OREGON STATE UNIVERSITY

2018 NASA SL TEAM

104 KERR ADMIN BLDG. # 1011

CORVALLIS, OR 97331

Preliminary Design Review

November 2nd, 2018

Contents

List of Tables

List of Figures

Acronym Dictionary

9DOF Nine Degree of Freedom. 16, 51, 52

AARD Advanced Retention Release Device. 102, 110

ABS Acrylonitrile Butadiene Styrene - A Common Thermoplastic Polymer. 9, 36, 37, 41, 91, 93, 94

AGL Above Ground Level. 15, 65, 67, 74, 76, 78, 79, 170, 174

AIAA American Institute of Aeronautics and Astronautics. 14, 203

APCP Ammonium Perchlorate Composite Propellant. 172

API Application Programming Interface. 46

ARRD Advanced Retention and Release Device. 67, 68, 106, 107, 109, 113, 178

ASL Above Sea Level. 73

ATF Alcohol, Tobacco, Firearms. 71

ATU Avionics Telemetry Unit. 43, 44, 46

BEAVS Blade Extending Apogee Variance System. 5, 26, 28, 44, 47, 48, 50–55

CAR Canadian Association of Rocketry. 172

CDR Critical Design Review. 167, 168, 172

CEMF Counter Electromotive Force, AKA Back EMF. 107

CFD Computational Fluid Dynamics. 51

CNC Computer Numerical Control. 31, 50

DC Direct Current. 46, 88

DDM Design Decision Matrix. 5, 6, 23, 26, 27, 29, 31, 32, 36, 37, 39, 41–43, 45–50, 54, 66, 67, 70, 71, 74, 85–87, 89–97, 102–106, 115, 118, 120

DORIS Doppler Orbitography and Radiopositioning Integrated by Satellite. 43

EMI Electromagnetic Interference. 107

FAA Federal Aviation Administration. 22, 171, 182

FHP Fore Hard Point. 10, 110, 222, 223

FMEA Failure Mode Effects Analysis. 122, 133, 182, 190

FN Foreign National. 167

FRR Flight Readiness Review. 47, 54, 168, 173, 179

GLONASS Global Navigation Satellite System. 43

GPS Global Positioning System. 43, 44, 46, 188

HDPE High-density polyethylene. 36, 39, 91

HPR High Powered Rocketry. 22, 102, 185

IMU Inertial Measurement Unit. 16, 51, 52

JHA Job Hazard Analysis. 9, 122, 123, 180, 190, 191

LED Light Emitting Diode. 191

LiPo Lithium Polymer. 45, 46, 191

LRR Launch Readiness Review. 179

MPRL Machine Product and Realization Laboratory. 180, 190

MSDS Material Safety Data Sheet. 182

NAR National Association of Rocketry. 22, 122, 125, 172, 182, 185, 190

NASA National Aeronautics and Space Administration. 122, 168, 171, 173, 182

NEMA National Electrical Manufacturers Association. 50

NFPA National Fire Protection Agency. 122

NTS not to scale. 8, 76–78

OROC Oregon Rocketry. 22, 182

OSGC Oregon Space Grant Consortium. 9, 22, 203

OSRT Oregon State Rocketry Team. 9, 14, 16, 22, 23, 25, 26, 28, 30–34, 41, 43, 44, 46, 47, 51–55, 58, 64, 66–68, 72, 79, 84, 85, 101–104, 106, 112, 115, 122, 124, 125, 133, 141, 149, 166–169, 180, 181, 183–186, 188, 203, 206

OSU Oregon State University. 44, 65, 66, 122, 173, 180, 190, 203

PCB Printed Circuit Board. 9, 51, 52, 109

PDR Preliminary Design Review. 97, 167, 170

PEARS Payload Ejection and Retention System. 6, 9, 10, 21, 90, 94, 101, 102, 104–106, 110–113, 219–222

PID Proportional-Integral-Derivative. 53, 54

PLA Polylactic Acid. 36, 37, 39, 40, 91

PLEC Payload Ejection Controller. 9, 10, 21, 104–111, 222

PPE Personal Protective Equipment. 165, 180, 191

RF Radio-Frequency. 5, 27–29, 34, 35, 38–41, 43–46

RPM Rotations per Minute. 89, 90

RRC3 Rocket Recovery Controller 3. 65, 66, 76, 78, 175

RSO Range Safety Officer. 112, 159, 164, 172, 174, 180, 182

SCAR Soil Collection and Retention. 6, 10, 22, 86, 96, 98, 100, 101, 113, 224–228

SL Student Launch. 56

SO Safety Officer. 190

SPDT Single Pole Double Throw. 9, 105, 106, 110, 111

STEM Science, Technology, Engineering and Mathematics. 112, 167, 168

TRA Tripoli Rocketry Association, Inc.. 22, 172, 182, 185, 190

USLI University Student Launch Initiative. 16, 55, 57, 65, 81, 84, 112, 172

1 Summary of PDR Report

1.1 Team Summary

Table 1: Team Summary Chart

Team Name	Oregon State Rocketry Team
Mailing Address	104 Kerr Admin Bldg #1011 Corvallis, OR 97331
Name of Mentor	Joe Bevier
NAR/TRA Number, Certification Level	NAR #87559 Level 3, TRA #12578 Level 3
Contact Information	joebevier@gmail.com, (503) 475-1589

1.1.1 Team Members and Organization

The Oregon State Rocketry Team (OSRT) consists of 37 members from the schools of Mechanical Engineering, Electrical Engineering, and Computer Science. The team strives to involve members of the campus American Institute of Aeronautics and Astronautics (AIAA) chapter.

Due to the multifaceted nature of this project, it has been broken up into three sub-teams according to technical design, with the following team descriptions:

Structures - Responsible for design and fabrication of the airframe and all internal components necessary for a successful launch and payload recovery. This team will also be in charge of implementing a proper motor while considering safety and handling before and after each launch. Key responsibilities include mass and stress analysis to ensure altitude precision, understanding key propulsive features to ensure reliability, and monitoring of the effects of design improvements.

Aerodynamics & Recovery - Responsible for the electronics behind aerodynamic stability, all parachute systems for recovery systems, and design of stability measures. Key requirements are to ensure a safe landing, monitor kinetic energy requirements, and fabricate electrical and mechanical hardware to ensure aerodynamic flight.

Payload - Responsible for the design, fabrication, and testing of a rover capable of traveling at least ten feet from the launch vehicle and collecting soil samples. Key responsibilities include meeting all customer requirements, designing a rover that reliably functions, and rigorous testing prior to the final launch.

The team consists of a team lead with three sub-teams. Each sub-team lead oversees a project team and additional testing project teams. The additional project teams include developing a test-bed for recovery ejection methods, implementing data-logging features to the airframe, creating a test method for ejection of the rover, and rapidly developing a rover prototype. The team construction can be seen in Figure 1.

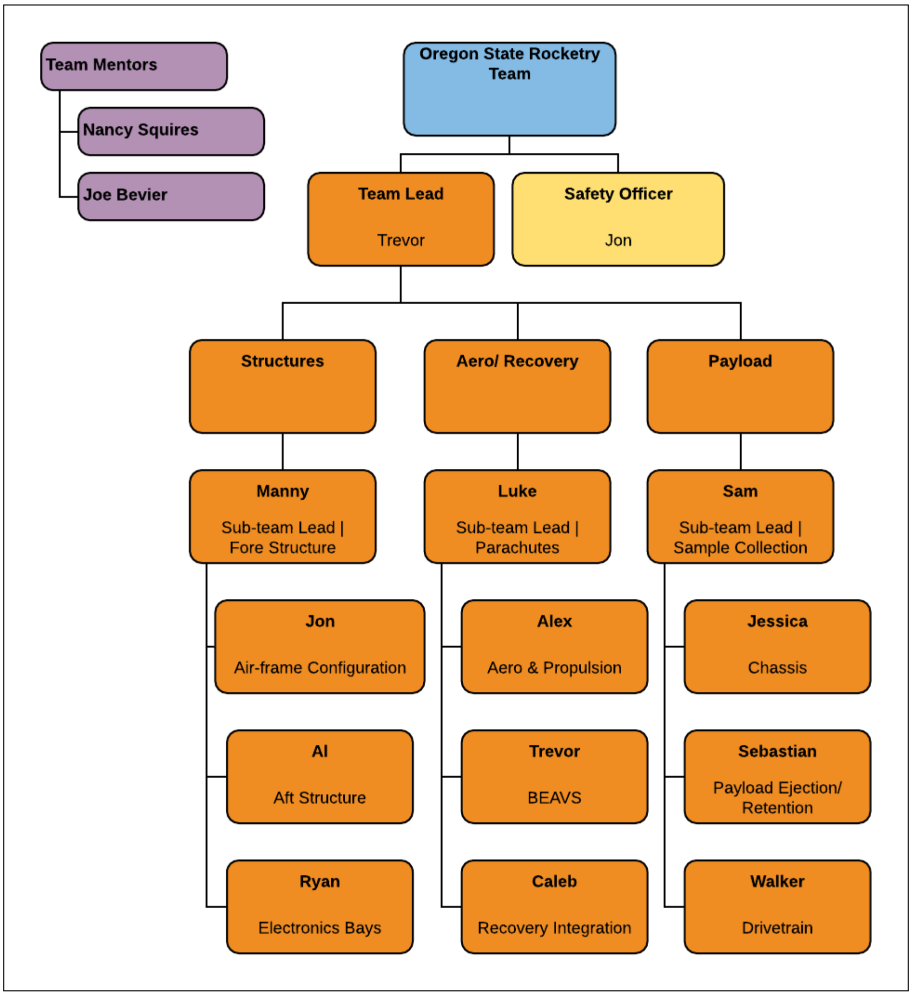

Figure 1: Team Organization

1.2 Vehicle Summary

The major vehicle characteristics are shown in Table 2. The launch vehicle airframe is a 6.25 in. inner diameter carbon fiber tube. The nosecone is a 5:1 ogive and attached to the base are four trapezoidal fins. Both the nosecone and fins are fiberglass. The launch vehicle is recovered in two independent sections: the aft section with the motor, and the fore section with the payload. At apogee, the launch vehicle will have two simultaneous separation events: the aft section will separate from the fore section and release a first drogue parachute, and the fore section will separate from the nosecone and release a second drogue parachute. At 500 ft Above Ground Level (AGL), the main parachutes, which are both held in with two tender descenders in series, are released. The drogue parachutes are cruciform, with 1.5 ft. in diameter. The main parachutes are toroidal, and the fore and aft are different sizes: the fore is 10 ft. in diameter, and the aft is 7 ft. in diameter. The launch vehicle will exit the 12 ft rail at 84.6 ft/s. The motor to be used is a Cesaroni L2375-WT 75 mm motor.

Table 2: Launch Vehicle Characteristics

Length	Weight	Motor
100 in.	54.9 lbs.	L2375-WT

1.3 Payload Summary

The OSRT has chosen to complete option two for the University Student Launch Initiative (USLI) payload: a deployable rover. The rover will be contained within the fore section of the airframe. Upon landing, the rover will be ejected from the airframe using black powder charges. The rover will have two coaxial, independently driven wheels with a chassis suspended between them. A spring-loaded stabilizer arm will act as a third point of contact with the ground. An Arduino Teensy 3.6 development board will autonomously control the motors to move the rover, receiving input from a sensor array including active sonar, passive sonar, and a Nine Degree of Freedom (9DOF) Inertial Measurement Unit (IMU). An auger will be mounted in the center of the chassis. When the rover is deployed the auger will periodically gather soil samples and store them in an internal containment unit. After collection, the rover will autonomously drive to a Scientific Base Station where it will perform an additional scientific experiment.

1.4 Flysheet

Milestone Review Flysheet 2018-2019

Institution	Oregon State University

Milestone	PDR

Vehicle Properties

Total Length (in)	100
Diameter (in)	6.25
Gross Lift Off Weigh (lb)	54.9
Airframe Material(s)	Carbon Fiber, Fiberglass
Fin Material and Thickness (in)	Carbon Fiber
Coupler Length(s)/Shoulder Length(s) (in)	12.5 / 6.25

Motor Properties

Motor Brand/Designation	Cesaroni L2375-WT
Max/Average Thrust (lb)	586.3 / 533.7
Total Impulse (lbf-s)	1102.67
Mass Before/After Burn (lb)	9.71 / 4.06
Liftoff Thrust (lb)	553.5
Motor Retention Method	Threaded Retainer

Stability Analysis

Center of Pressure (in. from nose)	72.051
Center of Gravity (in. from nose)	58.548
Static Stability Margin (on pad)	2.1
Static Stability Margin (at rail exit)	2.1
Thrust-to-Weight Ratio	9.77
Rail Size/Type and Length (in)	1515 / 144
Rail Exit Velocity (ft/s)	84.6

Ascent Analysis

Maximum Velocity (ft/s)	596
Maximum Mach Number	0.54
Maximum Acceleration (ft/s^2)	322
Target Apogee (ft)	4500
Predicted Apogee (From Sim.) (ft)	4797

Recovery System Properties - Overall

Total Descent Time (s)	71 (fore), 72 (aft)
Total Drift in 20 mph winds (ft)	2092 (fore), 2113 (aft)

Recovery System Properties - Energetics

Ejection System Energetics (ex. Black Powder)		Black Powder
Energetics Mass - Drogue Chute (grams)	Primary	2.12
	Backup	3
Energetics Mass - Main Chute (grams)	Primary	0.33
	Backup	0.33
Energetics Mass - Other (grams) - If Applicable	Primary	5.5
	Backup	8.25

