zag Deployment

In the stowed configuration of this design, the solar panels would be folded on top of each other such that the solar cells of each panel would be facing the other's. A series of these folds would be mounted on each side of the rover's body. During deployment, the series of folded panels on each side would begin to perform 180 degree flips at the same time extending the structure outward in a zig zag fashion from the rover's body and exposing the solar panels. A picture of the Projecta Folding Solar Panel Kit, a similar design to one used on some current space craft, is shown below in Figure 77.

Figure 77: Zig zag foldable solar panels.

An advantage of this design is the ability to reference spacecraft that utilize this design. A disadvantage is the potential difficulty integrating the design into the body structure of the rover.

5.3.4.5 Trade Study Categories

Two mandatory requirements were set for this study. The first requirement stated that the design be achievable to create, test, and implement within one season. The second requirement stated that the system must satisfy NASA's definition of foldable being that the surface area of exposed solar

panels increases. The categories considered for the study are listed and explained below in Table 48.

Category	Description
Integration	Ease of integration of the rover into the launch vehicle.
Solar Array Area	Potential increase in exposed solar panel surface area.
Simplicity of Design	Simplicity of the mechanical and electrical systems required to operate the design.
Affordability	The cost effectiveness of the design.
Payload Weight	Effect of the design on the overall weight of the payload.
Availability of Useable Panels	Availability to purchase or create solar panels that would fit the design.

Table 48: Foldable Solar Panels trade study categories and descriptions.

5.3.4.6 Results

The four designs explained above were compared using a Kepner Tregoe table shown below in Table 49.

Foldable Solar Panels Trade Study									
Options:		180 Degree Flip		Tower Rotate		Tent Style/Origami		Zig Zag	
Mandatory Requirements									
Achievable within 1 season		YES		YES		YES		YES	
Satisfies NASA requirement of foldable		YES		YES		YES		YES	
Categories	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Integration	25.00%	9	2.25	8	2	5	1.25	6	1.5
Solar Array Area	25.00%	5	1.25	9	2.25	10	2.5	7	1.75
Simplicity of Design	15.00%	8	1.2	7	1.05	3	0.45	7	1.05
Affordability	15.00%	8	1.2	7	1.05	6	0.9	7	1.05
Payload Weight	15.00%	8	1.2	6	0.9	7	1.05	7	1.05
Availability of Useable Panels	5.00%	10	0.5	10	0.5	10	0.5	10	0.5
Total Score	**100%**	76.00%		77.50%		66.50%		69.00%	

Table 49: Foldable solar panels trade study.

The results of the trade study have determined that a tower that will be raised, followed by stacked solar panels being deployed by means of rotation, will be pursued as the design of the foldable solar panels. This design will provide a high gain in exposed solar panel surface area and ensure unobstructed deployment of the solar panels.

5.4 Payload Subsystems

Based on the results of the system level trade studies performed, the following subsystems were derived from the main subsystems. This will provide division of labor and ensure system success. The subsystems and their relation to the main subsystems are shown below in Figure 78. The list of subsystems and an overview of each is shown in Table 50.

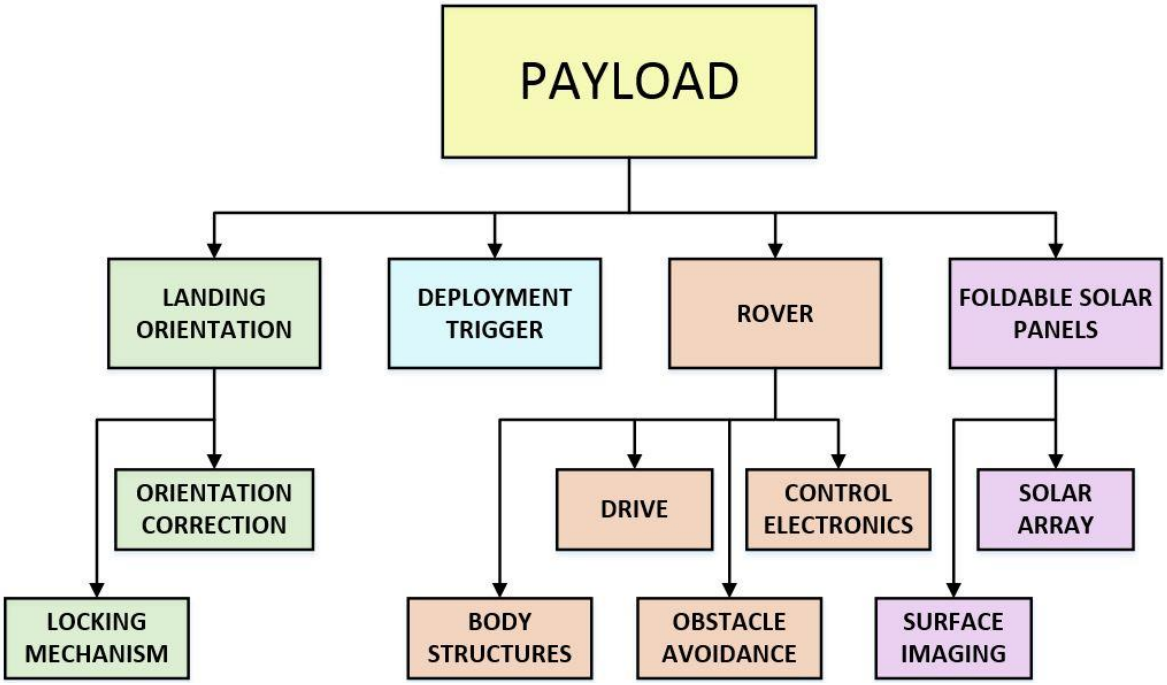


Figure 78: Payload subsystems.

Payload Subsystem	Subsystem Overview
Rover Orientation Correction Systems (ROCS)	The ROCS will be responsible for ensuring proper upright orientation upon landing prior to advancement.
Rover Locking Mechanism (RLM)	The RLM will be responsible for retaining the rover in the launch vehicle for the entirety of the flight.
Deployment Trigger System (DTS)	The DTS will be responsible for sending the deployment signal to the rover upon receiving approval to proceed from the RSO. The signal will then activate the rover's control scheme.
Rover Body Structures (RBS)	The RBS will be responsible for providing structural support for electronics as well as the solar array structure.
Rover Drive System (RDS)	The RDS will be responsible for advancing the rover payload five feet away from the launch vehicle.

Obstacle Avoidance System (OAS)	The OAS will be responsible for the recognition of objects in front of the rover that may hinder forward motion.
Solar Array System (SAS)	The SAS will be responsible for supporting and deploying the foldable solar panels.
Surface Imaging System (SIS)	The SIS will be responsible for collecting image data of the rover and ground area surrounding the payload after deployment of the SAS.
Control Electronics System (CES)	The CES will be responsible for the control scheme that will govern operation of all electronic components of the payload.

Table 50: Payload subsystems and overviews.

5.5 Rover Orientation Correction System (ROCS)

The Rover Orientation Correction System (ROCS), responsible for ensuring correct orientation upon landing of the payload bay, is comprised of an AFT End Thrust Bearing, a FWD End Support Bearing, and a Bridging Sled which supports the rover during flight. The ROCS is displayed fully assembled below in Figure 79.

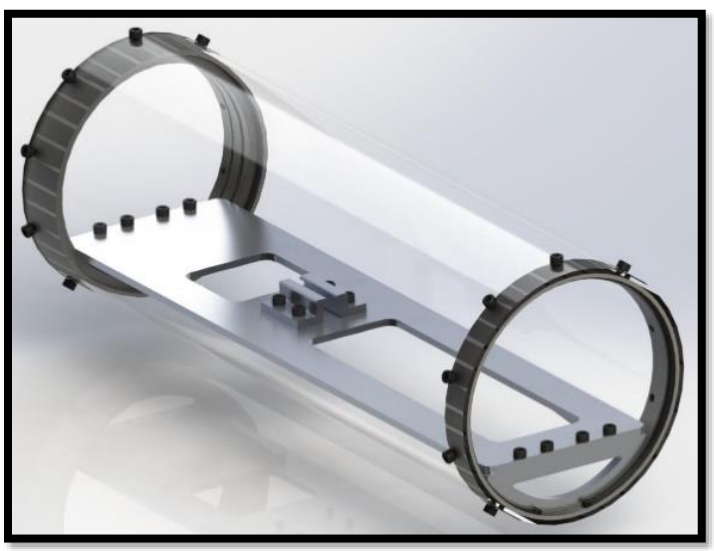


Figure 79: Rover Orientation Correction System.

5.5.1 AFT End Thrust Bearing (AETB)

The primary purpose of the AETB is to support and allow free rotation of the payload, while remaining structurally sound under applied loads experienced during vehicle flight. A bill of materials for the AETB is shown below in Table 51.

Quantity	Description	Material	Purpose

1	Outer Ring	D2 Tool Steel	House AETB components, to directly fasten AETB to airframe, to provide ball bearings a smooth surface to translate through the use of a raceway, to absorb critical forces.
1	Primary Inner Ring	D2 Tool Steel	To allow free rotation of the payload, to support bridging sled and rover, to provide ball bearings a smooth surface to rotate on through the use of a raceway.
1	Secondary Inner Ring	D2 Tool Steel	To provide ball bearings a smooth surface to rotate on through the use of a raceway, to absorb critical forces.
50	0.157 in. Ball Bearing	AISI 1010 Carbon Steel	To reduce friction between rotating components.
80	0.0625 in.Ball Bearing	AISI 1010 Carbon Steel	To reduce friction between rotating components, to prevent primary inner ring from bottoming out on outer ring.
2	Ball Bearing Retention Ring	316 Stainless Steel	To keep ball bearings separated.
1	Snap Ring	Spring Steel	To prevent AETB assembly from separating, to absorb critical forces.

Table 51: AFT End Thrust Bearing components.

An exploded view and section view, shown in Figure 80 Figure 81spectively, of the AETB preliminary design are shown below to display the individual components and method in which the parts will be integrated with one another. the individual components and method in which the parts will be integrated with one another.

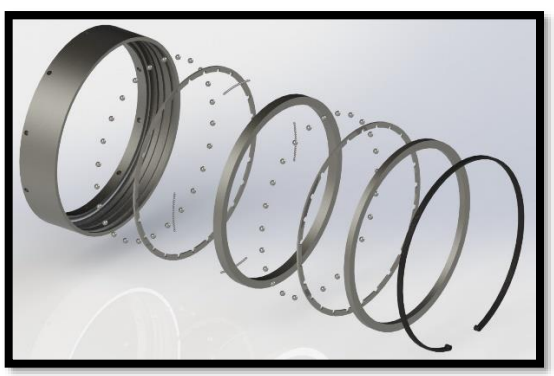


Figure 80: Exploded view of AETB.

Figure 81: Section view of AETB.

5.5.1.1 AFT End Thrust Bearing Design Notes

When the vehicle is in its launch orientation and descending after separation, the AETB will support the majority of the bridging sled and rover masses. The maximum forces the payload will experience during vehicle flight are induced by liftoff and recovery opening force. The AETB is designed to absorb these forces while remaining structurally sound.

The 0.0625 in. ball bearings are spatially separated to prevent the primary inner ring from bottoming out onto the outer ring. When constructing a tangent line connecting the ending and beginning ball bearings of two quarter segments, the connecting tangent line does not intersect the outer diameter of the inner ring with which the 0.0625 in. ball bearings come into contact.

5.5.2 FWD End Support Bearing (FESB)

The primary purpose of the FESB is to assist the AETB by supporting and allowing free rotation of the payload while providing clearance for the rover to exit the launch vehicle. A bill of materials for the support bearing is shown below in Table 52.

Quantity	Description	Material	Purpose
1	Outer Ring	D2 Tool Steel	House FESB components, to directly fasten FESB to airframe.
1	Inner Ring	D2 Tool Steel	To allow free rotation of the payload, to support bridging sled and rover, to provide ball bearings a smooth surface to rotate on through the use of a groove.
100	1/16 Ball Bearings	AISI 1010 Carbon Steel	To reduce friction between rotating components, to prevent primary inner ring from bottoming out on outer ring.
1	Snap Ring	Spring Steel	To prevent FESB assembly from separating.

Table 52: FWD End support bearing materials.

An exploded view and section view, shown in Figure 82 and Figure 83 respectively, of the FESB preliminary design are shown below to display the individual components and method in which the parts will be integrated with one another.

Figure 82: FESB exploded view.

Figure 83: FESB section view.

5.5.2.1 FWD End Support Bearing Design Notes

The FESB is designed with clearance between the inner ring and its translation boundaries: the snap ring and the outer ring. The mounting location of the bridging sled sets the clearance between the inner ring and its translation boundaries along the central axis of rotation. The clearances account for the potential deflection of the crescent mounting plates which the bridge sled mounts to, ensuring that the inner ring walls never come into contact with its boundary walls. By restricting the FESB's inner ring translation along the central axis of the payload bay, the AETB absorbs the majority of any force component along the central axis of rotation. All force components perpendicular to the central axis of rotation are shared between the AETB and FESB.

During flight and under the forces induced by liftoff and recovery opening force, the FESB's clearances mitigate much of the mechanical impact that the AETB experiences. The FESB also reduces the deflection of the bridging sled by adding support on the forward end of the sled. The FESB ball bearings are spatially separated in the same manner as the AETB described above.

5.5.3 Bridging Sled System (BSS)

The primary purpose of the Bridging Sled System (BSS) is to provide a stable surface for the rover to mount to. The BSS also mechanically prevents the rover from translating perpendicular to the central axis of the payload bay. A bill of materials for the Bridging Sled is shown below in Table 53.

Quantity	Description	Material	Purpose
1	Bridging Sled	6061-T6 Aluminum	To provide a stable mounting surface for intended payload, to provide a surface on which the rover tracks can build traction on.
2	Aft End Support Ribs	6061-T6 Aluminum	To provide support on the aft end of the bridging sled to reduce single point bending.

2	Fwd End Support Ribs	6061-T6 Aluminum	To provide support on the forward end of the bridging sled to reduce single point bending.
1	Aft End Crescent Mounting Plate	6061-T6 Aluminum	To allow the mounting of the bridging sled to the AETB primary inner ring.
1	Fwd End Crescent Mounting Plate	6061-T6 Aluminum	To allow the mounting of the bridging sled to the FESB inner ring.
2	Female T-Slot mount	6061-T6 Aluminum	To provide retaining surfaces for male T-slot, to prevent translation perpendicular to the central axis of the payload bay.

Table 53: Bridging Sled System Components.

An exploded view and full assembly model of the BSS preliminary design are shown in Figure 84 and Figure 85 respectively to display the individual components and the method in which the parts will be integrated with one another.

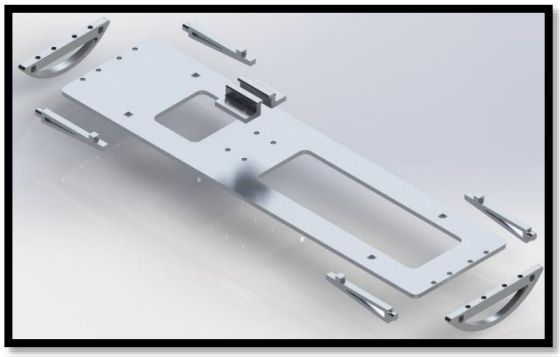


Figure 84: Exploded view of BSS.

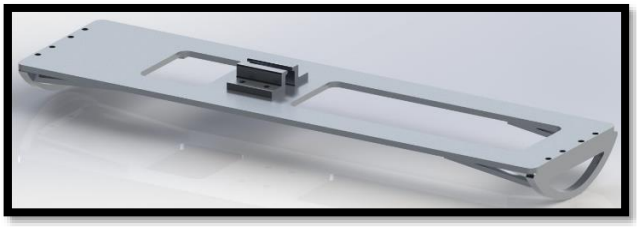


Figure 85: Fully assembled view of BSS.

5.5.3.1 Bridging Sled System Design Notes

The BSS will provide structural support for the rover and provide a rigid surface for the rover's tracks to transfer power to. During vehicle flight, the payload bay will be subjected to forces from multiple directions. While the AETB is designed to absorb the majority of the forces propagating through the central axis of the payload bay, the bridging sled is also impacted by the same forces. The forward and aft end crescent mounting plates will be doweled and epoxied to their respective inner rings on the FESB and AETB. The mounting point will be designed to withstand critical shear forces without yielding.

5.5.3.2 Support Ribs

To reduce static deflection and deflection caused by forces perpendicular to the central axis of the payload bay, forward and aft end support ribs will be used. The support ribs are situated directly under the point of contact between the rover and bridging sled and extend inward towards the center of the bridging sled. In doing so, the distance between the center of gravity of the rover and its nearest rigid support is minimized. By reducing the distance between the center of gravity of the rover and its nearest rigid support, single point bending is reduced. An aft end support rib is shown below in Figure 86.

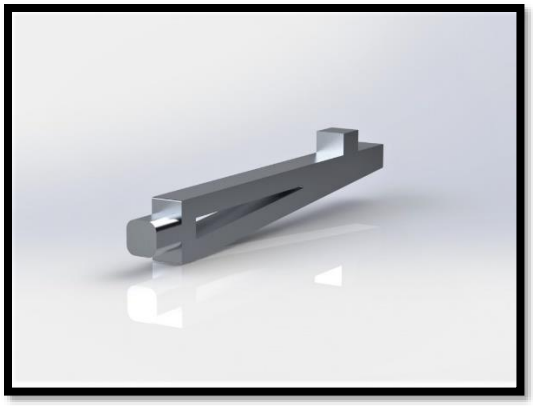


Figure 86: AFT End Support Rib

5.6 Rover Locking Mechanism (RLM)

The Rover Locking Mechanism (RLM) consists of two separate sub-system assemblies that work in conjunction with one another to greatly reduce the translational movement of the rover. The two sub-system assemblies of the RLM, the solenoid with its armature support bracket and female T-slot connection, and the axes of translation are displayed in Figure 87. This axes applies to the rest of the current section.

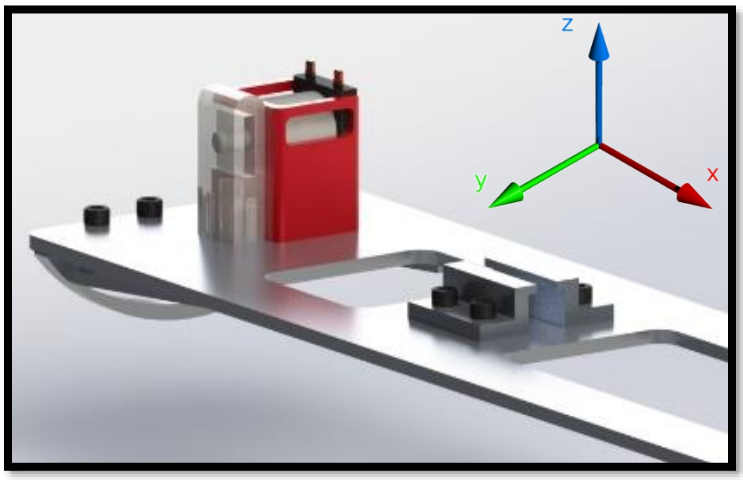


Figure 87: Rover Locking Mechanism fastened to BSS.

5.6.1 Solenoid and Armature Support Bracket

The solenoid and armature support bracket restrict translation along the X, Y, and Z axis of the payload bay when the solenoid arm is fully engaged with the armature support bracket. The solenoid arm, when engaged, restricts movement along the X and Z axis. The armature support bracket restricts movement in the Y and Z direction. A 6061-T6 aluminum latch is fastened with a clearance hole for the solenoid arm that slides between the armature support bracket. When the solenoid arm is engaged, the latch's translational and rotational movement along the X, Y, and Z axis is restricted. Since the latch is directly fastened to the chassis of the rover, the rover is

restricted in the amount it can translate or rotate along the X, Y, and Z axis. The solenoid, armature support bracket, and rover latch are displayed below in Figure 88.

Figure 88: Solenoid, Armature Support Bracket, and Rover Latch assembly

5.6.1.1 Solenoid Design Notes

The solenoid is designed to pull a load rather than push. The armature of the solenoid is .25 in. and made from steel. The solenoid operates on an open and close configuration and controlled by the electronic system onboard the rover.

5.6.1.2 Armature Support Bracket and Rover Latch Design Notes

The clearance hole for the solenoid arm in both the armature support bracket and rover latch measure 0.252 in. A clearance of 0.002 in. minimizes translation and impedes the deflection of the solenoid arm. A clearance of 0.005 in. exists between all four walls of the armature support bracket and rover latch. 0.005 in. of clearance per side effectively holds the rover in place while providing enough clearance to minimize the chances of hanging when the solenoid arm disengages, and the rover attempts to drive out of the payload bay. Both the armature support bracket and rover latch are made of 6061-T6 aluminum.

5.6.2 Male T-slot nut and Female T-slot

The female T-slot brackets situated on the bridging sled will work in conjunction with the male T-slot nut fastened to the chassis of the rover. When the male T-slot nut and female T-slot are engaged, translation of the rover in the direction of the Y and Z axis will be minimized. A front view of the preliminary design of the male T-slot nut engaged with the female T-slot is shown below in Figure 89.

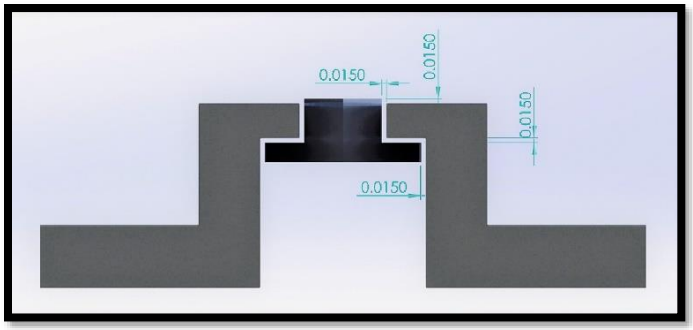


Figure 89: Male T- slot nut and Female T-slot engagement.

5.6.2.1 Male T-slot Nut and Female T-slot Design Notes

The T-slot assembly will aid the solenoid and armature mechanism by adding a second point of contact closer to the payloads forward end, where the translation and rotation of the payload along the Y and Z axis will be restricted. The stress and bending moment induced on the armature support bracket and rover latch is greatly reduced through the minimized rover chassis displacement in the Z direction. The T-slot assembly will allow the rover to translate along the X direction while maintaining a clearance of 0.015 in. to prevent catching on the female T-slot bracket. Both male and female T-slots are made of 6061-T6 aluminum.

5.6.3 RLM Safety

Sufficient factors of safety must be achieved for all parts of the system as the RLM will be responsible for retaining the rover in the launch vehicle during descent. Factor of safety verifications will be achieved through further design analysis and rigorous testing. Design changes to the RLM will be driven by the intent to reduce risk of failure.

5.7 Deployment Trigger System (DTS)

The Deployment Trigger System (DTS) will consist of a receiver module and antenna that will be responsible for allowing the team to deploy the rover after gaining RSO permission to proceed. The receiver module will reside in the payload recovery bay coupler. Wires will pass through the bulk plate into the payload bay and connect, via a pull-apart mechanism, into a software serial port running on the CES control board described in section 5.13.2.1.

5.7.1 Receiver Module Trade Study

A receiver module will be needed to receive the radio signal sent by a team member to trigger deployment of the rover. The following three options were considered to serve as the receiver module: HC-12, XBee-Pro 900 XSC S3B Wire, RFD900+ Telemetry Radio.

5.7.1.1 Trade Study Categories

Two mandatory requirements were set for this trade study. The first requirement stated that the receiver module be capable of receiving a signal sent from at least 2500 ft. away. This is to account for the maximum allowable recovery range of the launch field. The second requirement stated that

the receiver communicate to the control board via a serial protocol available on the control board. The categories considered for this study are listed and described below in Table 54.

Category	Description
Max Range	Maximum reception range achievable.
Ease of Communication	Ease with which the communication between the receiver module and control board can be achieved.
Affordability	The cost effectiveness of the module.

Table 54: Receiver modules trade study categories.

5.7.1.2 Results

These wireless modules were compared using the Kepner Tregoe table shown below in Table 55.

Wireless Modules Trade Study							
Options:		HC-12		XBee		RFD 900+	
Mandatory Requirements							
2500 feet range		YES		YES		YES	
Serial communication protocol		YES		YES		YES	
Categories	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Max Range	40.00%	6	2.4	8	3.2	9	3.6
Ease of Communication	30.00%	9	2.7	6	1.8	8	2.4
Affordability	30.00%	9	2.7	7	2.1	5	1.5
Total Score	100.00%	**78.00%**		**71.00%**		**75.00%**	

Table 55: Wireless module trade study.

Based on these results, we will design the Deployment Trigger System with the intent of using the HC-12 transmitter/receiver module. The transmitter will be held by a team member behind the flight line and will not cross it unless given permission by the RSO to do so.

5.7.2 Pull-apart Mechanism Trade Study

Upon receiving the deployment signal, the rover will begin to drive forward and exit the launch vehicle. The wires connecting the receiver module to the control board will disconnect as they are not needed after the signal is received and would otherwise hinder forward motion of the rover. The pull-apart mechanism will be responsible for ensuring ease of detachment of these wires.

A trade study was performed to determine the preferred design. The following three designs were considered to act as the pull-apart mechanism: jumper wires, 3.5mm jack, magnetic connector.

5.7.2.1 Jumper wire

Male jumper wires coming from the receiver module would connect into female jumper wires securely fixed to the rover body. This connection provides flexibility in wire configurations, low cost, and low separation force. However, this low separation force increases the risk of premature detachment. Jumper wires are shown below in Figure 90.

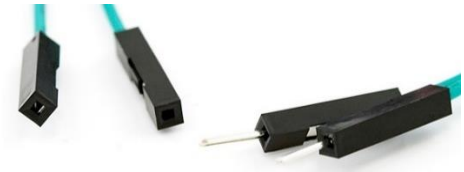


Figure 90: Jumper wires.

5.7.2.2 3.5mm Jack

3.5mm jacks are commonly used as audio connectors. These jacks are capable of four wire connection which corresponds to the number of wires needed for the receiver module to connect to the control board. This design provides a very compact and reliable connection method. However, this design has a high separation force required to pull apart male and female connectors presenting the risk of impeding the forward motion of the rover. A female and male 3.5mm jack are shown below in Figure 91.

Figure 91: Male and female 3.5mm jack.

5.7.2.3 Magnetic Connector

A set of magnetic connectors would be secured to the rover and the ROCS. The connectors would provide reliable connection and a low separation force can be achieved. A magnetic wire connector is shown below in Figure 92.

Figure 92: Four wire magnetic connector.

5.7.2.4 Trade Study Categories

Two mandatory requirements were set for this trade study. The first requirement stated that the connection method be capable of four wire connections. The second requirement stated that the connector be capable of disconnecting solely due to a pulling force with no unlatching required. The categories considered for this study are listed and described in Table 56.

Category	Description
Reliability	Ability to resist twisting and vibrations without disconnecting.
Ease of removal	Ease with which the rover can pull apart the connector.

Ease of connection	How easily the wires can be connected correctly during vehicle assembly.
Flexibility	Ability to reconfigure wires.
Affordability	Cost effectiveness of the design.

Table 56: Connector design trade study categories.

5.7.2.5 Connector Types Trade Study

Connector types were compared using the Kepner Tregoe table shown below in Table 57.

Connector Types Trade Study							
Options:		Jumper Wires		3.5mm Jack		Magnetic Connector	
Mandatory Requirements							
At least 4 separate wires		YES		YES		YES	
Can be unplugged by pulling		YES		YES		YES	
Categories	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Reliability	30.00%	6	1.8	9	2.7	8	2.4
Ease of removal	30.00%	9	2.7	6	1.8	8	2.4
Ease of connection	15.00%	5	0.75	9	1.35	8	1.2
Flexibility	15.00%	8	1.2	6	0.9	6	0.9
Affordability	10.00%	9	0.9	7	0.7	5	0.5
Total Score	100.00%		**73.50%**		**74.50%**		**74.00%**

Table 57: Connector type trade study.

Based on the results of the trade study, the pull-apart mechanism will be designed with the intent of using a 3.5mm jack connector. Testing will be done to verify that separation force of this jack can be exceeded by the rover.

5.7.3 Antenna Design

The antenna for the receiver module will be securely mounted to the airframe of the payload bay per result of the DTS trade study performed in section 5.3.3. Inherent complications are imposed by mounting the antenna to the exterior of the launch vehicle. Among these complications are the following:

- Unpredictability of landing orientation of the payload bay.
- Carbon fiber acting as a large conductive body that the antenna will be mounted to.
- The close proximity to the ground affecting signal radiation patterns.

Omnidirectionality is required to mitigate the complication of unreliable landing orientation of the payload bay. The antenna will experience ground losses due to the carbon fiber body and close proximity to the ground. Five designs considered for the antenna configuration were: multiple parallel dipole antennas, open loop antenna, slot antenna, spiral antenna, and multiple perpendicular monopole antennas.

5.7.3.1 Multiple Parallel Dipole Antennas

Multiple parallel dipole antennas would require three dipole antennas running parallel to the airframe mounted 120 degrees apart from each other. This will ensure that at least one antenna will be orientated such that line of sight could be achieved between the receiver and transmitter independent of the orientation of the payload bay. This design is represented below in Figure 93.

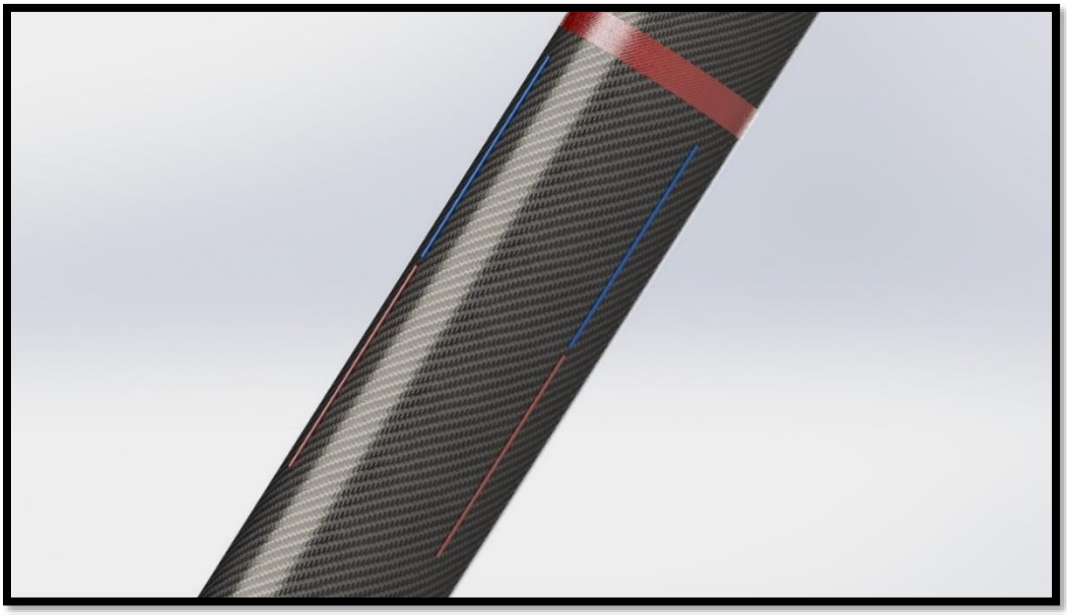


Figure 93: Multiple parallel dipole antennas.

5.7.3.2 Open Loop Antenna

In this design, the active wire and ground wire wrap around the circumference of the airframe in opposite directions and nearly meet on the opposite side. The properties of such an antenna depend greatly on location along the airframe and orientation relative to the ground. This design is represented below in Figure 94.

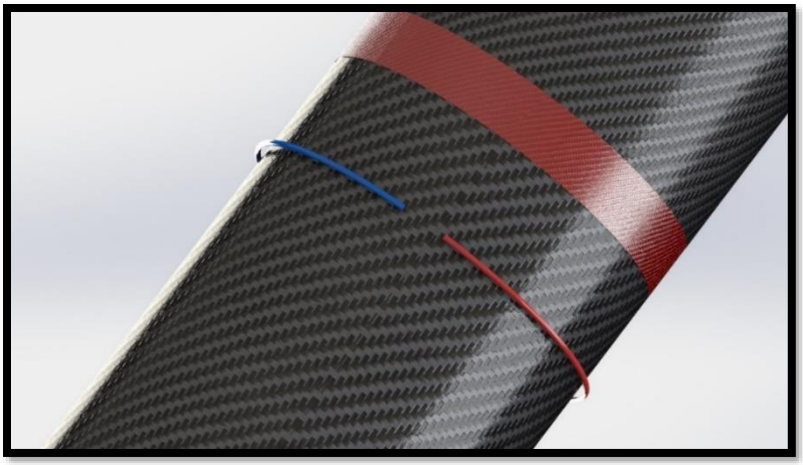


Figure 94: Loop antenna.

5.7.3.3 Slot Antennas

This design uses copper sheet cut to form a slot antenna that can be wrapped around the exterior of the airframe. This would provide large surface area for signal reception and negate the carbon fiber conductive body effects due to a thin insulation layer applied between the antenna and airframe. The slot antenna design is represented below in Figure 95.

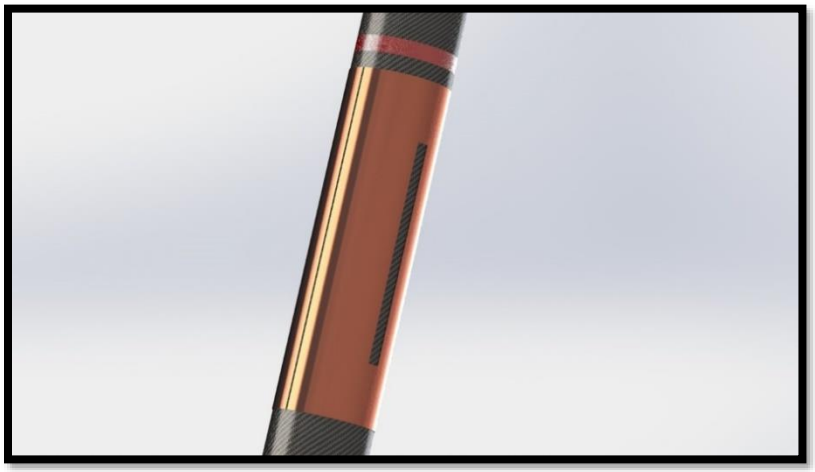


Figure 95: Slot antenna.

5.7.3.4 Spiral Antenna

This design utilizes a single antenna helically wrapped around the circumference of the airframe. Performance characteristics of this antenna design are not currently known and require simulation and testing to more fully determine its validity. A rendering of a spiral antenna with ground loop is shown in Figure 96.

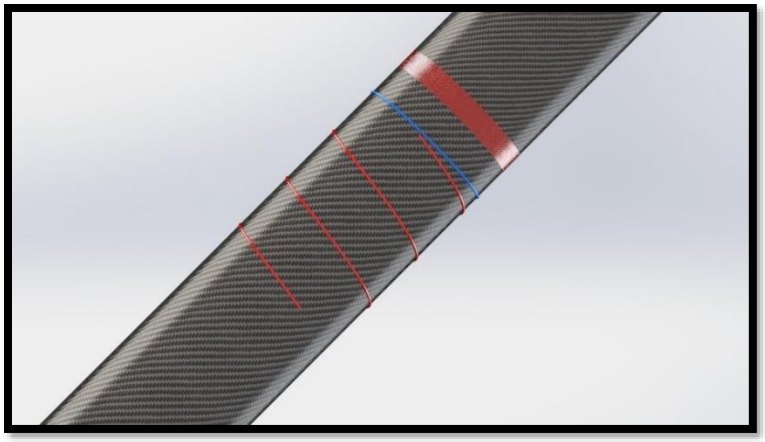


Figure 96: Spiral antenna.

5.7.3.5 *Multiple Perpendicular Monopole Antenna*

A monopole antenna perpendicular to the airframe would provide the best performance and simplicity, but would cause large drag affecting the flight of the launch vehicle. Additionally, it would likely be damaged severely on landing.

5.7.3.6 Further Research Required

As this is a unique application of different antenna designs, ANSYS simulations and physical testing is required to further understand the performance of the antenna configurations and determine the most optimal design for the DTS.

5.8 Rover Body Structure (RBS)

The RBS will be responsible for providing support for all the components in the rover including the rover drive system. It will also serve as the electronics bay of the rover. The RBS is shown below in Figure 97.

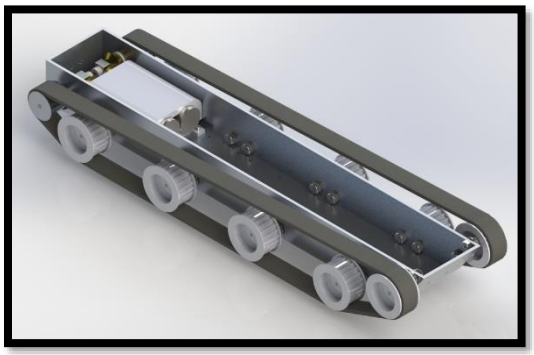


Figure 97: RBS rendering.

5.8.1 Rover Body Structure Trade Study

Three materials were considered for the material of the rover body. These three materials were wood, carbon fiber, and aluminum sheet metal. All three considerations were made with the assumption that the rover body will have a thickness of 0.100 in.

5.8.1.1 Wooden Rover Body

A wooden rover body would be composed of seven laser cut wooden panels that would then be assembled together using adhesive. A jig would have to be configured to ensure that the panels are perfectly aligned. The advantages of this solution are that it would be a low-cost and low-weight structure. The disadvantages of this solution are that it has very low structural strength and the need to construct a jig for perfect alignment.

5.8.1.2 Carbon Fiber Rover Body

A carbon fiber rover body would be composed of seven water jetted panels that would be assembled using adhesive. This solution would also need a jig to insure the correct alignment of the panels. The advantages of this solution are that it would be a high-strength and low-weight solution. The disadvantages of this solution are its affordability and the necessity of a jig for alignment.

5.8.1.3 Aluminum Rover Body

An aluminum sheet metal rover body would be a single sheet of aluminum that is water jetted and would be bent using a press brake to create the sides. The sides would then be welded together. The advantages of this solution are that it would be a high-strength and low-cost solution. This solution also would not need any jig for alignment since it will be a continuous sheet formed by a CNC press brake. The disadvantage of this solution is that it would add more weight to the payload relative to the other two considerations.

5.8.1.4 Trade Study Categories

A mandatory requirement was set for this trade study. The requirement stated that the design and fabrication of the method used by achievable within a single season. The categories considered for this study are listed and described below in Table 58.

Category	Description
Integration	Ease of integration of mounts and fasteners with the Rover Body Structure.
Structural Strength	The modulus of elasticity of the material.
Manufacturability	The ability to make the body structure with materials and machinery that the team has access to in a timely manner.
Affordability	The cost effectiveness of the design.
Payload Weight	The effect on the overall weight of the payload.
Availability of Materials	Ease of acquisition of materials in a timely manner.

Table 58: Rover Body Structures trade study categories.

5.8.1.5 Results

The designs explained above were analyzed using the Kepner Tregoe table shown below in Table 59.

Rover Body Structure Trade Study							
Options:		Wood		Carbon Fiber		Aluminum Sheet	
Mandatory Requirements							
Achievable within 1 season		YES		YES		YES	
Categories	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Integration	20.00%	5	1	6	1.2	10	2
Structural Strength	25.00%	2	0.5	10	2.5	6	1.5
Manufacturability	20.00%	10	2	7	1.4	8	1.6
Affordability	15.00%	10	1.5	1	0.15	10	1.5
Payload Weight	15.00%	10	1.5	7	1.05	5	0.75
Availability of Material	5.00%	10	0.5	1	0.05	10	0.5
Total Score	**100%**	65.00%		63.00%		73.50%	

Table 59: Rover body structure trade study.

From the results of the trade study it was concluded that an aluminum sheet rover body structure is the best option. The main factors behind this result were the aluminum sheets affordability, availability, and integration. The fact that the entire rover body structure, when made from aluminum, can be a single sheet that is water-jetted with precise holes for mounts makes this option the most advantageous.

5.8.2 Electronics Mounts

The main electronics controlling the rover and its functions will be secured to the rover by directly fastening them to the body with screws or custom designed mounts. The mounts are designed to secure the electronics during flight.

5.8.2.1 Motor Battery and Controller Battery Mount

A battery mount will be 3D printed to secure the motor battery and the controller battery described in section 5.13.4.6 and 5.13.2.7 respectively. The battery mount will be attached to the RBS by four 4-40 screws. The lid of the battery mount will be secured to the top by five 0-80 screws. A rendering of this mount is shown below in Figure 98.

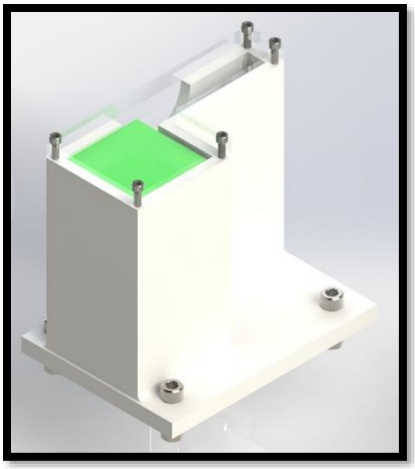


Figure 98: Battery mount.

5.9 Rover Drive System (RDS)

The primary purpose of the RDS is to translate the rover at least five feet from the launch vehicle. A bill of materials for the RDS is shown below in Table 60.

Quantity	Description
2	Planetary Gear Drive Motor
1	Motor Mount
2	4mm Stainless Steel Shaft
4	4mm Bore Shaft Mount Bevel Gear
2	T5 Series Timing Belt Pulley
1	Bushing
2	Delrin Washer
2	4-40 Socket Head Cap Screws
2	4mm Shaft Collar
2	Polyurethane T5-DL Timing Belt
10	Delrin T5 Series Passive Pulley
10	Stainless Steel Two-Bolt Flange-Mounted Ball Bearing

Table 60: Rover Drive System Components.

5.9.1 Rover Drive System Trade Study

Two designs were considered for the Rover Drive System. The two designs consisted were a belt driven system and a chain driven system.

5.9.1.1 Belt Drive System

A belt driven system was researched as an option for the drive train. This system would utilize a pulley and polyurethane belt to drive the rover. A belt driven system would provide high traction due to rubber's high coefficient of friction on most materials that the rover could possibly drive on. A belt system would also provide the largest clearance between the ROCS and the rover due to the availability of small width belts. The decrease in width allows for increased height as the

restricting geometry of the ROCS is a circle. This will allow for easy integration into the ROCS. Another advantage of the belt drive system is that it minimizes the amount of moving parts in the system which makes it more simplistic and reliable.

5.9.1.2 Chain Drive System

A chain drive system was considered as an option for this system. A chain driven system would utilize a sprocket and roller chain to drive the rover. An advantage to this system is that customized treads could be attached to the roller chain. The disadvantages of this design are that it would add excessive weight and size to the payload. Another disadvantage of this system is the possibility of the chain derailing from the sprocket which would lead to the failure of the mission.

5.9.1.3 Trade Study Categories

A trade study was conducted to determine which option the team should pursue. One mandatory requirement was set for this study. The mandatory requirement stated that the option must be achievable to design and manufacture within one season. The categories are listed and described below in Table 61.

Category	Description
Integration	How much clearance the drive system will have with the ROCS.
Reliability	The likelihood that the system will perform successfully.
Mechanism Simplicity	The overall simplicity of the design of the drive system.
Reusability	The ability of the drive system to be used multiple times.
Cost	The cost effectiveness of the design.
Weight	The effect on the overall weight of the payload.

Table 61: RDS Trade Study Categories.

5.9.1.4 Results

The four designs explained above were analyzed using the Kepner Tregoe table shown below in Table 62.

Drive System Trade Study					
Options:		Chain Drive		Belt Drive	
Mandatory Requirements					
Achievable within 1 season		YES		YES	
Categories	**Weights**	**Value**	**Score**	**Value**	**Score**
Integration	40.00%	5	2	7	2.8
Reliability	20.00%	4	0.8	7	1.4
Mechanism Simplicity	15.00%	5	0.75	9	1.35
Reusability	5.00%	5	0.25	8	0.4
Cost	10.00%	3	0.3	5	0.5

Weight	10.00%	3	0.3	6	0.6
Total Score	**100%**	44.00%		70.50%	

Table 62: Drive system trade study.

Based on the results of the trade study, the team concluded that a belt driven system was the best option to pursue. In all categories, the belt drive system was superior to a chain driven system.

5.9.2 Rover Track Design

The belts that will drive the rover will be T5 double sided timing belts that have a pitch of 0.197 in. and width of 0.63 in. This belt was chosen because of its compact design and availability. Using a belt with teeth minimizes the possibility of the belt slipping. Having teeth on the outside of the belt was chosen to improve traction on all terrains. The belt will be made of polyurethane type TPU-ST2 which has a shore hardness of 85 A. The coefficient of friction of this belt on polished steel is 0.77, which is similar to rubber's coefficient of friction on polished steel. Polyurethane was chosen for its high wear resistance and flexibility. The timing belt is shown below in Figure 99.

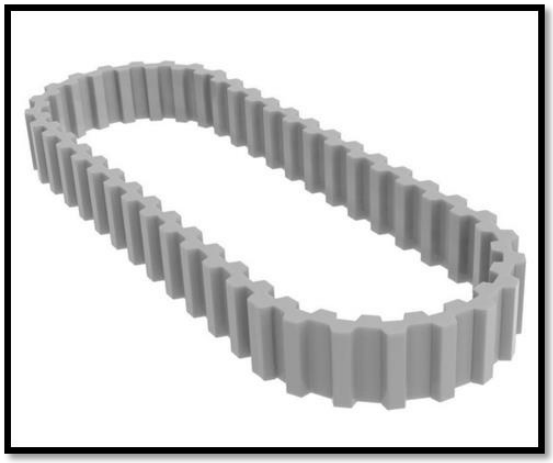


Figure 99: Rover Track Design.

5.9.3 Pulley Configuration

Pulleys will support the T5 timing belts. The pulleys will be configured in a manner to maximize the climbing ability of the rover. Each configuration will be composed of six pulleys. The forward most pulley will be elevated to produce a slope that will improve the rover's ability to overcome obstacles. An illustration of the pully configuration is shown in Figure 100.

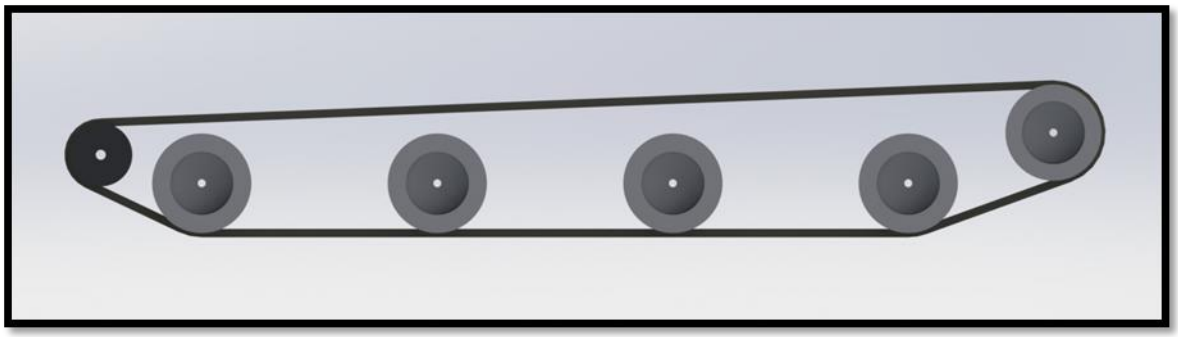


Figure 100: Pulley configuration.

5.9.3.1 Drive Pulley Design

The drive pulleys will be responsible for driving the timing belt. The drive pulleys that were chosen, have a 0.197 in. pitch to match the timing belts. These pulleys are made of machined aluminum and are designed to be mounted via press fit. An illustration of the drive pully is shown in Figure 101.

Figure 101: Drive pulley design.

5.9.3.2 Passive Pulley Design

The passive pulleys will be responsible for keeping the timing belts aligned during operation. These pulleys will be machined out of Delrin and will have a 5mm pitch to match the timing belts. These pulleys will be mounted via press fit to bearings that will be mounted to the outside of the rover body. An illustration of the passive pully design is shown in Figure 102.

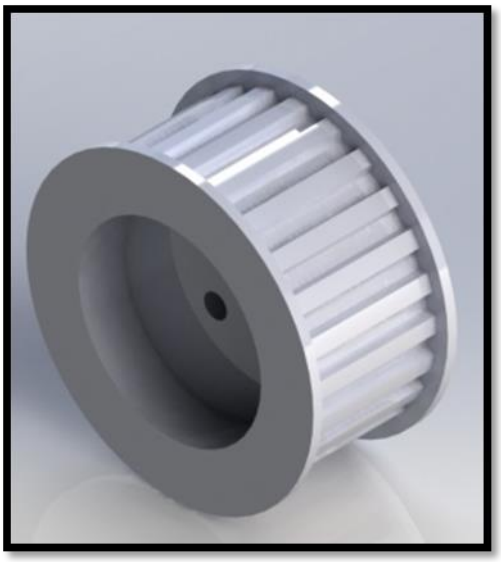


Figure 102: Passive Pulley Design.

5.9.3.3 Pulley Bearings

The bearings will be stainless steel two-bolt flange-mounted ball bearings. These bearings will be mounted to the outside of the rover body and will be responsible for allowing the passive pulleys to rotate freely. The shaft connecting the pulleys to the bearings is a 0.125 in. diameter, 0.75 in. long steel. The shaft will be press fit to the bearings. The bearings have a dynamic radial load capacity of 125 lbs. An illustration of the bearings is shown in Figure 103.

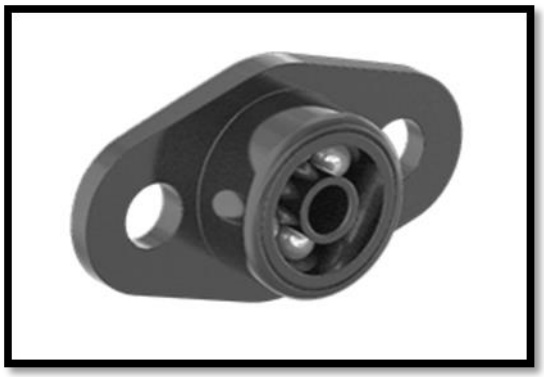


Figure 103: Pulley Bearing.

5.9.4 Main Drive Motors

The two main drive motors will serve as the source of propulsion for the rover. The motors will be mounted in the rear of the rover to drive the center of gravity of the rover towards the rear. For the purposes of this mission, maintaining forward motion is more valuable than the speed of the rover and therefor a motor with low RPM and high torque should be chosen. The main drive motors

have been selected from a line of premium planetary gear motors. The motor is shown below in Figure 104 followed by specifications in Table 63.

Figure 104: Actobotics planetary gear motor.

Actobotics Planetary Gear Motor Relevant Specifications	
Characteristics	
Technical Dimensions	
Total Weight (lbs)	0.22
Shaft Diameter and Length (in.)	Ø0.157 x 0.602
Motor Dimensions (in.)	Ø0.866 x 2.95
Operation	
Operative Voltage	12V
RPM	52
Stall Torque (ft-lb)	1.52

Table 63: Main drive motor specifications.

5.9.5 Bevel Gears

The primary function of the bevel gears is to transfer the rotational motion of the drive motors 90 degrees to the shaft of the drive pulleys. This function will allow for mounting the drive motors parallel to the drive pulley and for a more compact design of the overall rover. The bevel gear configuration is shown in Figure 105.

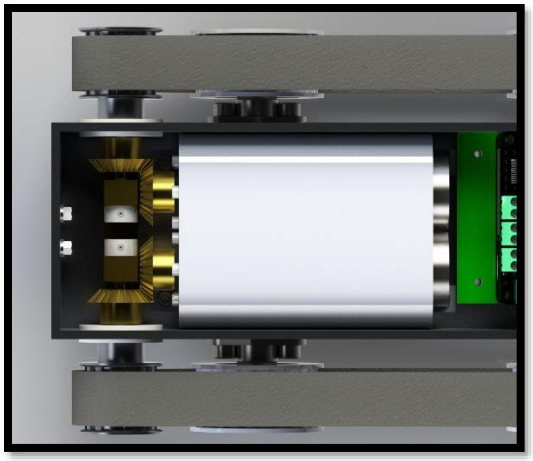


Figure 105: Bevel gear configuration.

5.9.6 Motor Mount

The motor mount will be responsible for retaining the motors during flight and while driving the rover. The motor mount will be machined out of Delrin and mounted to the bottom of the RBS by four 4-40 bolts. Each motor will be mounted to the front of the motor mount by four M2 screws. The motor mount is shown below in Figure 106.

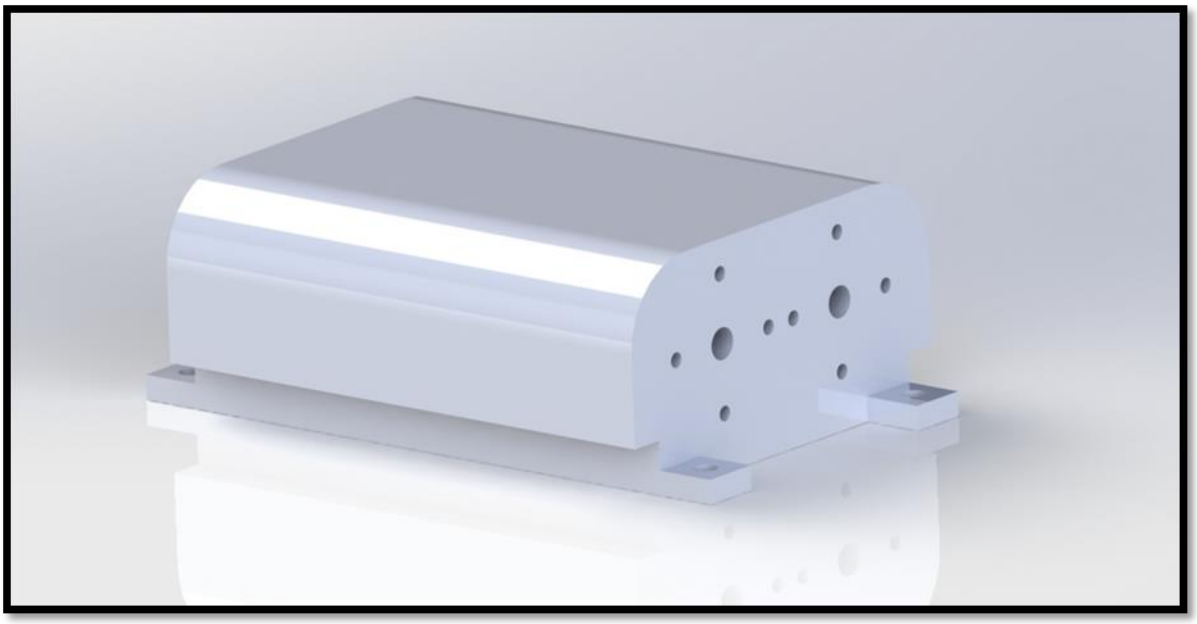


Figure 106: Rover motor mount.

5.9.7 Rover Drive System Bracket

The RDS bracket will be responsible for supporting both drive pulley axels. This bracket will be custom machined out of brass to ensure minimal wear and high rigidity. The bracket will be mounted to the rover body via two 4-40 bolts. The bracket is shown below in Figure 107.

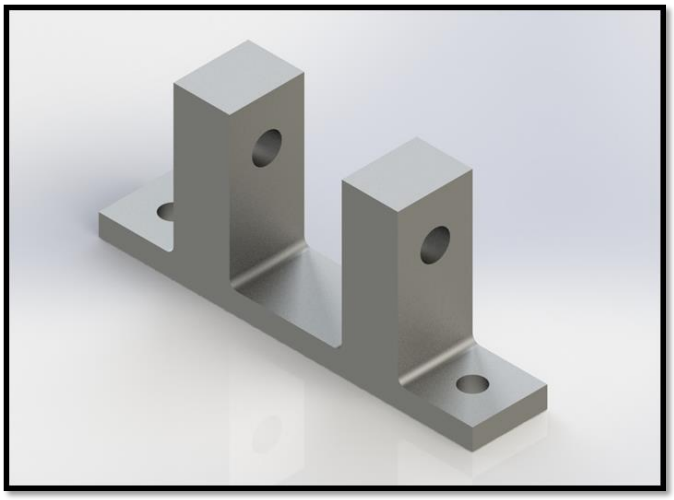


Figure 107: RDS Bracket.

5.10 Obstacle Avoidance System (OAS)

The Obstacle Avoidance System will be responsible for determining if there is an object in the path of the rover that will stop the rover from reaching its destination of five feet away from the payload bay. This will be achieved by using a distance sensor on the front of the rover, sweeping the sensor from left to right giving it a field of view, and analyzing the data to determine if an obstacle taller than the rover is in the immediate path of the rover. If the data concludes that an obstacle is present, the Control Electronics System control scheme will determine a course of action as described in section 5.13.1. This section will outline the operation and selection of devices for the OAS.

5.10.1 Distance Sensor Selection

The OAS will use a distance sensor with sufficient range and accuracy to detect objects in the immediate path of the rover within the distance left to travel in order for the rover to achieve its final destination. Three general options were considered for the type of distance sensor to be chosen: Lidar sensors, IR range sensors, and sonar sensor. A description of each type of sensor and justification for the decision to use or not use that type is outlined below.

5.10.1.1 IR Range Sensors

IR sensors emit a field of infrared radiation and look for fluctuations in the return field. As opposed to a focused narrow emission, IR sensors provide based on a cone of emission. These sensors operate well at short range; however, their output is highly non-linear as distance from the target increases. This leads to much more difficult data analysis techniques. For these drawbacks, an IR sensor is not a good choice for the OAS distance sensor.

5.10.1.2 Sonar Sensors

Sonar is an acronym for “sound navigation and ranging.” This type of sensor emits ultrasonic sound waves and measures the sound that is reflected back to the sensor. Sonar is typically used in marine applications as the sound waves attenuate less traveling through water. Being that these

sensors operate based on reflected sound waves, they are high susceptible to interference in open air applications. For this reason, a sonar range sensor is not a good choice for the OAS distance sensor

5.10.1.3 Lidar Sensor

Lidar is a combination of the terms “light” and “radar” and is commonly used as an acronym for light detection and ranging. This method utilized a very finely focused invisible laser source and matched sensor to determine the distance between two objects. Lidar sensors have very narrow field of view, but high accuracy for short and long distances in ambient and low light. The data stream fed back from the sensor can be done at a very high rate and no conversions are necessary to determine distance. This fits the needs of the OAS distance sensor.

5.10.2 Lidar Sensor Trade Study

Two lidar sensors were considered to act as the OAS distance sensor. The two sensors were the VL53L0X and VL6180X. A trade study was conducted to determine the optimal lidar sensor for the OAS.

5.10.2.1 Trade Study Categories

A mandatory requirement was set for this trade study. The requirement stated that the sensor must be capable of detecting objects within five feet of the sensor. The categories considered for this study are listed and explained below in Table 64.

Category	Description
Accuracy	Percentage error associated with the accuracy of the sensor.
Range	The typical range that the sensor can detect objects in with acceptable error.
Power Consumption	Power draw from the module during normal operation.
Affordability	The cost effectiveness of the sensor.

Table 64: Lidar sensor trade study categories.

5.10.2.2 Results

The four designs explained above were analyzed using a Kepner Tregoe table which is shown below in Table 65.

Lidar Sensor Trade Study					
Options:		VL53L0X		VL6180X	
Mandatory Requirements					
Can detect objects within five feet of sensor		YES		YES	
Categories	**Weights**	**Value**	**Score**	**Value**	**Score**
Accuracy	30.00%	7	2.1	7	2.1
Range	30.00%	9	2.7	4	1.2

Power Consumption	20.00%	4	0.8	9	1.8
Affordability	20.00%	8	1.6	9	1.8
Total Score	**100%**	72.00%		69.00%	

Table 65: Lidar sensor trade study.

The VL53L0X has been selected as the distance sensor of the Obstacle Avoidance System due to its longer range. This has been determined due to the fact that nearly all aspects of the sensors are the same except the cost and range. The cost difference to achieve a much greater range is very low leading to the decision of the sensor with longer range being selected. The sensor is shown below in Figure 108 followed by specifications in Table 66.

Figure 108: VL53L0X lidar sensor.

Adafruit VL53L0X Lidar Sensor Relevant Specs	
Characteristics	
Technical Dimensions	
Total Weight (lbs)	0.0029
Dimensions (in.)	0.8 x 0.7 x 0.1
Modules	
Accuracy	Outdoor: 6% - 9%
Lidar Range (in.)	1.97 - 47.24

Table 66: Lidar sensor specifications.

5.10.3 Field of View

Due to the finely focused laser emitted by the lidar, a wider field of view is desired to maximize the data collected without needing to use the main drive motors to turn the rover for the lidar to see a different view if an obstacle is detected. To achieve this field of view, the sensor will be mounted on top of a small servo motor that can be controlled by the CES. When the sensor detects an object in the immediate path of the rover, the small servo will rotate the sensor from left to right 180 degrees to determine if a path with no obstacle exists within the sensors field of view. The field of view to be achieved is displayed in blue below in Figure 109.

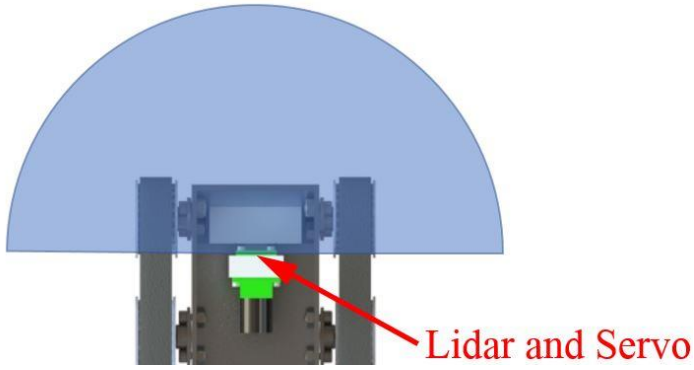


Figure 109: OAS field of view.

5.10.3.1 OAS servo

The OAS servo motor will be mounted to the front of the rover with sufficient clearance from other rover structures allowing it to rotate the sensor freely. Four small servo motors have been considered for the OAS servo motor. TowerPro servos have both a small package and are easily integrated with the CES control board described in section 5.13.2.1 using readily available libraries.

5.10.3.2 OAS servo Trade Study

A trade study was conducted to determine the motor to be selected. The following four TowerPro servo options were considered for the motor: SG92R, SG-5010, MG92B, and SG51R.

5.10.3.3 Trade study Categories

A mandatory requirement was set for this trade study. The requirement stated that the motor be capable of 180 degrees of rotation. The categories considered for the study are listed and explained below in Table 67.

Category	Description
Size	Overall size of the microcontroller board.
Weight	Effect of the microcontroller on the overall weight of the payload.
Operating Voltage	The recommended voltage to power the motor and its relation to the voltage output of the CES control board.
Affordability	The cost effectiveness of the sensor.
Torque	Torque rating of the motor.

Table 67: OAS servo trade study categories.

5.10.3.4 Results

The four motors were analyzed using a Kepner Tregoe table which is shown below in Table 68.

OAS Servo Trade Study				
Options:	SG92R	SG-5010	MG92B	SG51R
Mandatory Requirements				
180 degree rotation	YES	YES	YES	YES

Categories	Weights	Value	Score	Value	Score	Value	Score	Value	Score
Size	40.00%	8	3.2	6	2.4	7	2.8	8	3.2
Weight	30.00%	9	2.7	4	1.2	8	2.4	10	3
Operaing Voltage	15.00%	10	1.5	5	0.75	5	0.75	5	0.75
Affordability	10.00%	9	0.9	5	0.5	5	0.5	9	0.9
Torque	5.00%	4	0.2	8	0.4	6	0.3	2	0.1
Total Score	**100%**	85.00%		52.50%		67.50%		79.50%	

Table 68: OAS servo trade study.

The SG92R has been selected as the OAS servo motor. This motor will provide sufficient torque and rotation while maintaining a low footprint onboard the rover and low cost. The motor is shown below in Figure 110.

Figure 110: SG92R servo motor.

5.10.4 Mounting

The SG92R servo motor will be secured to the front of the rover using a high strength Velcro strip for removability. This will allow for more efficient testing and adjustable positioning to determine the optimal location of the assembly. A custom designed mount will be attached to the shaft of the rover. The OAS lidar sensor will be secured to this mount allowing the motor to rotate the sensor to achieve the field of view described above. The preliminary design of this mount is shown below in Figure 111.

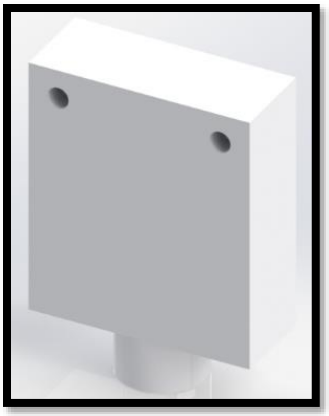


Figure 111: Lidar sensor mount.

5.11 Solar Array System (SAS)

The Solar Array System will be responsible for satisfying requirement 4.5.4 of the SOW. Based on the trade study performed in section 5.3.4, the SAS will consist of a tower assembly and a set of solar cell panels supported and unfolded by the tower assembly.

5.11.1 Solar Panels

The solar cell panels will be responsible for harvesting solar energy and using the power generated as an input to the Control Electronics System. Thicker, rigid solar panels were not considered for this design of the SAS to save space, weight, and allow flexibility of the panels. The efficiency of the panels is also considered low priority as the panels will not be used to power any systems directly. For these reasons, the panel dimensions and flexibility are the primary factors in choosing a solar panel.

PowerFilm Solar supplies cut-able panels in various length and width dimensions with a thickness of 0.00866 inches making them very flexible. A panel with high wattage while maintaining low weight is desired. A list of the smallest available panels to display the relationship between weight and wattage is provided below in Table 69.

Model	Wattage (W)	Weight (oz)	Dimensions (in)
SP3-37	0.066	0.03	2.52 x 1.45
MP3-37	0.15	0.04	4.49 x 1.45
MPT3.6-75	0.18	0.06	2.91 x 2.87
MPT3.6-150	0.36	0.1	2.91 x 5.75

Table 69: Solar panel specs.

As wattage continues to increase, the weight of the panel also increases. The MPT3.6-150 has been selected due to its high wattage, low weight, and large dimensions allowing for down-sizing the dimensions if deemed necessary. The MPT3.6-150 solar panels are shown below in Figure 112.

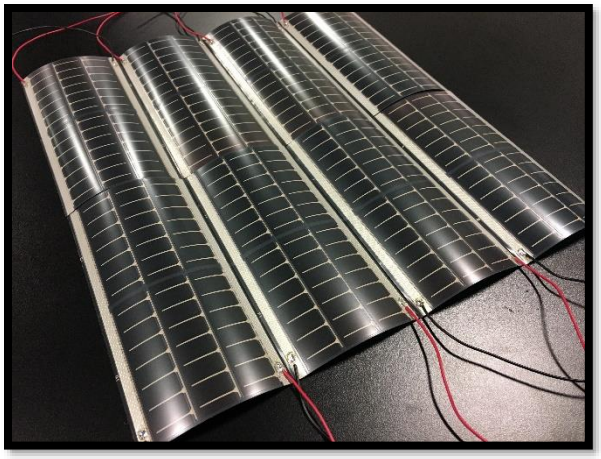


Figure 112: PowerFilm Solar MPT3.6-150 solar panels.

5.11.2 Deployment Motor

The deployment motor will be responsible for unfolding the solar array after the tower assembly, described below in section 5.11.3, will depress a limit switch until the pin is engaged at which time, it will release the limit switch. This switch will be used by the Control Electronics System to confirm successful actuation of the tower assembly. At this time, the CES will power a small motor inside the tower assembly housing that will unfold the solar panels from each other. The deployment motor is shown inside the tower assembly housing below in FIGURE#.

Figure 113: Deployment motor inside tower base.

A single shaft, high torque, minimum dimension motor is desirable for this design to ensure the motor will fit inside the tower base and provide sufficient torque to unfold the solar panels. This motor will be driven by the same 11.1V LiPo battery, described in section 5.13.4.6, that will power the main drive motors and as such, only 12V motors were considered. The following three Pololu metal gear motor lines were considered to choose this motor from: Micro Metal Gearmotors, 20D mm Metal Gearmotors, and 25D mm Metal Gearmotors.

The Pololu Micro Metal Gearmotors line of motors provides significantly smaller dimensions than the other two motor lines with comparable torque and RPM ranges. This system calls for a high torque, low RPM motor to unfold the panels without damaging them or any other nearby system. For this reason, the highest torque, lowest RPM motor of the 12V Micro Metal Gearmotors line has been selected for the deployment motor. The chosen motor is shown below in Figure 114 followed by specifications in Table 70.

Figure 114: Pololu 1000:1 Micro Metal Gearmotor.

1000:1 Micro Metal Gearmotor Relevant Specs	
Characteristics	
Technical Dimensions	
Total Weight (lbs)	0.023
Shaft Diameter and Length (in.)	Ø0.118 x 0.354
Dimensions (in.)	0.39 x 0.47 x 1.16
Operation	
Operative Voltage	12V
RPM	32
Stall Torque (ft-lb)	0.65

Table 70: Deployment motor specifications.

5.11.2.1 Shaft Extension

The 0.354 in. shaft of the motor must be extended to drive the uppermost panel support arm, described in 5.11.3.5. The 0.118 in. diameter shaft will be coupled to a 0.236 in. diameter D shaft of the deployment motor with a set screw shaft coupler, shown below in Figure 115. The larger diameter will give more contact area to distribute the panel support arm actuation load.

Figure 115: Set screw shaft coupler.

Metal washers will be used as spacers and will be pressed onto the coupler once the extension shaft is installed to allow for secure component mounting. The spacers are shown below in Figure 116.