Recovery System Properties - Recovery Electronics

Primary Altimeter Make/Model	PerfectFlite, StratoLoggerCF
Secondary Altimeter Make/Model	Missleworks, RRC3
Other Altimeters (if applicable)	Jolly Logic, AltimeterThree
Rocket Locator (Make/Model)	X-Bee Pro 900HP
Additional Locators (if applicable)	Sparkfun Venus GPS
Transmitting Frequencies (all - vehicle and payload)	CC1200: 433 MHz Xbee PRO 900HP: 900 MHz
Describe Redundancy Plan (batteries, switches, etc.)	Two altimeters for each section, separate batteries for each altimeter, separate charges for each altimeter, two chute releases per main chute
Pad Stay Time (Launch Configuration)	Altimeters: 8+ hours Tracking Unit: 3 hours

Recovery System Properties - Drogue Parachute

Manufacturer/Model	Top Flight Recovery / XTEAR-18			
Size or Diameter (in or ft)	18 in. (fore) / 18 in. (aft)			
Main Altimeter Deployment Setting	Apogee			
Backup Altimeter Deployment Setting	Apogee +1 s			
Velocity at Deployment (ft/s)	1.7			
Terminal Velocity (ft/s)	146 (fore) /127 (aft)			
Recovery Harness Material, Size, and Type (examples - 1/2 in. tubular Nylon or 1 in. flat Kevlar strap)	1 in. Nylon Web			
Recovery Harness Length (ft)	30 (fore) / 30 (aft)			
Harness/Airframe Interfaces	3/8 in. forged steel eyebolts connected to altimeter bulkheads			
Kinetic Energy of Each Section (Ft-lbs)	Section 1	Section 2	Section 3	Section 4
	5137 (fore)	3444 (aft)	419.3 (nosecone)	N/A

Recovery System Properties - Main Parachute

Manufacturer/Model	Fruity Chutes Toroidal			
Size or Diameter (in or ft)	10 ft (fore) / 8 ft (aft)			
Main Altimeter Deployment Setting (ft)	525			
Backup Altimeter Deployment Setting (ft)	500			
Velocity at Deployment (ft/s)	146 (fore) / 127 (aft)			
Terminal Velocity (ft/s)	12.02 (fore) / 12.95 (aft)			
Recovery Harness Material, Size, and Type (examples - 1/2 in. tubular Nylon or 1 in. flat Kevlar strap)	1 in. Nylon Web			
Recovery Harness Length (ft)	15			
Harness/Airframe Interfaces	3/8 in. forged steel eyebolts connected to altimeter bulkheads			
Kinetic Energy of Each Section (Ft-lbs)	Section 1	Section 2	Section 3	Section 4
	60.20 (fore)	52.40 (aft)	4.913 (nosecone)	N/A

Milestone Review Flysheet 2018-2019

Institution	Oregon State University	Milestone	PDR

Payload

Payload 1 (official payload)	Overview The rover will be contained within the fore section of the airframe. Upon landing, the rover will be ejected from the airframe using black powder charges. The rover will have two coaxial, independently driven wheels with a chassis suspended between them. A spring-loaded stabilizer arm will act as a third point of contact with the ground. An Arduino Teensy 3.6 development board will autonomously control the motors to move the rover, receiving input from a sensor array including active sonar, passive sonar, and a nine-degree-of-freedom IMU. An auger will be mounted in the center of the chassis. When the rover is deployed the auger will periodically gather soil samples and store them in an internal containment unit. After collection, the rover will autonomously drive to a Scientific Base Station where it will perform an additional scientific experiment.
Payload 2 (non-scored payload)	Overview None

Test Plans, Status, and Results

Ejection Charge Tests	Sub-Scale Test Plan: After final launch vehicle assembly with bulkheads and recovery system, a remote ignition system will be used to ensure proper seperation and parachute ejection with selected amount of black powder. Full-Scale Test Plan: After final launch vehicle assembly with bulkheads and recovery system, a remote ignition system will be used to ensure proper seperation and parachute ejection with selected amount of black powder.
Sub-scale Test Flights	Test Plan: Sub-scale launch vehicle will be constructed with the same stability margin as the full scale launch vehicle. The same ejection controllers will be used in sub-scale flights as the full scale launch vehicle. Status: Planned test flight November 17th, 2018 & December 15th, 2018. Results: N/A
Vehicle Demon-stration Flights	Test Plan: The full scale launch vehicle will be manufactured from final design choices. Altitude and decent calculations will be calculated and verfied with simulations. The full scale launch vehicle will be flown multiple times in January and February 2019 before NASA Student Launch competition launch day April 6th, 2019. Status: Preliminary design of launch vehicle is complete. Results: N/A
Payload Demon-stration Flights	Test Plan: The competition payload is planned to fly in all full scale flights. The payload demonstration flights will be the same flights as the vehicle demonstration flights. Status: Preliminary design of launch vehicle is complete. Results: N/A

Milestone Review Flysheet 2018-2019

Institution	Oregon State University	Milestone	PDR

Transmitter #1			
Location of transmitter:	Nosecone		
Purpose of transmitter:	Tracking/Telemetry		
Brand	Digi	RF Output Power (mW)	250
Model	Xbee PRO 900HP	Specific Frequency used by team (MHz)	900
Handshake or frequency hopping? (explain)	Frequency hopping, 400KHz wide channels		
Distance to closest e-match or altimeter (in)	6		
Description of shielding plan:	Conductive spray paint RF shielding around recovery electronics to ensure no interference with recovery electronics and to ensure that ejection takes place at the correct altitude.		

Transmitter #2			
Location of transmitter:	Nosecone		
Purpose of transmitter:	Long Range Tracking/Telemetry		
Brand	Texas Instruments	RF Output Power (mW)	40
Model	CC1200	Specific Frequency used by team (MHz)	433
Handshake or frequency hopping? (explain)	Frequency hopping		
Distance to closest e-match or altimeter (in)	6		
Description of shielding plan:	Conductive spray paint RF shielding around recovery electronics to ensure no interference with recovery electronics and to ensure that ejection takes place at the correct altitude.		

Transmitter #3			
Location of transmitter:	Aft section of airframe directly above the motor		
Purpose of transmitter:	Tracking/Telemetry		
Brand	Digi	RF Output Power (mW)	250
Model	Xbee PRO 900HP	Specific Frequency used by team (MHz)	900
Handshake or frequency hopping? (explain)	Frequency hopping, 400KHz wide channels		
Distance to closest e-match or altimeter (in)	4		
Description of shielding plan:	Conductive spray paint RF shielding around recovery electronics to ensure no interference with recovery electronics and to ensure that ejection takes place at the correct altitude.		

Transmitter #4			
Location of transmitter:	Aft section of airframe directly above the motor		
Purpose of transmitter:	Long Range Tracking/Telemetry		
Brand	Texas Instruments	RF Output Power (mW)	40
Model	CC1200	Specific Frequency used by team (MHz)	433
Handshake or frequency hopping? (explain)	Frequency hopping		
Distance to closest e-match or altimeter (in)	4		
Description of shielding plan:	Conductive spray paint RF shielding around recovery electronics to ensure no interference with recovery electronics and to ensure that ejection takes place at the correct altitude.		

Milestone Review Flysheet 2018-2019

Institution	Oregon State University	Milestone	PDR

Transmitter #5			
Location of transmitter:	Fore section above payload bay		
Purpose of transmitter:	Payload Ejection		
Brand	Digi	RF Output Power (mW)	250
Model	Xbee PRO 900HP	Specific Frequency used by team (MHz)	900
Handshake or frequency hopping? (explain)	Frequency hopping, 400KHz wide channels		
Distance to closest e-match or altimeter (in)	6		
Description of shielding plan:	Conductive spray paint RF shielding around recovery electronics to ensure no interference with recovery electronics and to ensure that ejection takes place at the correct altitude.		

Transmitter #6			
Location of transmitter:			
Purpose of transmitter:			
Brand		RF Output Power (mW)	
Model		Specific Frequency used by team (MHz)	
Handshake or frequency hopping? (explain)			
Distance to closest e-match or altimeter (in)			
Description of shielding plan:			

Additional Comments

None

2 Changes Since Proposal

2.1 Launch Vehicle Criteria

The launch vehicle has had several design changes since the proposal submission. The aspects changed include the airframe inner diameter, the motor, the recovery layout, and the overall weight of the launch vehicle. The airframe inner diameter has been increased to 6.25 in. from the proposed 5.2 in. The primary reason for this was to accommodate extra room for the payload and its auger. The new motor choice is the Cesaroni L2375-WT L-class motor. It was chosen because of the high thrust to weight ratio it provides, and the reason for the change was the increase in the projected weight of the launch vehicle. The fore and the aft section will be falling to the ground independently of each other, so two recovery systems are necessary. The new recovery layout includes a single compartment for containment of the parachutes and shock cords. The fore and aft sections both have a single compartment for containment. Lastly, the weight of the launch vehicle has increased to 54.90 lbf, which is an increase of 15.35 lbf.

2.2 Payload Criteria

The chassis design has been updated to be easily disassembled and reassembled for component testing and replacement. The cross bars connecting the two trusses of the chassis have been replaced with aluminum bars that are tapped at both ends. Screws are then passed through holes in the corner connection blocks in the two trusses and screwed into the aluminum cross bars from either end.

The evolution of other subsystems have given rise to a number of changes to the drivetrain. First, it was necessary to increase the diameter of the wheels in response to the larger airframe. Since the airframe diameter was increased to 6.250 in., the wheel diameter increased to 6 in. This gives $1/4$ in. for the foam wheel wrap and the carbon fiber sleeve which contains the rover. In an effort to maintain stability along with this change, the wheel width was also increased from 0.75 in. to 1.00 in. The mounting block sub-assembly, which fixes each half of the drivetrain to either end of the chassis, has developed significantly since the proposal. The rest of the drivetrain subsystem saw only minor changes to its design with respect to that submitted in the proposal.

The Payload Ejection and Retention System (PEARS) has had many small design changes, but the overall mechanisms for both retention and ejection have remained the same since the proposal. The most significant change is the development of a system to externally arm the PEARS once fully integrated into the airframe, and ready to prepare on the launch pad. This arming system consists of a PEARS switch to shunt the electrical circuit, an additive manufactured mount for the switch and Payload Ejection Controller (PLEC), and a pass-through bulkhead fixed in the airframe to create a pressure seal between the payload ejection charge and the arming hole.

The payload soil collection/retention system has acquired the acronym Soil Collection and Retention (SCAR). The SCAR will continue to use an auger for soil retriever, but a lead screw will no longer be used to feed the auger into and out of the ground. Instead, a tube with helical and vertical channel cuts will be used (see the SCAR Design Review section for details). This will reduce weight and volume.

Also, the SCAR will use a different system to close the doors of the soil container. Instead of using spring-assisted doors, the SCAR will use motor-operated doors that pivot near the center top and bottom, center positions of the container. Doors will be used on the top and the bottom so that soil can be collected and deposited.

The release mechanism allows for an additional scientific experiment to be performed on the sample. The specific details of such an experiment are still being finalized, but the team anticipates having a standalone station to which the rover can travel and deposit its soil sample.

2.3 Project Plan

Additional

There have been minor changes to the project schedule. Due to a shortened launch window by the Federal Aviation Administration (FAA), OSRT chose not to attend the public launch event on October 13th, 2018 at the Oregon Rocketry (OROC) launch site in Brothers, OR. This choice was made due to significant concern about launch opportunities throughout the day. This event was the team's planned date for launching Level 1 High Powered Rocketry (HPR) Certification Rockets for certification by National Association of Rocketry (NAR) or Tripoli Rocketry Association, Inc. (TRA). The team instead plans to attempt certifications after our subscale launch on November 10th, 2018. Other than this, only minor changes and updates have been made to the project schedule.

The team's budgeting plan has remained largely the same. It has submitted its grant proposal to Oregon Space Grant Consortium (OSGC) and is communicating with a variety of organizations seeking corporate sponsorships and partnerships. Specific part costing information has been updated to reflect accurate costs of the system throughout design changes.

3 Vehicle Criteria

3.1 Design Justification Methodology

To evaluate design alternatives, OSRT used a Design Decision Matrix (DDM), like the one found in Table 4. The table identifies the system requirements and specifications which are important for the system to function. Each of these system requirements were then assigned a weight from 1-10, where 1 is not a very important requirement and 10 is a critical requirement. A baseline design was arbitrarily selected from the design alternatives. The baseline design did not hold any significance, other than serving as a standard by which to compare other design alternatives. The rubric for the scoring system displayed in Table 3 was then used to compare all alternatives. Based on the rating system used, the baseline design is signified by being rated with 3's for all requirements. The requirement weight was multiplied by the design specific rating, and totaled. The design with the highest total rating represents the design which has been critically evaluated as being the leading design choice according to the specified criteria.

Table 3: Rating system for DDM on a 1-5 scale.