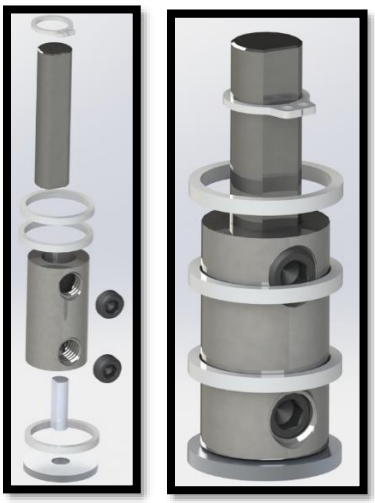


Figure 116: Motor shaft extension and spacers.

5.11.3 Tower Assembly

The tower assembly will rotate about a hinge and then deploy the solar array after the rover reaches its final destination. Array deployment requires the panels to be above the walls of the rover for proper deployment. However, the diameter of the airframe would restrict the number of panels that can be deployed if the array were to remain fixed in the upright configuration The assembly is shown in Figure 119.

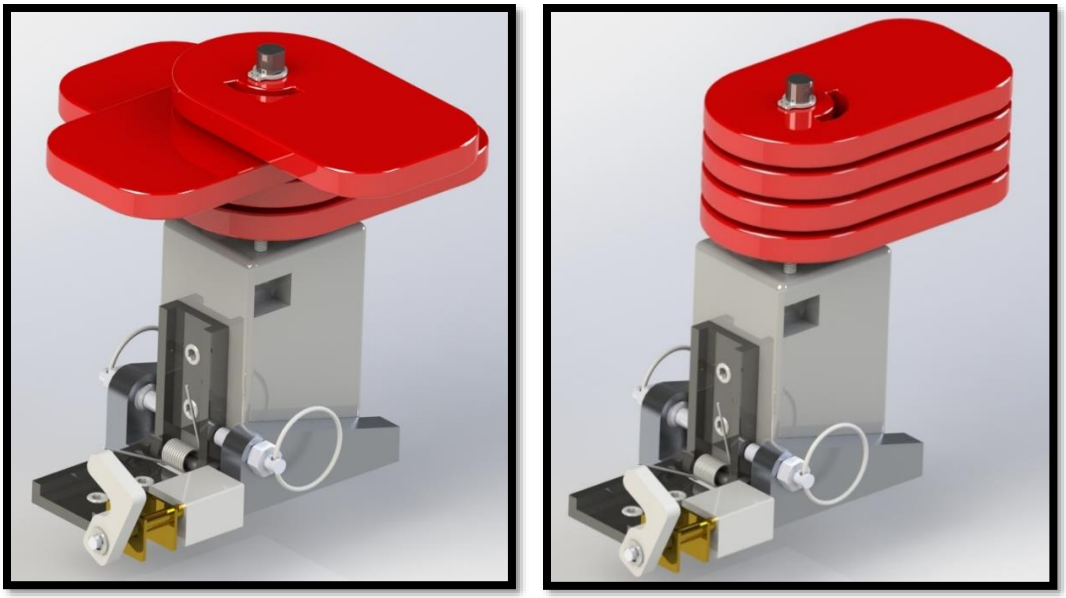


Figure 117: Actuated and stowed solar tower assembly.

5.11.3.1 Tower Base

The tower base will be the motor housing as well as the solar array mount. The deployment motor will fit into the slot in the top of the tower base and the motor power wires will exit the tower base through an aft-facing hole. The tower base is shown in Figure 118.

Figure 118: Tower base with wire chanel facing forward.

5.11.3.2 Spring Hinge

The spring hinge is an assembly of two hinge plates, a torsion spring, and a pin. The tower base will mount to the bottom of the rover with the spring hinge. The assembly is shown below in Figure 119.

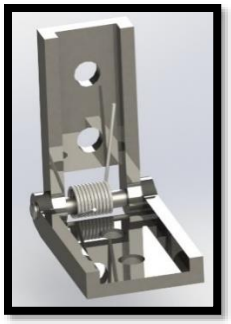


Figure 119: Spring hinge assembly.

The hinge will mount to the inside of the rover body with two 4-40 screws. The spring will allow the tower to actuate from the stowed flight position, shown in Figure 120, to the deployed position, as shown in Figure 121.

Figure 120: Stowed tower base position.

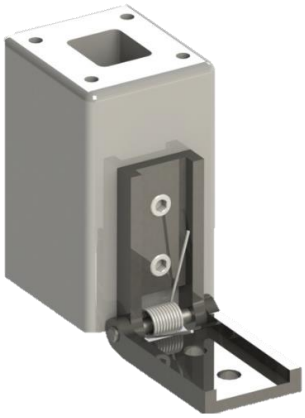


Figure 121: Deployed tower base position.

5.11.3.3 Locking Base

The locking base will consist of two retractable spring plungers and the base mount. The base mount will be fixed to the rover body with four 4-40 screws and the spring plungers will thread into the sides of the base, as shown in Figure 122.

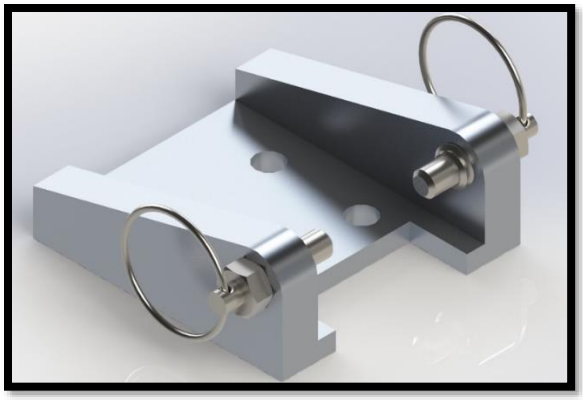


Figure 122: Solar tower locking base.

The spring plungers will be compressed when the tower base is stowed and will release once the tower base is fully actuated. Once the forward face of the tower base clears the spring plunger, it will not be able to return to the stowed position as the plungers will lock. The locked plunger is shown in Figure 123

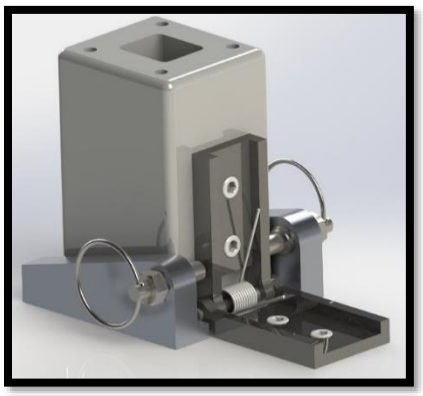


Figure 123: Spring plungers released and the tower base locked.

5.11.3.4 Tower Locking Motor

The tower locking motor will be the same motor used for the deployment motor. The locking motor will be mounted using a custom designed mounting bracket and the available mounting holes on the top plate of the motor.

An L-shaped attachment will be mounted onto the motor with retaining rings. The attachment will fit into a slot in the tower base, locking the base until the rover has reached its final destination. The motor with the attachment is shown in the locked position in Figure 124 and the unlocked position in Figure 125.

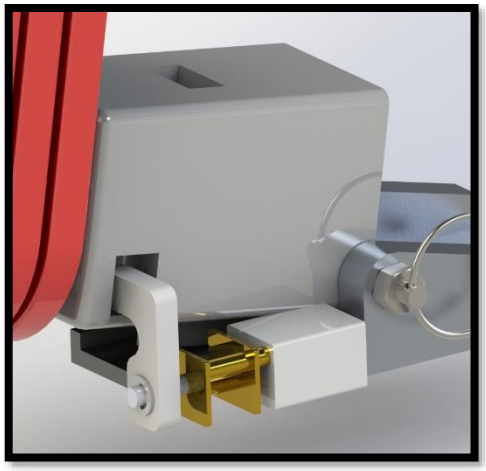


Figure 124: Locking motor locked.

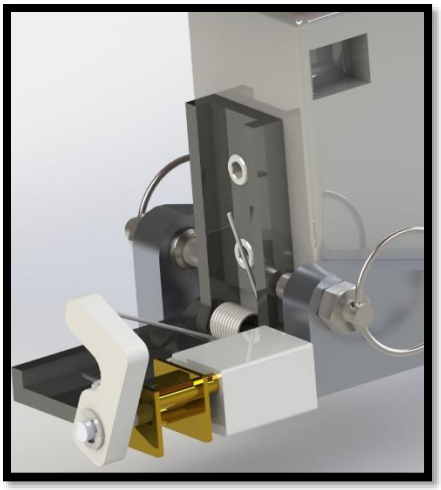


Figure 125: locking motor unlocked.

The attachment will rotate out of the slot when the solar array is ready to deploy, allowing the torsion spring will then actuate the tower into place.

5.11.3.5 Panel Support Arm

The solar panels will be fixed to the top of the panel support arms. The solar panels will overhang the edges of the panel support arms to take advantage of their flexibility. The edges of the solar panels will be bent around the support arm while in the stowed position and the panels will expand flat after array deployment. The bending will allow the panels to fit inside the rover body.

Each arm will consist of a central shaft hole, a peg slot, and a towing peg on one face. The peg slots will be oriented at different angels for each arm, as shown in Figure 126.

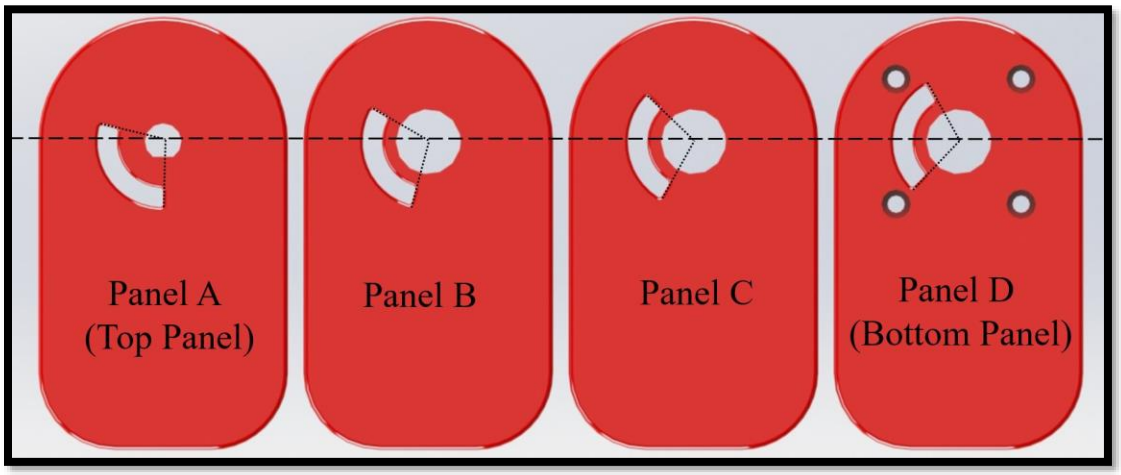


Figure 126: Peg slot angles for diferent arms from top most arm to bottom fixed arm.

The arms will stack vertically and rest on the spacers that are pressed onto the shaft extension coupler. The top panel will be secured with a retaining ring. The arms will stow in the orientation

shown in Figure 127. The peg locations were outlined to clarify their position and insertion depth.

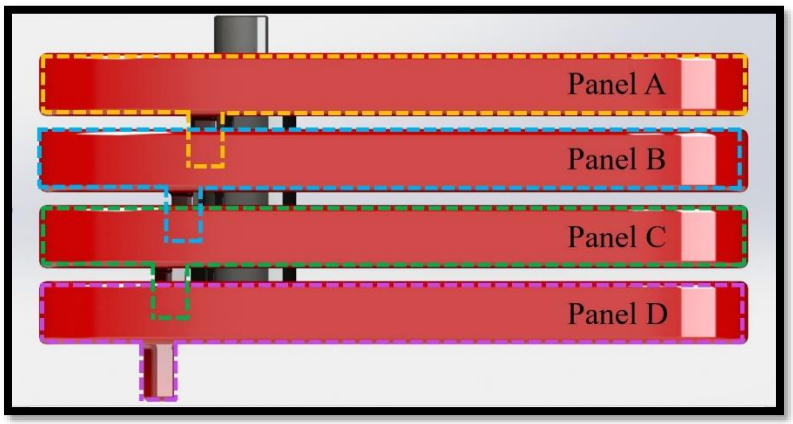


Figure 127: Panel support arms stacked and peg locations outlined.

The change in the peg slot angle will allow each towing peg to protrude into the peg slot of the support arm below it. A towing peg is shown in Figure 128.

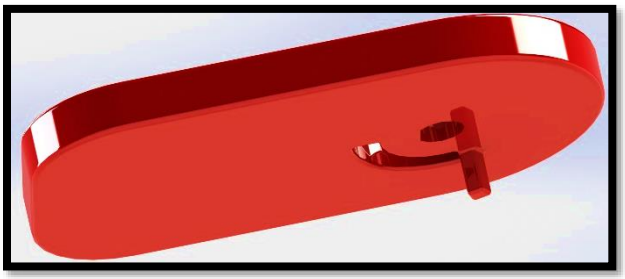


Figure 128: Isometric view of the arm and towing peg.

The bottom arm, panel D, will be mounted to the tower base with four 4-40 screws and will not have a towing peg.

As the top panel, panel A, is driven by the motor, the towing peg will follow the peg slot of panel B until the peg contacts the end of the slot. After the towing peg of panel A hits the end of panel B's peg slot, panel A will cause panel B to rotate. When panel B's towing peg contacts the end of panel C's slot, it will cause panel C to rotate until it contacts the end of panel D's slot.

Figure 129 shows the order of arm actuation, beginning with the top driven panel, panel A. The arms have been lengthened to make the deployment order clear.

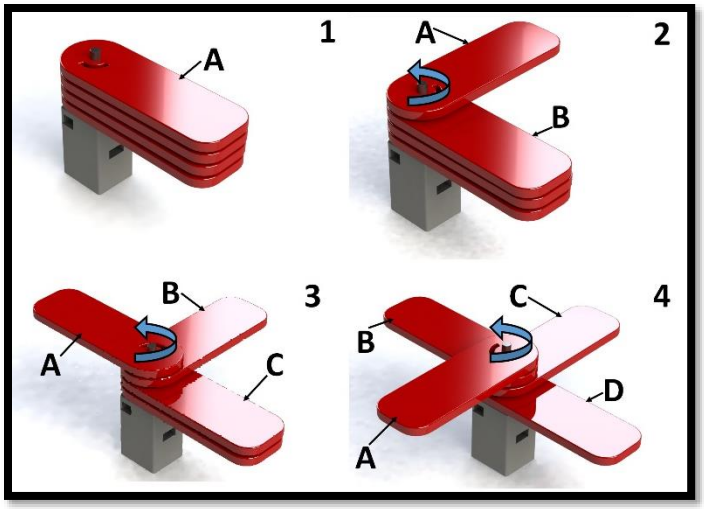


Figure 129: Panel support arm actuation order.

Figure 130 and Figure 131 show the towing peg locations of two adjacent panels before and after rotation.

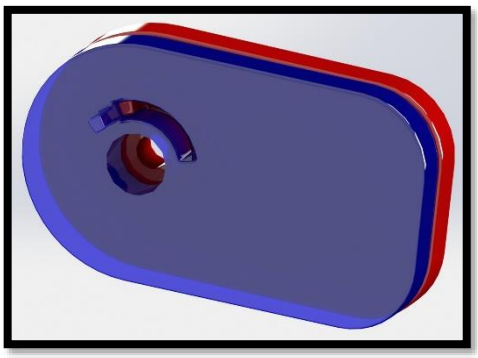


Figure 130: Two panel support arms in the stowed position.

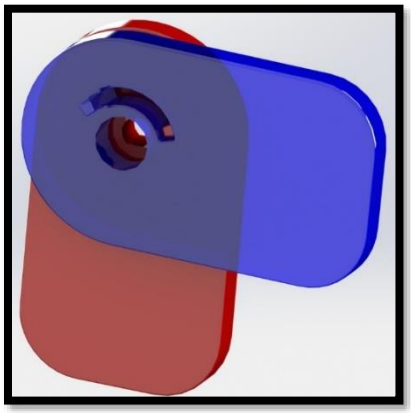


Figure 131:Two panel support arms in the deployed position.

5.11.4 Interface with the CES

The power generated by the solar panels will be fed into the control board of the CES. The voltage of the interfacing circuit will be read as an input signal by the control board. The preliminary circuit design for this operation is shown below in Figure 132.

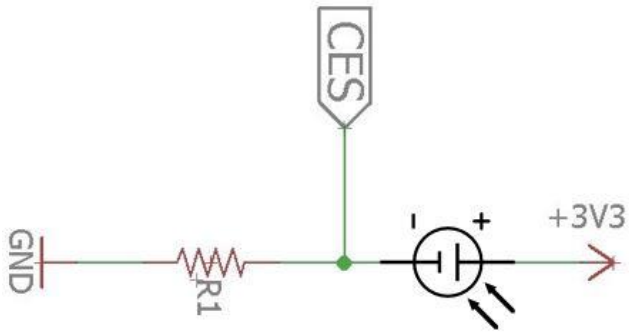


Figure 132: Solar power voltage divider.

This circuit makes use of the properties of solar panels increasing conductivity and decreasing resistance as more light is absorbed. The panels will act effectively act as a potentiometer controlled by light absorption in this circuit. The output of the voltage divider will be read by the control board to determine that the panels have deployed properly. A nearly zero volts low input will be seen by the control board in low light prior to deployment and a near 3.3V high input will be seen in daylight after deployment. The CES will use this input to trigger the Surface Imaging System.

5.12 Surface Imaging System (SIS)

The Surface Imaging System has been added to the payload as a secondary mission. This mission will not affect in any way the ability of the payload to complete its primary mission. The system has been added to embrace the mindset of the rover challenge of deploying an autonomous rover on another planet to collect data about that planet, of which images provide great scientific value. The SIS will be responsible for using the power harvested by the SAS as a trigger to take images of the rover and surrounding ground area and storing the images on the CES data logging board's microSD card, described in section 5.13.2.5, for analysis after retrieval of the payload.

5.12.1 Camera Module

The camera module will be responsible for taking the images and relaying them to the microSD card. Multiple camera modules are available that utilize I2C protocol to initialize sensors on the camera module and SPI to send and receive data to and from a master. This is the method that the CES control board, described in section 5.13.2.1, uses to send and receiver data.

5.12.2 Camera Module Trade Study

Three camera modules were considered for the SIS. These cameras were considered for the SIS camera module due to their small size, ability to be integrated with the CES control board, and open source libraries available. The following cameras modules were considered: ArduCAM Mini 2MP OV2640, the ArduCAM Mini 5MP OV5642, and the PTC08 TTL Serial JPEG camera.

5.12.2.1 Trade Study Categories

Two mandatory requirements were set for this trade study. The first requirement stated that available open source libraries be readily available for referencing control and configuration

software. The second requirement stated that control and configuration of the camera needs to be achievable by the CES control board. The categories considered for this trade study are listed and explained below in Table 71.

Category	Description
Weight	Effect of the microcontroller on the overall weight of the payload.
Image Quality	Active array size and image resolution.
Size	Overall size of the microcontroller board.
Affordability	The cost effectiveness of the sensor.
Power Consumption	Power draw from the module during normal operation.

Table 71: Camera module trade study categories.

5.12.2.2 Results

The three designs explained above were analyzed using the Kepner Tregoe table shown below in Table 72.

Camera Module Trade Study							
Options:		ArduCAM OV2640		ArduCAM OV5642		PTC08	
Mandatory Requirements							
Available open source libraries		YES		YES		YES	
Controlling and configuring is achievable with CES control board		YES		YES		YES	
Categories	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Weight	30.00%	7	2.1	6	1.8	7	2.1
Image Quality	25.00%	5	1.25	10	2.5	2	0.5
Size	15.00%	10	1.5	10	1.5	10	1.5
Afforddability	15.00%	9	1.35	7	1.05	7	1.05
Power Consumption	15.00%	8	1.2	5	0.75	8	1.2
Total Score	**100%**	74.00%		76.00%		63.50%	

Table 72: Camera module trade study.

Based on the results of the trade study, the ArduCAM Mini 5MP OV5642 camera has been selected as the SIS camera module. The resolution category has been weighted heavily due to the similarity in other specifications of these camera modules. The OV5642 provides far better resolution than the other two modules. This will allow more clear, higher quality images to be taken providing better data for analysis.

5.12.3 ArduCAM Mini 5MP OV5642

The selected camera module followed by its specifications are shown below in Figure 133 and Table 73.

Figure 133: ArduCAM Mini 5MP OV5642.

ArduCAM Mini Camera Module Relevant Specs	
Characteristics	
Technical Dimensions	
Total Weight (lbs)	0.0441
Active Array Size	2592 x 1944
Dimensions (in.)	1.34 x 0.94
Modules	
OV5642 Image Sensor	5MP image, 3-10fps video
Power Consumption (mA)	Normal: 390, Low power: 20
Compression	JPEG

Table 73: Camera module specifications.

5.12.4 Field of View Extension

The field of view, and thus data collected, will be increased by mounting the camera module on the back side of the OAS lidar sensor mount. By mounting the camera in the front of the rover with the lens facing the rear, the camera will provide data on both the rover's state after completion of its primary mission and the ground in the surrounding area of the rover. By mounting the camera to the same OAS servo motor in section 5.10.3.4, a single motor can provide a drastically increased field of view for both the lidar sensor and camera. The servo will pan the camera 180 degrees allowing it to take images of a much wider range of the rover and surrounding ground. The increased field of view is represented in blue below in Figure 134.

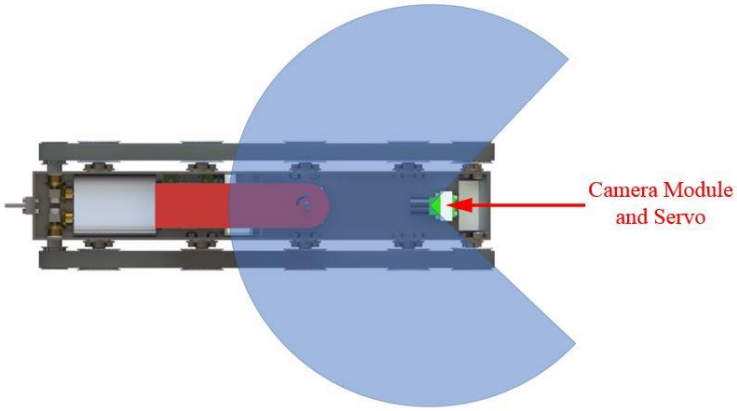


Figure 134: SIS field of view.

5.12.4.1 Mounting

The preliminary design of the interface between the camera module (green), the electronics mount (white), and the servo (grey) is shown below in Figure 135.

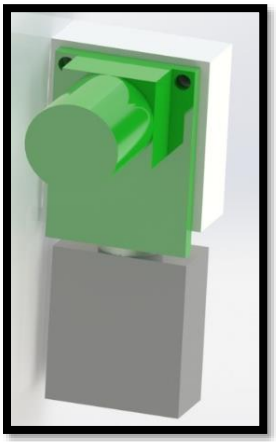


Figure 135: Camera module mount.

5.12.5 Interface with CES

The SIS camera module requires a 5V supply voltage and draws 390mA during operation. The CES control board and Solar Array System are not capable of supplying this power. For this reason, the control board will control a "normally open" transistor switch that will allow the 11.1V motor battery, described in section 5.13.4.6, to output through a voltage regulator circuit, stepping the voltage down to a 5V input supply to the camera module. The schematic design of the regulator circuit is shown below in Figure 136.

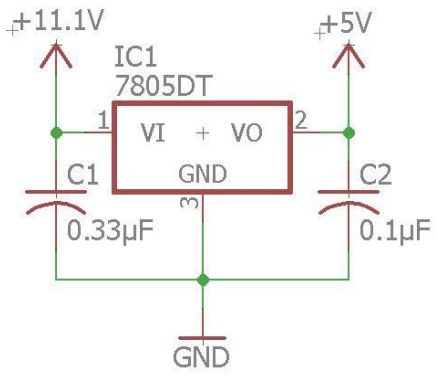


Figure 136: Voltage regulator circuit.

The power generated by the solar panels will be fed into the control board as an input that will only be high if the panels have been deployed successfully. The preliminary schematic for this circuit is described in section 5.11.4.

Upon recognizing this input, the control board will "close" the transistor switch, allowing the motor battery to be fed through the regulator circuit and to the camera module. The control board will then initialize the camera module and begin taking pictures.

5.13 Control Electronics System (CES)

The Control Electronics System will be contained in the body of the rover and perform the master control scheme for the payload. A microcontroller system with adequate processing power, general purpose input/output (GPIO), and communication protocol will be needed to run the control scheme, handle inputs from sensors, and handle outputs to the various motors on the payload. This section will detail the operation of the CES during each phase of the mission.

5.13.1 Control Scheme

The control scheme software will be constructed with safety as the highest priority. All systems will be in a safe configuration while unpowered. In the event of an abort mission at any point, the CES will shut down, effectively locking the system in a safe state. A process diagram for the control scheme is shown below in Figure 137.

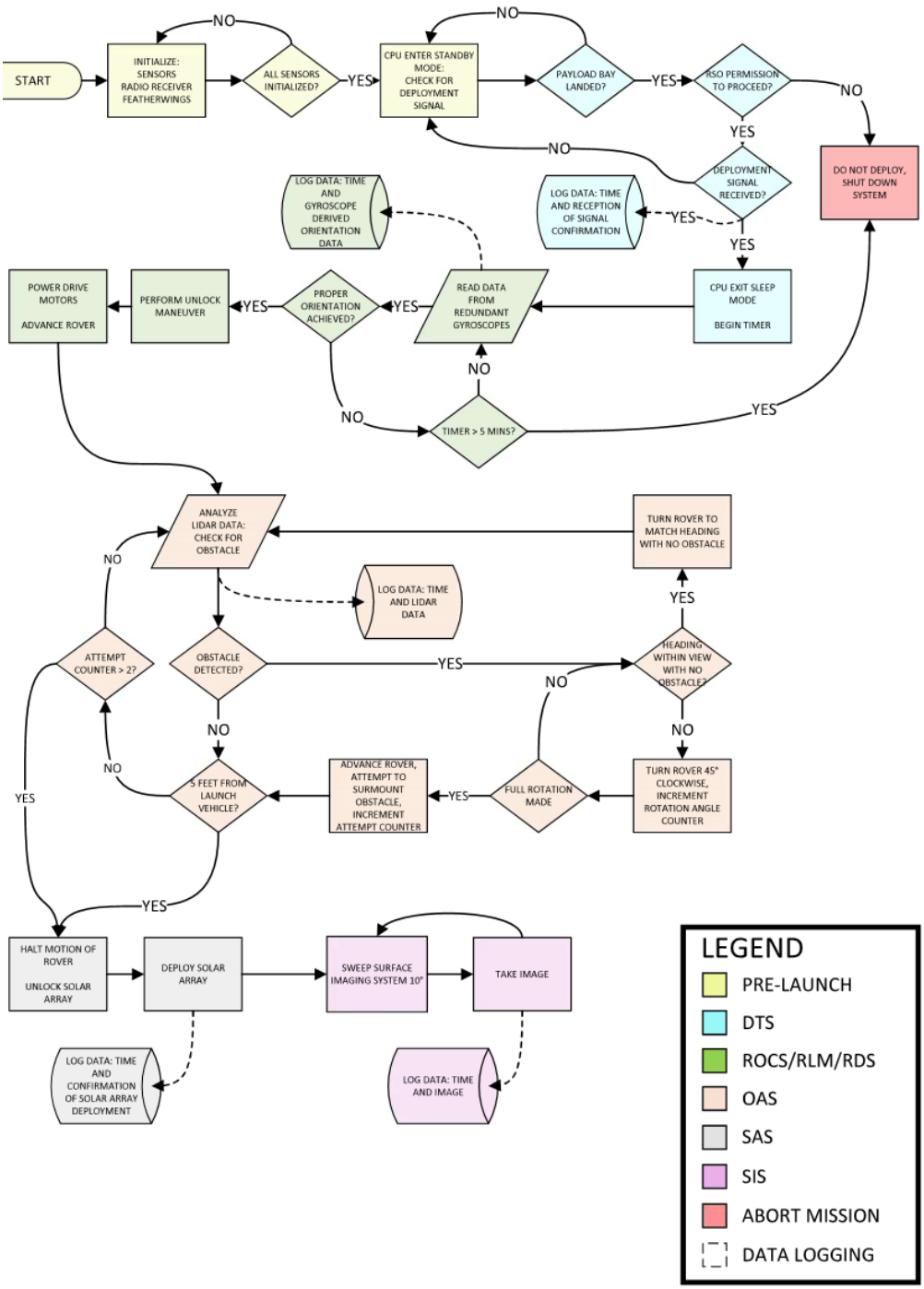


Figure 137: CES control scheme process diagram.

The controls have been divided into phases based on the system that will be operating during each phase of the mission.

5.13.2 Pre-Launch Phase

The pre-launch phase of the CES mission will consist of final software and hardware checks before integration with the launch vehicle. All batteries used for pre-launch testing will be replaced by

fully-charged batteries as the last operation before integration. This will ensure that the lifetime of the control electronics will exceed the duration of the mission.

5.13.2.1 Control Board

The CES control board will be powered on at this point. The control board refers to the microcontroller chosen to run the control scheme for the payload.

5.13.2.2 Control Board Trade Study

The following three options were considered to serve as the control board of the electronics system: Teensy 3.6, Raspberry Pi 3, Adafruit Feather M0 Bluefruit LE. A trade study was conducted to determine the optimal microcontroller for the CES control board. An emphasis has been put on the size of the microcontroller due to the space limitations inside the rover body.

5.13.2.3 Control Board Trade Study Categories

A mandatory requirement was set for this trade study that the control board needed to be capable of both SPI and I2C communication protocols. The categories considered for this study are listed and explained below in Table 74.

Category	Description
Size	Overall size of the microcontroller board.
Affordability	The cost effectiveness of the sensor.
Available GPIO	Number of available general purpose input/output pins the board has.
Weight	Effect of the microcontroller on the overall weight of the payload.
Processor	The ratings of the processor of the microcontroller.

Table 74: Control board trade study categories.

5.13.2.4 Results

The four control boards were analyzed using the Kepner Tregoe table shown below in Table 75.

Control Board Trade Study							
Options:		Teensy 3.6		Raspberry Pi 3		Feather Bluefruit LE	
Mandatory Requirements							
I2C and SPI communication		YES		YES		YES	
Categories	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Size	40.00%	6	2.4	3	1.2	9	3.6
Affordability	30.00%	7	2.1	6	1.8	7	2.1
Available GPIO	15.00%	10	1.5	9	1.35	7	1.05
Weight	10.00%	9	0.9	3	0.3	8	0.8
Processor	5.00%	7	0.35	10	0.5	6	0.3
Total Score	**100%**	72.50%		51.50%		78.50%	

Table 75: Control board trade study.

The Adafruit Feather M0 Bluefruit LE has been selected for the control board. The footprint area of the Raspberry Pi 3 board is over 4.25 times larger than that of the Teensy 3.6 and Feather. The Teensy 3.6 has a large length dimension relative to its width to account for the 62 available GPIO pins which would extend the length of the rover. This large number of GPIO pins is unnecessary for the control board and does not justify the increase in length. While the Teensy does contain onboard datalogging, it does not have Bluetooth control capabilities.

The Feather maintains a small footprint with sufficient GPIO along with Bluetooth control capability. Bluetooth control will be used during testing to provide ease of control and hands-off operation. The microcontroller is shown below in Figure 138 followed by specifications in Table 76.

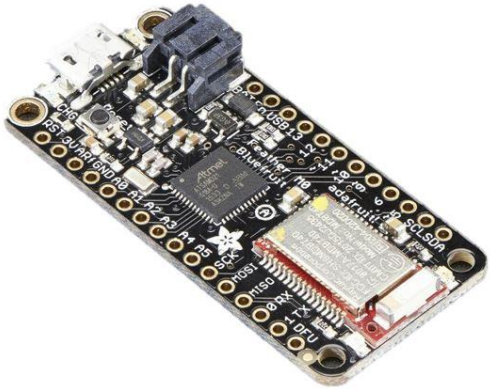


Figure 138: Adafruit Feather M0 Bluefruit LE.

Adafruit Feather M0 Bluefruit LE Analysis	
Characteristics	
Technical Dimensions	
Total Weight (lbs)	0.0126
Number of GPIO Pins	20
Dimensions (in.)	2.0 x 0.9 x 0.28
Modules	
Voltage Regulator	3.3V, 500mA peak
Power Consumption	92mA
Processor	ATSAMD21G18 ARM Cortex m0
Bluefruit LE Transmission Module	nRF51822
LiPo Charger	100mA charger

Table 76: Control board specifications.

5.13.2.5 Data logging Board

The selection of the Feather M0 Bluefruit as the control board requires a FeatherWing stackable add on to be chosen for the purposes of logging data as the control board does not have a mircoSD card onboard. The data logging board will be initialized and configured by the control board during this phase of the mission. The data logging board will be responsible for storing all data and images collected through the mission on a single microSD card. The Adafruit FeatherWing Adalogger has been selected for its stackable compatibility with the control board minimizing footprint and data logging capability with a battery backed real time clock for accurate data timestamps. The board is shown below in Figure 139.

Figure 139: Adafruit FeatherWing Adalogger.

5.13.2.6 Motor Driver Board

A motor driver board will be initialized and configured by the control board during this phase of the mission. The motor drive board will be responsible for driving the 4 inductively powered devices of the payload: two main drive motors, RLM solenoid, and SAS deployment motor. Similar to the data logging board, a motor driver board that is stackable and compatible with the Feather M0 Bluefruit and the FeatherWing Adalogger will save space inside the body of the rover. The DC Motor + Stepper FeatherWing satisfies these criteria and as such has been selected for the Motor Driver Board. The board is shown stacked on top of a Feather microcontroller below in Figure 140.

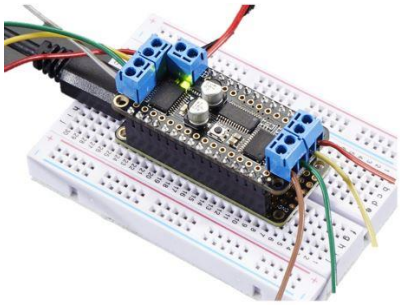


Figure 140: Adafruit FeatherWing motor driver board.

The motor driver contains four full H-Bridges, giving it the ability to drive four brushed DC motors in either direction with a 12-bit speed selection. This will be necessary for maneuvering the rover as the two main drive motors can be driven in opposite directions to turn the rover in place. Further specifications are shown below in Table 77.

Adafruit FeatherWing Motor Shield Relevant Specs	
Configurations	
Maximum DC Brushed Motors	4 bi-directional
Maximum Stepper Motors	2 unipolar or bipolar
Characteristics	
Technical Dimensions	
Total Weight (lbs)	0.0101
Dimensions (in.)	2 x 0.9 x 0.06
Modules	
Speed Control	12-bit speed selector
H-Bridges, TB6612 Chipset	1.2A per bridge, 4.5-13.5VDC

Table 77: FeatherWing motor shield specifications.

5.13.2.7 Controller Battery

The controller battery will be responsible for supplying power to the control board throughout the duration of the flight and payload mission. A 3.7V Lithium Polymer (LiPo) or Lithium Ion (LiIon) battery will be chosen dictated by the necessary 3.7V input for the Feather M0 Bluefruit microcontroller. The milliamp-hour (mAh) rating of the battery will dictate the amount of time the battery is able to supply adequate power to the control board. A projected battery life can be found with

$$Battery\ Life = \frac{Battery\ Capacity}{Device\ Consumption} * 0.7 \quad \textbf{(37)}$$

where the battery capacity is given in milliamp-hours, device consumption is given in milliamps, and the resultant battery life is given in hours. A list of compatible batteries for this microcontroller with different mAh ratings along with their respective projected lifetimes, weights, and volumes is shown below in Table 78.

Milliamp-hour Rating (mAh)	Projected Lifetime (hours)	Weight (lbs)	Volume (in^3)
100	0.76	0.0066	0.081
150	1.14	0.01	0.11
350	2.66	0.017	0.246
500	3.8	0.023	0.306
1200	9.13	0.05	0.936
2000	15.21	0.075	1.01
2200	16.74	0.10	1.05

2500	19.02	0.12	1.53
4400	33.48	0.21	2.88
6600	50.22	0.34	4.03

Table 78: Controller battery lifetimes and sizes.

As displayed, all values increase as the mAh rating of the batteries increase. The minimum required lifetime for the mission has been determined to be three hours accounting for pad time, flight, and payload mission times. Minimum dimensions and weights are desirable. These factors have led to the selection of the 500mAh LiPo for the controller battery. The battery is shown below in Figure 141.

Figure 141: 3.7V 500mAh controller battery.

The battery is fully rechargeable using the onboard battery charging circuit of the control board. This battery connects to the Feather microcontroller via a JST jack.

5.13.2.8 Initialization

The conclusion of the pre-launch phase will be final integration of the payload into the launch vehicle and start-up of the control board. At this time, the control scheme will be initiated and the control board will initialize all sensors and additional boards. The control board will then fall into standby mode to conserve power.

5.13.3 DTS Phase

The DTS phase of the CES mission extends throughout the flight of the launch vehicle. The Deployment Trigger System receiver module will continuously loop waiting for the deployment signal to be sent from the transmitter on the ground. The unique signal will ensure that no premature deployment signal is recognized.

Permission from the RSO to continue the mission is required for the deployment signal to be sent. After receiving of the deployment signal, the control board will exit standby mode and continue the control scheme.

5.13.4 ROCS/RLM/RDS Phase

This phase of the mission consists of the landing of the payload bay, orientation correction, unlocking of the rover, and initiation of the drive motors to advance the rover and exit the payload bay.

5.13.4.1 ROCS Orientation Check

Two onboard gyroscopes will be reading orientation data during this phase of the mission. This data will be stored on the microSD card and checked by the control board to ensure that proper orientation has been achieved prior to unlocking the rover as a redundant check. The deployment signal and proper orientation readings from both gyroscopes will need to be attained prior to unlocking the rover.

5.13.4.2 Gyroscope Trade Study

The gyroscopes considered for this operation were: Flora 9-DOF, Adafruit BNO055 9-DOF IMU, and L3GD20H Triple-Axis Gyro. A trade study was conducted to determine the gyroscope that would be selected. An emphasis has been put on the accuracy of the sensor since these sensors will be a safety measure.

5.13.4.3 Trade Study Categories

Two mandatory requirements were set for this study. The first requirement stated that a 3-axis gyroscope would be present on the sensor board. The second requirement stated that the communication protocol used would be I2C to allow for stacking of sensors. The categories considered for this study are listed and explained below in Table 79.

Category	Description
Accuracy	Percentage error associated with the accuracy of the sensor.
Affordability	The cost effectiveness of the sensor.
Size	Overall size of the board that the sensor is mounted on.
Degrees per second	Rate of the gyroscope.

Table 79: Gyroscope trade study categories.

5.13.4.4 Results

The four control boards were analyzed using a Kepner Tregoe table which is shown below in Table 80.

Gyroscope Trade Study							
Options:		BNO055 9-DOF		L3GD20H		Flora 9-DOF	
Mandatory Requirements							
3-axis gyroscope sensor is present on board		YES		YES		YES	
I2C comm allowing for stacking		YES		YES		YES	
Categories	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Accuracy	50.00%	9	4.5	8	4	8	4
Affordability	20.00%	10	2	7	1.4	6	1.2
Size	20.00%	7	1.4	7	1.4	9	1.8
Degrees per second	10.00%	10	1	10	1	10	1

Total Score	**100%**	89.00%	78.00%	80.00%

Table 80: Gyroscope trade study.

The BNO055 9-DOF Inertial Measurement Unit has been selected for the pair of sensors for the orientation check. The Flora 9-DOF has a slightly smaller footprint than the other two boards, however the performance of all the sensors is very similar for the purposes of this project. For that reason, accuracy and affordability took larger weight for the study. The accuracy of the BNO055 slightly surpasses that of the other two sensors. The team also has BNO055 IMUs from past projects making its cost receive a value of 10 for this the trade study. This led to the selection of the BNO055. The sensor is shown below in Figure 142 followed by specifications in Table 81.

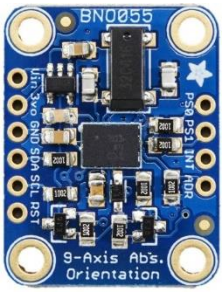


Figure 142: BNO055 9-DOF IMU.

Adafruit BNO055 9DOF IMU Relevant Specs	
Configuration	
Gyroscope only mode	
Characteristics	
Technical Dimensions	
Total Weight (lbs)	0.0066
Dimensions (in.)	1.1 x 0.8 x 0.2
Modules	
Gyroscope	2000 dps
Accuracy	0.05%

Table 81: BNO055 9-DOF IMU specifications.

5.13.4.5 RLM Unlocking

After receiving both the deployment signal and confirmation of proper orientation from the two IMUs, the control board will energize the coil of the RLM solenoid causing the armature to retract. This retraction will allow the rover to move forward freely and continue its mission.

5.13.4.6 RDS Advance

After unlocking the RLM, the CES will energize the two main drive motors of the rover advancing the rover forward out of the payload bay. The motors will be driven through the FeatherWing motor driver described above in section 5.13.2.6 allowing bi-directional control. Selection of the main drive motors is described in section 5.9.4. These motors have a 12V nominal operating voltage and 210mA no load, 4.9A stall current draw. These motors will have the largest current

consumption under load compared to the other motors and systems on the rover and therefor dictates the selection of a battery.

The battery has been selected from multiple batteries with similar performance characteristics, but different dimensions. The Storm 11.1V 400mAh 50C LiPo Battery Pack has been selected as it has minimum dimensions for its performance specifications. The battery was also selected for its rechargeability and high discharge rate. The capacity has been selected to be 400mAh which is projected to allow the motors to be powered in worst case scenario stall for approximately 3.5 minutes. The battery is shown below in Figure 143.

Figure 143: Storm 11.1V 400mAh 50C LiPo battery.

The preliminary method of recognizing the point at which the rover reaches at least five feet from the launch vehicle is calculating the distance traveled based on the time that the length of time that the motors have been driven. This method requires testing of translation speed and operation of the rover on different terrains to be viable.

5.13.5 OAS Phase

The OAS phase of the CES mission will consist of a continuous check for insurmountable obstacles in front of the rover and course adjustment accordingly. The CES will store the data from the OAS on the microSD card of the FeatehrWing Adalogger and analyze that data to determine if the rover needs to turn to avoid an obstacle. Selection of the sensor and interface with the CES is described in section 5.10.

5.13.6 SAS Phase

The SAS phase of the CES mission will consist of the unlocking and deployment of the foldable solar array. The control board will halt the main drive motors of the rover followed by unlocking of the SAS. The CES will use a limit switch to recognize upright locking of the solar tower assembly. After confirmation, the control electronics will power the deployment motor, unfolding the solar array. Selection of components and interface with the CES is described in section 5.11.

The conclusion of the SAS phase of the CES mission will be achieving full deployment of the Solar Array System. This is also the conclusion of the payload's primary mission.

5.13.7 SIS Phase

The SIS phase of the CES mission will consist of harvesting solar power by means of the SAS and using it to trigger the Surface Imaging System's camera module. The SIS will take images of the rover and surrounding ground and store them on the FeatherWing Adalogger's microSD card for

analysis after retrieval of the payload. Selection of components and interface with the CES is described in section 5.12.

This phase will continue to run until a team member is allowed onto the launch field to retrieve the rover. This marks the completion of the control scheme and the payload's secondary mission.

5.14 Preliminary Payload Specifications.

The preliminary design of the payload integrated with the ROCS in its flight configuration is shown below in Figure 144.

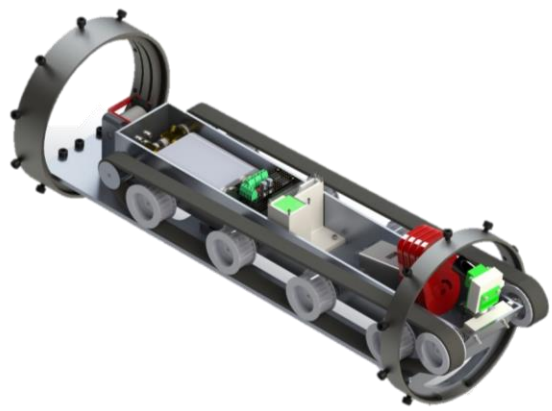


Figure 144: Fully integrated payload.

The overall weights and dimensions of the payload in its entirety are listed below in Table 82.

Assembly	Weight (lbs)	Dimensions (WxHxL)
ROCS	4.57	ID: Ø5.587 x 17.9
Rover (Stowed)	4.69	4.7 x 4.05 x 17.9
Rover (Deployed)	4.69	4.7 x 4.11 x 17.9
Total Payload:	9.26	Length: 19.6

Table 82: Total payload weights and dimensions.

5.15 Testing

All systems outlined above relating to the payload will be rigorously tested to verify all requirements set forth in 6.1.5.1.1.This section will outline the items to be tested and demonstrated for each subsystem to achieve mission success. Adequate safety measures will be put in place during all tests to ensure the safety of all team members.

5.15.1 ROCS Test Campaign

This campaign will be conducted to verify functionality of the Rover Orientation Correction System in all aspects relating to integration and performance of intended subsystem mission.

5.15.1.1 Subsystem testing

Test procedures will be developed and adhered to. Repeatability of performance stressed during testing as this is a critical system to mission success. Any changes to the system will require testing to be conducting again to verify the change.

5.15.1.2 Safety Considerations

Safety glasses will be worn during all testing that involves moving objects. Long sleeves and gloves will be worn when handling the carbon fiber payload bay as splinters are common. Caution must be taken to avoid pinch points. No team member will be standing down-hill of the test article during ground roll testing. Caution must be taken while using power tools to construct testing apparatus.

5.15.1.3 Items to be Tested

The test to be performed, a description of the test, and the requirement that the test will verify for the ROCS are shown below in Table 83.

Demonstration	**Description**	**Requirements Verified**
Integration Test	The ROCS will be inserted and removed from the payload multiple times following any changes to the system. This will demonstrate that this is a fully removeable system that can be integrated quickly for launches.	ROCS.1
Test	**Description**	**Requirements Verified**
Ground Roll Test	The ROCS will be inserted in the payload bay and rolls down a slope. The orientation of the sled will be noted after the payload bay comes to rest. A ramp will be constructed for consistency of test conditions.	ROCS.2

Table 83: ROCS testing plan.

5.15.2 RLM Test Campaign

This campaign will be conducted to verify functionality of the Rover Locking Mechanism in all aspects relating to integration and performance of subsystem mission.

5.15.2.1 Subsystem Testing

Repeatability will be stressed during testing of the RLM as this is a critical system to mission success. Any changes to the system will require testing to be conducted again to verify the change.

5.15.2.2 Safety Considerations

Safety glasses will be worn during all testing that involves moving objects. Grounding mats and wrist straps will be used when handling electronics to avoid static discharge. Caution must be taken to avoid pinch points. Power to any electrical components will be easily cut in the event that an emergency shutdown is needed.

5.15.2.3 Items to be Tested

The tests to be performed, a description of the test, and the requirement that the test will verify for the RLM are shown below in Table 84.

Demonstration	Description	Requirements Verified
Lock Release Test	The RLM will be engaged locking the rover to the ROCS. The assembly will be held in different orientations to ensure that the RLM is in fact locked. The solenoid of the RLM with then be powered and expected to release the rover.	RLM.3

Table 84: RLM testing plan.

5.15.3 DTS Test Campaign

This campaign will be conducted to verify functionality of the Deployment Trigger System in all aspects relating to integration and performance of subsystem mission.

5.15.3.1 Subsystem Testing

Repeatability will be stressed during testing of the RLM as this is a critical system to mission success. Any changes to the system will require testing to be conducted again to verify the change.

5.15.3.2 Safety Considerations

Grounding mats and wrist straps will be used when handling electronics to avoid static discharge. RF frequencies will remain in the range of legal frequencies with no certification required. Power to any electrical components will be easily cut in the event that an emergency shutdown is needed.

5.15.3.3 Items to be Tested

The tests to be performed, a description of the test, and the requirement that the test will verify for the DTS are shown below in Table 85.

Demonstration	Description	Requirements Verified
Range Test	The receiver module will be mounted in its intended flight configuration and separated from the transmitter by a linear distance of 2500 feet. The deployment signal will be sent from the transmitter module and expected to be recognized by the receiver module.	DTS.1
Detachment Test	The receiver module will be mounted in its intended flight configuration. The three wires connecting the CES to the DTS will be connected. The rover will be driven forward and the connection between the CES and DTS will be expected to disconnect completely.	DTS.2

Table 85: DTS testing plan.

5.15.4 RBS Test Campaign

This campaign will be conducted to verify functionality of the Rover Body Structures in all aspects relating to integration and performance of subsystem mission.

5.15.4.1 Subsystem Testing

Repeatability will be stressed during testing of the DTS as this is a critical system to mission success. Any changes to the system will require testing to be conducted again to verify the change.

5.15.4.2 Safety Considerations

Safety glasses will be worn during all testing involving moving objects. Inspections will involve minimal interaction with the system to verify the requirement.

5.15.4.3 Items to be Tested

The tests to be performed, a description of the test, and the requirement that the test will verify for the RBS are shown below in Table 86.

Inspection	Description	Requirements Verified
Electronics Housing Test	All electronics will be mounted onboard the rover in their intended flight configuration. Adequate clearance for manipulation of wires and ease of removal of components will be expected.	RBS.1
Ground Clearance Test	The rover drive system will be assembled with the RBS. Sufficient ground clearance such that only the drive belt treads are making contact with the ground will be expected.	RBS.3

Table 86: RBS testing plan.

5.15.5 RDS Test Campaign

This campaign will be conducted to verify functionality of the Rover Drive System in all aspects relating to integration and performance of subsystem mission.

5.15.5.1 Subsystem Testing

Repeatability will be stressed during testing of the RDS as this is a critical system to mission success. Any changes to the system will require testing to be conducted again to verify the change.

5.15.5.2 Safety Considerations

Safety glasses will be worn during all testing involving moving objects. Caution must be taken near rotating parts and to avoid pinch points. Power to any electrical components will be easily cut in the event that an emergency shutdown is needed.

5.15.5.3 Items to be Tested

The tests to be performed, a description of the test, and the requirement that the test will verify for the RDS are shown below in Table 87.

Demonstration	Description	Requirements Verified

Torque Test	The rover will be loaded with weight representative of the projected final weight. The RDS will be commanded to operate. The drive motors will be expected to provide sufficient torque to advance the rover forward.	RDS.1
Test	**Description**	**Requirements Verified**
Surmounting Obstacles Test	Various sizes and shapes of obstacles will be placed in front of the rover. The rover will be expected to surmount a sufficiently small object. This test will determine the limit of obstacles the rover can surmount and therefor the minimum sized obstacle the OAS must be capable of recognizing.	RDS.2
Sloped Terrain Test	The rover will be set on a slope of varying grade and will be expected to maintain forward motion over the slope. This will determine the maximum slope the rover can overcome.	RDS.3
Traction Test	The rover will be placed on different ground surfaces. The ground surfaces will be manipulated to emulate possible ground conditions of the launch field accounting for weather.	RDS.4

Table 87: RDS testing plan.

5.15.6 CES Test Campaign

This campaign will be conducted to verify functionality of the Control Electronics System in all aspects relating to integration and performance of subsystem mission.

5.15.6.1 Subsystem Testing

Repeatability will be stressed during testing of the CES as this is a critical system to mission success. Any changes to the system will require testing to be conducted again to verify the change.

5.15.6.2 Safety Considerations

Safety glasses will be worn during all testing involving moving objects. Grounding mats and wrist straps will be used when handling electronics to avoid static discharge. Caution must be taken near rotating parts and to avoid pinch points. Power to any electrical components will be easily cut in the event that an emergency shutdown is needed.

5.15.6.3 Items to be Tested

The tests to be performed, a description of the test, and the requirement that the test will verify for the CES are shown below in Table 88.

Demonstration	**Description**	**Requirements Verified**

Deployment Signal Recognition Test	The CES will be connected to the DTS. The deployment signal will be sent and the CES will be expected to recognize this unique signal.	CES.1
RLM Release Test	The CES will be connected to the RLM. Software will allow the CES to be controlled by a team member's cellphone via Bluetooth. User input will instruct the CES to unlock. The CES will be expected to unlock the RLM.	CES.3
Motion and Maneuverability Control Test	The CES will be connected to the RDS. Software will allow the CES to be controlled by a team member's cellphone via Bluetooth. User input will instruct the CES to drive the RDS motors for forward, reverse, and turning motion. The CES will be expected to maintain control of the RDS.	CES.4
SAS Deployment Test	The CES will be connected to the SAS. Software will allow the CES to be controlled by a team member's cellphone via Bluetooth. User input will instruct the CES to deploy the SAS.	CES.6
Data Logging Test	The CES will be fed a stream of input data from sensors and the SIS. The CES will be expected to store all data efficiently on an onboard microSD card for later analysis.	CES.7
Battery Life Test	The CES will be powered and left running a program that will constantly output the voltage of the battery. The system will be left running until insufficient power is being provided by the battery, shutting down the board. The time elapsed from power up to shutdown will be expected to exceed three hours.	CES.8
Test	**Description**	**Requirements Verified**
Orientation Confirmation Test	The CES will be integrated into a round tube. The tube will be rolled into different orientations, both proper and non-proper orientation for deployment of the rover. The CES will be expected to distinguish proper from non-proper orientation for deployment.	CES.2
Obstacle Avoidance Control Test	The CES will be connected to the OAS and RDS. Objects will be placed in front of and around the assembly. The CES will be expected to recognize data sent from the OAS, determine a path of avoidance, and control the RDS to meet that path.	CES.5

Table 88: CES testing plan.

5.15.7 OAS Test Campaign

This campaign will be conducted to verify functionality of the Obstacle Avoidance System in all aspects relating to integration and performance of subsystem mission.

5.15.7.1 Subsystem Testing

The OAS will be extensively ground tested to confirm operation and refine the avoidance algorithm. Any changes to the system will require testing to be conducted again to verify the change.

5.15.7.2 Safety Considerations

Grounding mats and wrist straps will be used when handling electronics to avoid static discharge. Power to any electrical components will be easily cut in the event that an emergency shutdown is needed.

5.15.7.3 Items to be Tested

The tests to be performed, a description of the test, and the requirement that the test will verify for the OAS are shown below in Table 89.

Test	Description	Requirements Verified
Obstacle Recognition Test	The OAS will be integrated with the CES. Obstacles will be placed in front of and around the OAS. The OAS will be expected to recognize these objects and relay the data back to the CES. This test will be used to determine range and accuracy of the OAS for the avoidance path algorithm.	OAS.1

Table 89: OAS testing plan.

5.15.8 SAS Test Campaign

This campaign will be conducted to verify functionality of the Solar Array System in all aspects relating to integration and performance of subsystem mission.

5.15.8.1 Subsystem Testing

Repeatability will be stressed during testing of the SAS as this is a critical system to mission success. Any changes to the system will require testing to be conducted again to verify the change.

5.15.8.2 Safety Considerations

Safety glasses will be worn during all testing involving moving objects. Caution must be taken to avoid pinch points. Power to any electrical components will be easily cut in the event that an emergency shutdown is needed.

5.15.8.3 Items to be Tested

The tests to be performed, a description of the test, and the requirement that the test will verify for the SAS are shown below in Table 90.

Demonstration	Description	Requirements Verified

Clearance Test	The SAS will be integrated into the rover. Sufficient clearance for the panels from all surrounding components will be expected.	SAS.2
Upright Lock Test	The SAS will be instructed to deploy. The system will be expected to remain locked in its final deployed configuration until manual unlock is performed by a team member.	SAS.3
Unfold Test	The SAS will be instructed to unfold the solar panels. The solar panels will be expected to separate from each other with minimal restriction and exposed solar panel surface area will be expected to increase.	SAS.4

Table 90: SAS testing plan.

5.15.9 SIS Test Campaign

This campaign will be conducted to verify functionality of the Surface Imaging System in all aspects relating to integration and performance of subsystem mission.

5.15.9.1 Subsystem Testing

The SIS will be extensively ground tested to confirm operation. Any changes to the system will require testing to be conducted again to verify the change.

5.15.9.2 Safety Considerations

Grounding mats and wrist straps will be used when handling electronics to avoid static discharge. Power to any electrical components will be easily cut in the event that an emergency shutdown is needed.

5.15.9.3 Items to be Tested

The tests to be performed, a description of the test, and the requirement that the test will verify for the SIS are shown below in Table 91.

Demonstration	Description	Requirements Verified
Image Capture Test	The SIS will be connected to the CES. Software will allow the CES to be controlled by a team member's cellphone via Bluetooth. The SIS will be instructed by the CES upon user input being sent to take a picture and store it on a microSD card.	SIS.1

Table 91: SIS testing plan.

5.15.10 Full-Scale Flight Test Campaign

This test campaign will be conducted to verify that characteristics of all systems in flight align with ground testing results. The entire system will be integrated into the launch vehicle and operate for the entire duration of the mission as is intended at competition.

5.15.10.1 Subsystem Testing

All subsystems will be tested during full-scale flight tests. This campaign will also be used to confirm analysis conducted on systems prior to flight.

5.15.10.2 Safety Considerations

All safety measures for high powered rocket launches will be strictly adhered to in the interest of safety for all team members.

5.15.10.3 Items to be Tested

Full-scale flight tests will confirm functionality during the flight of the launch vehicle and produce data that will be used to further analyze all subsystems of the payload. Requirements to be verified by full-scale flight tests are listed below in Table 92.

Test	Description	Requirements Verified
Full-Scale Flight Tests	This test series will demonstrate all subsystems of the payload's ability to perform the entirety of its designed mission throughout a flight of the launch vehicle.	ROCS.3 RLM.1 RLM.2 RBS.2 SAS.1

Table 92: Full-scale flight testing plan.

6 Project Plan

6.1 Requirements Verification

6.1.1 Launch Vehicle Requirements Verification

Each launch vehicle Statement of Work requirement, and said requirement's method of verification, are shown below in Table 93. The methods of verification were derived using the NASA Systems Engineering Handbook and consist of Inspection, Analysis, Demonstration, and Test.

Requirement Number	Requirement Description	Method of Verification
2.1	The vehicle will deliver the payload to an apogee altitude of 5,280 feet above ground level (AGL).	**Analysis:** The launch vehicle shall be designed to reach an apogee altitude of 5,280 feet AGL. Several OpenRocket simulations as well as hand calculations will be performed to ensure the ideal motor is selected. The VDS will be tested to ensure an accurate altitude is achieved.
2.2	The vehicle will carry one commercially available, barometric altimeter for recording the official altitude used in determining the altitude award winner.	**Inspection:** A PerfectFlite StratoLogger CF altimeter will be used to record the official apogee altitude for the competition flight.
2.3	Each altimeter will be armed by a dedicated arming switch that is accessible from the exterior of the rocket airframe when the rocket is in the launch configuration on the launch pad.	**Inspection:** The altimeters shall utilize a 6-32 PCB Screw-Switch purchased from Missile-Works. The screw switch shall be mounted on the altimeter sled with a small hole drilled into the airframe to provide access to the switch. The screw switch holes shall be placed opposite from the rail buttons to ensure the launch rail will not block access.
2.4	Each altimeter will have a dedicated power supply.	**Inspection:** Each altimeter shall be powered by a new Duracell 9-Volt battery.
2.5	Each arming switch will be capable of being locked in the ON position for launch (i.e. cannot be disarmed due to flight forces).	**Demonstration:** Each screw switch shall be tightened to its maximum preload to ensure the screw won't back out of its threads.
2.6	The launch vehicle will be designed to be recoverable and reusable. Reusable	**Demonstration:** The parachutes for each section of the launch vehicle shall

	is defined as being able to launch again on the same day without repairs or modifications.	be designed specifically for that section's mass. Through proper material selection and parachute deployment, the launch vehicle will be fully reusable. This will ensure that all sections of the launch vehicle will land under the maximum allowable kinetic energy without receiving any damage. In the event that a fin receives damage from landing, the Removable Fin System (RFS) allows for quick replacement of any fin.
2.7	The launch vehicle will have a maximum of four (4) independent sections.	**Inspection:** The launch vehicle shall deploy the rover payload without splitting into more than four independent sections during recovery.
2.8	The launch vehicle will be limited to a single stage.	**Analysis:** The limited apogee altitude of 5,280 feet AGL eliminates the need for a launch vehicle with more than a single stage. Therefore, the motor selected shall be capable of delivering the launch vehicle to 5,280 feet AGL with a single stage.
2.9	The launch vehicle will be capable of being prepared for flight at the launch site within 3 hours of the time the Federal Aviation Administration flight waiver opens.	**Demonstration:** The launch vehicle shall be designed to be assembled quickly. The rover payload shall be designed to be separate from the rest of the launch vehicle during assembly. Prior to competition, the team shall practice assembling the launch vehicle, and rover payload, in its entirety in under 3 hours.
2.10	The launch vehicle will be capable of remaining in launch-ready configuration at the pad for a minimum of 1 hour without losing the functionality of any critical on-board components.	**Test:** All electronics onboard the launch vehicle shall be tested in simulated launch ready conditions to ensure their maximum battery life is greater than one hour.
2.11	The launch vehicle will be capable of being launched by a standard 12-volt direct current firing system. The firing system will be provided by the NASA-designated Range Services Provider.	**Inspection:** The launch vehicle will utilize proven launch igniters purchased from Wildman Rocketry for all test flights prior to competition. The igniters are designed to ignite the vehicle's motor by use of a standard 12-volt direct current firing system.

2.12	The launch vehicle will require no external circuitry or special ground support equipment to initiate launch (other than what is provided by Range Services).	**Analysis:** The launch vehicle will not require external circuitry or special ground support equipment to initiate launch. This is in accordance with requirement 2.12 in the statement of work.
2.13	The launch vehicle will 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).	**Inspection:** The team will use an AeroTech L2200-G motor for the full-scale launch vehicle. This is in accordance with requirement 2.13 in the statement of work.
2.14	Pressure vessels on the vehicle will be approved by the RSO.	**Inspection:** The current design of the launch vehicle does not require the use of any pressure vessels. If the design changes to include such a system, NASA and the RSO will be notified, and the criteria mentioned in the Statement of Work will be met.
2.15	The total impulse provided by a College and/or University launch vehicle will not exceed 5,120 Newton-seconds (L-class).	**Inspection:** The total impulse of the AeroTech L2200-G is 5,104 Newton-seconds.
2.16	The launch vehicle will 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.	**Analysis:** Several OpenRocket simulations shall be completed to verify that the static stability margin at the point of rail exit is greater than 2.0. Hand calculations shall also be performed to verify that the OpenRocket simulations are accurate.
2.17	The launch vehicle will accelerate to a minimum velocity of 52 fps at rail exit.	**Analysis:** Several OpenRocket simulations shall be completed to verify that the velocity at the point of rail exit is greater than 52 fps. Hand calculations shall also be performed to verify that the OpenRocket simulations are accurate.
2.18	All teams will successfully launch and recover a subscale model of their rocket prior to CDR.	**Test:** A 1:2 scaled model of the launch vehicle shall be designed and launched prior to CDR.
2.19	All teams will successfully launch and recover their full-scale rocket prior to	**Test:** The team plans to conduct several full-scale test flights

	FRR in its final flight configuration. The rocket flown at FRR must be the same rocket to be flown on launch day.	throughout the season. Test flights will verify that the launch vehicle is structurally sound and stable. Test flights will also verify if the recovery system and VDS preform as designed.
2.19.1	The vehicle and recovery system will have functioned as designed.	**Test:** For a full-scale flight test to be considered a success, all parachutes must deploy properly, and all structural components of the launch vehicle must not be damaged. Cameras will be mounted internally in the recovery bays to provide visual confirmation of recovery systems performing nominally.
2.19.2.1	If the payload is not flown, mass simulators will be used to simulate the payload mass.	**Analysis:** The launch vehicle shall be designed to accept payload mass simulators in the form of steel ballast.
2.19.2.1.1	The mass simulators will be located in the same approximate location on the rocket as the missing payload mass.	**Analysis:** The launch vehicle shall be designed to accept payload mass simulators in the form of steel ballast. The steel ballast will be located in the payload bay where the rover payload would be if it were flown.
2.20	Any structural protuberance on the rocket will be located aft of the burnout center of gravity.	**Analysis:** OpenRocket and CFD simulations will be performed to ensure that the VDS drag blades are located aft of the burnout center of gravity.
2.21.1	The launch vehicle will not utilize forward canards.	**Inspection:** The launch vehicle shall not utilize forward canards. This is in accordance with requirement 2.21.1 in the statement of work.
2.21.2	The launch vehicle will not utilize forward firing motors.	**Inspection:** The launch vehicle shall not utilize forward firing motors. This in accordance with requirement 2.21.1 in the statement of work.
2.21.3	The launch vehicle will not utilize motors that expel titanium sponges (Sparky, Skidmark, MetalStorm, etc.)	**Inspection:** The launch vehicle will utilize an AeroTech L2200 Mojave Green motor.
2.21.4	The launch vehicle will not utilize hybrid motors.	**Inspection:** The launch vehicle will utilize an AeroTech L2200 Mojave Green motor.
2.21.5	The launch vehicle will not utilize a cluster of motors.	**Inspection:** The launch vehicle will utilize a single AeroTech L2200 Mojave Green motor.

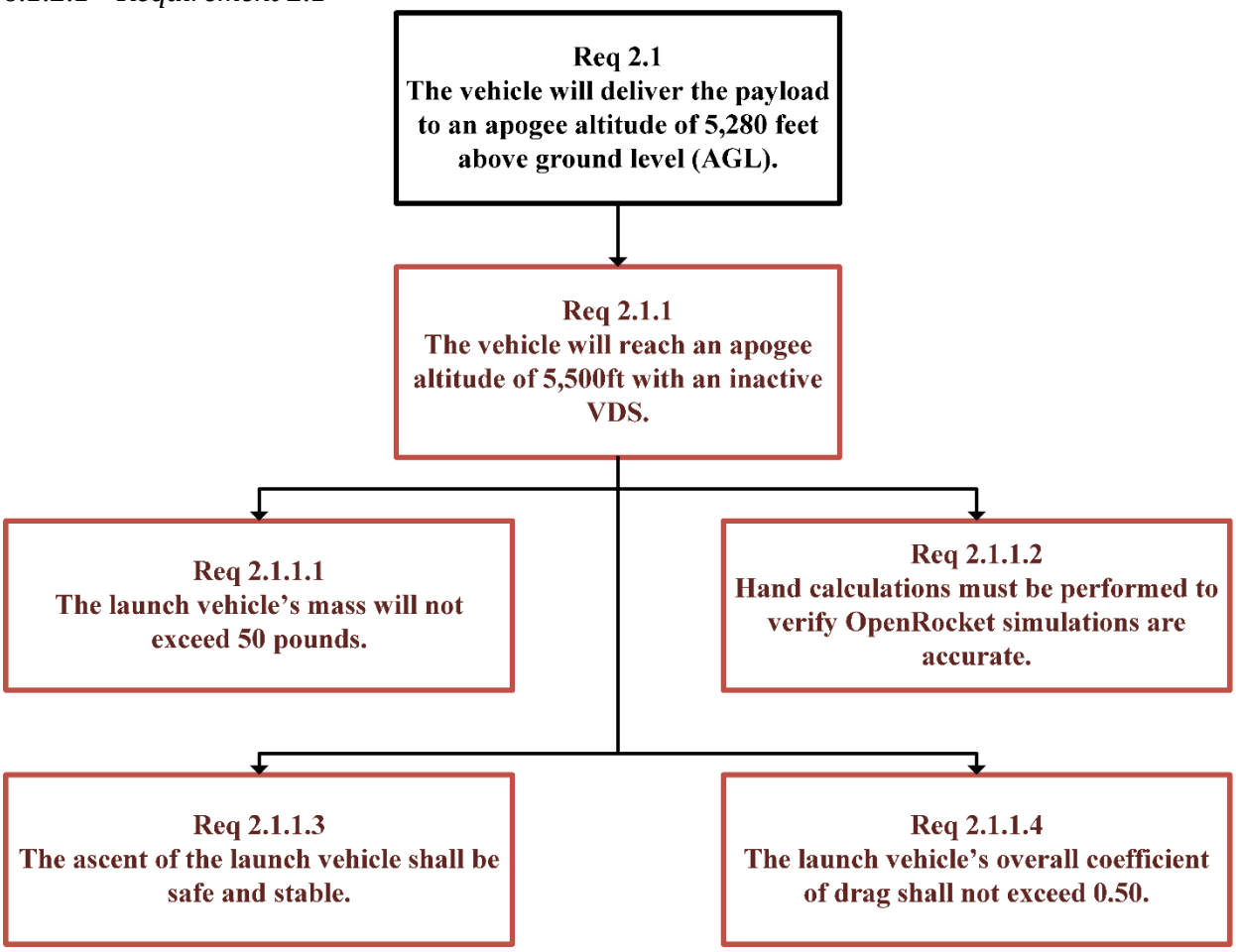
2.21.6	The launch vehicle will not utilize friction fitting for motors.	**Inspection:** The motor will be secured to the launch vehicle with an aluminum motor retainer, fastened with three stainless steel shoulder bolts.
2.21.7	The launch vehicle will not exceed Mach 1 at any point during flight.	**Analysis:** Several OpenRocket simulations shall be completed to verify that the velocity of the launch vehicle doesn't exceed Mach 1 during the flight. Hand calculations shall also be performed to verify that the OpenRocket simulations are accurate.
2.21.8	Vehicle ballast will not exceed 10% of the total weight of the rocket.	**Inspection:** The launch vehicle will not utilize any ballast.

Table 93: Statement of Work requirements descriptions and methods of verification.

6.1.2 Team Derived Launch Vehicle Requirements Verification

In the following sections, the Statement of Work Requirements are accompanied by the team's own Team Derived Requirements. Each requirement is included with a visual flow chart, outlining where a team derived requirement originated from, along with a requirements verification table. The methods of verification were derived from the NASA Systems Engineering Handbook and include Inspection, Analysis, Demonstration, and Test.