Rating	**Rating Definition**
1	Significantly worse than baseline
2	Worse than baseline
3	Baseline
4	Better than baseline
5	Significantly better than baseline

Table 4: Generic DDM

Design		**Idea 1**		**Idea 2**		**Idea 3**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Specification 1							
Specification 2							
Specification 3							
Specification 4							
Total							

3.2 Launch Vehicle

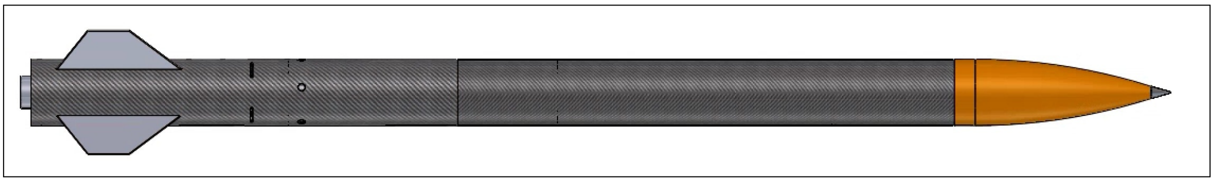

Figure 2: Launch Vehicle CAD Model

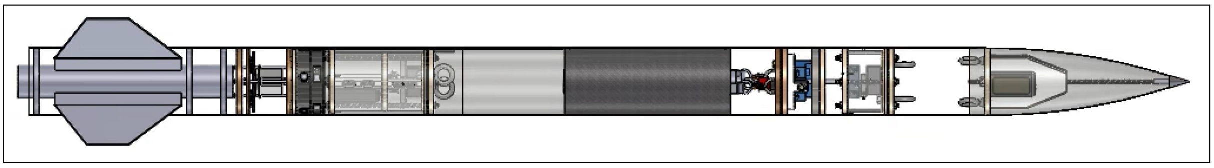

Figure 3: Launch Vehicle Components CAD Model

3.2.1 Mission Statement & Success Criteria

The launch vehicle will successfully deliver the payload to the target altitude, deploy recovery systems at apogee, and safely land the airframe on the ground within 2,500 ft of the launch site. The vehicle will remain reusable throughout the entire process. The mission will be determined a success for the launch vehicle when the following criteria has been met:

- The launch vehicle travels in a stable configuration toward apogee
- The launch vehicle reaches the specified height within tolerance range
- The airframe successfully separates into two recovery sections
- The drogue chutes deploy successfully
- The main chutes deploy successfully
- Both airframe sections land on the ground, without causing structural damage
- The payload ejects successfully

3.2.2 Subscale

A subscale launch vehicle will be manufactured and flown to test recovery systems and aerodynamic properties of the launch vehicle. The subscale launch vehicle will be 75 in. in total length, with an AeroTech K1100T. The OpenRocket model can be seen in Figure 4. The airframe will be manufactured of a 4 in. diameter fiberglass tube. The fins and nosecone of the subscale launch vehicle are designed to have a similar subscale stability as the full scale launch vehicle.

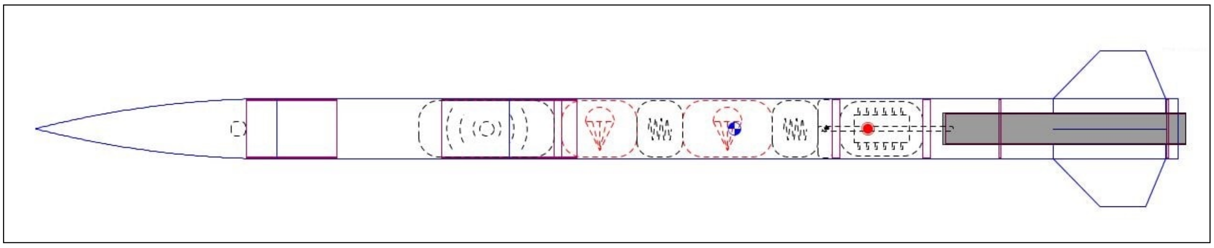

Figure 4: OpenRocket model of the subscale launch vehicle.

The structures sub-team is largely responsible for the manufacturing of the subscale launch vehicle. The OSRT will create the airframe, which consists mainly of fiberglass. The design is meant to test many aspects of the full scale airframe such as: fin design, altimeters, recovery system, landing kinetic energy simulations, aerodynamic simulations, and many of the manufacturing processes which will be used on the full scale launch vehicle.

At this point, OSRT has cut fin slots and fins, sanded all components, and began assembly. To cut the fin slots, OSRT mounted the body to a milling machine in a rotary fixture. The rotary fixture allowed for precise positioning of the airframe so the slots were separated exactly 90° apart. Mounting the body so that it was level and straight relative to the machine was the most challenging problem encountered with this method. Throughout the manufacturing and assembly process, the team has been developing and updating manufacturing plans that can be used on the full scale launch vehicle. The subscale airframe has proved to be beneficial before it has even been launched as it has helped to prepare the OSRT for future manufacturing. Figure 5 shows some of the team's progress on the subscale launch vehicle.

Figure 5: Manufacturing the subscale centering ring assembly.

Over the course of the next two weeks, subscale manufacturing will be completed so that the launch date of November seventeenth can be achieved. Construction from a structural point of view is nearly done and the electrical and computer science team will soon take over. One aspect that the OSRT is particularly excited about testing is the Blade Extending Apogee Variance System (BEAVS). A small version of the BEAVS will be implemented into the subscale in order to analyze how accurate it can control apogee and what aspects can be improved.

3.2.3 Airframe Structures

The launch vehicle is designed to withstand all forces that it may see during launch, separation, and landing. To ensure everything stays together, OSRT will use G5000 RocketPoxy for all structural bonding within the launch vehicle. The justifications can be seen in Table 5. This is an epoxy which OSRT has used previously on several airframes and seen no failures.

Table 5: Epoxy DDM

Design		**G5000 RocketPoxy**		**J-B Weld**		**Mid-Cure**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Bond Strength	9	5	45	5	45	3	27
Cure Time	4	3	12	1	4	3	12
Weight	7	5	35	4	28	3	21
Total		**92**		**77**		**60**	

3.2.3.1 Nosecone

The goal in specifying the shape of the nosecone is to reduce the overall drag of the launch vehicle. The OSRT launch vehicle will remain in subsonic speeds throughout the flight. Simulated by OpenRocket, the fastest the launch vehicle will travel will be Mach 0.53. Three nosecone shapes were investigated for the launch vehicle: conical, ogive, and Von Karman. Table 6 displays the weights and results from the investigation.

An ogive nosecone was chosen for the OSRT launch vehicle. At subsonic speeds, the drag coefficient is comparable to that of the Von Karman nosecone. With the selection of 6.25 in. airframe diameter, any of the nosecone decisions will require manufacturing work by OSRT. The conic nosecone is the simplest manufacturing work of the three. However, the simplicity in ordering a ogive nosecone is the key factor in choosing an ogive nosecone. There are multiple options in purchasing an ogive that can be modified for this specific airframe.

The material that will be used for the nosecone will be fiberglass. The high compression strength of fiberglass and its toughness makes it ideal for the constant forces experienced during the launch. The nosecone houses

Table 6: Nosecone Shape DDM

Design		Ogive		Conic		Von Karman	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Drag Coefficient	10	3	30	2	20	3	30
Ease of Manufacturing	7	3	21	4	28	3	21
Ease of Purchasing	5	4	20	3	15	3	15
Total		**71**		**63**		**66**	

avionics and will be transmitting radio frequencies so that the team can track the position of the fore section of the launch vehicle. Fiberglass has Radio-Frequency (RF) transparency, which will allow for a secure connection to the ground station. The primary alternative to fiberglass is carbon fiber which provide strength and reduced weight but does not allow for RF penetration.

The decision was made based on a balance of strength, manufacturability, RF transparency, ease of purchase, and weight as detailed in Table 7. The engineering requirements used for the nosecone material selection were as follows:

- Strength: Necessary to maximize to reduce probability of nosecone deformation while under launch loading conditions.
- Ease of Manufacturing: A qualitative measurement of the ease of dealing with the specific material.
- RF Transparency: Necessary to allow constant communication with the ground station.
- Ease of Purchase: Measurement of lowest cost along with its availability to purchase commercially.
- Weight: Necessary to minimize to reduce overall airframe weight.

Table 7: Nosecone Material DDM

Design		Fiberglass		Plastic		Carbon Fiber	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Strength	8	3	24	2	16	4	32
Ease of Manufacturing	9	3	27	3	27	3	27
RF Transparent	9	3	27	3	27	1	9
Ease of Purchasing	5	3	15	2	10	3	15
Weight	7	3	21	4	28	4	28
Total		**114**		**108**		**111**	

3.2.3.2 Body Tube

A launch vehicle body tube made up of carbon fiber and fiberglass will be utilized. The body tube will transition from carbon fiber to fiberglass using a filament wound manufacturing process. This allows the body tube to be lightweight, structurally sound, and RF transparent in the areas it is needed. The inner diameter of the body tube will be 6.25 in. This diameter was chosen to accommodate a larger rover size.

On the aft end of the launch vehicle, the BEAVS system will be integrated, which requires slots to be cut in the body tube. To maintain structural integrity, these slots will be placed within a carbon fiber section of the body tube. Additional holes will be placed in the aft end of the launch vehicle for the Camera System integration. These holes will be placed within a fiberglass section, as to maintain live streaming capabilities. Within the aft end of the launch vehicle will be the motor, which will be mounted inside of a fiberglass motor tube.

There were several materials considered to manufacture the body of the airframe: carbon fiber, fiberglass, and blue tube. Carbon fiber is the strongest material that was debated. It provides the team a sturdy base, which will have full capabilities to withstand the forces experienced during flight. It is also the most aesthetic of these materials and is a very common choice for high powered rocketry. However, it is also the most expensive of these choices. Another, more serious concern, is that carbon fiber is not RF transparent, meaning that no electronics will be able to send or receive a signal.

Fiberglass is RF transparent and also comes at a lower cost. The drawbacks are that it weighs more and has a higher flexibility. In most cases with airframes, higher flexibility is a poor quality to have. However, upon landing, carbon fiber is more easily damaged than fiberglass.

Blue tube is the cheapest of the three and, while it is not the strongest, it should provide enough strength to withstand the applied forces. It is also RF transparent so sending and receiving signals would not be an issue.

To decide upon a material, each of the types of airframe were scored based on how well a material fulfills the criteria as well as the importance of that criteria, as detailed in Table 8. A list of the requirements of the launch vehicle body is as follows:

- Strength: A key aspect that needs to be met and is measured using Young's Modulus.
- Ease of Manufacturing: Whatever material is chosen, it is key that it have the ability to modify and fit the needs of the OSRT.
- Thermal Conductivity: A high value is needed to keep the temperature of the all electronics and components low to reduce the chances of overheating.
- Ease of Purchasing: Minimize to allow funding to be placed elsewhere.
- Weight: The weight of the material is the most important factor in determining the airframe material.

Table 8: Body Tube Material DDM

Design		**Fiberglass**		**Carbon Fiber**		**Blue Tube**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Strength	8	4	32	5	40	3	24
Ease of Manufacturing	7	2	14	3	21	3	21
Thermal Conductivity	3	5	15	4	12	3	9
Ease of Purchasing	3	4	12	4	12	3	9
Weight	9	3	27	4	36	3	27
Total		**100**		**121**		**90**	

Based on the properties of carbon fiber, specifically the strength to weight ratio, the body will be mostly comprised of this material. The material is not RF transparent, which will call for fiberglass couplers to be used to maintain constant ground station communication during flight. For sections of the launch vehicle that do not land on a coupler, carbon fiber tube will transition to fiberglass using a blend, then transition back to carbon fiber to maximize the usage of this material as shown in Figure 6, (30 in. of the airframe will be fiberglass and 11 in. will be carbon fiber, with a 1 in. transition). However, picking a lightweight material will help reduce the weight of the rocket, causing an improvement in launch performance and reliability of the recovery system. The weight reduction will allow for more freedom when designing the rover. Finally, the high strength of carbon fiber will prevent the body from deforming, reducing the possibility of broken components.

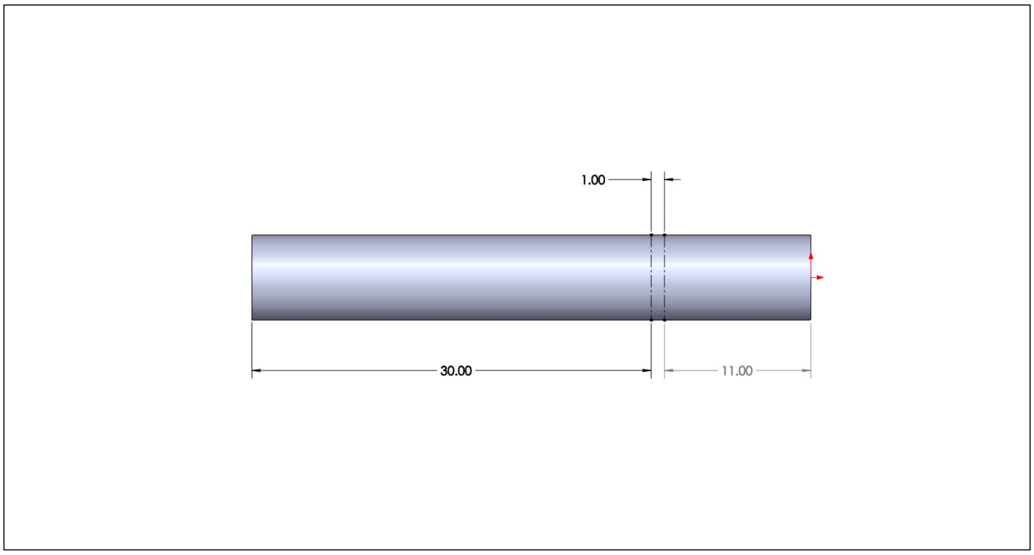

Figure 6: Fore Airframe: Fiberglass to Carbon Fiber Transition

3.2.3.3 Fins

OSRT has chosen four trapezoidal fins for the launch vehicle. This shape can be seen in Figure 7.