6.1.2.1 Requirement 2.1



Requirement Number	Requirement Description	Method of Verification
2.1	The vehicle will deliver the payload to an apogee altitude of 5,280 feet above ground level (AGL).	**Analysis:** The launch vehicle shall be designed to reach an apogee altitude of 5,280 feet AGL. Several OpenRocket simulations as well as hand calculations will be performed to ensure the ideal motor is selected. The VDS will be tested to ensure an accurate altitude is achieved.
2.1.1	The vehicle will reach an apogee of 5,500ft AGL with an inactive VDS.	**Analysis:** The launch vehicle shall be designed using lightweight materials and the proper motor to overshoot the target altitude with an inactive VDS.
2.1.1.1	The launch vehicle's mass will not exceed 50 pounds.	**Analysis:** The launch vehicle shall be designed to use lightweight materials and the masses of each component

		shall be recorded during the design phase.
2.1.1.2	Hand calculations must be performed to verify OpenRocket simulations are accurate.	**Analysis:** Hand calculations shall be performed prior to CDR to ensure the OpenRocket simulations are accurate.
2.1.1.3	The ascent of the launch vehicle shall be safe and stable.	**Analysis:** The launch vehicle shall be designed to ascend safely. Safety checklists shall be written to ensure that no assembly steps will be missed.
2.1.1.4	The launch vehicle's overall coefficient of drag shall not exceed 0.50.	**Analysis:** CFD simulations will simulate flight conditions and compute the coefficient of drag of the entire launch vehicle

6.1.2.1.1 Derivation of Requirement 2.1.1

To ensure that the VDS is given the opportunity to reduce the apogee altitude to precisely 5,280ft, the launch vehicle shall be designed to reach an apogee altitude of 5,500ft. The altitude of 5,500ft was chosen to minimize the risk of overshooting the 5,600ft waiver in effect at our most commonly used launch field for test flights. The apogee altitude of 5,500ft provides a 100ft buffer in the event of a VDS failure.

6.1.2.1.2 Derivation of Requirement 2.1.1.1

To ensure that the launch vehicle will be able to reach an apogee of 5,500ft, a weight limit is necessary. Through running OpenRocket and MATLAB Simulink simulations, with an overall coefficient of drag of 0.5 and an AeroTech L2200 motor, a weight of 50 pounds was found to be the maximum weight the launch vehicle could weigh and still reach 5,500ft altitude.

6.1.2.1.3 Derivation of Requirement 2.1.1.2

As is standard in engineering, hand calculations will be performed to verify that the simulations are within a reasonable accuracy. Not only does this second calculation method verify that the simulations are accurate, but also that they were set up correctly.

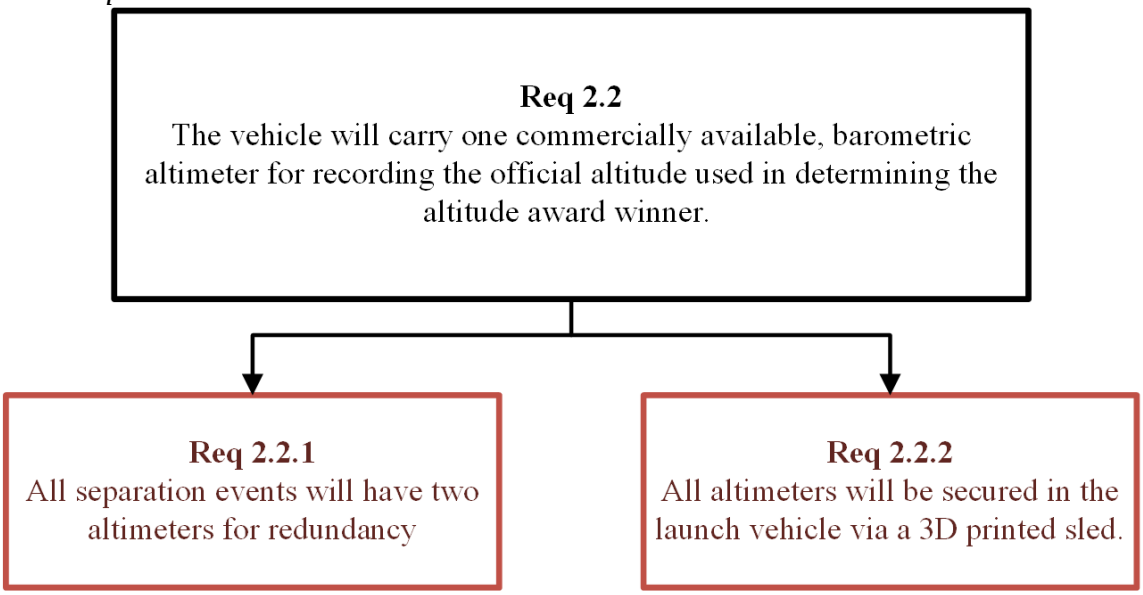
6.1.2.1.4 Derivation of Requirement 2.1.1.3

Safety is the highest priority during the design and construction of the launch vehicle. For the mission to be considered a success, the launch vehicle is required to safely ascend to 5,280 feet.

6.1.2.1.5 Derivation of Requirement 2.1.1.4

In the past, the team has determined that the coefficient of drag for a similarly sized launch vehicle was 0.5. By decreasing the coefficient of drag of the rocket, the launch vehicle is more efficient and thus requires a smaller motor to achieve the target apogee altitude. The full-scale launch vehicle is required to be optimized to minimize the coefficient of drag.

6.1.2.2 Requirement 2.2



Requirement Number	Requirement Description	Method of Verification
2.2	The vehicle will carry one commercially available, barometric altimeter for recording the official altitude used in determining the altitude award winner.	**Inspection:** A PerfectFlite StratoLogger CF altimeter will be used to record the official apogee altitude for the competition flight.
2.2.1	All separation events will have two altimeters for redundancy.	**Inspection:** Two PerfectFlite StratoLogger CF altimeters will be used to initiate separation events during recovery. The altimeters will be programmed to be delayed relative to each other to prevent over pressurization of the airframe.
2.2.2	All altimeters will be secured in the launch vehicle via a 3D printed sled.	**Analysis**: The launch vehicle shall be designed to accept 3D printed sleds for holding each altimeter. All 3D printed sleds will be designed prior to CDR.

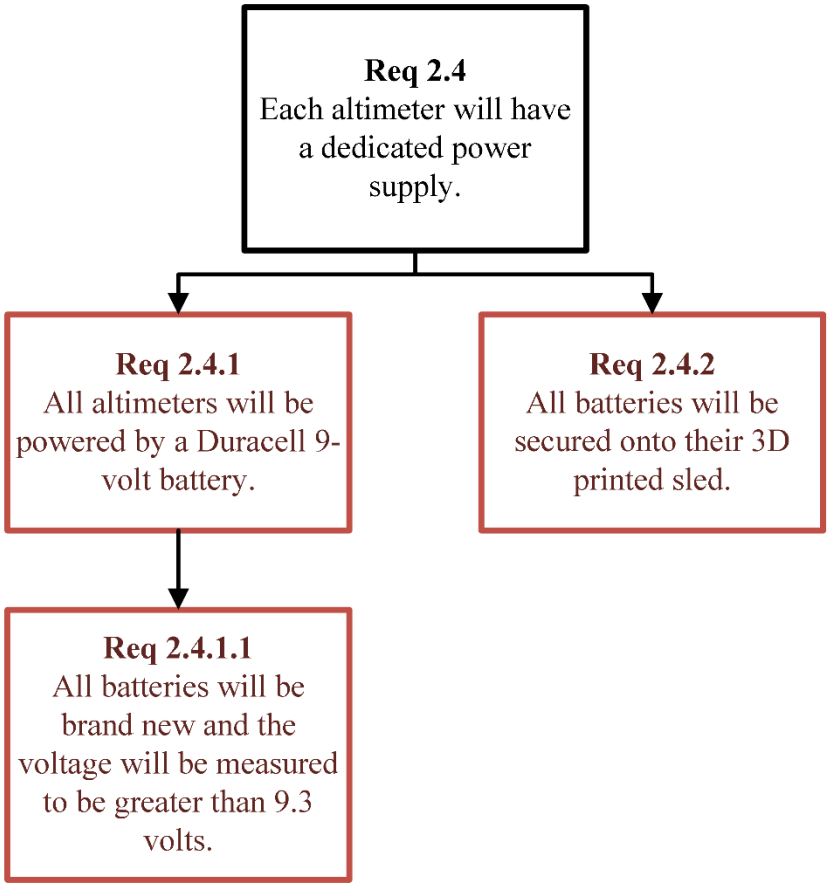
6.1.2.2.1 Derivation of Requirement 2.2.1

To reduce the likelihood of a separation failure, each separation event will be required to have two altimeters responsible for activating their respective black powder charge. This requirement exists to reduce the risk of the launch vehicle entering a ballistic state due to an electronics or e-match failure. The separation charges will be programmed with a delay relative to each other to prevent over-pressurization of the airframe.

6.1.2.2.2 Derivation of Requirement 2.2.2

To prevent the altimeters from disconnecting from their batteries, the altimeters will be required to be secured in the launch vehicle by a 3D printed sled. This requirement also prevents the altimeters from becoming damaged during flight and jeopardizing the success of the mission. The 3D printed sleds shall be designed to withstand the forces experienced during a flight.

6.1.2.3 Requirement 2.4



Requirement Number	Requirement Description	Method of Verification
2.4	Each altimeter will have a dedicated power supply.	**Inspection:** Each Stratologger altimeter shall be powered by a dedicated 9-Volt battery.
2.4.1	All altimeters will be powered by a Duracell 9-volt battery.	**Inspection:** Each altimeter shall be powered by a Duracell brand 9-Volt battery. Due to their internally welded contact points, Duracell brand batteries are well suited for the high accelerations experienced during flight.
2.4.1.1	All batteries will be brand new and the voltage will be measured to be greater than 9.3 volts.	**Test:** Each battery will be new and measured with a digital multi-meter to

		assure that the voltage is greater than 9.3 volts.
2.4.2	All batteries will be secured onto their 3D printed sled.	**Inspection:** Each battery will be secured to its 3D printed sled using a cover secured to the sled via four screws.

6.1.2.3.1 Derivation of Requirement 2.4.1

Due to the high acceleration experienced during flight, the altimeters will be required to be powered by Duracell brand 9-volt batteries. Duracell welds the cells inside the battery and therefore they can withstand the forces experienced during flight. Using other brands of batteries, with non-welded cells, could result in a loss of power to the altimeters upon liftoff, thus resulting in mission failure.

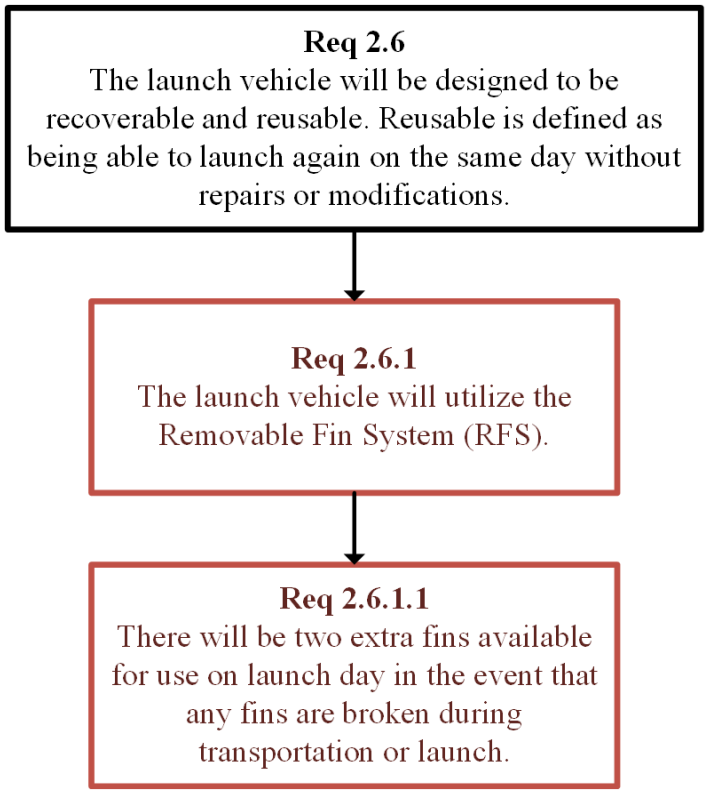
6.1.2.3.2 Derivation of Requirement 2.4.1.1

To ensure that the altimeters will receive enough power to accurately measure altitude and ignite the e-matches, all batteries used to power altimeters shall be brand new and measured to have greater than 9.3 volts. This requirement increases safety and the overall likelihood of mission success.

6.1.2.3.3 Derivation of Requirement 2.4.2

To prevent the batteries from disconnecting from the altimeters during flight, the batteries will be required to be secured in the launch vehicle by a 3D printed sled. The batteries shall be secured to the sled via a laser cut lid made from wood. This requirement prevents the altimeters from becoming damaged during flight and jeopardizing the success of the mission. The 3D printed sleds shall be designed to withstand the forces experienced during a flight.

6.1.2.4 Requirement 2.6



Requirement Number	Requirement Description	Method of Verification
2.6	The launch vehicle will be designed to be recoverable and reusable. Reusable is defined as being able to launch again on the same day without repairs or modifications.	**Analysis:** The launch vehicle shall be designed to be fully recoverable and reusable. Several test launches will be conducted before competition to verify recoverability and reusability of the launch vehicle.
2.6.1	The launch vehicle will utilize the Removable Fin System (RFS).	**Analysis:** The launch vehicle will be designed to accommodate the RFS and to remove and install fins from the launch vehicle.
2.6.1.1	There will be two extra fins available for use on launch day if any fins are broken during transportation or launch of the launch vehicle.	**Inspection**: The team will prepare a minimum of two extra fins before launches and account for the added expense in the team budget. This will be in conjunction with requirement 2.6.1.

6.1.2.4.1 Derivation of Requirement 2.6.1

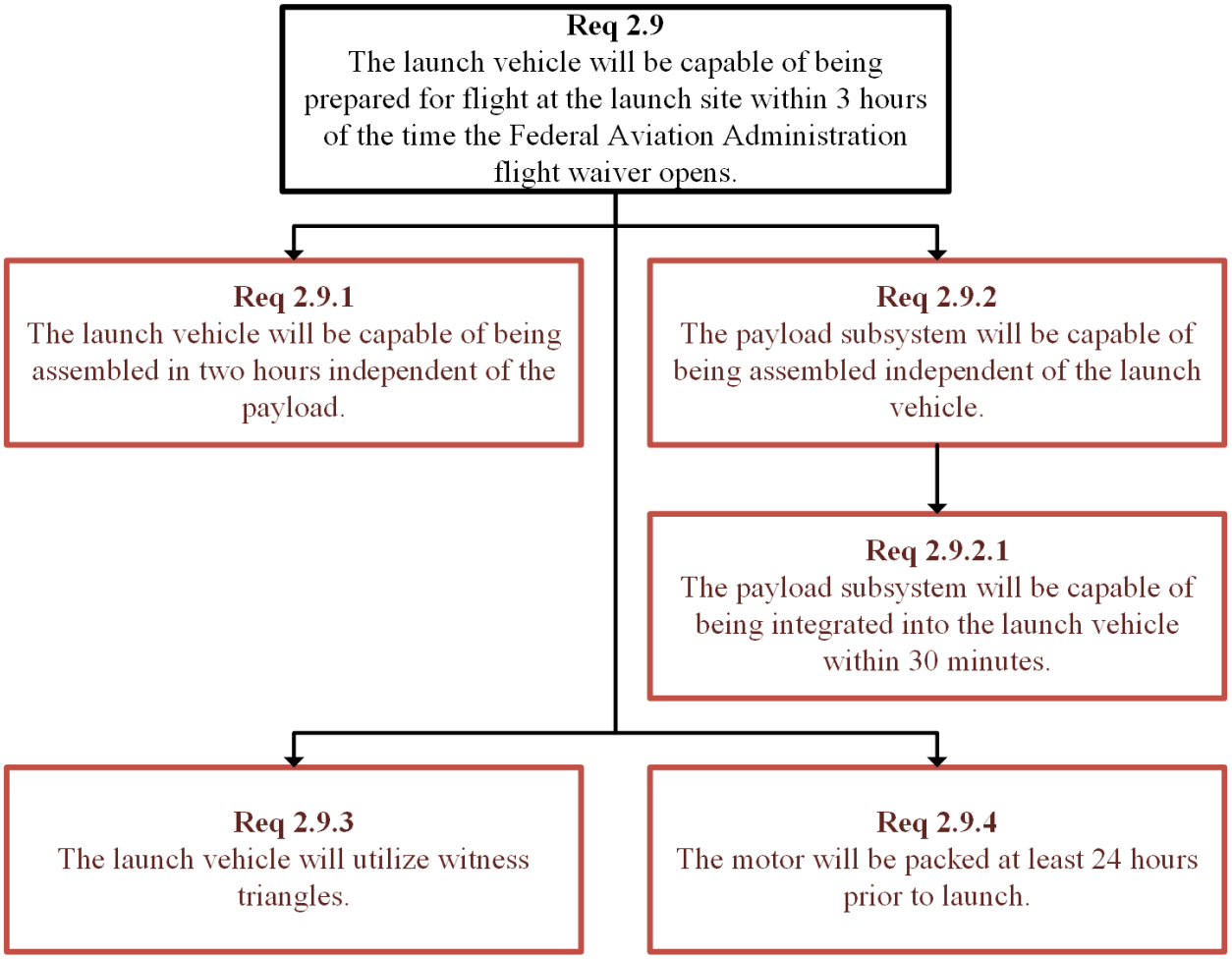
For the launch vehicle to be fully reusable, the ability to replace broken fins becomes a necessity. Without the RFS, a broken fin would require complete reconstruction of the booster section. This would be a setback to the team's schedule and budget. For this reason, the RFS is required to be

used on the launch vehicle to prepare for the possibility of a fin becoming damaged during a test flight.

6.1.2.4.2 Derivation of Requirement 2.6.1.1

In the event that a fin is damaged on launch day, the ability to replace the damaged fin quickly and easily would allow the launch vehicle to launch again on the same day. For this reason, it is required that the team has two extra fins prepared for flight on launch day. This requirement increases the likelihood of meeting Requirement 2.6.

6.1.2.5 Requirement 2.9



Requirement Number	Requirement Description	Method of Verification
2.9	The launch vehicle will be capable of being prepared for flight at the launch site within 3 hours of the time the Federal Aviation Administration flight waiver opens.	**Demonstration:** Assembly of the launch vehicle shall take less than 3 hours. A comprehensive checklist will be prepared by the team to assist in accurate and expedited vehicle assembly while preparing for flight.

2.9.1	The launch vehicle will be capable of being assembled in two hours independent of the payload.	**Demonstration:** The launch vehicle shall be designed to be assembled in less than two hours. A comprehensive checklist will be prepared by the team to assist in accurate and expedited vehicle assembly while preparing for flight. The team shall practice assembling the launch vehicle prior to competition to ensure Requirement 2.9.1 is met. This will be in accordance with Requirement 2.9.
2.9.2	The payload subsystem will be capable of being assembled independent of the launch vehicle.	**Analysis:** The launch vehicle shall be designed so that the rover payload can be assembled separate of the launch vehicle.
2.9.2.1	The payload subsystem will be capable of being integrated into the launch vehicle within 30 minutes.	**Demonstration:** The launch vehicle shall be designed so that the rover payload can be integrated into the launch vehicle within 30 minutes. The team shall practice integrating the payload into the launch vehicle to verify Requirement 2.9.2.1 is met.
2.9.3	The launch vehicle will utilize witness triangles.	**Inspection:** The launch vehicle shall utilize witness triangles on each coupler.
2.9.4	The motor will be packed at least 24 hours prior to the launch.	**Demonstration:** A minimum of 24 hours prior to launch, the launch vehicle's motor shall be packed.

6.1.2.5.1 Derivation of Requirement 2.9.1

To prepare for launch day and ensure that the launch vehicle meets Requirement 2.9, the launch vehicle is required to be capable of being assembled in two hours. This provides a buffer of one hour in the event of unexpected delays during assembly.

6.1.2.5.2 Derivation of Requirement 2.9.2

To prepare for launch day and ensure that the launch vehicle meets Requirement 2.9, the payload subsystem is required to be capable of being assembled independent of the launch vehicle. This requirement allows for the payload and vehicle sub-teams to work independently on launch day. This will result in an overall more efficient assembly process and ensure that no issues with the payload subsystem can affect the assembly time of the launch vehicle.

6.1.2.5.3 Derivation of Requirement 2.9.2.1

To prepare for launch day, and ensure that the launch vehicle meets Requirement 2.9, the payload subsystem is required to be capable of being integrated into the launch vehicle within 30 minutes.

This requirement will further increase the overall efficiency of the assembly process and increase the likelihood of the launch vehicle meeting Requirement 2.9.

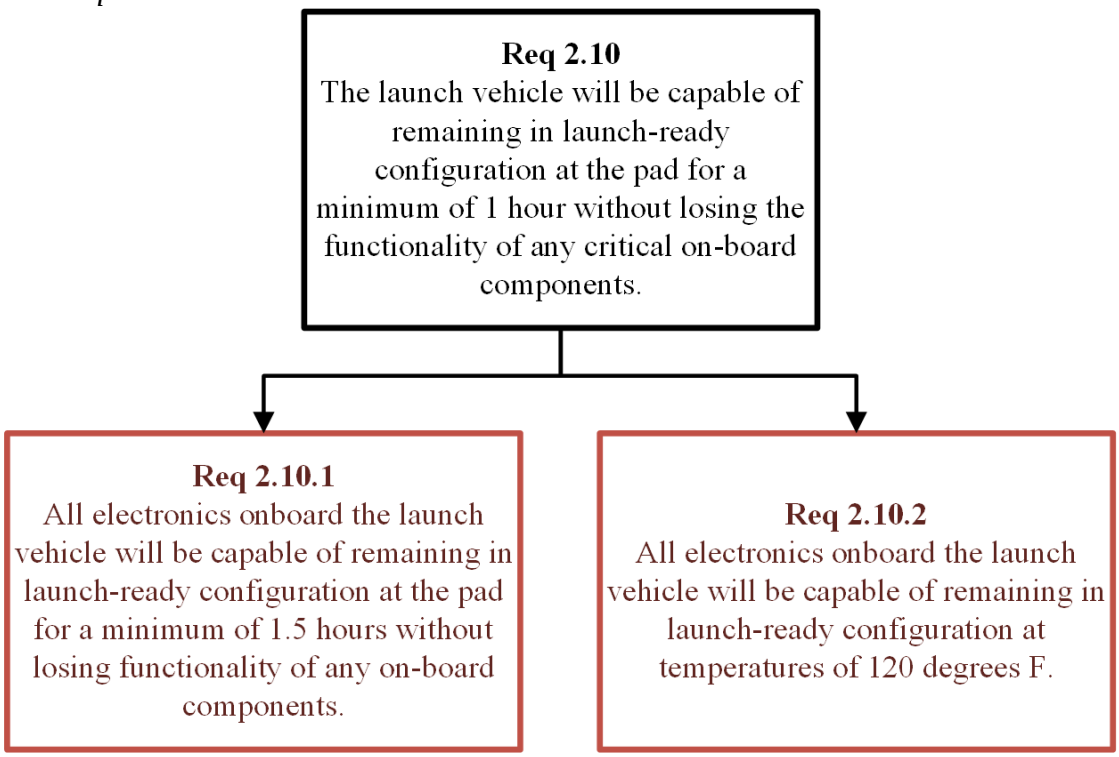
6.1.2.5.4 Derivation of Requirement 2.9.3

To prepare for launch day, and ensure that the launch vehicle meets Requirement 2.9, the launch vehicle is required to utilize witness rings on couplers and connecting sections of airframe. This will increase the overall efficiency of the assembly process as all venting/switch holes in the airframe will line up with their respective holes in the couplers.

6.1.2.5.5 Derivation of Requirement 2.9.4

To ensure that any glue used during the packing of the motor is fully cured at the time of launch, it is required that the motor be fully packed at least 24 hours prior to launch. This increases safety and the overall success of the mission by reducing the likelihood of a CATO.

6.1.2.6 Requirement 2.10



Requirement Number	Requirement Description	Method of Verification
2.10	The launch vehicle will be capable of remaining in launch-ready configuration on the pad for a minimum of 1 hour without losing the functionality of any critical on-board components.	**Demonstration:** The power supplies for the VDS and recovery electronics shall demonstrate the capability to exceed Requirement 2.10.

2.10.1	All electronics onboard the launch vehicle will be capable of remaining in launch-ready configuration at the pad for a minimum of 1.5 hours without losing functionality of any on-board components.	**Demonstration:** The power supplies for the VDS and recovery electronics shall demonstrate the capability to exceed Requirement 2.10.
2.10.2	All electronics onboard the launch vehicle will be capable of remaining in launch-ready configuration at temperatures of 120°F.	**Analysis:** The components chosen for the systems shall be considered for their ability to withstand high temperatures while still retaining functionality.

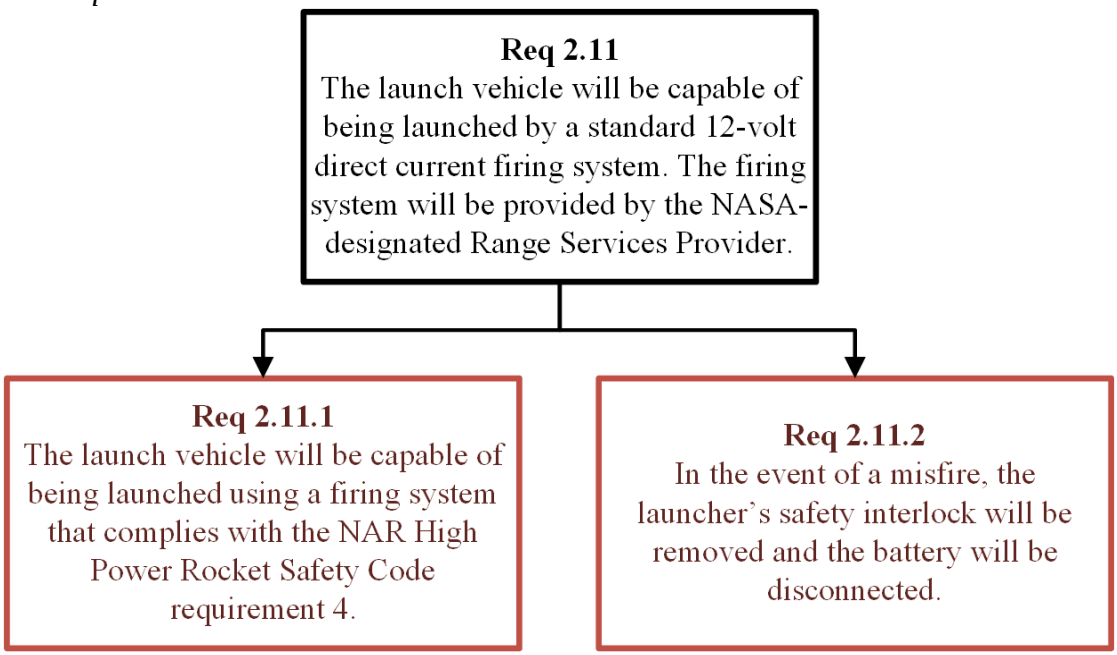
6.1.2.6.1 Derivation of Requirement 2.10.1

To prepare for unexpected delays once the launch vehicle is on the pad, it is required that all electronics onboard be capable of remaining in launch-ready configuration for a minimum of 1.5 hours. Requirement 2.10.1 increases the likelihood of the launch vehicle meeting Requirement 2.10 as it adds a 30-minute buffer to the one-hour requirement. Requirement 2.10.1 reduces the likelihood of any electronics failing on the pad and thus increases the likelihood of mission success.

6.1.2.6.2 Derivation of Requirement 2.10.2

To prepare for the possibility of high temperatures on launch day, all electronics on-board the launch vehicle must be capable of performing in temperatures of at least 120°F. This will ensure that in hot weather conditions, the launch vehicle electronics will perform nominally. Requirement 2.10.2 reduces the likelihood of any electronics failing on the pad and increases the likelihood of mission success.

6.1.2.7 Requirement 2.11

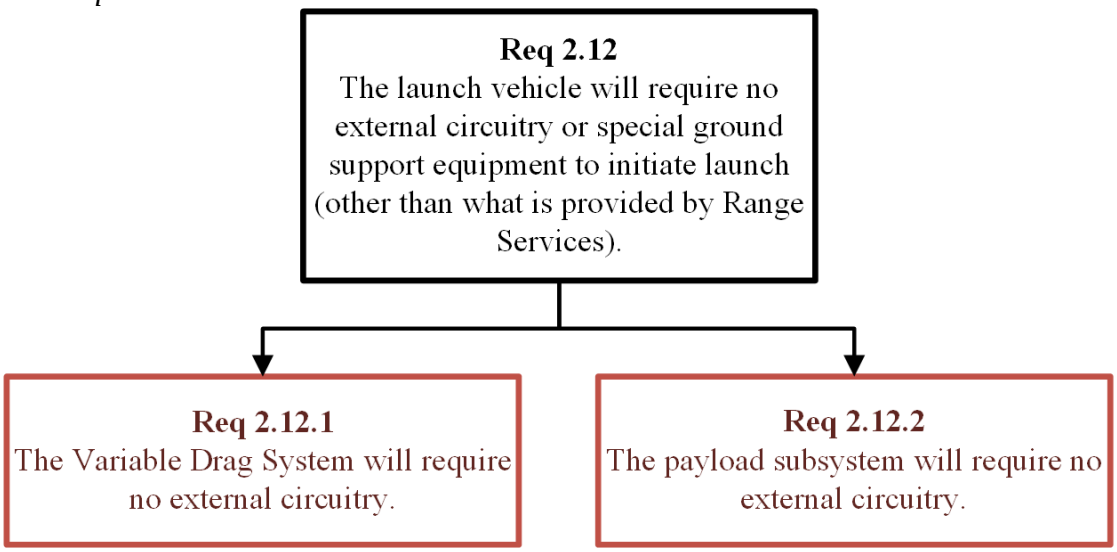


Requirement Number	Requirement Description	Method of Verification
2.11	The launch vehicle will be capable of being launched by a standard 12-volt direct current firing system. The firing system will be provided by the NASA-designated Range Services Provider.	**Inspection:** The chosen motor shall allow for the use of the standard 12-volt direct current firing system provided by the NASA-designated Range Service Provider.
2.11.1	The launch vehicle will be capable of being launched using a firing system that complies with the NAR High Power Rocket Safety Code requirement 4.	**Inspection:** The launch vehicle shall be designed to be launched by a firing system complying to NAR High Power Rocket Safety Code requirement 4. This is in accordance to Requirement 2.11.

6.1.2.7.1 Derivation of Requirement 2.11.1

To ensure a safe ignition of the launch vehicle's motor, it is required that the launch vehicle be capable of being launched via a firing system that complies with NAR High Power Rocket Safety Code Requirement 4. This will ensure that the launch vehicle can launch safely and meet Requirement 2.2 and Requirement 2.11.

6.1.2.8 Requirement 2.12



Requirement Number	Requirement Description	Method of Verification
2.12	The launch vehicle will require no external circuitry or special ground support equipment to initiate launch (other than what is provided by Range Services).	**Analysis:** The launch vehicle shall be designed for launch to be initiated only by an external source and shall not require any external support equipment. This is in accordance to requirement 2.11.

2.12.1	The Variable Drag System will require no external circuitry.	**Analysis:** The Variable Drag System shall be designed to not require any external circuitry and shall be a self-contained system.
2.12.2	The payload subsystem will require no external circuitry.	**Analysis:** The payload subsystem shall be designed to not require any external circuitry and shall be a self-contained system.

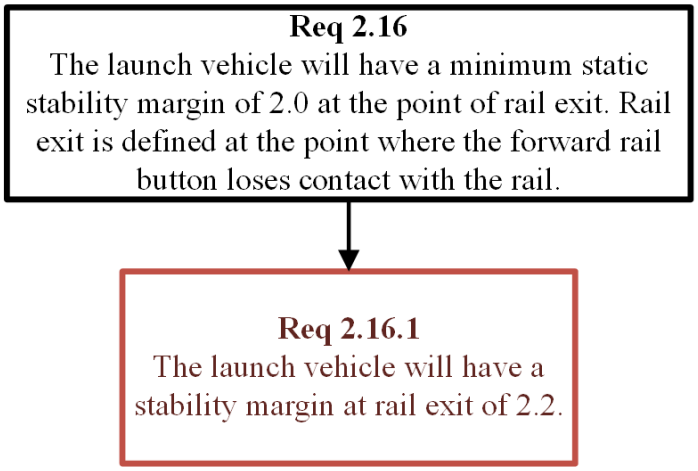
6.1.2.8.1 Derivation of Requirement 2.12.1

In the past, the VDS required an external cable leading from the booster to the nose cone. This cable added unnecessary risk in that it could have become tangled in the recovery equipment, or caused a separation failure. The external cable also added drag and mass to the launch vehicle. For these reasons, Requirement 2.12.1 has been introduced and requires that the VDS require no external circuitry. In this instance, any antennae system responsible for transmitting telemetry to the ground is not defined under external circuitry.

6.1.2.8.2 Derivation of Requirement 2.12.2

Any external circuitry would add drag that would inhibit the launch vehicle from reaching an apogee of 5,500ft. For this reason, the payload subsystem is required to include no external circuitry. In this instance, any antennae system responsible for deploying the payload upon landing is not defined under external circuitry.

6.1.2.9 Requirement 2.16



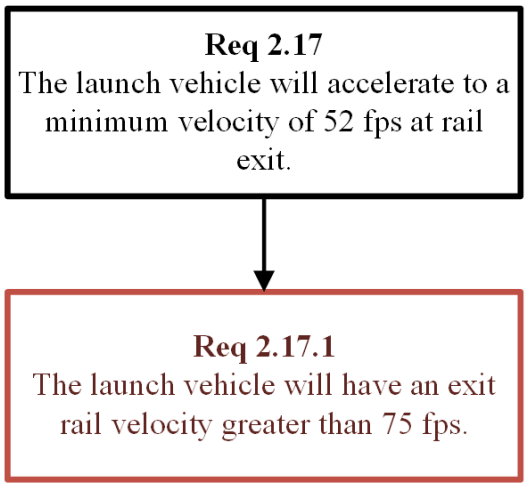
Requirement Number	Requirement Description	Method of Verification
2.16	The launch vehicle will have a minimum static stability margin of 2.0 at the point of rail exit. Rail exit is	**Analysis:** The launch vehicle shall be designed to have a stability of 2.0 at the point of rail exit. OpenRocket simulations shall be used to design the

	defined at the point where the forward rail button loses contact with the rail.	launch vehicle to the stability margin required and hand calculations shall be used to verify the stability.
2.16.1	The launch vehicle will have a stability margin at rail exit of 2.2.	**Analysis:** The launch vehicle shall be designed to have a stability margin of 2.2 upon rail exit.

6.1.2.9.1 Derivation of Requirement 2.16.1

The launch vehicle's stability margin can change dramatically with unexpected mass changes. Due to the fact that the masses for all components of the launch vehicle are estimates, requirement 2.16.1 requires that the launch vehicle be designed to have a stability margin of 2.2 at rail exit. This requirement ensures that the launch vehicle will fly stably and exceed Requirement 2.16.

6.1.2.10 Requirement 2.17

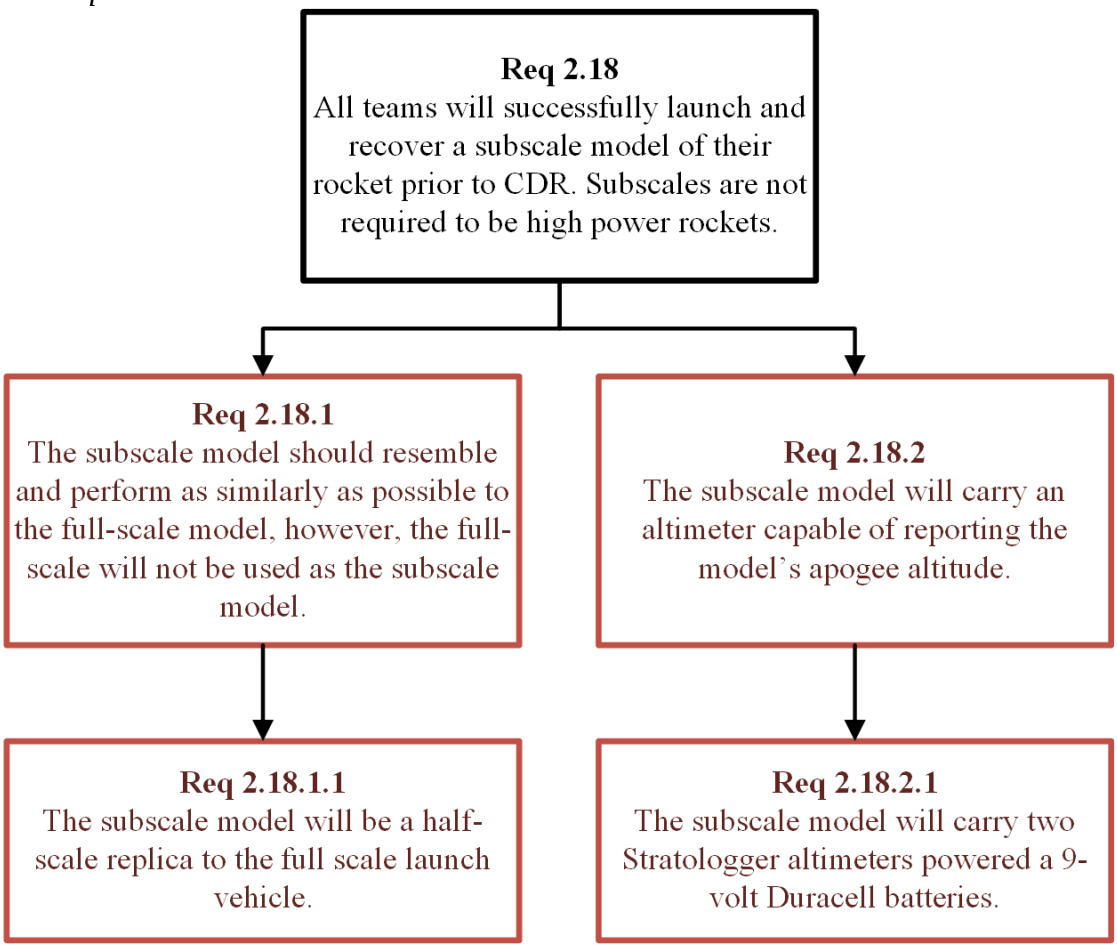


Requirement Number	Requirement Description	Method of Verification
2.17	The launch vehicle will accelerate to a minimum velocity of 52 fps at rail exit.	**Analysis:** The launch vehicle shall be designed to have a minimum velocity of 52 fps at the point of rail exit. OpenRocket simulations shall be used to select a motor to meet the requirement and hand calculations shall be used to verify.
2.17.1	The launch vehicle will have an exit rail velocity greater than 75 fps.	**Analysis:** The launch vehicle shall be designed to have an exit velocity of 75 fps at rail exit. OpenRocket simulations shall be used to select a motor to meet the requirement and hand calculations shall be used to verify.

6.1.2.10.1 Derivation of Requirement 2.17.1

Exit rail velocity can change dramatically with unexpected mass changes. As the masses for all components of the launch vehicle are estimates, Requirement 2.17.1 requires that the launch vehicle be designed to have an exit rail velocity greater than 75 fps. This requirement ensures that the launch vehicle will safely exit the rail and easily meet Requirement 2.17.

6.1.2.11 Requirement 2.18



Requirement Number	Requirement Description	Method of Verification
2.18	All teams will successfully launch and recover a subscale model of their rocket prior to CDR. Subscales are not required to be high power rockets.	**Test:** The team shall design, build, and test a subscale model of the rocket prior to CDR.
2.18.1	The subscale model should resemble and perform as similarly as possible to the full-scale model, however, the full-scale will not be used as the subscale model.	**Analysis:** The subscale model of the rocket shall be designed to closely follow the design of the full-scale model to ensure similar performance and flight characteristics.
2.18.1.1	The subscale model will be a half-scale replica to the full-scale launch vehicle.	**Analysis:** The subscale model shall be built as a 1:2 scale to be in order to test

		recovery design decisions and to identify potential design obstacles.
2.18.2	The subscale model will carry an altimeter capable of reporting the model's apogee altitude.	**Inspection:** Stratologger CF altimeters shall be mounted in the subscale model in order to report the model's apogee altitude.
2.18.2.1	The subscale model will carry two Stratologger altimeters powered by 9-volt Duracell batteries.	**Inspection:** The subscale model shall carry two Stratologger altimeters powered independently by new 9-volt Duracell batteries for redundancy.

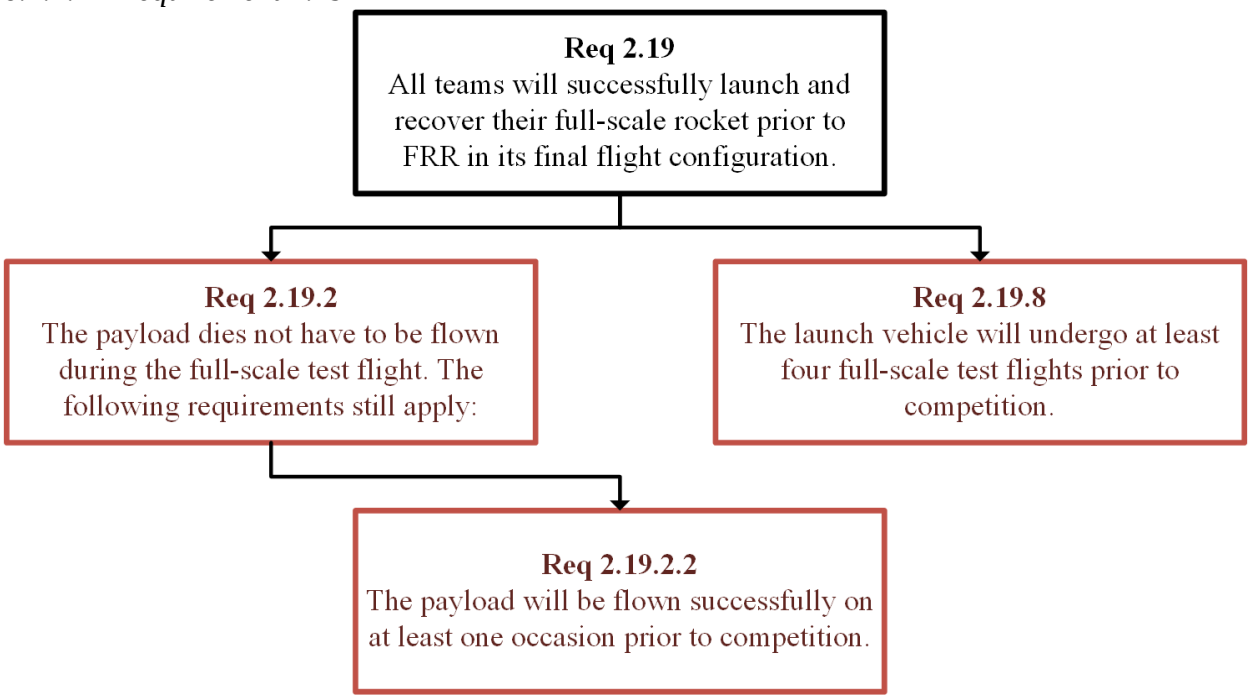
6.1.2.11.1 Derivation of Requirement 2.18.1.1

To design the subscale vehicle as closely to the full-scale vehicle as possible, the subscale will be required to be a ½ scale replica of the full-scale launch vehicle. This requirement helps ensure that no human errors are made when performing calculations relating to the dimensions of the subscale as it is exactly ½ of the full-scale.

6.1.2.11.2 Derivation of Requirement 2.18.2.1

In case of an unexpected altimeter failure, it is required that the subscale fly with two altimeters for redundancy. Due to the high acceleration experienced during flight, the altimeters will be required to be powered by Duracell brand 9-volt batteries. Duracell welds the cells inside the battery and therefore can withstand the forces experienced during flight. Using other brands of batteries with non-welded cells could result in a loss of power to the altimeters upon liftoff, thus resulting in mission failure.

6.1.2.12 Requirement 2.19



Requirement Number	Requirement Description	Method of Verification
2.19	All teams will successfully launch and recover their full-scale rocket prior to FRR in its final flight configuration.	**Test:** The team will conduct a test launch prior to FRR to verify the success of all launch vehicle systems.
2.19.2.2	The payload will successfully complete its mission on at least one occasion prior to competition.	**Test:** The rover will be tested to drive five feet and deploy its solar panels in a realistic environment prior to competition.
2.19.8	The launch vehicle will undergo at least four full-scale test flights prior to competition.	**Test:** Several full-scale launches will be conducted to identify potential problems and to verify repeatability of successful launch vehicle systems.

6.1.2.12.1 Derivation of Requirement 2.19.2.2

To ensure that the payload subsystem is in full working order by competition, it is required that the payload be flown at least once prior to competition. This will ensure that any issues with the payload deployment will be solved by competition.

6.1.2.12.2 Derivation of Requirement 2.19.8

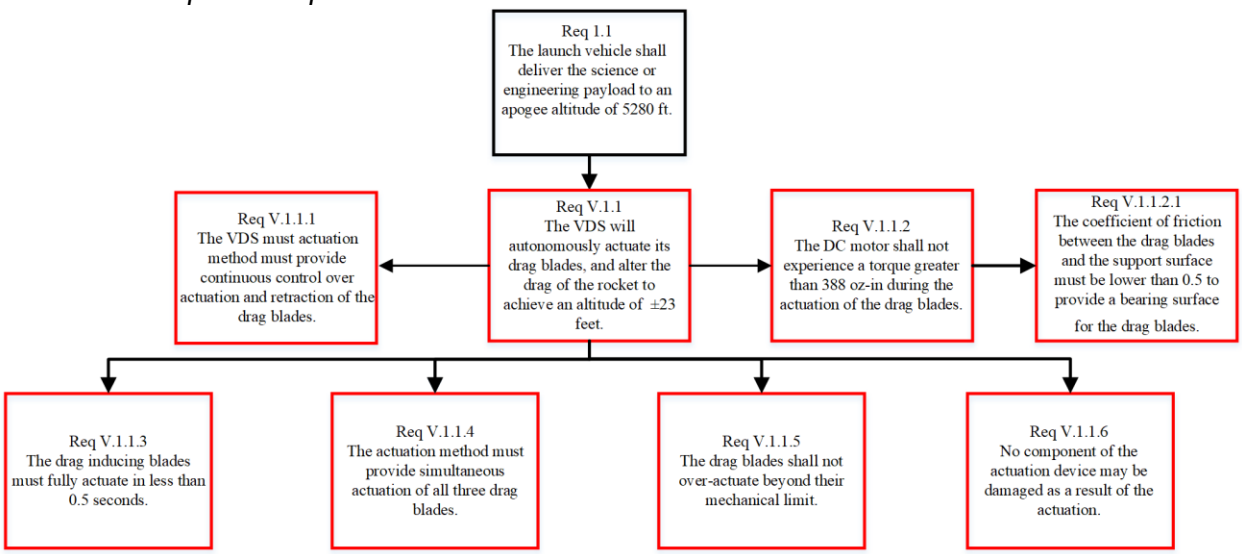
To be prepared for competition, the launch vehicle will be required to undergo at least four full-scale test flights. The first of which will serve as a control launch with an inactive VDS, the second will serve as a full-brake test for the VDS, and the third and fourth will serve as performance tests for the VDS and payload subsystems. These test flights will also ensure that any issues with recovery can be solved by competition.

6.1.2.13 Variable Drag System Requirements

Requirement number	Requirement	Method of Verification
1.1	The launch vehicle shall deliver the science or engineering payload to an apogee altitude of 5280 ft.	**Demonstration:** A flight altimeter will record the apogee altitude of the vehicle in each sub-scale and full-scale launch
2.2	The vehicle will carry one commercially available, barometric altimeter for recording the official altitude used in determining the altitude award winner. Teams will receive the maximum number of altitude points (5,280) if the official scoring altimeter reads a value of exactly 5280 feet AGL. The team will	**Demonstration:** The VDS will contain an aerospace grade barometric derived altimeter

	lose one point for every foot above or below the required altitude.	

6.1.2.13.1 Verification of Actuation Method



Requirement number	Requirement	Method of Verification
V.1.1	The VDS will autonomously actuate its drag blades, and alter the drag of the rocket to achieve an altitude of ±23 feet.	**Test:** The actuation method will be tested independently of the launch vehicle, as well as through several test launches to verify the system.
V.1.1.1	The VDS must actuation method must provide continuous control over actuation and retraction of the drag blades.	**Demonstration:** Three drag inducing blades set with radial gear teeth that mesh with a central spur gear will allow for continuous control by a single DC motor.
V.1.1.2	The drag inducing blades must fully actuate in less than 0.5 seconds.	**Analysis:** An actuation device which provides the fastest actuation speed will be chosen.
V.1.1.3	The DC motor shall not experience a torque greater than 388 oz-in during the actuation of the drag blades.	**Analysis:** The friction force between the drag blades and support plates will be calculated to determine the required torque to actuate the drag blades.
V.1.1.3.1	The coefficient of friction between the drag blades and the support	**Demonstration:** Delrin Acetal Resin was chosen for the bearing

	surface must be lower than 0.5 to provide a bearing surface for the drag blades.	surface because it has a coefficient of friction of approximately 0.3 with Aluminum.
V.1.1.4	The actuation method must provide simultaneous actuation of all three drag blades.	**Test:** A prototype of the VDS V3 gear assembly will be manufactured to verify that the design will provide a simultaneous actuation.
V.1.1.5	The drag blades shall not over-actuate beyond their mechanical limit.	**Demonstration:** Two limit switches will be fastened to the top VDS aluminum support plate, and communicate with the DC motor to prevent over actuation.
V.1.1.6	No component of the actuation device may be damaged as a result of the actuation.	**Analysis:** Finite Element Analysis was performed using ANSYS Workbench to verify the structural integrity of the gear mesh and drag blades.

Derivation of requirement V.1.1

The ability for the VDS to determine its own actuation is crucial to the success of the system. For the VDS to successfully slow the vehicle to ± 23ft of the target apogee, the drag blades must be able to control their own movement, and continuously alter their position throughout the coast phase of the launch.

Derivation of requirement V.1.1.1

Continuous control over drag blade actuation provides the VDS with full control over the magnitude of the drag force acting on the launch vehicle throughout the vehicle's ascent. The VDS can therefore alter the degree of drag blade actuation throughout the ascent to impart the drag force required to achieve an apogee of 5,280ft. ± 23ft.

Derivation of requirement V.1.1.3

To ensure that the VDS can precisely control the amount of drag force acting on the launch vehicle, it must be able to actuate the drag blades in less than 0.5 seconds. The high actuation speed allows the VDS to take full advantage of the control scheme designed to continuously alter the drag force on the vehicle.

Derivation of requirement V.1.1.3

The motor torque requirement for the actuation of the VDS drag blades was determined by performing a dynamic loading analysis on the VDS. The required torque is a function of the friction between the drag blades and the bearing surface and the moment arm over which the torque is applied. The maximum drag force was determined during test launches with the VDS V2 prototype to be approximately 20lbf. The maximum drag force was used to calculate the friction force with

$$f_k = D\mu \qquad \textbf{(38)}$$

where D is the maximum drag force and μ is the coefficient of friction between Delrin and Aluminum. The friction force is used to calculate the required torque with

$$\tau = f_k r \qquad \textbf{(39)}$$

where r is the distance from the centroid of the friction fore acting on the drag blade to the point of contact on the teeth of the central spur gear. The maximum torque required to actuate the drag blades with a safety factor of 2 and a gear inefficiency of 70% is 358oz-in. (22.375 in-lbs). Therefore, the DC motor must provide a minimum stall torque of 358oz-in. (22.375in-lbs.).

Derivation of requirement V.1.1.3.1

A minimal coefficient of friction between the drag blades and their bearing surface optimizes the VDS design by reducing the motor torque requirements and increasing actuation speed. Delrin Acetal Resin was chosen as the bearing plate material because it provides a coefficient of friction of 0.3 with Aluminum.

Derivation of requirement V.1.1.4

The VDS was designed to be a safe and reliable system with as few moving parts as possible. By simultaneously actuating the three drag blades with a single central spur gear, the risk of failure by any individual component is reduced.

Derivation of requirement V.1.1.5

To protect the motor from attempting to actuate the drag blades past their mechanical limit, limit switches will be implemented at the point of full actuation and full retraction of the drag blades.

Derivation of requirement V.1.1.6

The gear mesh between the central spur gear and the drag blades was designed against static failure. Fatigue failure was not considered as the gears will not be subjected to a significant number of loading cycles. The drag blades were also designed to withstand the maximum drag force they would encounter when actuated into the airflow surrounding the launch vehicle. The results of the Finite Element analysis are discussed further in section 3.4.8.4.1.5.

Requirement number	Requirement	Method of Verification
V.1.2	The VDS will telemetrically communicate its current state to a ground station during flight, without altering the path of the vehicle.	**Test:** This requirement will be verified in sub scale flight testing. The tests will verify whether the range and data transmission rates are of acceptable standards for the needs of the VDS.
V.1.3	The VDS shall have access to power controls externally from the vehicle.	**Demonstration:** The VDS bay will contain a port which grants access to the power source of the VDS in order to prevent power depletion during integration.
V.1.4	The VDS shall be capable of determining the state of the vehicle (i.e. altitude and velocity) with noise limits of no more than ± 5.0 m and ± 5.0 m/s respectively.	**Demonstration:** The VDS will take data points and employ the use of a built in sensory Kalman filter in order to reduce noise in data intake.

Derivation of requirement V.1.2

In order to validate whether the vehicle is take the trajectory that is anticipated, and also to identify the level of noise in the data, it is necessary to have the VDS deliver this data to a ground station in real time throughout the flight of the vehicle. The system will have no effect on the operation of the VDS software, nor will commands be signaled to the system from the ground. The sole purpose of this is to acquire data as accurately and as quickly as possible.

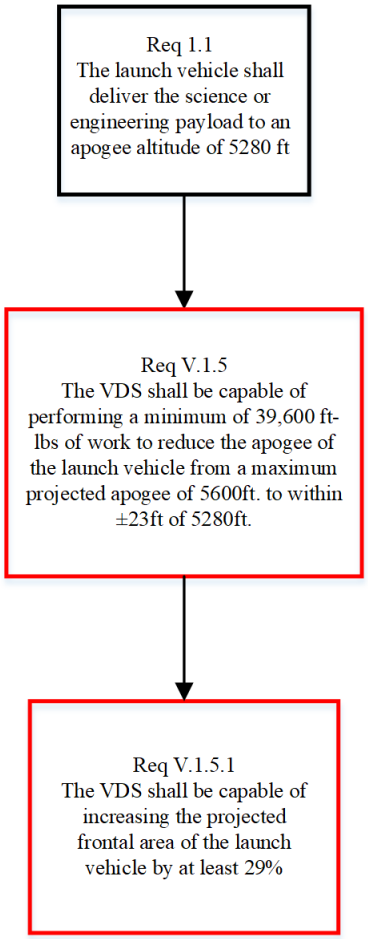
Derivation of requirement V.1.3

During testing of the VDS V2 it became apparent that power retention is a critical issue during integration of the vehicle. Because the VDS is the first system that is integrated, as it is contained within the booster section, the system must be active throughout the installment of both the payload as well as the recovery which occasionally is a lengthy process; this depletes the batteries of the system before the vehicle has begun launch. To mitigate this, there will be an external access port on the body of the vehicle from which a computer or alternative device is able to control and power the VDS until integration is complete and the rocket is ready for launch.

Derivation of requirement V.1.4

It is crucial for the VDS to be able to determine the state of the vehicle during ascent.

6.1.2.13.2 Verification of Braking Power Requirements



Requirement number	Requirement	Method of Verification
V.1.5	The VDS shall be capable of performing a minimum of 39,600 ft-lbs of work to reduce the apogee of the launch vehicle from a maximum projected apogee of 5600ft. to within ±23ft of 5280ft.	**Test:** Multiple launches will be conducted to verify that the VDS can perform the required amount of work on the launch vehicle.
V.1.5.1	The VDS shall be capable of increasing the projected frontal area of the launch vehicle by at least 29%	**Inspection:** Computer aided design software was used to verify that the VDS V3 will increase the projected area of the rocket by at least 29%

Derivation of requirement V.1.5

The VDS must be able to reduce the apogee of the launch vehicle from 5,600ft. to 5,280ft. The VDS V3 can reduce the apogee of the vehicle by 880ft., or by a magnitude of 2.75. This design provides greater assurance that slight variations in motor thrust, the weight of the launch vehicle, or weather conditions will not affect the ability of the VDS to deliver the launch vehicle to 5,280ft. To reduce the apogee by 880ft., the VDS must perform 39,600ft-lbs of work on the launch vehicle. The work requirement was calculated using

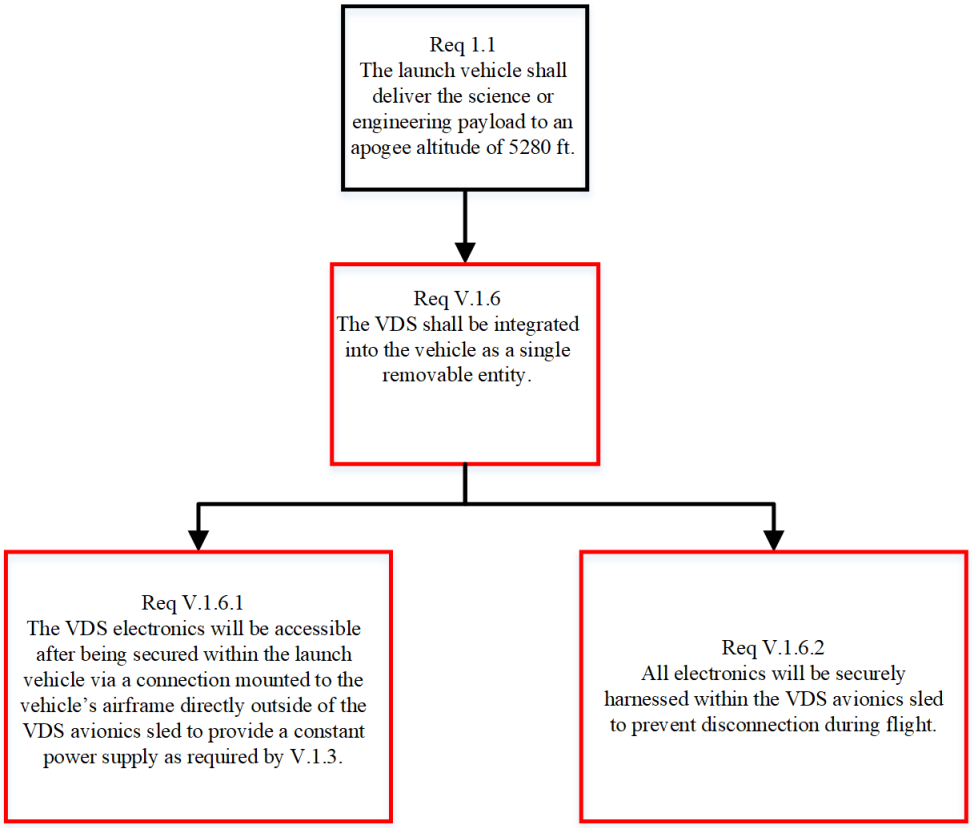
$$W = mg\Delta h \tag{40}$$

where m is the total mass of the launch vehicle, g is the gravitational constant, and h is the apogee of the vehicle.

Derivation of requirement V.1.5.1

The VDS V2 was proven capable of delivering the launch vehicle to 5,280ft. by increasing the projected frontal area of the launch vehicle by 29%. The VDS V3 was designed to increase the projected frontal area by 29%.

6.1.2.13.3 Verification of VDS Integration Method



Requirement number	Requirement	Method of Verification
V.1.6	The VDS shall be integrated into the vehicle as a single removable entity.	**Demonstration:** CAD software will be used to ensure that all components of the VDS fit within a single 6

		in. x 12 in. carbon fiber coupler.
V.1.6.1	The VDS electronics will be accessible after being secured within the launch vehicle via a connection mounted to the vehicle's airframe directly outside of the VDS avionics sled to provide a constant power supply as required by V.1.3.	**Test:** An audible signal will be produced by the VDS avionics when connected to a power supply.
V.1.6.2	All electronics will be securely harnessed within the VDS avionics sled to prevent disconnection during flight.	**Inspection:** All components will be visually inspected to ensure that all fasteners have been set in place before inserting the VDS into the launch vehicle.

Derivation of requirement V.1.6

Each subsystem must be designed with the intent to be efficiently integrated into the launch vehicle without negatively impacting any other subsystem. The chance of impacting the integration of any other subsystem is reduced by integrating the VDS into the vehicle as a single removable entity.

Derivation of requirement V.1.6.1

Due to the possibility of a long wait time on the launch pad, the VDS electronics may require an external power supply before launch. Mounting a connection port directly through the airframe surrounding the avionics sled provides an efficient method to maintain a power supply to the VDS electronics.

Derivation of requirement V.1.6.2

All electronics will be safely secured within the avionics sled to prevent malfunction or disconnection during flight using #10-40 screws. Mechanically fastening the electronics to a 3D printed sled provides a secure attachment method, and allows for removal if maintenance or alterations are necessary.

6.1.3 Team Derived Safety Requirements

Requirement Number	Requirement	Verification

Safety 1.1	Provide the team with an updated Safety Manual which includes team safety review plans, PPE requirements, emergency equipment, MSDS, machine operation instructions, FAA laws, and NAR and TRA regulations.	**Demonstration** The Safety Officer must keep a team Safety Manual updated with appropriate procedures prior to the subscale launch and all safety procedures prior to the first full-scale launch. The manual will be posted on the team website.
Safety 1.2	Make MSDS and Machine Cage operator manuals available and accessible to the team at all times.	**Demonstration** All MSDS are in the Safety Manual that will be continuously updated as the team uses new materials.
Safety 1.3	Require and confirm that all team members have read and agree to comply with all regulations set forth by the Safety Manual.	**Demonstration** The Safety Officer will ensure that each member of the team has agreed to follow the Safety Manual by the Preliminary Design Review. New members must agree to the regulations prior to beginning any work for the team.
Safety 1.4	Identify safety violations and take appropriate action to correct them.	**Demonstration** Team members that violate the Safety Manual will have manufacturing and launch day eligibilities revoked until they meet with the Safety Officer and agree to follow all of the established rules.
Safety 2.1	Participate in testing preparations and processes to ensure that risks are mitigated.	**Demonstration** The Safety Officer must sign off on each testing procedure before it is implemented.
Safety 3.1	Enforce proper use of Personal Protective Equipment (PPE) during manufacturing, construction, testing, and flight of the rocket.	**Inspection** The Safety Manual has proper safety techniques for construction and ground testing that each team member must sign.
Safety 4.1	Provide a plan for proper purchase, storage, transportation, and use of all energetic devices.	**Demonstration** All energetic devices will be transported by vehicle and kept inside a clearly identified explosives box.

Table 94. Safety Officer requirements and verifications.

6.1.4 Recovery Requirements

6.1.4.1 SOW Verifications

Requirement Number	Requirement Description	Method of Verification
3.1	The launch vehicle will 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.	**Analysis:** The StratologgerCFs shall be programed such that the payload section and booster section drogue parachutes will be deployed at apogee and both main parachutes will be deployed at 500 ft.
3.2	Each team must perform a successful ground ejection test for both the drogue and main parachutes. This must be done prior to the initial subscale and full-scale launches.	**Demonstration:** Black powder tests shall be repeated until nominal effects are observed a minimum of two times during initial testing and a minimum of one time before each flight of the launch vehicle.
3.3	At landing, each independent sections of the launch vehicle will have a maximum kinetic energy of 75 ft-lbf.	**Analysis:** All parachute sizes shall be calculated according to the kinetic energy requirement such that all sections of the launch vehicle shall land with a kinetic energy under 75 ft-lb.
3.4	The recovery system electrical circuits will be completely independent of any payload electrical circuits.	**Inspection:** All StratologgerCFs will be independent of any surrounding electrical circuits and will be individually powered by Duracell 9V batteries.
3.5	All recovery electronics will be powered by commercially available batteries.	**Inspection:** Each StratologgerCF shall be powered by a new Duracell 9-Volt battery. Duracell batteries are chosen because of their internally soldered leads which makes them highly reliable.

3.6	The recovery system will contain redundant, commercially available altimeters. The term "altimeters" includes both simple altimeters and more sophisticated flight computers.	**Inspection:** The recovery system shall contain four PerfectFlite StratologgerCF altimeters. Two StratologgerCFs will be included in each recovery bay for redundancy.
3.7	Motor ejection is not a permissible form of primary or secondary deployment.	**Inspection:** The launch vehicle shall use an AeroTech L2200 which does not include an ejection charge after the burnout of the propellant grain.
3.8	Removable shear pins will be used for both the main parachute compartment and the drogue parachute compartment.	**Inspection:** Three nylon 4-40 socket head cap screw shear pins shall be used for each recovery bay intended to separate.
3.9	Recovery area will be limited to a 2500 ft. radius from the launch pads.	**Analysis:** Drift calculations will be done to ensure that all components of the launch vehicle shall be recovered within 2500 ft. of the launch rail.
3.10	An electronic tracking device will be installed in the launch vehicle and will transmit the position of the tethered vehicle or any independent section to a ground receiver.	**Inspection:** Each independent section shall include a Trackimo GPS tracking device.
3.10.1	Any rocket section, or payload component, which lands untethered to the launch vehicle, will also carry an active electronic tracking device.	**Inspection:** A Trackimo GPS tracking device shall be secured in each independent section of the launch vehicle, and will transmit the position of the launch vehicle to a ground receiver. The payload shall also carry a GPS tracking device.
3.10.2	The electronic tracking device will be fully functional during the official flight on launch day.	**Demonstration:** The electronic tracking devices shall be tested and monitored during all test flights prior to the official flight to ensure that they are fully functional and used on the official launch day.

3.11	The recovery system electronics will not be adversely affected by any other on-board electronic devices during flight (from launch until landing).	**Analysis:** All on-board electronic devices shall be designed not to interfere or affect the recovery system electronics.
3.11.1	The recovery system altimeters will be physically located in a separate compartment within the vehicle from any other radio frequency transmitting device and/or magnetic wave producing device.	**Inspection:** All StratologgerCFs shall be placed in a separate section of the launch vehicle from any other radio frequency transmitting device and/or magnetic wave producing device.
3.11.2	The recovery system electronics will be shielded from all onboard transmitting devices, to avoid inadvertent excitation of the recovery system electronics.	**Analysis:** The recovery devices shall be designed to be shielded from all transmitting devices on the launch vehicle.
3.11.3	The recovery system electronics will 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.	**Analysis:** The recovery devices shall be designed to be shielded from all transmitting devices on the launch vehicle that generate magnetic waves.
3.11.4	The recovery system electronics will be shielded from any other onboard devices which may adversely affect the proper operation of the recovery system electronics.	**Demonstration:** The recovery system electronics will be tested during test flights of the launch vehicle to ensure that any other onboard electronic devices to not interfere with the performance of the recovery system electronics.

Table 95: SOW requirement verification

6.1.4.2 Team Derived Requirements

In addition to the Requirements dictated in the SOW outlined in Table , self-derived requirements have been made to ensure a safe recovery for the launch vehicle, the onboard electronics, and the crowd. These requirements will be represented as TDR.x.x to show they are self-derived and separate from the SOW requirements. These requirements are shown in Figure 145 below.

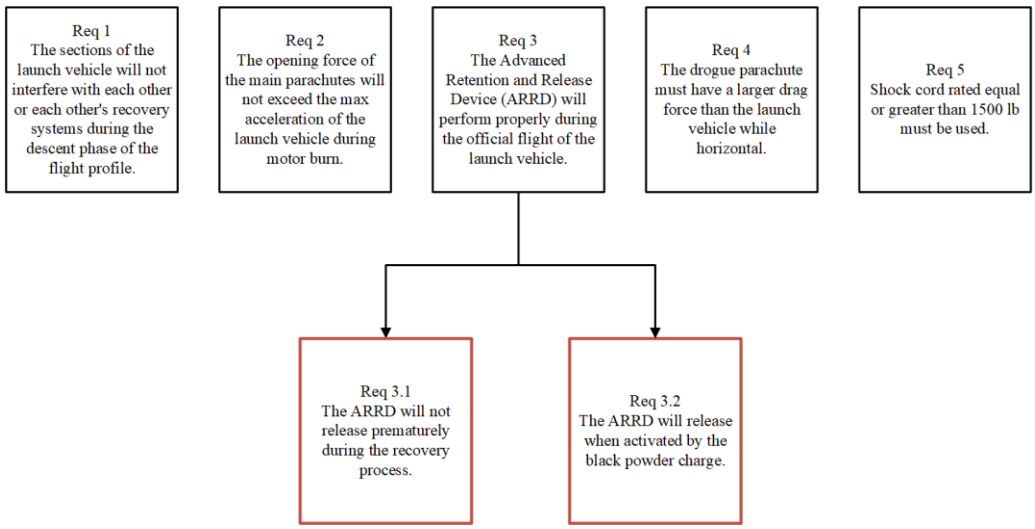


Figure 145: Self derived requirement flowchart

These requirements and verification are detailed below in Table 96.

Requirement Number	Requirement Description	Method of Verification
TDR1	The sections of the launch vehicle will not interfere with each other or each other's recovery systems during the descent phase of the flight profile.	**Demonstration:** The recovery system shall be designed so that the booster drogue parachute will be deployed at apogee, and after a two second delay, the payload section drogue parachute will be deployed. This will ensure that the payload and booster sections of the launch vehicle will descend at different altitudes to prevent interaction. This system shall be tested during test flights of the launch vehicle prior to the official competition flight.
TDR2	The opening force of the main parachutes will not exceed the max acceleration of the launch vehicle during motor burn.	**Test:** The main parachutes shall each have an opening force reduction ring which will create friction with the shroud lines and reduce the opening forces. This system shall be tested during test flights of the launch vehicle and data will be collected via sensors on-board the payload and booster sections of the launch vehicle. The data will be analyzed, and the size of the opening force reduction ring will be adjusted to ensure that the opening

		force of the main parachutes will not exceed ft-lb
TDR3	The Advanced Retention and Release Device (ARRD) will perform properly during the official flight of the launch vehicle.	**Demonstration:** Ground tests shall be done on the ARRD to ensure that it will perform properly during the official flight of the launch vehicle. Test flights prior to the competition flight shall also test the redundancy and reliability of the ARRD to ensure that it will perform as planned during the competition flight.
TDR3.1	The ARRD will not release prematurely during the recovery process.	**Demonstration:** The ARRD shall be load tested to at least 400 lbs to ensure that it will not release. If release occurs, the ARRD shall be reassembled and tested again to ensure that it will not release prematurely during the recovery process.
TDR3.2	The ARRD will release when activated by the black powder charge.	**Demonstration:** The ARRD shall be tested prior to the official flight of the launch vehicle to ensure that it will release when activated by the black powder charge.
TDR4	The drogue parachute must have a larger drag force than the launch vehicle while horizontal.	**Analysis:** The drogue parachute shall be designed to have a larger drag force than the launch vehicle while horizontal. ANSYS fluid simulation will be used to verify calculations.
TDR5	Shock cord used must be rated equal to or greater than 1500 lbs-f.	**Inspection:** Shock cord rated for at least 1500 lb shall be used during all flights of the launch vehicle.