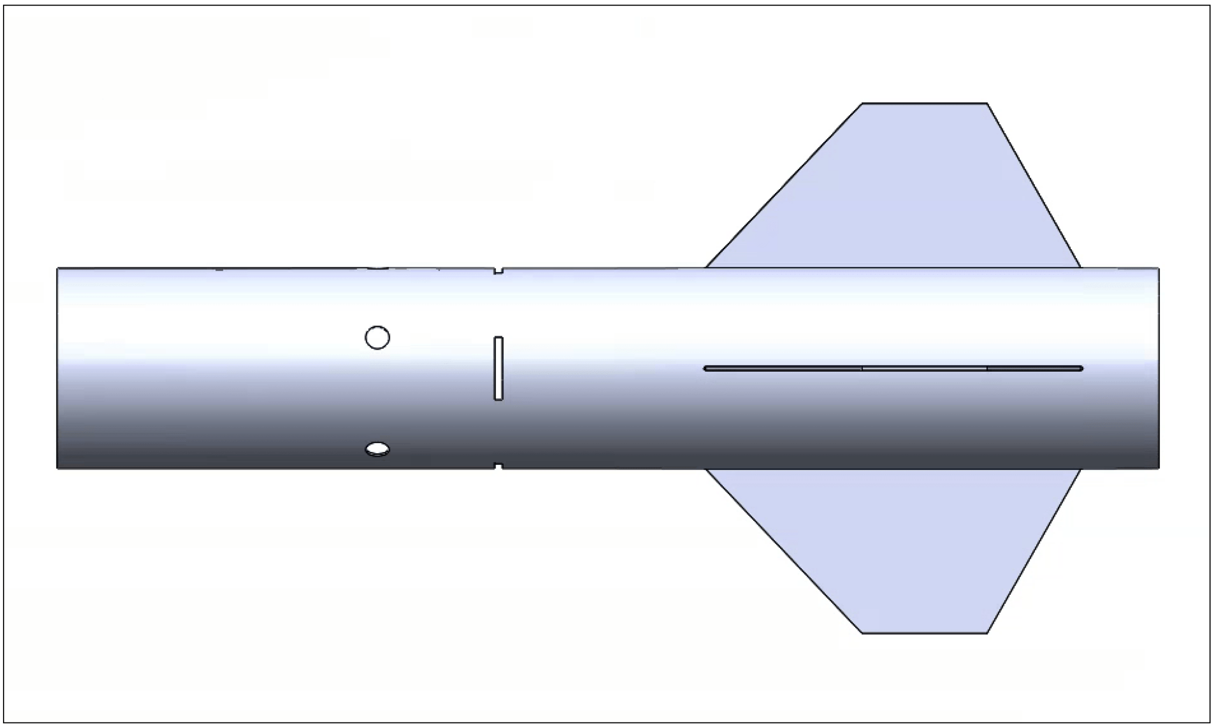

Figure 7: Model of Four Trapezoidal Fins

The trapezoidal fins were chosen to have a trailing edge towards the fore section of the launch vehicle. Along with the trailing edge angle, the aft end of the fins are located two in. forward from the bottom edge of the aft airframe. The combination of angle and location is to prevent possible damage to the fins upon landing of the aft airframe. The trailing edge angle and the forward location of the fins requires larger dimensions to maintain stability. If weight requirements become to large of an issue, OSRT will investigate lowering the weight of the fins by locating the trailing edge closer to the bottom of the airframe and with less of a trailing edge angle. In doing these two things, fins are able to be reduced in size while maintaining stability.

Ease of manufacturing has been one of the main deciding factors for the fins of the launch vehicle. For alignment, aligning 3 fins is more of a challenge than aligning 4 fins. With 3 fins, the human eye may have a difficult time identifying whether the fins are perfectly 120° apart from one another. With 4 fins, it is much easier to identify if the fins are out of alignment, as 90° is a very intuitive angle for the human eye. Additionally, as stated prior, manufacturing a fixture that aligns 4 fins is not as challenging as a fixture that aligns 3 fins. A fixture for 4 fins only requires 2 perpendicular cuts with a circle in the middle, whereas a fixture for 3 fins would require 3 cuts with a circle in the middle. The fixture that OSRT will use for ensuring fin alignment, while the epoxy cures, can be seen below in Figure 8. Because of angle propagation in manufacturing, this is the reason OSRT chose 4 fins over 3 fins. There are certainly arguments to be made for 3 fins being a better decision, but since manufacturing and fin alignment are so crucial, OSRT considered that the utmost important deciding factor, as seen in Table 9.

Table 9: Number of Fins DDM

Design		**4 Fins**		**3 Fins**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score
Stability	7	3	21	4	28
Accurate Alignment	9	3	27	1	9
Weight	5	3	15	4	20
Total		**63**		**57**	

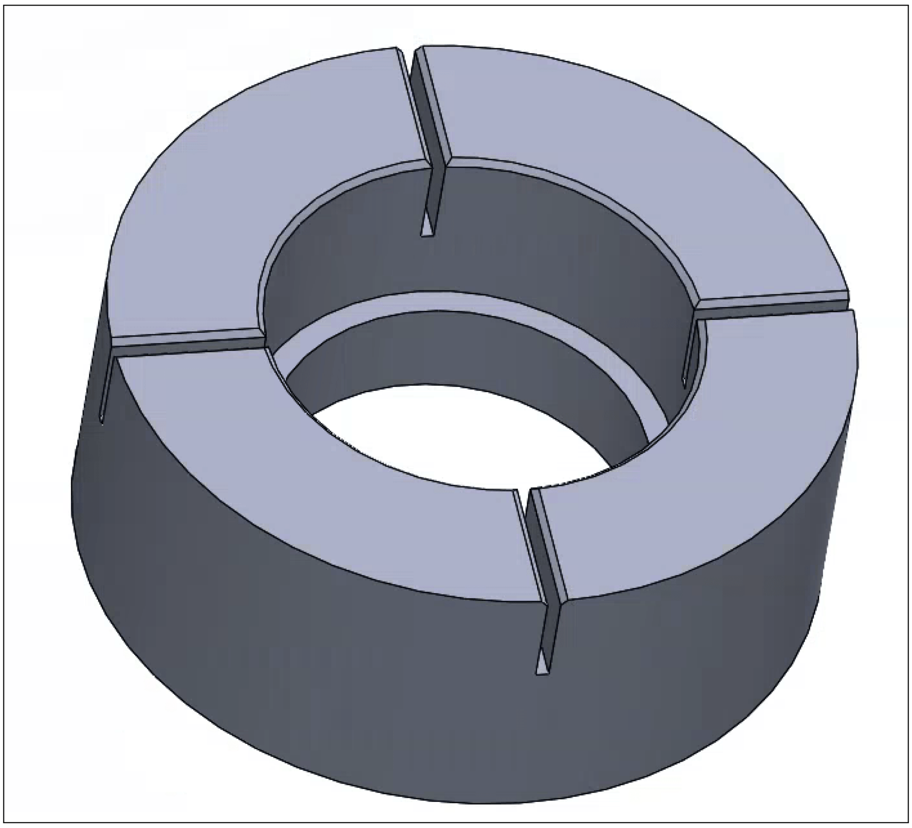

Figure 8: Fin Alignment Fixture

For shape, cutting a consistent trapezoid without a taper is reasonably simple to achieve. OSRT will use a series of non-marring clamps in a Computer Numerical Control (CNC) mill to secure the carbon fiber stock onto a piece of aluminum. After making the first cuts, which will create a datum plane, OSRT will locate that datum plane and make the angled cuts. This process should not take very long, as it only requires the initial setup and one fixture change to achieve the six cuts for each fin.

Deciding on fin material came down to two choices, mostly, which were carbon fiber and fiberglass. Compared to fiberglass, carbon fiber was the clear choice in all aspects except for ease of manufacturing. With these fins being a one-off production run, this criterion was weighted lower than the others. Another material option that was considered was wrapping carbon fiber around fiberglass, as was done in previous years. All things considered, carbon fiber will be the material of choice for OSRT, as seen below in Table 10.

Table 10: Fin Material DDM

Design		**Carbon Fiber**		**Fiberglass**		**Carbon Fiber Wrapped Fiberglass**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Strength	8	5	40	3	24	4	32
Manufacturability	6	2	12	3	18	1	6
Weight	7	5	35	3	21	4	28
Total		**87**		**63**		**66**	

3.2.3.4 Threaded Rods

There are three threaded rods that make up the main mounting system to connect the parachutes to the airframe; one in each section that separates on parachute deployment. All the rods are Grade 8 3/8 in. diameter threaded rods with have 16 threads per inch. Because the rods are responsible for retaining the components, including the parachutes, to the airframe on deployment, it is crucial to know just how much stress is being applied to the rods and whether or not they can withstand it. The maximum force happens when the main parachutes are deployed and has been calculated to be 1,350 lbf. At a 3/8 in. diameter, the cross-sectional tensile strength area is 0.0775 in^2. By using the stress equation, we can find the following stresses are applied to the threaded rod:

$$ExpectedMaximumStress = \frac{F}{A} = \frac{1350lbf}{0.0775in^2} = 17,500psi \tag{1}$$

$$SafetyFactor = \frac{TensileStrength}{ExpectedMaximumStress} = \frac{150,000psi}{17,500psi} = 8.57 \tag{2}$$

Based on these calculations, we can see that a Grade 8 $3/8$ in. threaded rod has a factor of safety at over 8.5, which is above the team requirement of five. This means that a threaded rod at with a $3/8$ in. diameter will be able to withstand the applied forces without adding too much unnecessary weight. The safety factor may appear to show that we can lower the diameter of the rod, however, due to the importance of this rod, we will not be using anything smaller.

3.2.3.5 Bulkheads

The launch vehicle will utilize several bulkheads, which will be made up of $1/2$ in. marine grade plywood. This plywood is lightweight, structurally sound, and easy to manufacture. The density of the marine grade plywood is 0.364 oz/in^3 , meaning each bulkhead will weigh approximately 1.053 lb. Previously, OSRT has experimented with more advanced bulkheads, such as honeycomb core sandwiched by carbon fiber. The results of these experiments have never produced bulkheads as reliable as marine grade plywood. Therefore, OSRT has decided the leading design option is the proven reliability of marine grade plywood, despite a small increase in overall weight. Additionally, OSRT has plenty of marine grade plywood from previous OSRT and OSRT usage, which reinforces the decision to use this material.

3.2.3.6 Motor Tube

The launch vehicle will utilize a fiberglass motor tube manufactured by Madcow Rocketry. In previous years, OSRT has had success with these motor tubes. Madcow Rocketry motor tubes are inexpensive, reliable, while maintaining strength. Other materials were considered, but OSRT decided that the weight benefits did not outweigh the cost difference.

3.2.3.7 Motor Retainer

At the end of the Motor Tube, OSRT will utilize an Anodized 6061 Aluminum Motor Retainer manufactured by AeroPack. The selected motor retainer has a rating up to, and including, L-class motors. These motor retainers have been used by OSRT in the past and have successfully held up to the forces applied to them. To ensure a secure retention of the motor, OSRT will utilize additional G5000 RocketPoxy on the grooves of the retainer. This will allow for many small fillets to be formed and guarantee the motor retainer is secured onto the fiberglass motor tube.

3.2.3.8 Centering Rings

The launch vehicle will utilize three centering rings to ensure the motor tube is parallel with the body tube. These centering rings will be made up of $1/2$ in. marine grade plywood. This plywood is lightweight, structurally sound, and easy to manufacture. See Section 3.2.3.5 for more explanation on material choice.

3.2.3.9 Fore Coupler

The fore coupler connects the nosecone to the fore airframe. This component will be 10 in. in total length; 8 in. in to the fore airframe and 2 in. into the nosecone. The inner diameter is 6.1070 in. and the wall thickness is 0.0715 in. The material selected for this component is fiberglass because the avionics bay in the nosecone needs RF transparency.

3.2.3.10 Canister

To make overall integration easier, the OSRT will implement a canister that will contain several components into one body. The canister also serves as the coupler for the aft section, extending 6.25 in. into the fore. The canister houses three main components: the parachutes, the ejection bay, and the 360° camera system. Packed parachutes rest on the ejection bay, which sits on top of the 360° camera system with a 3/8 in. threaded rod through the center. A rail is permanently mounted within the canister to position the components. The wall thickness is 0.0715 in., which makes the inner diameter 6.1070 in. It has a total length of 26 in. to allow enough space for the 360° camera system, the aft avionics and ejection bay, parachute packing, and 6.25 in. of acting coupler. and is made from fiberglass to provide avionics with an RF transparent material to send signals. Figure 9 shows the canister with the components that go within while Figure 10 displays the dimensions.

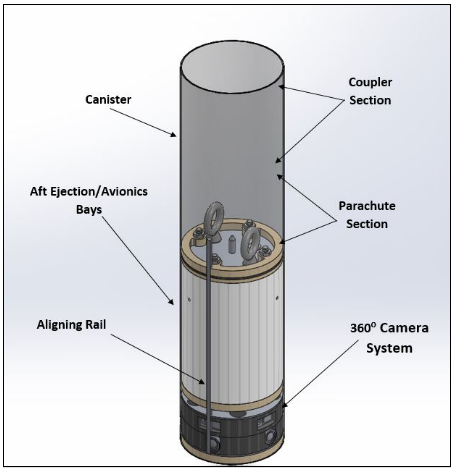

Figure 9: Aft Canister

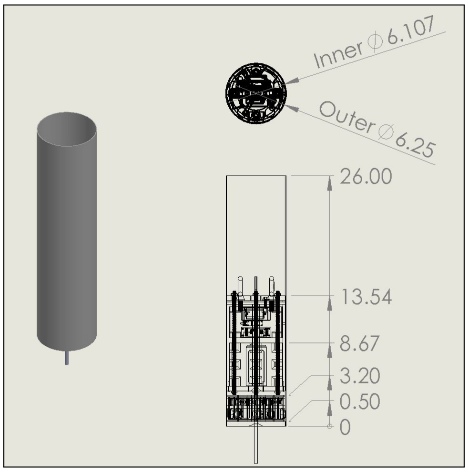

Figure 10: Aft Canister Drawing

3.2.4 Ejection and Avionics Bays

3.2.4.1 Fore Avionics Bay

The avionics are responsible for reporting the position of the launch vehicle, particularly after the parachutes have deployed and the airframe has safely landed. Avionics send a signal to the ground team so that the team can recover the airframe and deploy the rover. It is very important that the avionics can send the signal so, anything surrounding the avionics, needs to be RF transparent. Placing the avionics in the nosecone has two major advantages: It saves valuable space and it provides the avionics with an RF transparent casing. There are few other components that would fit as well in the nosecone and the fiberglass material that the nosecone is made from allows the avionics to send signals unimpeded.