Table 96: Self derived requirement verifications

6.1.5 Payload Requirements

6.1.5.1 Subsystem Requirements, Verification Plan, and Derivation

This section will provide the team derived requirements for each subsystem, the method for verification of each requirement, and a description of the derivations.

6.1.5.1.1 Rover Orientation Correction System (ROCS)

Requirement Number	Requirement	Verification Method
ROCS.1	The ROCS must be entirely removeable from the payload bay.	Demonstration The ROCS will be inserted and removed from the payload bay multiple times.
ROCS.2	The ROCS must reliably come to rest such that the rover is upright inside the payload bay.	Test Ground testing will be performed to verify requirement. See section 5.13.4.1
ROCS.3	The ROCS must withstand high loads experienced during flight remaining functional after experiencing these loads.	Analysis/Test Analysis will be done to ensure that the rocks will maintain a sufficient factor of safety. Full-scale flight tests will confirm analysis results.

Table 97: ROCS requirements.

Derivation of Requirement ROCS.1

The payload must be entirely enclosed in the airframe of the launch vehicle. This requirement has been imposed to drive a design that will minimize time to integrate the payload into the launch vehicle. This also provides ease of access to the entire payload for servicing and analysis of flight effects on the ROCS.

Derivation of Requirement ROCS.2

The current design of the rover determined by the trade studies performed requires proper orientation to be achieved prior to the rover beginning its mission. This requirement has been imposed to ensure that the rover will be able to continue with its mission. This also allows for the redundant checking of orientation before the RLM can be released using gyroscope sensors to ensure safety of all bystanders.

Derivation of Requirement ROCS.3

This requirement has been imposed in the interest of safety of all bystanders. The highest loads that the system will experience during flight will be determined and accounted for in the design.

6.1.5.1.2 Rover Locking Mechanism (RLM)

Requirement Number	Requirement	Verification Method
RLM.1	The RLM must retain the rover in the payload bay throughout flight.	Test The RLM will be engaged during multiple full-scale flights. See section 5.15.2
RLM.2	The RLM must sustain the high loads of liftoff, opening force, and landing.	Analysis/Test Analysis will be done to ensure that the RLM will maintain a sufficient factor of safety. Full-scale flights will also confirm verification. See section 5.13.3
RLM.3	The RLM must release the rover upon receiving the deployment signal.	Demonstration Ground testing will demonstrate that the RLM reliably releases the rover.

Table 98: RLM requirements.

Derivation of Requirement RLM.1

This requirement has been imposed in the interest of safety for all bystanders. This requirement will also be part of satisfying requirement 5.1.1 of the SOW ensuring that the rover will not deploy until the payload bay has landed.

Derivation of Requirement RLM.2

The RLM must be capable of sustaining functionality during and after exposure to maximum loads experience by the system during liftoff and payload bay main parachute opening force to ensure the safety of all bystanders.

Derivation of Requirement RLM.3

This requirement has been imposed to fulfill requirement 5.1.1 of the SOW. The rover must be able to be deployed to continue its primary mission of driving five feet and deploying the solar array. If this requirement were not met, the rover would not be able to deploy from the launch vehicle rendering the mission a failure.

6.1.5.1.3 Deployment Trigger System (DTS)

Requirement Number	Requirement	Verification Method
DTS.1	The DTS must maintain signal reception within a radius of 2500 ft. from the transmitter.	Demonstration Ground testing will demonstrate a range of at least 2500 ft.
DTS.2	The DTS receiver antenna must reliably detach from the receiver module without restricting motion of the rover.	Demonstration Demonstration of the rover detaching from the receiver module will be conducted.

Table 99: DTS requirements.

Derivation of Requirement DTS.1

This requirement has been imposed to due to NASA requirement 5.1.1 of the SOW. This will ensure that the rover will be capable of maintaining signal reception anywhere within the acceptable recovery radius having the transmitter stationed at the flight line.

Derivation of Requirement DTS.2

This requirement has been imposed due to the results of the deployment trigger trade study shown in section 6.1.5.1.4. The receiver module antenna will be securely fixed to the exterior of the airframe meaning that it must detach from the rover in order for the rover to drive forward.

6.1.5.1.4 Rover Body Structures (RBS)

Requirement Number	Requirement	Verification Method
RBS.1	The RBS must house all payload electronics, mechanical systems, and the SAS in a secure and accessible manner.	Inspection Inspection of the rover body will ensure adequate space for electronics and solar array.
RBS.2	The RBS must sustain loads experienced during flight and recovery.	Analysis/Test Analysis will be done to ensure the rover body can sustain flight loads. Full-scale launches will also confirm verification. See section 5.15.4
RBS.3	The RBS must maintain sufficient clearance above the ground.	Inspection Inspection will confirm clearance between the ground and rover body.

Table 100: RBS requirements.

Derivation of Requirement RBS.1

This requirement has been imposed for ease of access to all payload systems to reduce integration and servicing times. The results of the rover trade study shown in 5.8have determined that a design in which the electronics travel with the rover will be pursued requiring a structure capable of supporting the electronics.

Derivation of Requirement RBS.2

The body structure must be capable of maintaining integrity during and after maximum loads experienced during flight to ensure the safety of all bystanders.

Derivation of Requirement RBS.3

This requirement has been imposed as part of ensuring that the rover will be capable of achieving NASA requirement 5.1.1 of the SOW. The rover must maintain clearance above the ground so as to not bottom out before reaching five feet from the launch vehicle.

6.1.5.1.5 Rover Drive System (RDS)

Requirement Number	Requirement	Verification Method
RDS.1	The RDS must provide sufficient torque to drive the rover forward.	Demonstration Ground testing will demonstrate the ability of the RDS to provide sufficient torque.
RDS.2	The RDS must allow the rover to surmount an obstacle in front of the rover.	Test Ground testing will be performed to verify requirement. See section 5.15.5
RDS.3	The RDS must allow the rover to drive up and down a sloped terrain.	Test Ground testing will be performed to verify requirement. See section 5.15.
RDS.4	The RDS must maintain traction in any ground condition.	Test Ground testing will be performed to verify requirement. See section 5.15.

Table 101: RDS requirements.

Derivation of Requirement RDS.1

This requirement has been imposed to ensure that the motors chosen to drive the rover are capable of advancing the rover forward during its primary mission. This will be part of satisfying NASA requirement 4.5.3 of the SOW.

Derivation of Requirement RDS.2

This requirement has been imposed due to the unpredictability objects that will be scattered in the launch field. The rover must be able to account for objects that could hinder its forward motion. This will be part of satisfying NASA requirement 5.1.1of the SOW.

Derivation of Requirement RDS.3

This requirement has been imposed due to the unpredictability of the uneven terrain of the launch field. The rover must be capable of overcoming both positive and negative slopes. This will be part of satisfying NASA requirement 5.1.1 of the SOW.

Derivation of Requirement RDS.4

This requirement has been imposed due to the unpredictability of the condition of the launch field on launch day. Weather and farm work can have significant impact on the condition of the surface of the launch field. The rover must be capable of remaining of performing completing its mission

in any field condition. This will be part of satisfying NASA requirement 4.5.3 of the SOW. Statement of Work

6.1.5.1.6 Control Electronics System (CES)

Requirement Number	Requirement	Verification Method
CES.1	The CES must recognize reception of the deployment signal.	Demonstration Software logic will be tested to demonstrate repeatability of performance.
CES.2	The CES must confirm via two gyroscope sensors that proper orientation has been achieved after the deployment signal has been recognized.	Test Ground testing will be performed to confirm performance. See section 5.15.6
CES.3	The CES must release the RLM.	Demonstration Software logic will be tested to demonstrate repeatability of performance.
CES.4	The CES must control the forward motion and maneuverability of the rover.	Demonstration Software logic will be tested to demonstrate repeatability of performance.
CES.5	The CES must analyze the data sent from the OAS and determine the optimal drive path for the rover.	Test Ground testing will be performed to confirm performance. See section 5.15.6
CES.6	The CES must control deployment of the solar panels.	Demonstration Software logic will be tested to demonstrate repeatability of performance.
CES.7	The CES must log all data and picture on a microSD card.	Demonstration Software logic will be tested to demonstrate repeatability of performance.
CES.8	The CES controller battery lifetime must exceed three hours.	Demonstration The battery lifetime will be experimentally confirmed.

Table 102: CES requirements.

Derivation of Requirement CES.1

The control scheme of the rover must be configured in such a manner that the deployment of the rover is completely controlled by a team member with the transmitter of the DTS. This requirement will ensure that reception of the deployment signal will be a necessary condition to deploy the rover in the control software logic. This is done in the interest of safety for all bystanders to avoid deployment of the rover before landing.

Derivation of Requirement CES.2

It is necessary that multiple conditions be met before the CES deploys the rover. This requirement has been imposed as a safety measure to further ensure that no premature deployment of the rover will occur. The addition of this redundant condition in the control software logic will ensure the safety of all bystanders and success of the primary mission of the payload.

Derivation of Requirement CES.3

This requirement has been imposed to ensure that no system other than the CES be capable of unlocking the RLM and deploying the rover. This will minimize the risk of premature deployment.

Derivation of Requirement CES.4

This requirement has been imposed to allow the rover to move both forward and backward, and turn in any direction. This

Derivation of Requirement CES.5

This requirement has been imposed to ensure that in the event that the OAS recognizes an insurmountable object, the rover must be able to avoid that obstacle.

Derivation of Requirement CES.6

This requirement has been imposed to ensure that no system other than the CES be capable of allowing the solar array to deploy. Premature deployment of the solar array may hinder the motion of the rover and cause failure of the primary mission.

Derivation of Requirement CES.7

This requirement has been imposed for analysis of data collected during the mission of the payload. This data will then be collected and analyzed after recovery of the rover. This will allow the team to evaluate the performance of the payload during all phases of its mission and confirm mission success.

6.1.5.1.7 Obstacle Avoidance System (OAS)

Requirement Number	Requirement	Verification Method
OAS.1	The OAS must detect an obstruction to the forward motion of the rover.	Test Ground testing will be performed to confirm performance. See section 5.15.7

Table 103: OAS requirement.

Derivation of Requirement OAS.1

This requirement has been imposed due to the unpredictability objects that will be scattered in the launch field. In the case that the rover encounters an obstruction that it will not be able to be overcome by simply driving over it, the rover must be able to recognize this and perform an

avoidance maneuver accordingly. This will be part of satisfying NASA requirement 5.1.1of the SOW.

6.1.5.1.8 Solar Array System (SAS)

Requirement Number	Requirement	Verification Method
SAS.1	The SAS must hold the solar panels in place throughout the flight of the vehicle.	Test Full-scale flight tests will demonstrate the ability of the SAS to secure the panels prior to desired deployment.
SAS.2	The SAS must raise the solar panels providing clearance for the solar panels from any surrounding rover components.	Demonstration Ground testing will demonstrate repeatability of performance.
SAS.3	The SAS must lock in the upright configuration.	Demonstration Ground testing will demonstrate repeatability of performance.
SAS.4	The SAS must unfold the solar panels such that the exposed solar panel surface area increases.	Demonstration Ground testing will demonstrate repeatability of performance.

Table 104: SAS requiremets.

Derivation of Requirement SAS.1

This requirement has been imposed to protect the solar array from potential damage during flight. This will also ensure that the panels not prematurely deploy and restrict the rover's forward motion. This will be part of satisfying NASA requirement 5.1.1 of the SOW.

Derivation of Requirement SAS.2

This requirement has been imposed to ensure successful deployment of the solar panels. Successful deployment is measured as a success if the surface area of exposed solar cells increases due to the deployment. This will be part of satisfying NASA requirement 5.1.1of the SOW.

Derivation of Requirement SAS.3

This requirement has been imposed to ensure that the solar panels remain deployed after completion of the payload's primary mission for the panels to continue to harvest solar energy to be used by the SIS. This will be part of satisfying NASA requirement 5.1.1of the SOW.

Derivation of Requirement SAS.4

This requirement has been imposed too comply with NASA's definition of "foldable" being that foldable solar panels are ones that increase the surface area of exposed solar cells after deployment. This will be part of satisfying NASA requirement 5.1.1 of the SOW.

6.1.5.1.9 Surface Imaging System (SIS)

Requirement Number	Requirement	Verification Method
SIS.1	The SIS must use the power generated by the solar panels to take images of the rover and surrounding ground.	Demonstration Ground testing will demonstrate repeatability of performance.

Table 105: SIS requirement.

Derivation of Requirement SIS.1

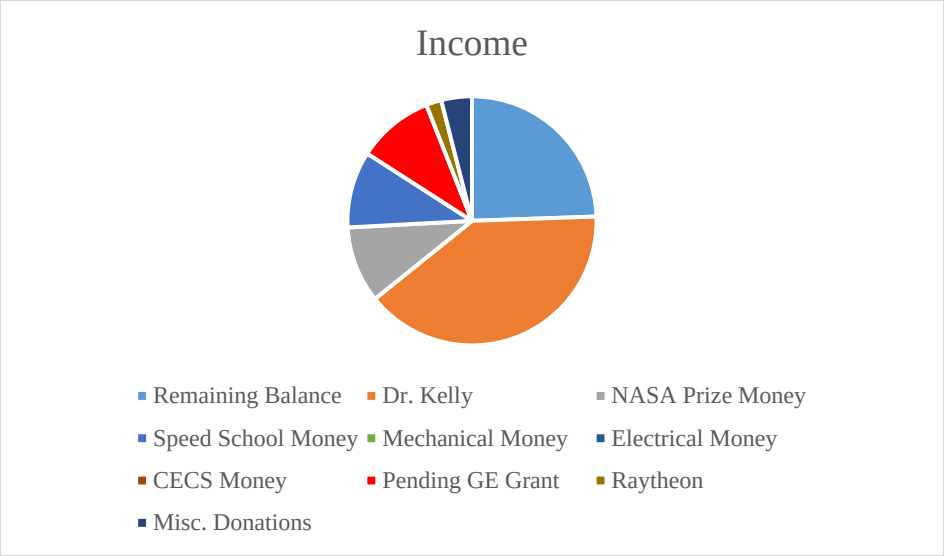
This requirement has been imposed in an attempt to embrace the mindset behind the autonomous rover payload challenge of deploying a rover on another planet and collecting data about that planet. This system will collect data that would be considered extremely valuable in that situation and store it for analysis at a later time.

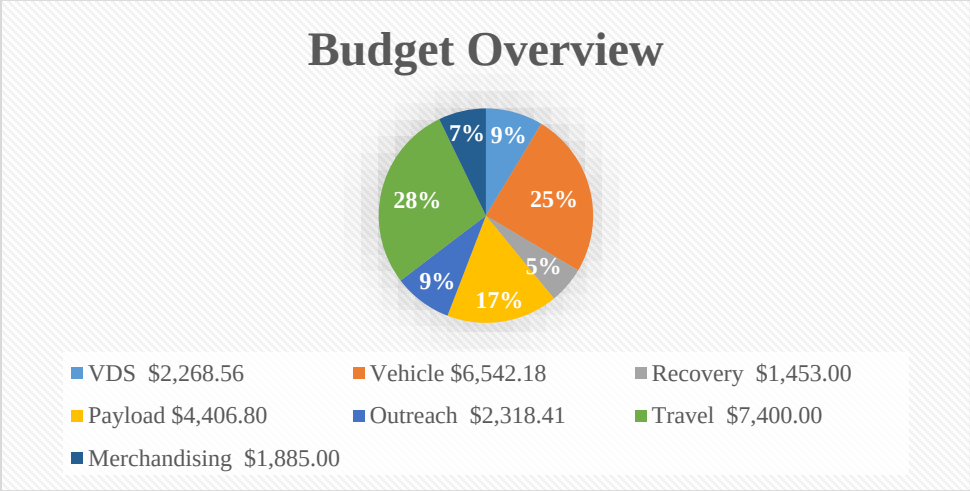
6.2 Budgeting and Timeline

6.2.1 Funding and Budget

The projected expenses and income for the 2017- 2018 season are broken down by subteams. These expenses are broken down by individual components and materials omitting taxes (due to tax exemption) along with projected overages to ensure enough money is allocated to each subteam to sustain themselves for the season even during unexpected circumstances such as last year's CATO which resulted in a drastic increase in last year's budget.

Budget	Total
VDS	$ 2,268.56
Vehicle	$6,542.18
Recovery	$1,453.00
Payload	$4,406.80
Outreach	$ 2,318.41
Travel	$ 7,400.00
Merchandising	$ 1,885.00
Team Improvements	$ 6,200.00
Total Projected Expenses	$ 32,473.95
Total Projected Carryover	$ 17,826.05
Total Projected Income	$ 50,300.00
Total Received Income	$ 45,300.00
Total unreceived Income	$ 5,000.00





Description	Quantity	Per Unit Cost	Total Cost
1/4" Thick 12" x 48" Delrin	1	$ 85.22	$ 85.22
1/8" Dowel Pins 3/4" Length (pkg of 25)	2	$ 10.63	$ 21.26
M3-16 mm Socket Head Cap Screws (pkg of 50)	1	$ 10.20	$10.20
SD/microSD 8Gb	2	$9.95	$19.90
Short Feather Male Headers - 12-pin and 16-pin Male Header Set	20	$0.50	$10.00
GPS	1	$39.95	$39.95
banana to alligator clip cables	1	$3.95	$3.95
Hall effect encoder cable	1	$5.00	$5.00
10-Pin Connector w/ Header, 0.1" Spacing	2	$2.90	$5.80
Encoder Cable	2	$1.11	$ 2.22
Male D-Sub Connector	2	$0.81	$ 1.62
Female D-Sub Connector	2	$0.87	$ 1.74
Power Switch	2	$2.11	$ 4.22

MegaMotor	1	$49.99	$ 49.99
Break Aqay Headers - 40-pin Male (Long Centered, PTH, 0.1")	2	$0.75	$1.50
Teensy Header Kit	2	$1.50	$ 3.00
10 Pin Header	6	$0.50	$3.00
7.4V Lipo Battery	2	$5.20	$10.40
11.1V Lipo Battery	2	$26.90	$ 53.80
12" x 24" 6061 Aluminum plate .125 thick	1	$48.55	$ 48.55
Quick Lock Seals	1	$18.00	$ 18.00
AndyMark Neverest 40 DC Motor	2	$28.00	$56.00
10 Pin Recepticle	2	$0.54	$1.08
10 Pin Insurance	2	$0.19	$ 0.38
Standoff	10	$0.53	$5.30
Copper PCB sheets	2	$6.00	$12.00
teensy header pin	10	$0.66	$ 6.60
Banana Plug Post (Black)	2	$0.35	$ 0.70
Banana Plug Post (Red)	2	$0.35	$0.70
Protoboard	2	$12.49	$ 24.98
Vectornav 100 IMU	2	$530.00	$1,060.00
Break Away Headers - 40-pin Male (Long Centered, PTH, 0.1")	2	$0.75	$1.50
Shipping Cost	1	$250.00	$250.00
Miscellaneous	1	$450.00	$ 450.00
Overall Total			**$ 2,268.56**

Description	Quantity	Price per	Total
6k Carbon fiber tow, 4lbs	4	$57.68	$230.72
Epoxy	3	$141.95	$425.85
6061-T6 Aluminum sheet	1	$160.72	$160.72
1/4"-20 threaded rod	4	$4.46	$17.84
1/4"-20 Hex Nuts	1	$6.74	$6.74
1/4" plywood	1	$12.60	$12.60
4-40 Black Nylon Shear Pins	1	$5.42	$5.42
18-8 Stainless Steel Shoulder Screw, 1" Long Shoulder, 1/4"-20 Thread	9	$6.06	$54.54
18-8 Stainless Steel Shoulder Screw, 1/4" Diameter 1" Long Shoulder, 10-32 Thread	9	$2.65	$23.85
Aerotech L2200G-PS	6	$249.99	$1,499.94
6" X 12" Carbon Fiber coupler	5	$110.00	$550.00
Featherweight screw switches	8	$5.00	$40.00
Paint-job/wrap	1	$400.00	$400.00
Glenmarc G5000 Epoxy	2	$69.99	$139.98

Stratologger CF	4	$49.46	$197.84
Rail buttons	3	$10.00	$30.00
E-Matches	2	$15.79	$31.58
Ejection Charge Canisters	50	$0.50	$25.00
Sub-scale Blue Tube (3" diameter, 48" length)	2	$29.95	$59.90
Sub-scale Blue Tube coupler	1	$9.95	$9.95
Sub-scale motor	2	$50.00	$100.00
Sub-scale motor retainer	1	$31.95	$31.95
Carbon fiber sheet 24" x 48" .125" thick	1	$422.00	$422.00
MDF for nose cone mold	1	$15.99	$15.99
7781 x 50" 8.9oz Fiberglass Prepreg Fabric	1	$47.05	$47.05
Shoulder Length Poly Gloves - 35" Clear	1	$17.00	$17.00
Shipping	1	$700.00	$700.00
Miscellaneous	1	$500.00	$500.00
Potential replacement parts	1	$750.00	$750.00
Heat shrink tape	2	$17.86	$35.72
Total			**$6,542.18**

Material	**Quantity**	**Cost**	**Total**
Large Spool of Thread	4	$ 30.00	$ 120.00
Reefing Rings	6	$ 3.00	$ 18.00
Silver Sharpies (36)	1	$ 29.00	$ 29.00
Sewing Machine Repair	1	$ 160.00	$ 160.00
Stronger Shockcord (ft)	200	$ 0.55	$ 110.00
Spectra Line	4	$ 30.00	$ 120.00
Possible CO2 deployment	4	$ 174.00	$ 696.00
Shipping	1	$ 200.00	$ 200.00
Total			**$ 1,453.00**

Description	**Qt**	**Price**	**Total**
Adafruit Feather M0 Bluefruit LE	2	$29.95	$59.90
Adafruit FeatherWing Motor Shield	2	$19.95	$39.90
3.7V 500mAh LiPo	2	$7.95	$15.90
Planetary Gear Drive Motor	2	$27.99	$55.98
11.1V 400mAh LiPo	2	$14.90	$29.80
BNO055 9DOF IMU	1	$34.95	$34.95
VL53L0X Distance Sensor	2	$13.95	$27.90
Flower Tower Motor	2	$50.00	$100.00
HC-12 433MHz Radio Module	3	$7.55	$22.65
Rover Locking System	1	$300.00	$300.00

PowerFilm Solar Panel	8	$9.29	$74.32
Micro Metal Gear Motor	2	$24.95	$49.90
Small Throw Push-Pull Solenoid	2	$9.95	$19.90
Large Throw Push-Pull Solenoid	2	$14.95	$29.90
Carbon Fiber Sheet	1	$1,000.00	$1,000.00
4mm 90 Degree Gear	8	$5.99	$47.92
ArduCAM Mini Camera Module	2	$39.99	$79.98
Adafruit FeatherWing Adalogger	2	$8.95	$17.90
Large Diameter Bearings	2	$300.00	$600.00
Tracks	2	$150.00	$300.00
Shipping	1	$500.00	$500.00
Miscelaneous Electronics	1	$1,000.00	$1,000.00
TOTAL			**$4,406.80**

Description	**Qt**	**Price**	**Total**
Starhwak Model Rocket Kit (pkg of 25)	3	$ 149.67	$ 449.01
Estes Tandem Model Rocket Launch set	2	$ 26.18	$ 52.36
1/2A3-4T Engine Bulk Pack (pkg of 24)	2	$ 57.79	$ 115.58
Scotch Tape (pkg of 3)	40	$ 4.74	$ 189.60
BristleBot Kit	50	$ 19.99	$ 999.50
Pipe wrench	1	$ 14.98	$ 14.98
PVC Cement	1	$ 9.10	$ 9.10
plasticweld	1	$ 7.86	$ 7.86
Standard Tire Valve	1	$ 4.82	$ 4.82
10' 6" PVC	2	$ 6.74	$ 13.48
2" slip fit t joint	1	$ 2.84	$ 2.84
2" end cap	2	$ 0.84	$ 1.68
Slim Wall Plug in 120 volt tire inflator	1	$ 19.99	$ 19.99
2" to 1" reducer bushing	1	$ 1.52	$ 1.52
1" to .5" reducer bushing	1	$ 1.09	$ 1.09
.5 in threaded male adapter	2	$ 0.44	$ 0.88
.5" x2' pvc pipe	1	$ 1.47	$ 1.47
sprinkler system valve	1	$ 15.27	$ 15.27
push button switch	1	$ 11.99	$ 11.99
battery lead 9v	1	$ 5.39	$ 5.39
Miscellaneous	1	$ 400.00	$ 400.00
Total			**$ 2,318.41**

Description	**Qt**	**Price**	**Total**
Competition Hotel	N/A	$ 6,000.00	$ 6,000.00

Competition Travel Gas	N/A	$ 750.00	$ 750.00
Trailer Rental	N/A	$ 150.00	$ 150.00
Test Launch Gas	N/A	$ 500.00	$ 500.00
Total			**$ 7,400.00**

Description	**Qt**	**Price**	**Total**
Safety Glasses Replacement	50	$ 4.00	$ 200.00
Motor Magazine	1	$ 500.00	$ 500.00
Launch Rail	N/A	$ 400.00	$ 400.00
Office Supplies	N/A	$ 250.00	$ 250.00
General Team Equiptement	N/A	$ 350.00	$ 350.00
Miscalalious Costs	N/A	$ 1,000.00	$ 1,000.00
Science Center Exhibit	N/A	$ 3,500.00	$ 3,500.00
Total			**$ 6,200.00**

Description	**Qt**	**Price**	**Total**
Shirts	30	$18.00	$540.00
Polos	40	$28.00	$1,120.00
Stickers	1,000	$0.15	$150.00
Business Cards	2,500	$0.03	$75.00
Total			**$ 1,885.00**

6.2.2 Project Plan

The project plan for this coming year is shown in Figure 146 with a team timeline displayed in Figure 147, with the individual timelines for each subteam shown below in: Figure 148, Figure 149, Figure 150, and Figure 151.

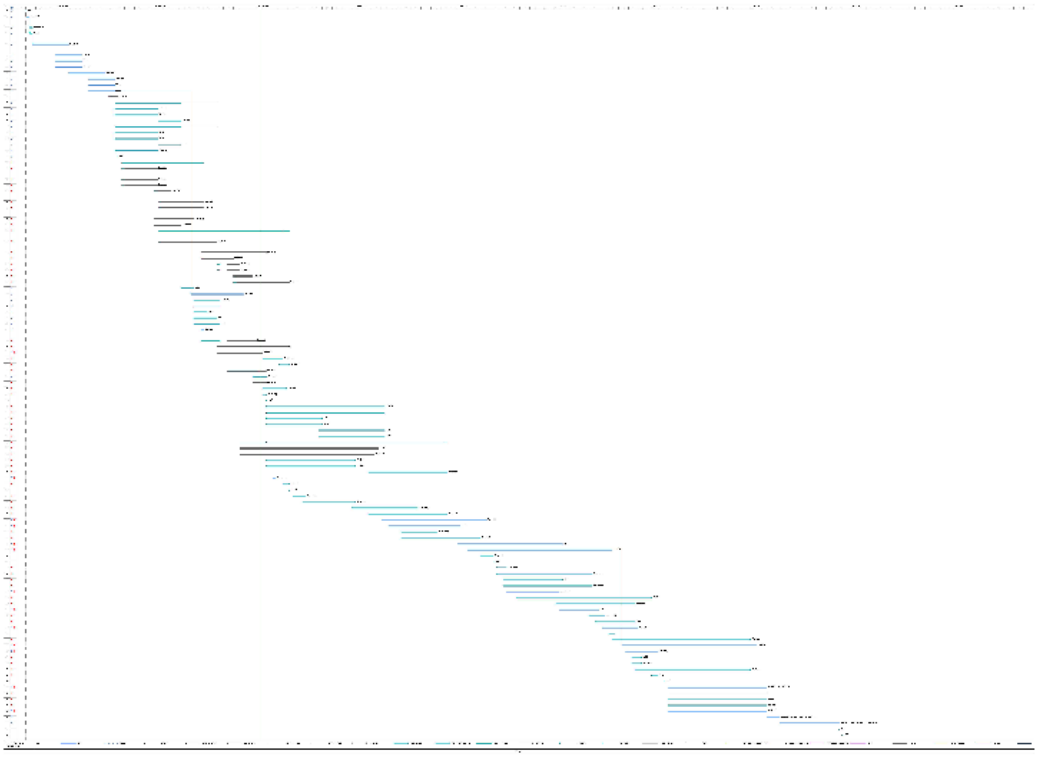


Figure 146: Team Gantt Chart

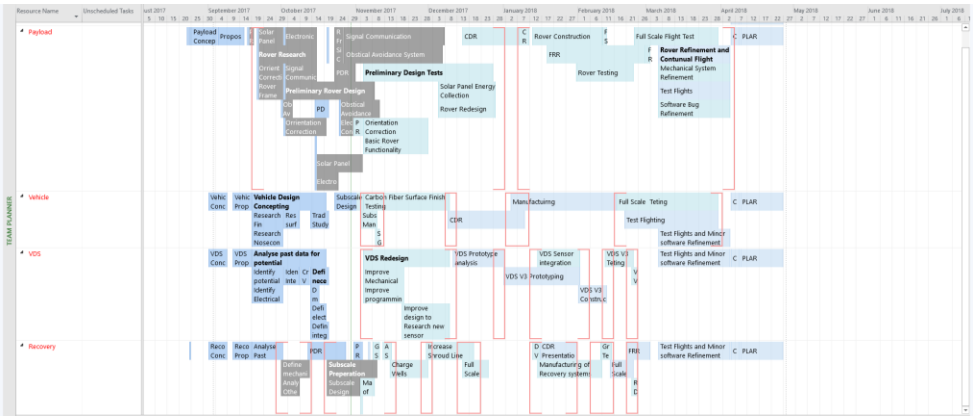


Figure 147: Subteam Project Timeline

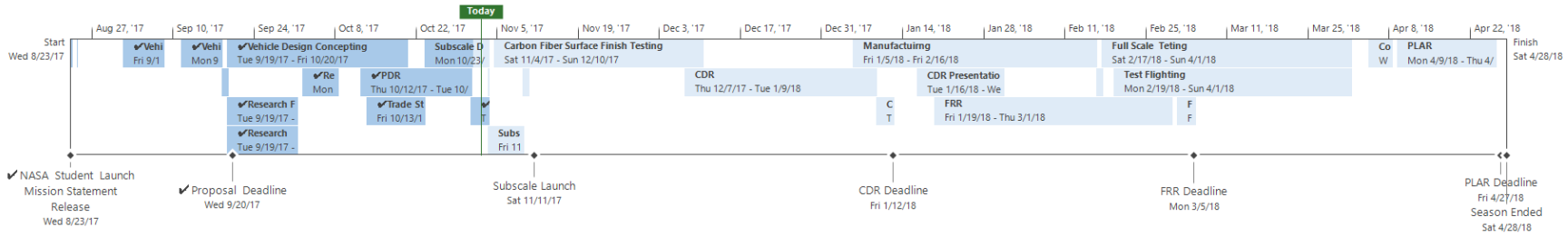


Figure 148: Vehicle Timeline

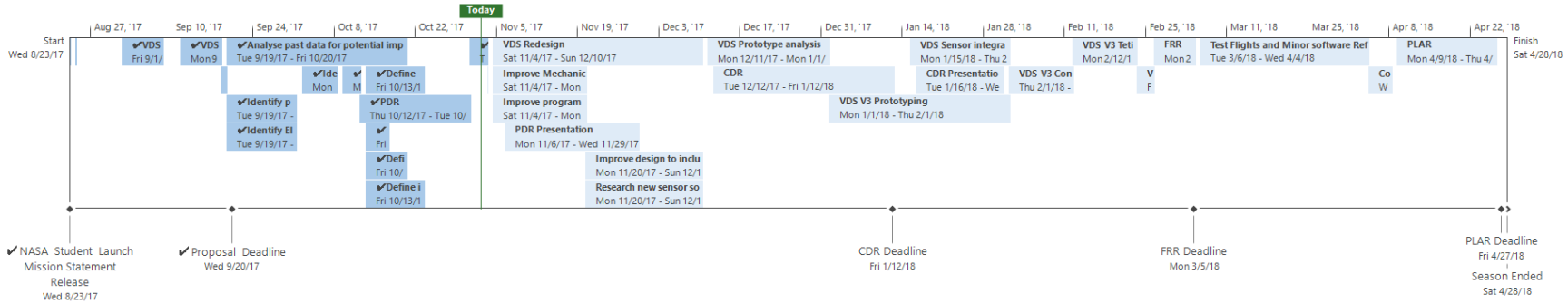


Figure 149: VDS Timeline

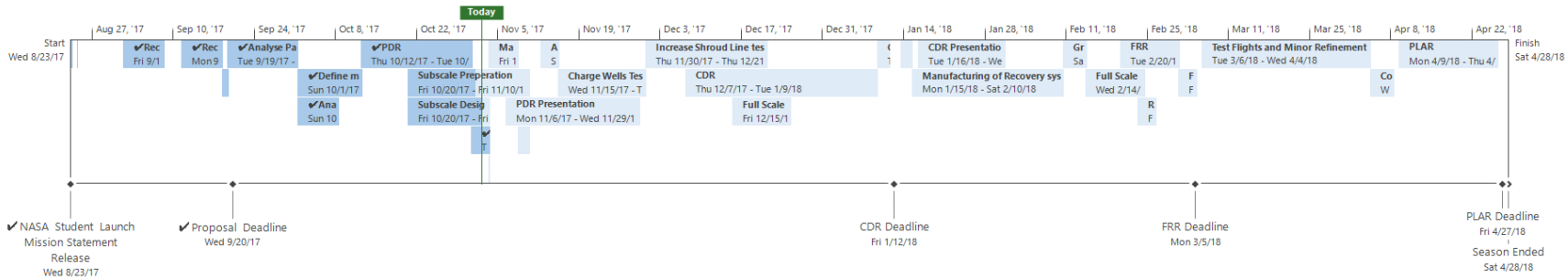


Figure 150: Recovey Timeline

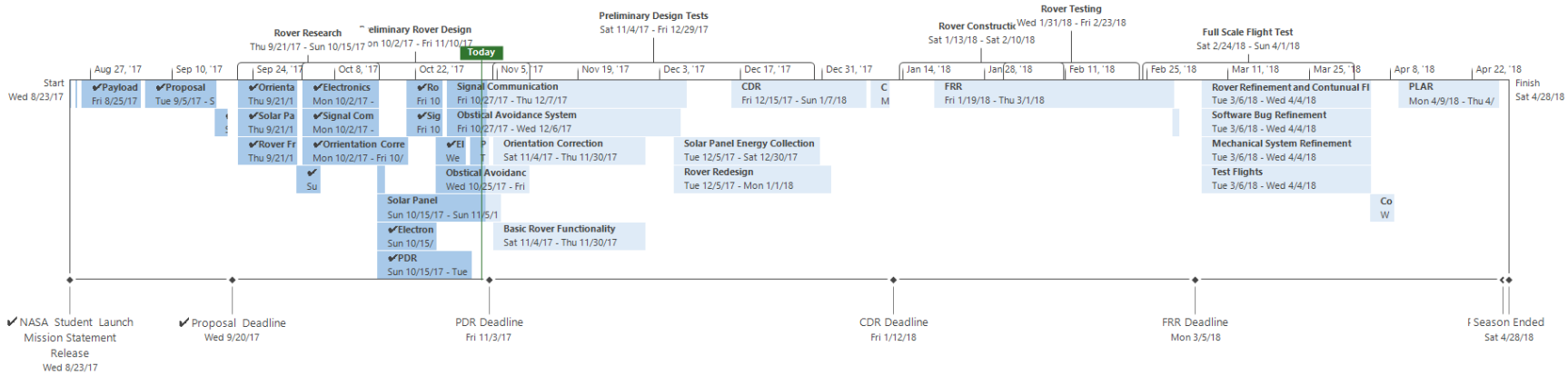


Figure 151: Payload Timeline

6.2.2.1 Educational Engagement

Throughout the last five years the team has had the opportunity the University of Louisville's River City Rocketry has had the opportunity to engage with over 8,000 students and adults in our local community. The team's outreach allows us to give back to the state of Kentucky by teaching the youth about engineering, math, technology, logical thinking and rocketry. River City Rocketry continues to develop new relationships through outlets like our website, all while further developing the relationships the team has built up over the years by striving to constantly improve the quality of education brought to every individual we engage with in our community.