An alternative design had the avionics and altimeters in the same body separated by a bulkhead and RF shielding. This design would mean it is nearly identical to the Aft Electronics Bay, which would mean simpler manufacturing. The avionics, along with the altimeters, would sit within the fore coupler. However, the surroundings still need to be RF transparent which means that another fiberglass section would need to be wound into the body of the airframe. However, locating the avionics within the nosecone provides benefits that outweigh the gain of similar manufacturing.

A 3D printed block that provides mounting points for electrical components. The block has a hole through the middle so that it can slide onto the $3/8$ in. threaded rod that goes through the nosecone. The block is also hollow when possible to conserve weight. There are two small holes in the nosecone bulkhead and in the bottom of the mounting block so that two screws can secure the avionics in position. Below the bulkhead are two eye bolts to provide a mounting point for the parachutes. These bolts are held in place with locknuts. Washers help to disperse the forces more evenly throughout the bulkhead. Figure 11 shows the 3D model of the fore avionics bay with transparent nosecone.

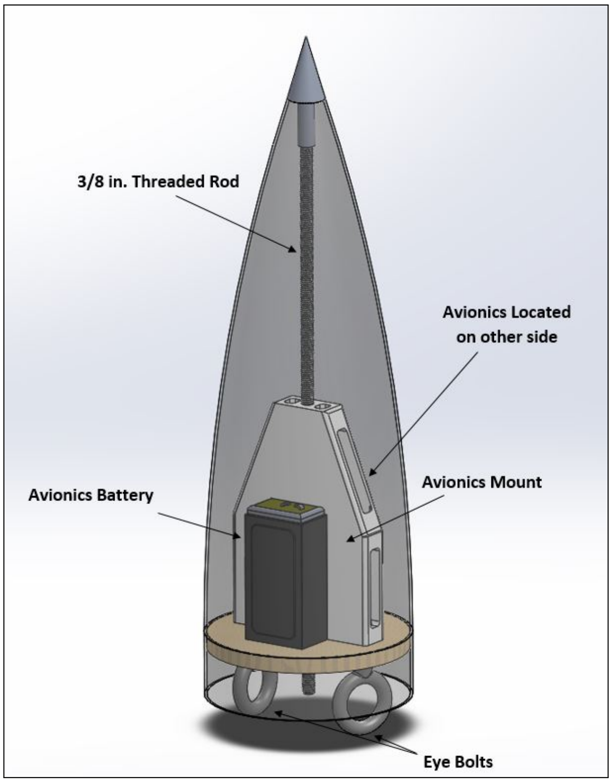

Figure 11: Fore Avionics Bay

Polylactic Acid (PLA), High-density polyethylene (HDPE), and Acrylonitrile Butadiene Styrene - A Common Thermoplastic Polymer (ABS) were considered as possible materials for the avionics mount. A scoring matrix was implemented to determine which material is best for the avionics mount. There are three main factors that were considered to choose which material best serves the needs of the avionics bay: weight, ease of manufacturing, and strength. Weight is the most important factor and must remain low. Table 11 shows the scores of the materials based on their fulfillment of the weighted factors.

Table 11: Fore Avionics Mount Material DDM

Material		**PLA**		**HDPE**		**ABS**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Weight	10	4	40	3	30	3	30
Ease of Manufacturing	5	4	20	4	20	3	15
Strength	3	3	9	3	9	3	9
Total		**69**		**59**		**54**	

From Table 11, PLA received the highest score. HDPE and ABS were very similar in all categories except for weight. ABS also fell short in ease of manufacturing. Both of these other materials would suffice as viable alternatives, but PLA is the leading design choice. With this taken into account, the weight of the fore avionics are 1.33 lbf including the battery. Figure 61 in the Appendix shows the critical dimensions of the for avionics.

3.2.4.2 Fore Ejection Bay

The major design considerations for the fore ejection bay are: difficulty arming the altimeters from outside the body, aligning the bay rotationally, faulty pressure seals, and safety concerns with arming the ejection charges. The switches to the altimeters are turned by Allen wrenches. If the switches were placed right against the mount within the bays, arming the altimeters from the exterior of the launch vehicle would be extremely difficult. By extending the switches so that they are just on the inside of the cylinder body, the altimeters are much easier to arm.

Once the bay is placed in the body, it will be difficult to rotate the bay due to the tight fit. That meant that the compartment needed to be perfectly aligned as it was slid into place. To fix this, a rail is mounted within the body that positions the bay. The bottom bulkhead has a small funnel-like shape to help the rail into position.

Above the bay, a pressure seal is needed to minimize the charges for ejection. There are several factors to consider when deciding the best design for the pressure seal: seal strength, removability, weight and manufacturability. Three designs were considered: a putty seal, a slightly smaller bulkhead with rubber gaskets filling the gap, and a bulkhead with four threaded rods that tighten down on a rubber ring. Table 12 shows how each design compared to others.

Table 12: Pressure Seal DDM

Material		**Threaded Rods**		**Gaskets**		**Putty**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Seal Strength	10	5	50	3	30	2	20
Removability	7	3	21	3	21	1	7
Weight	5	1	5	3	15	4	20
Manufacturability	3	2	6	3	9	4	12
Total		**82**		**75**		**59**	

Based on the DDM, the pressure seal with four threaded rods through the bay is the best way to seal the chamber. It provides the strongest seal, which is the main function of the seal. It is also removable, but it is heavy and will take some extra effort to manufacture. The gasket seal is a good alternative and, as a last resort, putty can be considered.

To increase the safety, a single pull, double throw shunting switch is attached to the upper bulkhead. This helps to eliminate the chance that the charges will fire early due to static build up lighting the e-match. The circuit to the e-match is grounded while the shunting switch is in the disarmed position, unlike a normal switch. Figure 12 shows the assembly of the fore ejection bay and Figure 13 shows the bay with a transparent body.

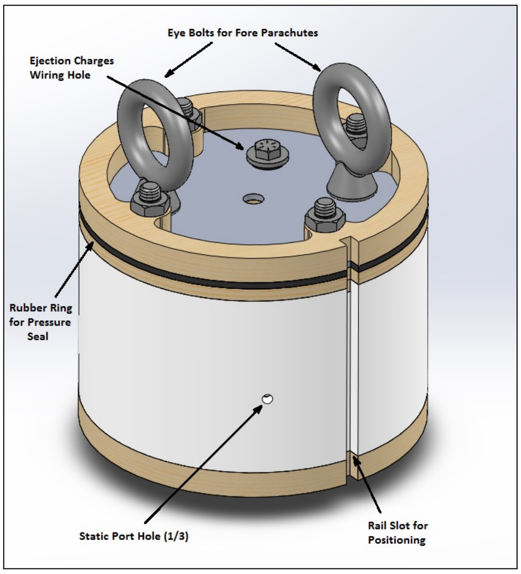

Figure 12: Fore Ejection Bay Assembly

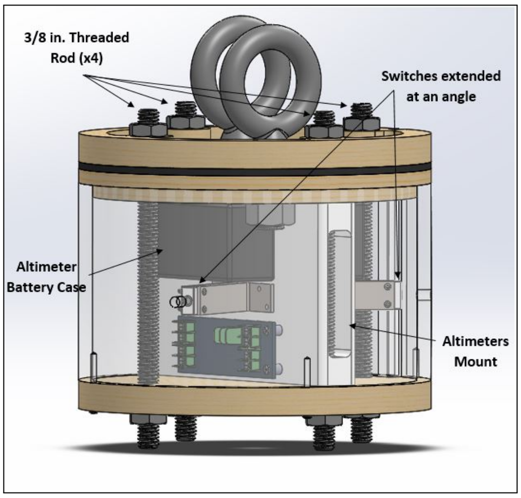

Figure 13: Fore Ejection Bay Transparent

Overall, this component weighs 1.51 lbf. A 3/8 in. threaded rod is centered through the bay and anchors it using a locknut. The locknut sits on top of a half in. wooden bulkhead and a custom aluminum washer that helps to disperse the forces produced by the deployment of the parachutes. Two eye bolts are mounted on the ejection bay to provide attachment points for the parachutes. The walls of the bay are 1/4 in. thick, are 3D printed, and made at an outer diameter of 6.25 in. There are three, 3/16 in. holes placed at 120° apart from each other. These holes serve several purposes: To allow access to the switches from the exterior of the launch vehicle, to eliminate unintentional pressure changes present when only two holes at 180° are involved, and as static port holes for the altimeters. The altimeter switches are mounted on 3D printed supports that are set at angles to align with the static port holes. These supports are connected to the main mounting point, which is a 3D printed block that provides a body to attach electronics and other small components. The block is hollowed out where possible to minimize the weight. Battery cases with screw closed covers are mounted on either side of the block. The bottom bulkhead is also made of half in. plywood. Four threaded rods go through the bay to connect and secure the removable seal and to hold the bottom bulkhead in place. All the inside wall are coated in conductive spray paint to make the compartment RF shielded. Figure 14 shows the Fore Ejection Bay integrated with the fore body. The bay slides in along a rail that is mounted to the inner wall of the airframe.

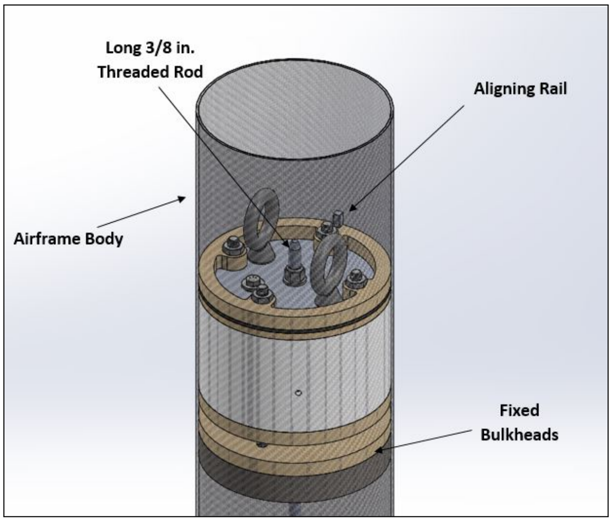

Figure 14: Fore Ejection Bay Integration

A scoring matrix helped to determine the best materials choice for the bay. The main point of concern for the bays are the walls of the body. There are several factors to be considered here when making a material choice: strength, weight, and manufacturing ease. Each of these factors is given a weight and the materials are scored according to how well they fulfill the requirements. The possible materials were aluminum, HDPE, and PLA. Table 13 shows the scoring matrix with the corresponding scores for each material.

Table 13: Fore Ejection Bay Material DDM

Material		**PLA**		**HDPE**		**Aluminum**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Strength	7	3	21	3	21	4	28
Ease of Manufacturing	5	3	15	3	15	1	5
Total		**76**		**66**		**53**	

PLA received the highest total score and will be the first choice for the body wall material. The lower weight of the PLA makes it a more desirable material. While the strength of aluminum is the highest, its extra weight and more difficult manufacturing causes the material to receive a relatively low score. The dimensions of the Fore Ejection Bay are shown in Figure 63 in the appendix.

3.2.4.3 Aft Ejection and Avionics Bays

The aft ejection bay combines the avionics and the altimeters bays into one component separated by a bulkhead coated in RF shielding spray paint. The overall design is the same as the Fore Ejection Bay with extra length for the avionics. The outer diameter is slightly smaller due to the whole bay being within the canister design. The bulkhead at the bottom of the altimeters bay must screw into the mounting block so that it can separate the avionics and the altimeters, creating a quality RF shield. The added length is five in. and is capped by a bulkhead just like the bottom bulkhead in the Fore Ejection Bay. Along the side of the bay is a key way. When integrated into the canister, the rail will slide within the slot and align the bay to the correct orientation. Figure 15 shows the Aft Ejection Bay assembly and Figure 16 shows how the altimeters are avionics are kept separate.

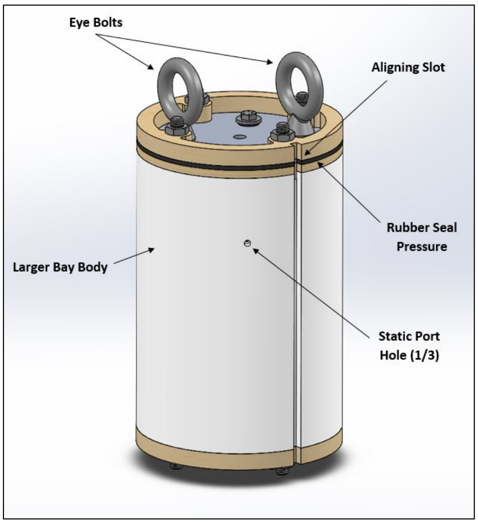

Figure 15: Aft Ejection Bay Assembly

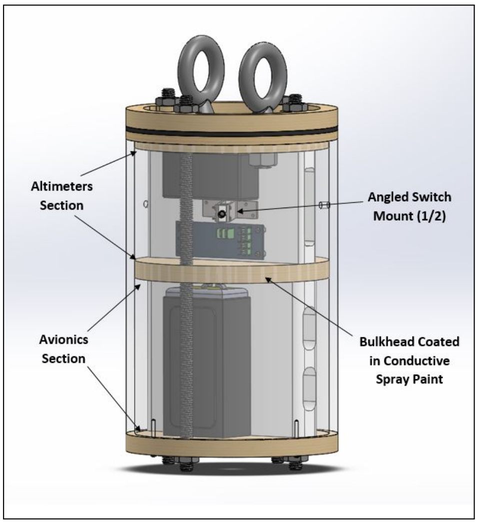

Figure 16: Aft Ejection Bay Transparent

As seen in Figure 16, the avionics and altimeters are completely separated wooden bulkhead with a conductive spray paint applied to the bulkhead. This serves to protect the altimeters from any signals given off by the avionics. The overall dimensions of the component are given in Figure 64 in the appendix. The bay weighs 2.21 lbf.

Just as in the Fore Ejection Bay, the walls are made of PLA at 1/4 in. thick. The requirements are all the same and PLA fulfills them well. However, to determine if conductive spray paint is truly the best material to use, a scoring matrix was applied to several other candidates; an aluminum tube and plates, and aluminum foil. The materials will be scored on three criteria: electrical conductivity, weight, and ease of integration. Table 14 displays the scoring breakdown along with the weights of each criteria.

Table 14: RF Shielding Material DDM

Material		Conductive Spray Paint		Aluminum Foil		Aluminum	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Electrical Conductivity	10	3	30	3	30	4	40
Weight	8	3	24	3	24	1	8
Ease of Integration	6	5	30	3	18	2	12
Total		**84**		**72**		**60**	

Table 14 confirms that a conductive spray paint is the best option. While it may not be as good against blocking extremely powerful signals, the barrier that it provides is plenty strong. Additionally, it is by far the easiest to integrate into the bays and is lightweight. If spray paint does prove that it is not a viable option, aluminum foil will be a strong alternative candidate.

3.2.5 Camera Bay

To increase social media presence, the OSRT will improve the filming capability of each launch. A 360° camera system will be integrated into aft section of the air frame. This will allow OSRT to create interactive 360° panorama video which can be posted online and on all social media. This system is composed of five GoPro cameras evenly angled to capture 360°, enclosed with two 3D printed discs to hold the cameras. Thin metal plates will be secured on the top and bottom of this module to provide durability. A threaded rod running through the middle of the module will hold all parts together. The system will be placed bellow the avionics bay in the canister for ease of assembly. Holes will be cut through both the canister and air frame to access the camera controls. To capture 360° video, the cameras will film in unison and then the film will be stitched together using software. To start and stop the cameras at the same time, shutter cables will be attached to the exterior ports of each camera. All five shutter cables will be attached to a manual trigger cable which will be accessible from the outside of the air frame. Using the trigger, all five cameras can be controlled in unison. An exploded view of he system can be seen below in Figure 17.

The materials used were selected to make the system lightweight. GoPro cameras themselves are very durable and do not need much extra protection. The metal plates use a lightweight aluminum alloy and the 3D printed body uses ABS because of its low density. The whole system will weigh approximately 1.75 lbf.

Multiple alternatives were considered when designing this system. The two main design concerns were the type of cameras to use and how to encase the body of the system. The decision matrix used to make final design choices is shown below in Table 15 and Table 16. Weight was the largest contributing factor when making these design choices. Using the decision matrix shown below, the final design was made. The most interesting design choice was the decision not to use two 360° GoPros instead of five standard GoPros. Two

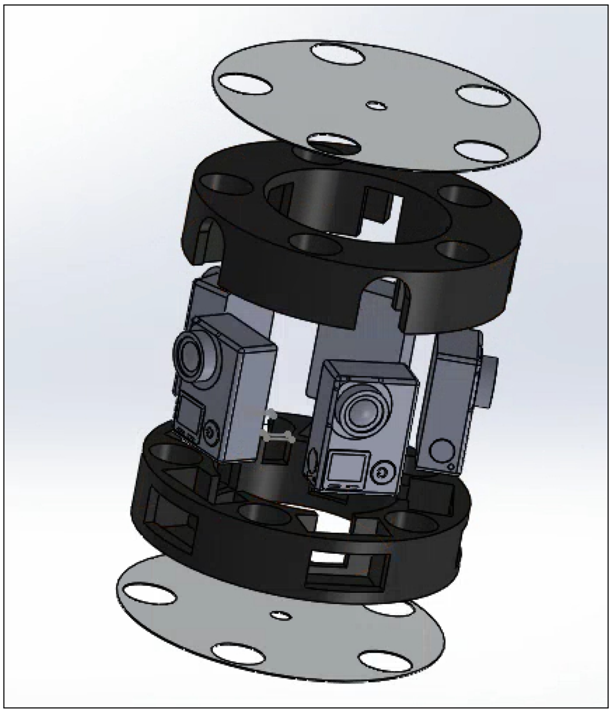

Figure 17: 360 Camera Exploded View

cameras will be lighter than five cameras but 360° cameras have a very spherical lens to allow for 360° filming. For these cameras to work, the lens would have to protrude out of the air frame. It was decided that this would compromise the aerodynamics of the launch vehicle too much and the decision to use five standard GoPros was made.

Table 15: 360° Camera DDM

Design		**Five GoPros**		**Two 360 Degree GoPros**		**Other brands**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Weight	5	3	15	4	20	2	10
Aerodynamics	4	3	12	2	8	3	12
Cost	3	3	9	2	6	4	12
Total		**36**		**34**		**34**	

Table 16: Camera Bay Material Decision Matrix

Design		**3D printed body, aluminum plates on ends**		**3D printed body, aluminum tube encasing whole system**		**All 3D printed parts**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Weight	5	3	15	1	5	4	20
Ease of Assembly	3	3	9	2	6	2	6
Durability	2	3	6	4	8	1	2
Total		**30**		**19**		**28**	

3.2.6 Avionics

Tracking of the launch vehicle while in flight is critical for safety, recovery, and launch data analysis. OSRT will be using an independent electronic system running code created by the OSRT software team. Ultimately, the goal of the OSRT software sub-team is to create a robust and feature-packed system tailored to our specific usage requirements. The launch vehicle telemetry and avionics system will primarily utilize a standalone Global Positioning System (GPS) unit and RF transceiver to transmit telemetry data to a remote ground station for the duration of the flight. These technologies were chosen due to their maturity, performance, and reliability at low cost, making them perfect for inclusion in a standalone avionics system under high duress. We will primarily be using GPS (as opposed to Global Navigation Satellite System (GLONASS) or Doppler Orbitography and Radiopositioning Integrated by Satellite (DORIS)) due to its prevalence and ease of implementation. GPS technology compares coordinate satellite data relative to orbits, after which the time data is transmitted to a ground station that uses the data to triangulate the position of the active device.

Table 17: Avionics System DDM

Design		**Existing OSRT developed Avionics Telemetry Unit (ATU)**		**Feature modification and updates to OSRT developed ATU**		**Consumer Avionics Kit**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Ease of Development	3	3	9	3	9	3	9
Integration with Current Systems	4	3	12	4	16	3	12
Feature Inclusion	4	3	12	5	20	3	12
Expenditure	2	3	6	4	8	3	6
Total		**39**		**53**		**49**	

Additionally, we will be implementing a data logger and control system to interface between the avionics subsystem and the BEAVS. If the team cannot validate the OSRT developed ATU through ground testing, followed by flight testing, a consumer avionics kit will be purchased. OSRT will evaluate consumer avionics kits more in depth if the need arises, however an Altus Metrum TeleMega is the leading choice for consumer avionics kits. This is the leading choice due to other team's experience at Oregon State University (OSU) with other rocketry teams.

3.2.6.1 GPS Components

The GPS unit provides all of the position data generation, allowing for launch vehicle tracking and real-time recovery. Critical design considerations include Rx sensitivity, power consumption, time to first fix, and position update frequency. High Rx sensitivity ensures that signals can be locked quickly despite environments with high signal attenuation and noise. Power consumption directly impacts the choice of delivery system, and lower power consumption significantly decreases the risk of in-flight failures due to power issues. Time to first fix defines the time required for a GPS system to acquire a fairly accurate signal lock, thus a faster time to first fix cuts down on preparation time during launch and lowers overall operating time and power consumption. A low time to first fix also ensures that a lost signal will be relocated quickly.

Position update frequency determines how accurate the data is and how useful post-launch analysis will be. A higher update frequency will provide more data points to run through scripts and other such analysis tools. Position update frequency is also critical to providing robust data due to the launch vehicle's high velocity during flight. Higher data refreshing resolutions ensure more accurate data collection and more information in the flight data, as well as an increase in the probability of correct positioning coordinate transmission over a fixed period of time. Low update frequencies run the risk of transmitting erroneous data or losing packets during transmission.

The secondary component within the GPS block is the antenna. The distance involved in GPS transmission renders the signal seen by commercial receivers very weak, (going as far down as -160 dBm). In response to this, OSRT has acquired a high gain active antenna for use in the GPS system that is capable of 30 dB of gain on the RF signal prior to base-band down conversion. This decreases data loss and improves the time to first fix and in-flight signal retention.

3.2.6.2 RF Transceivers

The in-flight avionics system will utilize two transceivers operating on 900 MHz and 433 MHz bands at maximum powers of 210 mW and 40 mW respectively. The inclusion of a second band was decided upon after experiencing electromagnetic interference in previous iterations of the system. The 900 MHz would work better if running at 250 mW, but competition rules stipulate that there is a 250 mW power cap over all frequencies.

Table 18: Transmission Frequency DDM

Frequency		**900 MHz**		**900 MHz + 433 MHz**		**2.4 GHz**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Ease of Use	3	3	9	2	6	3	9
Effective Distance	5	3	15	4	20	1	5
Ease of Development	4	3	12	3	12	3	12
Jamming Resistance	3	3	9	4	12	3	9
Total		**45**		**50**		**35**	

The 900 MHz transmission frequency is easy to use and unlicensed. The technology is well developed and robust. By contrast, 433 MHz is more difficult to use and requires a HAM radio license. Its infrequent use gives us transmission redundancy that ensures system robustness. 2.4 GHz transmission was also considered, but it has an incredibly limited range of only 105 feet (900 MHz, by contrast, provides a 60 mile line of sight distance at 1 watt on high gain antennae).

Design of the base station antenna was determined by the gain and antenna position relative to the launch vehicle. These constraints, as well as size, were the primary design concerns for the launch vehicle antenna as well; the orientation of the launch vehicle relative to the base station is indeterminate during launch, thus a wide radiation pattern must be broadcast to allow for ubiquitous signal transmission.

The avionics subsystem will be mounted in both the nosecone and aft airframe section in order to track both independent sections of the launch vehicle on descent. RF transparency of the launch vehicle is a requirement for both avionics subsystems, the launch vehicle design has been configured to account for these units. Additionally, steps have been taken to shield altimeters from external RF signals to ensure more accurate and reliable signal operation.

3.2.6.3 Power System

Design considerations for the power supply system include reliable supply voltage, battery cell durability, max current output, energy density, and size constraints for the avionics bay. Lithium Polymer (LiPo) cells in a prismatic pack were chosen due to high energy density and reduced volatility when compared to non-prismatic pack cells. Alkaline batteries are not viable due to low energy density and high weight; the possibility of running out of power during setup or in-flight is too great, proving LiPo a preferable alternative.

Table 19: Battery Options DDM

Battery Choice		**Alkaline**		**Prismatic LiPo**		**Standard LiPo**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Weight	5	3	15	4	20	4	20
Energy Density	4	3	12	5	20	4	16
Volatility	4	3	12	4	16	2	8
Total		**39**		**56**		**44**	

The chosen batteries are 7.4 V with the GPS receiver and the RF transceivers requiring a 3.3 V supply. The battery supply voltage will be regulated down to 3.3 V to ensure consistently low noise, thus preventing RF performance reductions or failures. The micro-controller signal processor requires a 5 V Direct Current (DC) supply, but noise is negligible (and thus regulation is easier).

3.2.6.4 Software and Control

The ATU will utilize a Teensy 3.6 microcontroller with an ARM Cortex-M4 processor. This device provides enough computational power to log data, parse I/O, and communicate via serial interface. It is faster, wastes less computational and memory resources, and fits the mission profile more effectively than the alternatives. An on board microSD card slot allows for data logging as a redundant backup to transmission, and saves extra real-time data for analysis after launch vehicle retrieval.

Table 20: Avionics Microcontroller DDM

Microcontroller Choice		**Arduino UNO**		**Teensy 3.6**		**Raspberry Pi**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Speed & Efficiency	5	3	15	5	25	2	10
Computational Power	4	3	12	3	12	4	16
Ease of Systems Integration	4	3	12	4	16	3	12
Total		**39**		**53**		**38**	

A majority of the program code will be written in C/C++ (more specifically, Arduino), with the ground station user interface being written in Python. These languages have been chosen due to team familiarity, solid Application Programming Interface (API) support for the devices that OSRT has settled on, and ease of use with the Teensy micro-controller. Analysis scripts will most likely be written in either Python or MATLAB due to those languages' strengths in data parsing and extrapolation.

3.2.7 Blade Extending Apogee Variance System

Several different methods to accurately reach the target altitude were analyzed by OSRT. The main method used to control apogee altitude is through motor selection. Motors can be quickly simulated using OpenRocket, and OSRT must simulate motor performance for flight. However, there is a limited number of motors which are manufactured, therefore accuracy is limited based on motor characteristics determined by the manufacturer. Additionally, if any weights in the launch vehicle differ from those used in the simulation, the apogee altitude will be significantly off.

In 2018, OSRT used motor selection exclusively to control apogee altitude. The results of motor selection to control apogee altitude were not ideal for OSRT because the weight of components of the launch vehicle increased during manufacturing. At Flight Readiness Review (FRR) in the 2018 competition, OSRT simulated the launch vehicle to fly 405 ft below the target altitude due to the difficulties of controlling apogee altitude through motor selection alone. Additionally, simulations predicted the apogee altitude to decrease by up to another 226 ft based on wind conditions. A suspected motor defect from the manufacturer caused the 2018 OSRT competition launch to be even further below target altitude than expected. Based on the poor performance through motor selection alone, OSRT wanted to explore options of additional methods to increase accuracy of achieving the declared target altitude of 4,500 ft.

One method OSRT considered to control apogee altitude beyond motor selection is to use ballast to increase the mass of the launch vehicle. When using ballast, the flight is simulated based on conditions at the launch pad and ballast is adjusted accordingly. This has a benefit of being easy to manufacture and implement. However, this method relies on the accuracy of the simulation and the conditions which are input. Inconsistent wind speed could still affect the apogee altitude. Additionally, OSRT is limited to a maximum ballast weight of 10% of the fully loaded launch vehicle.