6.2.2.1.1 Classroom Curriculum

The University of Louisville's River City Rocketry team has developed a variety of educational programs that will be incorporated into this year's outreach program. This list includes many of the different activities in which the team has participated in the past and will continue to do this year.

Raytheon MathMovesU

Over the past few years, River City Rocketry has partnered with Raytheon Missile Systems and the University of Louisville in order to put on MathMovesU. This event will take place at the Engineering Education Garage, and the intramural fileds. This event is outlined in the following steps below

Step 1: Introduction from RCR and Raytheon

Following check in, a brief presentation about River City Rocketry, what the team does along with going over the agenda for the afternoon. Following the team's presentation, Raytheon Missile systems will give a brief presentation on what they do and how STEM classes are immensely beneficial to building rockets in the future.

Step 2 Construction and Lesson of Estes Kits:

During this time the Team will go over the basics of how rockets operate and the science behind it, along with instructing the students on how to construct their rockets and the functionality of each component on the rocket. During this construction phase volunteers will be stationed throughout the room to assist with construction while a single member directs the students through the process with the aid of a PowerPoint.

Step 3 Safety Briefing and Launching of Estes kits.

Following the construction of the rockets the teams safety officer will perform a safety briefing for students and adults to ensure that all personal, volunteers, students and parents are safe throughout the event. While rockets are being launched everyone that is within the bounds of the intermural field will be required to wear safety glasses and be restricted to remain behind a pre-defined white line located over 100 ft away from the launch stations.

As each rocket is set up on the launch stand, one student per stand at a time will be allowed past the white line, where that student will work with the Range Safety Officer (RSO) to set up the rocket on the stand. Following the setup of the rocket and returning to behind the white line the RSO will give the student permission to insert the safety key. A countdown from 5 to 0 from the student will follow the insertion of the key. Following the student countdown, the students will ignite their rockets into the sky.

The steps and safety precautions used during the MathMovesU event by River City Rocketry is constant with all major rocketry building outreach events.

STEM night

This will be the second year River City Rocketry will be participating in STEM night at Farmer Elementary. The team will have a table on display of your past rockets. During this event last year, the team used bristle bots. During this event last year students got to choose from racing a prebuilt bot to learn how they operate and test their reaction time or to build their own bristle bot from scratch and learn about how small circuitry works and how it integrates into mechanical systems. This year the team plans to design a similar booth that will be both engaging and educational for the kids.

6.2.2.1.2 Outreach Opportunities

6.2.2.1.2.1 *Engineering Exposition (E-Expo)*

Since 2006, the J.B. Speed School of Engineering Student Council has hosted the largest student-run event on the University of Louisville's campus called the Engineering Exposition. The event is geared towards celebrating strides in engineering as well as getting the local youth interested in the field. During the event, the professional engineering societies on U of L's campus set up educational activities and scientific demonstrations for the elementary and middle school students to participate in.

The University of Louisville River City Rocketry Team will host its sixth annual bottle rocket competition for middle school students. Teams from local middle schools can participate in teams of up to three students to design and build their own water bottle rockets out of two-liter bottles and other allowable materials. Workshops will be held with schools interested to teach the students about the components of a rocket and aerodynamics in preparation for the competition. The students will get to show off their rockets at the E-Expo event and will conclude the day with the competition. Teams will compete for awards in highest altitude, best constructed rocket, and landing closest to the launch pad. This event has been a huge success in the past and many schools have voiced interest in continuing their involvement so we are looking for our best turnout yet this year. A previous paper rocket E-Expo event is shown in **Figure 152**.

Figure 152: Team member, Denny, building rockets with students at E-Expo 2016.

In addition to the water rocket competition, the team will host a paper rocket station for people of all ages. This has been the most popular station at the exposition in the past and we are looking to continue to build up that reputation.

6.2.2.1.2.2 *Boy Scouts and Cub Scouts:*

In the past, the University of Louisville River City Rocketry Team has worked with local Boy Scout and Cub Scout troops to assist the earning of the Space Exploration merit badge. The team has assisted in developing a program that meets the requirements to earn the merit badge. The scouts get to learn about the history of space, current space endeavors, and build and launch an Estes rocket. The team has plans to continue to work with these groups throughout the year with one event already scheduled for September 30th, 2017.

While cub scouts are not eligible to earn their merit badge, we still enjoy getting to teach them about rocketry. We have had the pleasure of working with scout troops in educating the kids about the fundamentals of rocketry, while also giving them the opportunity to build and launch their own paper rockets.

6.2.2.1.2.3 *Big Brothers Big Sisters Partnership*

Big Brothers Big Sisters is active in the Louisville community and is constantly striving to bring opportunities to underprivileged kids. The team recently put on a program at the Back to School Event for kids that had not yet been paired with a mentor through the program, shown in Figure 153. This is the second year in a row that the team has participated in this event. Both years, this event has been a huge success in brining STEM to under-privileged kids.

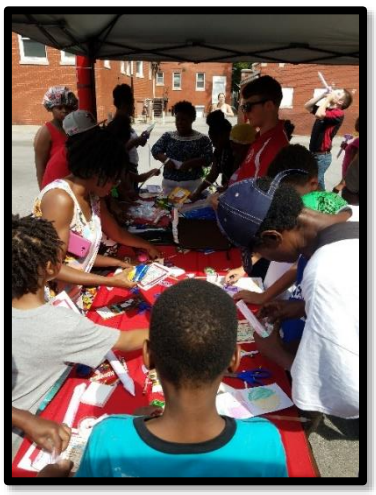


Figure 153: Big Brothers Big Sisters Back to School Event (2017).

6.2.2.1.2.4 *Louisville Mini-Maker Faire*

Annually, Louisville hosts a Mini-Maker Faire. The team always participates by taking the previous year's project out to show off to anyone attending the event. A variety of people attend this event ranging from small children to adults with experience in the field. This gives the team an opportunity to talk to the community about our project and what it does. This is an informal setting which is perfect for interacting with visitors and answering their questions about the project, what the team does, and about rocketry in general.

6.2.2.1.2.5 *Kentucky Science Center*

During the 2015-2016 season, the team first came in contact with Andrew Spence, manager of public programs and events, that assisted in several events in the Louisville area. For this season the team will participate in the Youth Science Summit, Advanced Manufacturing, and National Engineers' Week at Kentucky Science Center. The team will be able to reach out to hundreds of young rocketeers and teach them about rocketry, engineering, and skills needed to succeed as an engineer.

During this current season the team will build an exhibit at the science center, this exhibit will consist of a paper rocket building station, the rocket from the 2016-2017 season and informational plats about NSL, River City Rocketry and the science behind rocketry.

6.2.3 Conclusion

River City Rocketry is returning as motivated as ever to continue participating in the NASA Student Launch competition and are striving to continue to meet our standard by setting the following goals:

· To improve upon and set a standard for safety for years to come

· To engage 1500+ students in STEM centered engaging outreach events, encouraging enthusiasm for rocketry and the larger STEM fields.

· To continue to improve on the Variable Drag System (VDS) design to set a new standard for apogee accuracy in NSL flights.

· To design a reliable payload system to deploy solar arrays and easily integrates into the vehicle.

· To grow the team; expanding the team's cumulative knowledge of rocketry and ensuring a sustained continuous improvement in the team's ability to achieve its goals.

7 Appendix I - Safety Risk Assessment Table

7.1 Failure Modes and Analysis

7.1.1 Stability and Propulsion Failure Analysis

Stability and Propulsion Hazard Failure Analysis						
Hazard	**Cause/ Mechanism**	**Outcome**	**Severity Value**	**Probability**	**Risk Level**	**Mitigation**
Launch vehicle fails to lift off the launch rail	1. Damage to the launch rail 2. Damage to the launch rail buttons	1. Damage to launch vehicle 2. Improper motor burn that could cause a mission failure	4	5	Low	The launch rail and launch buttons will be inspected for smooth fit prior to launch
Defective igniter	Arrived from manufacturer with crushed head or wires that fail to complete the circuit	1. Motor will not ignite at all and 2. Motor will ignite improperly leading to a poor burn	4	4	Low	All motor igniters will be inspected prior to packing and prior to launch.

Motor CATO	1. Igniter inserted improperly. 2. Faulty motor or motor casing from manufacturer	1.Motor does not ignite, delaying the flight of the launch vehicle 2. Motor failure and launch vehicle will fail mission	1	4	Moderate	Motor casings will be thoroughly inspected upon arrival to the launch site, prior to motor packing, and after booster recovery. The team will also check the MESS for warnings about the competition motors.
Airframe failure	Airframe experiences higher stress than the material can support	Airframe ruptures during flight, resulting in mission failure	1	5	Moderate	The launch vehicle airframe will be designed to have an adequate factor of safety
Rail button shears off	Velocity at the point of the rail departure is lower than what is acceptable	The launch vehicle exits the rail with a large attitude angle, which will decrease the apogee altitude of the launch vehicle	2	4	Moderate	1.The launch vehicle will be aligned carefully on the launch rail 2. Rail buttons will be inspected prior to each launch and damaged rail buttons will not be used

Exit rail velocity is lower than 75 ft/s	Impulse of motor is not high enough for the full-scale launch vehicle. There is a high friction coefficient between rail buttons and launch rail	The launch vehicle will be unstable and be susceptible to large angles of attitude	1	5	Moderate	The motor will be selected to ensure that the proper exit rail velocity will be achieved
Inaccurate calculations or simulations	Vehicle experiences higher aerodynamic forces than expected	Possible fin flutter, unstable flight and larger angle of attitude	2	4	Moderate	The calculations will be reviewed during weekly meetings and will be cross checked with past calculations
Fins shearing off due to fin flutter	Drastic decrease in vehicle stability	Launch vehicle experiences unpredictable flight trajectory	1	4	Moderate	The fins will be analyzed to minimize or eliminate the risk of fin flutter

Premature separation of launch vehicle during flight	Altimeter malfunctioned and misread pressure data	The launch vehicle separates on ascent, causing catastrophic failure and possible injury to personnel	1	5	Moderate	In the case of a catastrophic failure, an airhorn will be used to warn the crowd.

Table 106: Stability and propulsion failure analysis.

7.1.2 Launch Vehicle Failure Analysis

Launch Vehicle Failure Mode Analysis						
Hazard	**Cause/ Mechanism**	**Outcome**	**Severity Value**	**Probability Value**	**Risk Level**	**Mitigation**
Centering ring failure	1. Epoxy is not properly applied to centering rings 2. The centering rings yield	Motor will continue to move through the vehicle, destroying the internals of the launch vehicle	1	4	Moderate	The centering ring will have FEA data showing that the ring will not fail. Two rings will be used to secure the motor

Motor retainer failure	1.Motor retainer did not have proper preload. 2. Motor retainer did not have proper thread engagement. 3. Motor retainer was not sized correctly for expected loads	Motor casing falls out of launch vehicle, resulting in mission failure and potential injury of personnel	1	5	Moderate	Pre launch procedures will be followed to ensure that components are correctly installed.
Descent velocity is higher than expected	Improper design/development of recovery system.	Vehicle components break upon landing, resulting in a mission failure	1	5	Moderate	The calculations will be repeated and compared against calculations from the past to check for similarities
Fin retainer fails	Fin retainer was improperly sized and was not able to withstand the load experienced during flight.	Fins fall out of launch vehicle during flight, resulting in mission failure	1	5	Moderate	The fit of the fins can be checked and ensured during pre launch procedures
Pinching	1.Coupler and airframe meet 2. Airframe into coupler 3. Loading the vehicle on the rail 4. Assembling the launch rail	1.Stress concentration, increase in shear and change in aerodynamics 2. Minor injury to a team member	2	5	Low	All team members will be reminded of how to avoid pinches from the launch vehicle during assembly

Dropped rocket	Improper lifting, storage, or carrying of the launch vehicle	Could result in microfractures in motor, resulting in complete mission failure	1	4	Moderate	At least two people must lift the rocket at during assembly. The airframe must remain on a rocket stand during integration
Carbon fiber splinters	Sharp airframe edges	Team members get splinters	4	2	Moderate	Ensure proper sanding during manufacturing to decrease sharp edges
BP charges ignite prematurely	1. Activating Stratos before rocket is on pad 2. Faulty programming or Strato fails to properly read altitude	1. BP goes off while team member is holding rocket, resulting in team injury 2. Rocket prematurely separates resulting in mission failure	1	5	Moderate	1. All Stratos will not be turned on until after the rocket is loaded on the pad. 2. Stratos programming will be done with at least 2 trained people present. Stratos will be tested to verify proper functionality prior to launch
Pinched recovery lines	Recovery system not properly packed	Recovery system fails resulting in the launch vehicle falling at terminal velocity	1	3	High	Shear pin holes must align to launch. Pinched recovery gear will prevent proper alignment

Table 107: Launch vehicle failure mode analysis.

7.1.3 VDS Failure Analysis

VDS Failure Mode Analysis						
Hazard	**Cause/Mechanism**	**Outcome**	**Severity Value**	**Probability Value**	**Risk Level**	**Mitigation**

7.4 V battery death	Improper charging	1. If the battery dies prior to launch, the drag blades would not potentially actuate during flight 2. If the battery dies during ascent the rocket will not reach the intended height	3	4	Low	The battery will be charged throughout integration up until the rocket leaves for the launch rail
Time variable overflow	Extended run time	VDS drag blazes could potentially actuate on rail, leading to increased rail friction, rail button shear and lower than expected exit velocity	1	4	Moderate	If time on rail is excessive, VDS can be restarted removing the issue of the variable overflow
VDS VN-100 or other sensors are affected by transmitting antenna	Sensor exposed to transmitting antenna	VDS drag blazes could potentially actuate on rail, leading to increased rail friction, rail button shear and lower than expected exit velocity	1	4	Moderate	Extensive testing will be conducted to determine the level of risk
11.1V battery death	Improper charging	1. If the battery dies prior to launch, the drag blades would not potentially actuate during flight 2. If the battery dies during ascent the rocket will not reach the intended height	3	4	Low	The battery will be charged throughout integration up until the rocket leaves for the launch rail

Broken gearbox	VDS blades remained actuated during recovery	Permanent damage to VDS assembly Hazard to crowd if recovery is unsuccessful	2	4	Moderate	VDS is programmed to retract blades after apogee The team is currently investigating recovery force reduction
Sensor error due to DC motor feedback	Improperly isolated circuits	VDS actuates too early, launch vehicle undershoots altitude resulting in mission failure	2	4	Moderate	New sensors have a built-in sensor filter to eliminate noise Signal line noise from motor encoder reduced
Pressure phenomenon from open-ended propulsion bay causes altitude error	Vacuum formed under propulsion bay	VDS actuates too early, launch vehicle undershoots altitude resulting in mission failure	2	2	High	Electronics bay will be airtight from the actuation bay to prevent possible interference
Slow speed SD card causes delay in data reading	Installed the wrong SD card	VDS fails to respond to accurate real time data resulting in imprecise system function and higher altitude than anticipated.	2	4	Moderate	This will be mitigated through pre-flight check lists
Sharp blade edges	Burs may result from blade manufacturing	Minor injuries to personnel	4	5	Low	Edges will be deburred prior to VDS assembly

Table 108: VDS failure analysis.

7.1.4 Recovery Failure Analysis

Recovery Vehicle Failure Mode Analysis						
Hazard	**Cause/Mechanism**	**Outcome**	**Severity Value**	**Probability Value**	**Risk Level**	**Mitigation**

Launch vehicle sections fail to separate	1.Insufficient pressure to break shear pins 2.Coupling fitting is too tight	Launch vehicle follows unsafe projectile flight profile	1	5	Moderate	1. Launch vehicle body designed to separate under pressurization from black powder charge 2. Coupling between sections of the launch vehicle body reduced to have loose fit
Altimeter or E-match Failure	One or more parachutes fail to deploy	Launch vehicle follows unsafe ballistic flight profile	1	5	Moderate	Redundant altimeters and E-matches included to decrease risk
Parachute fails to open	1. One or more parachutes entangled in shroud lines 2. Main parachute constrained by storage bag	Launch vehicle follows unsafe ballistic flight profile	1	4	Moderate	1. Deployment bags designed to organize packing and reduce chance of failure 2. Deployment bags designed with extra room to organize packing and reduce chance of deployment failure
Launch vehicle descends too rapidly under parachute	1. Parachute is under minimum appropriate size 2. One or more parachutes fail to properly deploy	Launch vehicle is damaged by greater than anticipated impact force	2	5	Low	1. Parachutes designed to be appropriately sized during both nominal and off nominal conditions 2. Recovery Systems carefully inspected prior to packing and launch
Launch vehicle descends too slowly under parachute	Parachute is over maximum appropriate size	Launch vehicle will drift farther than anticipated	3	3	Moderate	Parachutes designed to be appropriately sized during both nominal and off

						nominal conditions
Parachute material or seam fails	Potential partial or total failure of recovery system	Damage to launch vehicle	2	5	Low	Recovery Systems carefully inspected prior to packing and launch
Parachute or suspension lines burnt during deployment	Potential partial or total failure of recovery system	Damage to launch vehicle	2	5	Low	Careful packing and use of fireproof wadding to reduce likelihood of occurrence
Separation of recovery system from launch vehicle	1. Bulkead is dislodged 2. Parachute disconnects from U-bolt 3. One or more parachutes fail to properly deploy, causing greater than anticipated opening forces	Launch vehicle follows unsafe ballistic flight profile	1	5	Moderate	Cables and bulkhead designed to withstand expected loads with a reasonable factor of safety
Main parachute fails to open	ARRD fails to deploy	Launch vehicle is damaged by greater than anticipated impact force	2	5	Low	Redundant altimeters and E-matches included to decrease risk
ARRD deploys prematurely	ARRD is not assembled properly	Launch vehicle will drift farther than anticipated	3	4	Low	ARRD will be tested to decrease risk
Charge well does not vent properly	Vent holes in airframe are not properly sized	Ejection charge will create too much pressure in the airframe	4	5	Low	Vent holes in airframe will be sized correctly
Shock cord snaps due to opening force	1. Opening force is too large 2. Opening force reduction ring does not work properly	Section of the launch vehicle follows unsafe ballistic flight profile	1	3	High	Opening force reduction ring and parachute canopy will be sized properly

Zippering of the airframe	Main parachute is deployed at an off-nominal angle	Damage to the launch vehicle	4	5	Low	Drogue phase of descent will keep the recovery bay in the correct orientation

Table 109: Recovery vehicle failure mode analysis.

7.1.5 Payload Failure Analysis

Recovery Vehicle Failure Mode Analysis						
Hazard	**Cause**	**Outcome**	**Severity Level**	**Probability Level**	**Risk Level**	**Mitigation**
Pinch points	1. Rotation of rocs during handling and integration testing 2. Timing belt and drive system 3. SAS locking mechanism and panel actuation 4. Locking mechanism	Minor injuries to team members damage to wiring	4	3	Low	Construction and assembly procedures will remind the team to be aware of pinching
Collision	1. Rocs ground roll test 2. Drive system testing	1. Damage to the rover 2. Minor harm to personnel	4	4	Low	Members must be attentive during rover testing
Static discharge	Sensor and control electronics testing	Damage to electronics	3	5	Low	Members will use a grounding mat and the wrist strap when testing electronics
Premature deployment	Premature extraneous signal not transmitted by the team deploys the rover prior to the bay landing safely	The rover may fall out of the open end of the payload bay	1	4	Moderate	The payload will have a locking mechanism, two gyroscopes, and a unique deployment

						signal. The locking mechanism will remain locked while unpowered.
Failed mechanical locking system	1. Cannot withstand liftoff loads 2. Cannot withstand opening force loads 3. Cannot withstand landing loads 4. Solenoid retraction prevented due to loading from rover weight	The rover may fall out of the open end of the payload bay	2	4	Moderate	The mechanical locking system will be tested extensively
Unreceived deployment signal	1. Rover lands out of range 2. Receiver antenna is damaged 3. Obstructed receiver transmitter line-of-sight	The rover will not deploy. Failed payload mission	2	3	Moderate	Simulations and field testing will be conducted on multiple antenna configurations. Measures will be taken to ensure that the range can exceed 2500 ft.
Obstructed rover path	1. Obstacle blocking end of bay 2. Contact with interior of payload bay 3. DTS receiver module does not detach	1. The rover will not deploy. Failed payload mission 2. The rover will deploy but it may not reach 5 ft. Away. Failed payload mission	3	3	Moderate	The lidar system will allow the rover to determine alternate, non-obstructed paths

Component falls out of payload bay during recovery	Vibrations loosen components	1. Component is a risk to the crowd or the team 2. The payload will not perform as intended. Possible failed payload mission	2	4	Moderate	The payload will have a locking mechanism, two gyroscopes, and a unique deployment signal. The locking mechanism will remain locked while unpowered.
ROCS fails	1. Bearing misalignment 2. Opening force	1. Rover falls out of the payload bay during recovery 2. Improper orientation for deployment	2	4	Moderate	Calculations, simulations, and extensive testing will be conducted prior to flight

Table 110: Payload Risk Assessment.

7.1.6 Project Management Risks

Integration	Likelihood	Impact	Project Impacts	Mitigations
Increased complexity of design	Medium	High	Redesign or delayed lights	The VDS and payload were designed with integration in mind including the ability to charge batteries while the systems are already integrated and having the systems be easily removable
Shipping delays or unavailability	Low	Medium	Project delays	The team has the ability to machine components a in house or purchase them from multiple vendors
Delayed component verifications	Medium	High	Severe delays or project failures due to missed deadlines	Verification testing has already been outlined and possible problems and critical design functions have been identified

Overscheduled team members	Medium	Medium	Severe delays or project failures due to missed deadlines	Gantt charts and project breakdown show clear path to completion
Lack of team member availability due to classes and University required co-ops	High	Medium	Lack of expertise and labor to complete designs on time	The team recruited a variety of engineering disciplines and school years to account for the gaps that co-ops can cause in the team
Document and model sharing issues	High	High	Lost, damaged, or overridden files require work to be unnecessarily repeated	The team uses Own Cloud to store models and SharePoint to store documents. Both platforms have syncing errors and conflicts regularly. The team is researching other ways to share documents
Exceeding project budget	Low	High	Severe delays or project failures due to missed deadlines	The team is actively seeking sponsors. The members ordering must pay attention to detail in shipping also eliminates the risk of mistakes
Manufacturing equipment is not available	Low	Low	Delayed manufacturing and verification schedules	The team has a variety of workspaces with ample equipment available for use at almost any time

Table 111: Project risk identifications and mitigations.

7.1.7 Workshop Hazard Assessment

Workshop Risk Assessment						
Hazard	**Cause**	**Outcome**	**Severity Value**	**Probability Value**	**Risk Level**	**Mitigation**

Using power tools and hand tools such as blades, saws, drills, etc.	Improper training on tools and other lab equipment.	1. Mild to severe cuts or burns to personnel. 2. Damage to rocket or components of the rocket. 3. Damage to the equipment	2	4	Low	1. Individuals must be trained on the tool being used. Those not trained should not attempt to learn on their own and should find a trained individual to instruct them. 2. Safety glasses must be worn whenever using power tools. 3. Sweep or vacuum up shavings to avoid cuts from debris.
Sanding or grinding materials.	1. Improper use of PPE. 2. Improper training on the use of a Dremel tool.	1. Mild to severe rash. 2. Irritated eyes, nose or throat with the potential to aggravate asthma. 3. Mild to severe cuts or burns from a Dremel tool and sanding wheel.	3	3	Low	1. Long sleeves should be worn whenever sanding or grinding materials. 2. Proper PPE should be utilized such as safety glasses and dust masks with the appropriate filtration required. 3. Individuals must be trained on the tool being used. Those not trained should not attempt to learn on their own and should find a trained individual to instruct them.
Machining equipment including CNC, lathe, and saws.	Improper training on tools and other lab equipment.	1. Damage to the equipment. 2. Damage to materials being machined. 3. Potentially severe cuts or burns to personnel.	1	5	Low	The machining equipment available to the team requires safety training and certification for each team member. Each piece of equipment has a Job Safety and Sequence Instruction card that details task steps, safety instructions, and ergonomic reminders.

Working with chemical components resulting in mild to severe chemical burns on skin or eyes, lung damage due to inhalation of toxic fumes, or chemical spills.	1.Chemical splash 2.Chemical fumes	1. Mild to severe burns on skin or eyes. 2. Lung damage or asthma aggravation due to inhalation of fumes,	2	4	Low	MSDS documents will be readily available at all times and will be thoroughly reviewed prior to working with any chemical. Each member must acknowledge the hazards that accompany working with these chemicals. All chemical containers will be marked to identify appropriate precautions that need to be taken. Nitrile gloves are available and shall be used when handling hazardous materials.
Damage to equipment while soldering.	1. Soldering iron is too hot 2. Prolonged contact with heated iron	1.The equipment could become unusable. 2. Parts of the circuit get damaged and become inoperable.	3	3	Low	1. The temperature on the soldering iron will be controlled and set to a level that will not damage components. 2. For temperature sensitive components sockets will be used to solder ICs to.
Dangerous fumes while soldering.	1. Use of loaded solder can produce toxic fumes 2. Leaving soldering iron too long on plastic could cause plastic to melt producing toxic fumes	1. Inhalation of toxic fumes could make team members sick. 2. Lung irritation may occur.	3	3	Low	Team members will be trained how to solder and will follow all safety protocols related to soldering.

Potential burns to team members while soldering.	Team members do not pay attention while soldering.	Team member could suffer minor to severe burns	4	3	Low	Team members will be trained how to solder and will follow all safety protocols related to soldering.
Overcurrent from power source while testing.	Failure to correctly regulate power to circuits during testing.	Team members could suffer electrical shocks which could cause burns to heart arrhythmia	2	4	Low	The circuits will be analyzed before they are powered to ensure they don't pull too much power. Power supplies will also be set to the correct levels.
Use of cutting fluid.	Use cutting fluid when machining metals.	Contains carcinogens.	1	5	Low	Safety glasses shall be worn whenever machining metals.
Use of white lithium grease.	Used while installing motor and on ball screws.	1. Irritation to skin and eyes. 2. Respiratory irritation.	3	4	Low	1. Nitrile gloves and safety glasses are to be worn when applying grease. 2. When applying grease, it should be done in a well-ventilated area to avoid inhaling fumes.
High voltage shock.	Improper use of welding equipment.	Death or severe injury.	1	5	Low	All team members are required to be trained and certified on the equipment prior to use. Any time team members are welding, there must be at least two certified people present.
Bit breaks on mill.	Spindle speed set too high	Injury to personnel and damage to equipment and/or rocket component.	2	5	Low	All team members are required to be trained and certified on the mill prior to use. If personnel is uncertain about the proper settings, they are to consult an experienced member prior to operation.
Metal shards.	Using equipment to machine metal parts.	Metal splinters in skin or eyes.	2	5	Low	Team members must wear long sleeves, long pants, and safety glasses when machining metal parts.

Table 112: Workshop Risk Assessment.

7.1.8 Environmental Hazards to the Launch Vehicle Risk Assessment

Environmental Hazards to Launch Vehicle Risk Assessment						
Hazard	**Cause**	**Outcome**	**Severity Value**	**Probability Value**	**Risk Level**	**Mitigation**
Low cloud cover.	N/A	Unable to test entire system.	1	4	Low	When planning test launches, the forecast should be monitored in order to launch on a day where weather does not prohibit launching or testing the entire system.
Rain.	N/A	1. Unable to launch. 2. Damage to electrical components and systems in the rocket.	1	4	Moderate	1. When planning test launches, the forecast should be monitored to launch on a day where weather does not prohibit launching or testing the entire system. 2. Have a plan to place electrical components in water tight bags. Have a location prepared to store the entire rocket to prevent water damage.
High winds.	N/A	1. Must launch at high angle, reducing altitude achieved. 2. Increased drifting. 3. Unable to launch.	1	4	Moderate	1,2,3. When planning test launches, the forecast should be monitored in order to launch on a day where weather does not prohibit launching or testing the entire system. If high winds are present but allowable for launch, the time of launch should be planned for the time of day with the lowest winds.

Trees.	N/A	1. Damage to rocket or parachutes. 2. Rocket components must be retrieved by professionals.	1	4	Moderate	Launching with high winds should be avoided to to avoid drifting long distances. Drift calculations have been computed, so we can estimate how far each component of the rocket will drift with a specific wind velocity. The rocket should not be launched if trees are within the estimated drift radius.
Swampy ground.	N/A	Irretrievable rocket components.	1	4	Moderate	The rocket should not be launched if there is swampy ground within the predicted drift radius.
Ponds, creeks, and other bodies of water.	N/A	1. Loss of rocket components. 2. Damaged electronics.	1	4	Moderate	Launching with high winds should be avoided in order to avoid drifting long distances. The rocket should not be launched if a body of water is within the estimated drift radius. Should the rocket be submerged in water, it should be retrieved immediately and any electrical components salvaged. Electrical components are to be tested for complete functionality prior to reuse.

Extremely cold temperatures.	N/A	1. Completely discharged batteries will cause electrical failures and fail to set off black powder charges, inducing critical events. 2. Rocket will not separate as easily.	1	5	Low	1. Batteries will be checked for charge prior to launch to ensure there is enough charge to power the flight. If the flight is delayed, batteries will should be rechecked and replaced as necessary. 2. If the temperatures are below normal launch temperature, black powder charges should be tested to ensure that the pressurization is enough to separate the rocket. If this test is successful, the rocket should be safe to launch.
Extremely high temperatures.	N/A	1. Prolonged heat exposure could reduce the performance of or degrade electronics, lead to over discharging, or cause an explosion in LiPo batteries. 2. Adhesives could degrade and lead to possible electrical malfunctions.	1	5	Low	The rocket should not be exposed to sun for long periods of time. If the rocket must be worked on for long periods of time, shelter should be sought.
Humidity.	N/A	Moisture in the motor or black powder charges may prevent them from igniting.	1	5	Low	Motors and black powder should be stored in a location free from moisture.

UV exposure.	N/A	Possibly weakening materials or adhesives.	4	4	Low	The rocket should not be exposed to sun for long periods of time. If the rocket must be worked on for long periods of time, shelter should be sought.

Table 113: Environmental hazards to the launch vehicle visk assessment.

7.1.9 Launch Vehicle Hazards to the Environment Assessment

Launch Vehicle Hazards to Environment Assessment						
Hazard	**Cause/ Mechanism**	**Outcome**	**Severity Value**	**Probability Value**	**Risk Level**	**Mitigation**
Harmful substances permeating into the ground or water.	Improper disposal of batteries or chemicals.	Impure soil and water can have negative effects on the environment that in turn, work their way into humans, causing illness.	4	3	Low	Batteries and other chemicals should be disposed of properly in accordance with the MSDS sheets. Should a spill occur, proper measure are to be followed in accordance with the MSDS sheets and any EHS standards.
Release of hydrogen chloride into the atmosphere.	Burning of composite motors.	Hydrogen chloride dissociates in water forming hydrochloric acid.	4	1	Moderate	While the probability of hydrochloric acid forming is high, the amount that would be produced over the course of a season is negligible. Fewer than six motors are predicted to be fired during the year.

Release of reactive chemicals.	Burning of composite motors.	Reactive chemicals work to deplete ozone layer.	4	1	Moderate	While the probability of releasing reactive chemicals into the environment is high, the quantity released will result in negligible effects. Fewer than six motors are predicted to be fired during the year.
Release of toxic fumes in the air.	Burning of ammonium perchlorate motors.	Biodegradation.	4	1	Moderate	Ammonium perchlorate will be burned in small quantities and infrequently. The toxins released will cause minimal degradation.
Production of styrene gas.	The fiberglass that is used in the vehicle body is manufactured by a second party.	Toxic air emissions.	4	1	Moderate	Productions methods for fiberglass produces toxic air pollutants, particularly styrene, which evaporate during the curing process. Due to the quantity of fiberglass utilized on the rocket, the amount of pollutants produced throughout manufacturing process will have a negligible effect on the environment.
Spray painting.	The rocket will be spray painted.	1. Water contamination. 2. Emissions to environment.	2	5	Low	All spray painting operations will be performed in a paint booth. This prevents any overspray from entering the water system or air.
Soldering wires.	All wires will be soldered together to retain strength and proper connection.	1. Air contamination 2. Ground contamination	4	1	Low	The amount of vapor from the soldering process is at such a low quantities that no action will be needed.

Use of lead acid battery leakage.	Old or damaged housing to battery	1. Acid will leak onto the ground and get into the water system. 2. Chemical reaction with organic material that could potentially cause a fire.	3	4	Low	1. We are using new batteries that have been factory inspected and tested. 2. Proper lifting and storing procedures in accordance with manufacturer's specifications will be adhered to.
Plastic waste material.	Plastic used in the production of electrical components and wiring.	1. Sharp plastic material produced when shaving down plastic components could harm animals if ingested. 2. Plastic may enter the water system through a drain.	3	5	Low	All plastic material will be disposed of in proper waste receptacles.
Wire waste material.	Wire material used in the production of electrical components.	Sharp bits of wire being ingested by an animal if improperly disposed of.	3	5	Low	All wire material will be disposed of in proper waste receptacles.
CO_2 emissions.	Travel to launch sites and competition.	Contribution to greenhouse effect and to global warming	4	1	Moderate	While the effects of CO_2 emissions cannot be reversed, the amount produced is negligible.

Table 114: Launch vehicle hazards to the environment.