The other method which OSRT considered to control apogee altitude is to control the drag profile of the launch vehicle based on feedback from live data during flight. This reduces the launch vehicle's susceptibility to be affected by changes in weather conditions or motor performance between flights. An active control system could be combined with a ballast system, which is the method OSRT chose to consider. This method has a drawback of increased complexity, and difficulty of manufacture.

To evaluate each of these outlined methods, the DDM shown in Table 21 was used.

Based on the results of the DDM, OSRT is pursuing motor selection, ballast, and active control of the drag profile of the launch vehicle to control apogee altitude. The motor selected will be capable of carrying the launch vehicle above target altitude, ballast will be used to reduce apogee altitude based on launch day conditions, and control of the drag profile will be used to achieve increased precision. The overall system which will be used to achieve a target altitude is called the Blade Extending Apogee Variance System (BEAVS). The BEAVS is composed of a mechanical system, an electrical system, and a control system.

Table 21: Target Altitude DDM

Design		**Motor selection**		**Motor selection and ballast**		**Motor selection, ballast, active control of drag profile**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Mission performance	5	3	15	4	20	5	25
Reliability	5	3	15	4	20	5	25
Ease of Development	3	3	9	2	6	1	3
Weight	2	3	6	3	6	2	4
Total		**45**		**52**		**57**	

3.2.7.1 Mechanical System

The mechanical system of the BEAVS is broken down into two different subsystems to control apogee altitude: a passive and an active system.

The passive BEAVS is a coupled pair of ballast bays. These ballast bays are separated into the fore and aft sections, one below and one above the center of gravity. The mass contained within these ballast bays can be easily adjusted which will result in a consistent center of gravity location. A consistent center of gravity location is beneficial because it will allow for the launch vehicle to maintain the same stability while changing the amount of ballast. The active system will adjust the drag profile of the launch vehicle during flight to achieve the desired apogee altitude. Ballast weights are shown in both fore and aft sections as shown in Table 22. Ballast calculations between fore and aft were determined based on the ratio of distances of the ballast bays to the center of gravity of the launch vehicle. The distance to the aft ballast bay is 48.3% of the distance to the fore ballast bay, which means the mass in the fore ballast bay should be 48.3% of the mass in the aft ballast bay to maintain stability. Simulations were performed in OpenRocket, with results displayed in Table 22.

Table 22: Simulated launches with ballast.

Wind Speed (mph)	**Aft Ballast (lbf)**	**Fore Ballast (lbf)**	**Stability (calibers)**	**Predicted Apogee (ft)**
0	1.45	0.70	2.10	4500
5	1.42	0.68	2.10	4500
10	1.32	0.64	2.10	4500
15	1.17	0.57	2.10	4500
20	0.99	0.47	2.10	4500

The active system design had numerous design options considered. The first option considered was to have flaps which rest on the outside of the airframe that can be actuated to be perpendicular to the airframe.

Because the flaps can be manufactured to any size, this option represents the largest potential increase in cross sectional area. However, the flaps being actuated in this manner has constant force applied to any device used to actuate the flaps.

Alternatively, blades could be extended from the interior of the airframe which increase the cross sectional area. While the increase in cross sectional area is limited by the diameter of the airframe, a significant increase in cross sectional area can still be obtained. Shown in Figure 18 and Figure 19 are the three design options which were considered for blades extending from the interior of the airframe.

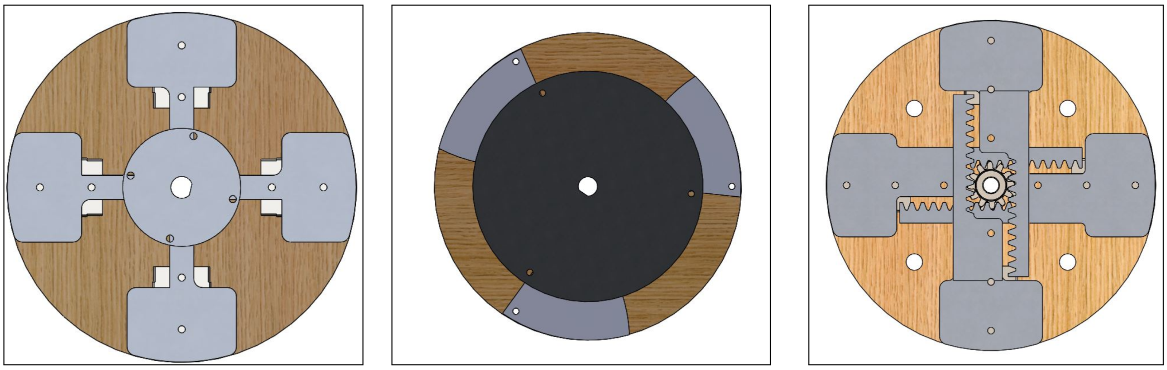

Figure 18: A visualization of three options considered for active control of drag profile in their closed positions.

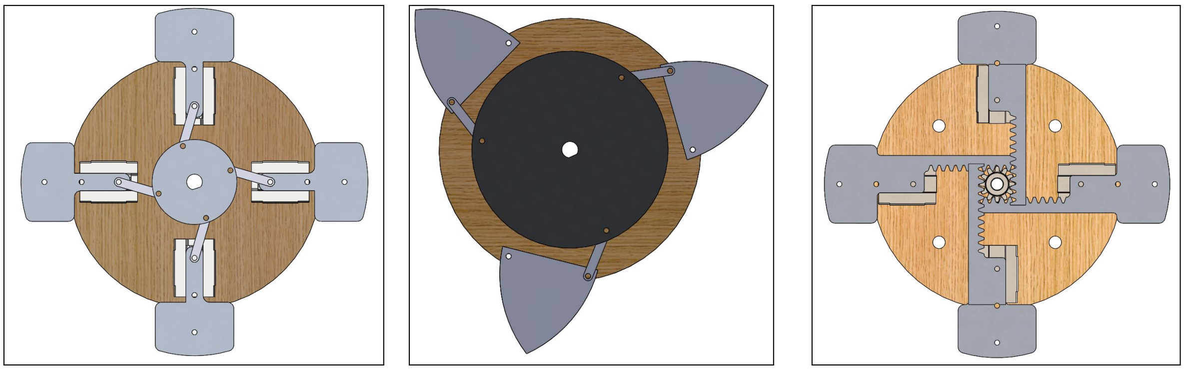

Figure 19: A visualization of three options considered for active control of drag profile in their opened positions.

All four of the potential active system designs that were described were compared using a DDM, shown in Table 23.

Table 23: Active control of drag profile DDM.

Design		**Flaps actuated from airframe**		**Blades rotate out of airframe with linkage**		**Blades extending out of airframe with linkage**		**Blades extend out of airframe with rack and pinion**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Actuation Reliability	5	3	15	3	15	3	15	3	15
Actuation Force Required	4	3	12	5	20	5	20	5	20
Increase in cross sectional area	4	3	12	1	4	2	8	2	8
Number of Components	2	3	6	4	8	3	6	4	8
Ease of Manufacture	2	3	6	4	8	4	8	4	8
Weight	2	3	6	3	6	3	6	3	6
Total		57		61		63		**65**	

Based on the results of the DDM, the selected design uses a rack and pinion system to extend blades from the interior of the airframe. This results in a 32% increase in cross sectional area with the blades fully deployed.

The BEAVS will extend 4 blades through slots cut into the airframe. The blades will be manufactured from 1/8 in. aluminum plate using a CNC mill. The blades will attach to a linear bearing on a 7 millimeter guide rail with three M2 fasteners. The blades will have a set of teeth which create a rack and pinion system with a central drive gear. The central drive gear will operate all four fins simultaneously. The linear bearings will be mounted to a removable bulkhead made of 1/2 in. aerospace grade plywood. The bulkhead will be just fore of the motor casing, and will provide the structure for how the system is retained within the launch vehicle. Four 8-32 threaded rods will extend through the bulkhead towards the fore section of the launch vehicle. These four threaded rods will tie into a bulkhead on the fore side of BEAVS with nylon lock nuts, which will adapt to a 3/8 threaded rod to attach the remainder of components in the aft section. A National Electrical Manufacturers Association (NEMA) 23 stepper motor will be directly attached to the central drive gear. Fore of the NEMA 23 stepper motor is the aft ballast weights of the passive system of the BEAVS. The fore ballast bay is located on the fore section hard point.

The blades were designed to account for a 32.0% increase cross sectional area of the test launch vehicle. Due to the self optimizing control scheme, the 32.0% increase in the cross sectional area was used to approximate a drag coefficient of the launch vehicle by acting as a direct multiplier of the drag coefficient before deployment. The test launch vehicle being flown has a drag coefficient of approximately 0.51 according to OpenRocket simulations, therefore when the blades are fully deployed it is estimated that the drag coefficient is 0.67. This is expected to be a conservative estimate, due to fluid flow separation effects when the blades are extended from the airframe. OSRT is currently developing a Computational Fluid Dynamics (CFD) simulation in STAR-CCM+ which will provide a more accurate result for the drag coefficient with and without blade deployment. Due to the self optimizing nature of the BEAVS control system, there is allowable error within the drag coefficient to still achieve desired performance of the system. Figure 20 and Figure 21 depicts the completed design for the system.

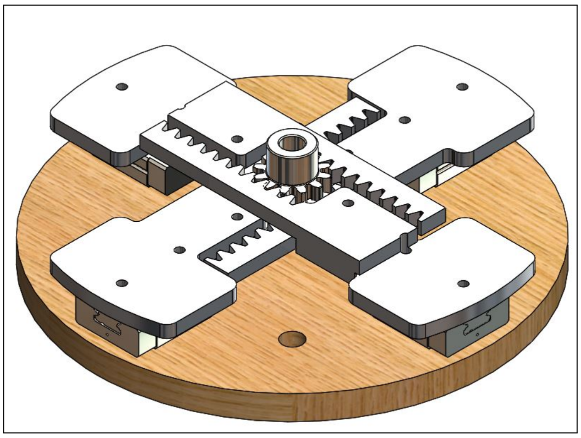

Figure 20: Selected active drag profile design

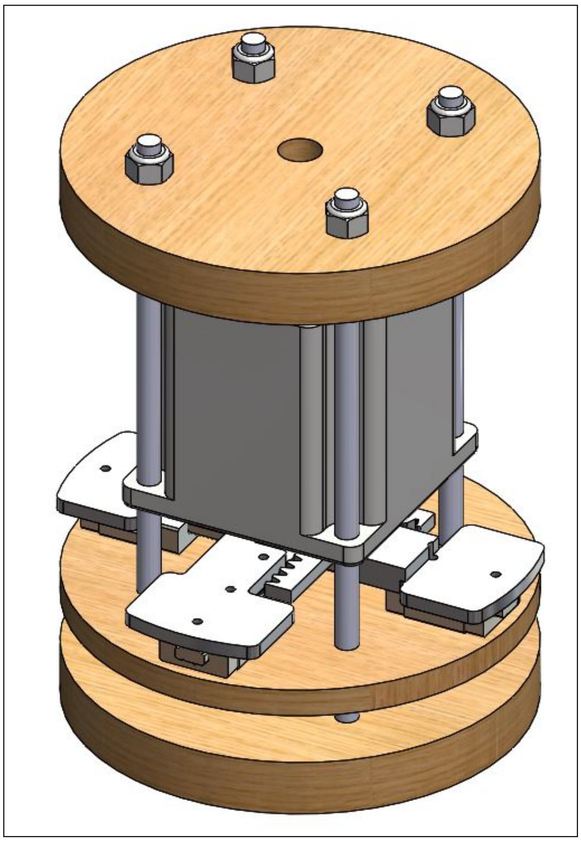

Figure 21: Full assembly that is placed in the launch vehicle

The BEAVS will be placed as the furthest aft component excluding the motor. This location will be aft of the center of pressure. When the blades deploy, the center of pressure will be lowered, increasing stability. The increase in stability assures that in the event of system failure the launch vehicle will not pose a safety risk.

3.2.7.2 Electrical System

The BEAVS will consist of a Printed Circuit Board (PCB) which is designed by OSRT and consists of a 9DOF accelerometer, an IMU, a barometric pressure sensor, a motor driver, a rotary encoder, a servo motor, and a microcontroller. The software for the BEAVS will be written in C/C++ to control the sensor data acquisition and filter the data. The data will be filtered using a Kalman filter to reduce the effects of noise in the measurements. The filtered data will then be used to predict apogee altitude based on current flight characteristics. Based off of the predicted apogee altitude, the PCB will send a signal to a motor driver which will be used to control the exact position of the blades during flight.

The majority of components for the electrical system were selected due to availability to OSRT. There were no known issues with the electrical hardware used for the avionics system used in 2018. Therefore, OSRT has decided to proceed with these sensors to accelerate preliminary design and testing schedule. Therefore, incorporating these sensors into the BEAVS allows for more rapid development and testing of the mechanical system and control system.

The electrical components which are available to use on the BEAVS to OSRT from 2018 include:

- Microcontroller - Teensy 3.6
- Barometric Pressure Sensor - MPL3115A2
- 9DOF IMU - BNO055
- Accelerometer - ADXL377

Because these components have already been purchased, they will be used for initial testing. The data from each flight will be carefully reviewed to determine if any of these components is performing below expectations.

Additional components for the electrical system are required for the electrical system of the BEAVS. The motor driver, motor, rotary encoder, these components have been elected to be:

- Motor - NEMA 23 capable of 297 oz-in holding torque
- Motor Driver - TB6600 Stepper Motor Driver
- Rotary Encoder - Bourns 3362P-1-103LF 10 kΩ Potentiometer

The rotary encoder was selected to be a rotating potentiometer as opposed to a standard rotary encoder. The function of the rotary encoder is not to determine position of the blades for this system, the stepper motor is capable of providing reliable position accuracy. The rotating potentiometer will be mounted to the motor shaft of the BEAVS and measured voltage data will be logged. As the motor rotates, voltage will change based off of the resistance of the potentiometer. This can be used to determine mechanical movement of the central drive gear during flight. If for some reason the stepper motor is overloaded and begins skipping steps, it will be noticeable in the post flight data analysis. Other solutions to this problem are possible, but based on ease of implementation and accuracy required, this component was chosen.

3.2.7.3 Control System

The BEAVS will have a control scheme which is reliant on a varying set point to provide in-flight optimization. The control system has been reduced to a one degree of freedom control system, where the control variable is apogee altitude. The control system has only two possible states: blades retracted or blades deployed (note: this assumes blades deploy quickly enough to be negligible). With the blades deployed, the drag coefficient of the launch vehicle is increased, which lowers the expected apogee altitude. With blades retracted, the expected apogee altitude does not change. Therefore, the only output of the control system is to lower the expected apogee altitude. This presents a problem that if the control scheme overshoots the desired apogee altitude, there is no way to recover from that error to reach the desired apogee altitude. OSRT broke this problem down into two different ways which the problem could be solved.

Several options for the BEAVS control system have been evaluated. These options include: a basic Proportional-Integral-Derivative (PID) control loop with a long duty cycle, a varying set point PID control loop with a long duty cycle, and a linear multivariate control system. The primary risks identified with BEAVS are the development time and testing required to have a reliable system.

The most traditional control method which OSRT was familiar with, designing a PID control loop and optimizing the parameters through testing in order to find PID coefficients such that the system is critically damped. This method presented a large flaw: the best way to test these parameters is to fly the launch vehicle and analyze the results of the test numerous times. This is expensive per test, and may require more test flights than OSRT can reasonably conduct before the competition.

The next option which OSRT looked at was to use a varying set point control scheme. In this control scheme, the desired apogee altitude value after blade deployment changes during flight. The set point is to be reduced through a number of steps that gradually approach the overall desired apogee altitude value. After each blade deployment cycle, the PID parameters are updated based on the error between the set point and the expected apogee altitude. This method drastically reduces the number of tests required to dial in the PID parameters of the control system.

The final option which OSRT considered is a multivariate control system. This is a multiple input, single output control scheme which OSRT feels is more adequately capable of accurately achieving the desired apogee altitude.

Table 24 displays the evaluation of the different control schemes considered.

Table 24: BEAVS Control System DDM

Design		Traditional PID Loop		Varying set point PID Loop		Multivariate Control System	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Accuracy	2	3	6	4	8	5	10
Tuning Required	2	3	6	5	10	5	10
Ease of development	5	3	15	2	10	1	5
Total		**27**		**28**		**25**	

The selected design is a varying set point PID loop. This will reduce testing, increase accuracy, and OSRT believes it is reasonable to develop the system within the available timeline. The self optimizing control scheme will utilize only proportional control during initial testing of the BEAVS for simplicity. If it is determined to be necessary to achieve desired accuracy, integral and derivative control can be added in at a later date.

The proportional control will be achieved through duty cycle adjustment. A duty cycle of 1 second will be utilized, with the blade deployment time required to reach the set point being a percentage of the duty cycle. The percentage of the duty cycle will be multiplied by the proportional control parameter to determine actual blade deployment time. Based on the 2017-2018 OSRT flight data, BEAVS should have at least 10 duty cycles to be completed during flight.

3.2.7.4 Testing

Despite the BEAVS being a self optimizing system during flight, OSRT will validate the concept of BEAVS through a series of tests. The initial testing phase will be the most rigorous. If the BEAVS is unable to pass the testing phase with the desired accuracy from the active system before competition, the BEAVS will be simplified to only include the passive system for mission assurance. The potential simplification to only the passive system will be performed by manufacturing but not arming the system on launch day. This allows for the testing phase on the BEAVS test airframe to extend into the launch vehicle manufacturing phase if required. At least one succesful launch with the active system armed will be performed by OSRT before FRR for the system to be armed at the competition launch.

The testing will take place on a Mad Dog DD fiberglass airframe which has 4 in. body diameter. This airframe was selected due to its availability to the OSRT: it was used as the 2017-2018 OSRT subscale launch vehicle. Some modifications to the airframe are being made to prepare it for testing of the BEAVS. It will, however, allow OSRT to begin testing the system as early as possible. OSRT plans to test the BEAVS on at least 3 flights in the Mad Dog DD launch vehicle before the active system is integrated into the full scale launch vehicle. Once integrated into the full scale launch vehicle, the BEAVS will undergo additional testing to ensure it maintains performance in the new airframe. The rigorous and continuous testing of the BEAVS will allow OSRT to provide an additional level of mission assurance by proving reliability of the active portion of the BEAVS.

In addition to the testing of the active portion of the BEAVS, the passive portion of the BEAVS must be tested in all potential ballasted conditions prior to launch at competition based on USLI rules. The BEAVS will be tested with at minimum 3 ballast configurations, sized for wind conditions of 0 mph, 10 mph, and 20 mph. If feasible, OSRT hopes to test ballast conditions for 0 mph, 5 mph, 10 mph, 15 mph, and 20 mph.

3.2.8 Motor alternatives

The planned motor is the Cesaroni L2375-WT L-class motor. The primary reason behind this motor selection is the projected altitude of 4797 ft. simulated by OpenRocket. The projected altitude is desired to be above the target altitude of 4,500 ft, allowing future design choices ability to increase slightly in mass if required. The BEAVS with an amount of ballast within the Student Launch criteria can be added at launch to further refine apogee altitude to 4,500 ft. The L2375-WT current motor selection has a thrust to weight ratio of over 10 for the current launch vehicle. A high thrust to weight ratio will help prevent the launch vehicle from being impacted by launch day conditions. The L2375-WT manufacturer specifications can be seen in Table 25.

Table 25: Current Motor Specifications

Impulse (N-s)	Loaded Mass (lbf)	Empty Mass (lbf)	Max Thrust (lbf)	Burn Time (s)	Thrust-Weight Ratio	Cost ($)
4905	9.17	4.06	586.3	2.07	10.68	331.99

Many AeroTech and Cesaroni motors were considered and simulated using OpenRocket. Motor alternatives can be seen in Table 26. Projected altitude of all three alternative motors is well below the target altitude of 4500 ft. Future design work to lower the weight of the entire launch vehicle can make the motor alternatives more viable options. It is desirable that the thrust to weight ratio be higher than the values presented in Table 26.

Table 26: Motor Alternatives

Manufacturer	Model	Predicted Altitude (ft)	Thrust/Weight
AeroTech	L952W	4125	4.11
AeroTech	L1300R	3838	6.14
Cesaroni	L1090SS	3960	6.36

3.3 Recovery Subsystem

3.3.1 Component Level Design

3.3.1.1 Number of Recovery Sections

The number of recovered sections is the number of sections that land separately, untethered and under different parachutes. A single recovery section would keep all different sections of the launch vehicle tethered together during recovery process. Using a single section would require fewer separation points on the launch vehicle, simplifying the structure of the airframe. This design option requires fewer ejection controllers and only one telemetry unit. This greatly simplifies the avionics and ground station system. With one section, all the launch vehicle weight would land under a single set of parachutes, this means the size of the parachute would need to be increased to meet the landing kinetic energy requirement set in the Student Launch (SL) handbook. A larger parachute also causes the packing volume to become more of a concern, and is more difficult to achieve a controlled and successful deployment of the parachute

Multiple recovered sections increase the complexity to the airframe, avionics, and recovery system, but reduces the weight each set of parachutes needs to support. This decrease in mass allows for the launch vehicle sections to fall at a greater velocity and decreases the drift radius of the launch vehicle. As mass increases, landing velocity decreases, and the required parachute size increases. If weight is increased too much, multiple launch vehicle sections must be used because of volume limitations and parachute availability. The main engineering requirements for the number of recovered sections are used in Table 27 and are the following:

- Landing Kinetic Energy: The lighter the rocket section, the easier it is to stay under the required kinetic energy limits.
- Recovery Radius: Reduced weight increases landing velocity and reduces the effective drift radius of the rocket during descent.
- Ease of Integration (Structures): More sections create more airframe parts and coupler tubes for structures to manufacture and assemble.
- Ease of Integration (Recovery): Additional sections require their own telemetry, ejection control, and parachute harness, which will double the work required for recovery.
- Ease of Integration (Payload): More sections can give payload ejection an advantage by putting the payload closer to an open end and by removing anything in the airframe blocking the exit.

Table 27: Number of Recovery Sections

Design		One Section		Two Sections	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score
Landing Kinetic Energy	4	3	12	5	20
Recovery Radius	2	3	6	5	10
Ease of Integration (Structures)	1	3	3	2	2
Ease of Integration (Recovery)	1	3	3	5	5
Total		**24**		**37**	

It was determined that a single recovered section is impractical for the current weight predictions of the launch vehicle because of the kinetic energy requirement in the USLI handbook. The launch vehicle will be recovered in two independent and untethered sections. The projected weight of the empty launch vehicle is 46.9 lbf. This would require a landing velocity no greater than 13.4 ft/s and a parachute size of 14 ft in diameter for a single recovered section to be used so that the fore and aft section would land under the maximum allowed kinetic energy. A parachute this large would be difficult to pack and is less likely to have a controlled and successful deployment. Using two sections will reduce the mass, allowing for a greater landing velocity which will decrease the size of parachutes needed and the drift radius.

3.3.1.2 Number of Parachute Compartments

Each untethered section of the launch vehicle is required by USLI rules to land with a dual deployment system. This means a drogue parachute will be used at apogee, then a main parachute deploys at a preset, lower altitude.

A single-compartment stores both the main and drogue parachutes in one bay. The drogue parachute will be ejected during separation at apogee. The main will be retained using Tender Descenders. At a lower altitude, an altimeter ignites the black powder inside the Tender Descenders, allowing the main parachute to be pulled out by the drogue parachute. Once the parachute shroud lines are fully extended, the lines become taught and the deployment bag is pulled off the main parachute, allowing for inflation. Having inflation outside of the launch vehicle allows for the parachute to be protected by the deployment bag until it is out of the airframe and away from heat. Using one compartment reduces the number of bulkheads in the airframe and will simplify airframe and lower weight.

A two-compartment configuration stores the main and drogue parachute in separate bays with its own set of separation and ejection charges. Because each charge is required to eject less weight, the charges will be more reliable and can be smaller. A two compartment system will also simplify the bridle configuration. The engineering requirements used for the number of parachute compartments in Table 28 are as follows:

- Inflation Control: The control over the main deployment to minimize snatch load.
- Extraction Control: The control over the parachutes deployment to reduce the chance of entanglement.
- Ease of Integration (Structures): The simplicity of designing the air frame with the given number of parachute compartments.
- Ease of Integration (Recover): The simplicity of designing the parachute ejection and retention system with the given number of parachute compartments.

Table 28: Number of Parachute Compartments

Design		**Two-Compartments**		**One-Compartment**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score
Inflation Control	3	3	9	4	12
Extraction Control	3	3	9	4	12
Ease of Integration (Structures)	2	3	6	3	6
Ease of Integration (Recovery)	2	3	6	4	8
Total		**30**		**38**	

A single recovery compartment will be used for the fore and aft sections of the airframe. This design will be used for the added simplicity of design, including the reduced weight of requiring fewer bulkheads. This design also allows for better control of the main parachute, ensuring a manageable snatch load, and a reliable ejection.

3.3.1.3 Canopy Shape

A canopy shape must be selected before the size can be calculated. Parachutes will not be manufactured by OSRT, so only parachutes that are available to purchase online will be considered. The parachutes considered are flat sheet, panel style, cruciform, elliptical, and toroidal.

A flat sheet parachute is constructed out of a flat sheet of material. These parachutes have stability that is okay at low speeds and poor at high speeds. Flat sheet parachutes are relatively cheap, but have poor performance compared to other types. These parachutes have a low coefficient of drag that peaks at 0.7, and typically consist of six to eight shroud lines. This parachute is shown in Figure 22.

Figure 22: Flat Sheet Parachute

Panel style parachutes are medium cost parachutes that have a tendency to rotate or spin. These parachutes can be used as a main parachute, but have a lower coefficient of drag than elliptical style parachutes at 1.1. This type of parachute is shown in Figure 23

Figure 23: Flat Panel Style Parachute

Cruciform parachutes are shaped like a cross that have good stability at any speed, but have a very low coefficient of drag. They make good high speed drogue parachutes, but would require a very large diameter to be used as a main parachute. A cruciform parachute is shown in Figure 24.

Figure 24: Cruciform Parachute

Elliptical parachutes are medium cost parachutes that have the shape of a bisected hollow sphere with a spill hole in the center. This shape has a large coefficient of drag around 1.6. This type of parachute consists of 12 shroud lines around the perimeter of the parachute which reduces the chance of entanglement upon ejection. Elliptical parachutes have good stability and are widely available on the market. An elliptical parachute is show in Figure 25.

Figure 25: Elliptical Parachute

A toroidal parachute is a high cost parachute that has a similar circular shape as an elliptical parachute with a spill hole in the center. This parachute has a second set of shroud lines around the spill hole. This design applies load differently than the elliptical parachute to increase the coefficient of drag by flattening out the top of the parachute. Toroidal parachutes have a coefficient of drag of 2.2. These added shroud lines increase the chance of entanglement, but result in a higher coefficient of drag that allows for a smaller parachute to be used. A toroidal parachute is shown in Figure 26

Figure 26: Flat Toroidal Parachute

The main engineering requirements for the canopy shape used in Table 29 are as follows:

- Coefficient of drag: as the coefficient of drag increases, a smaller parachute can be used. this makes integration for recovery and structures easier.
- Cost: Keeping costs low is preferred
- Performance: This is the ratio between descent rate and the weight of the parachute. It is preferred to use the lowest weight option
- Entanglement Susceptibility: The parachute should have resistance against entanglement to ensure a reliable ejection and inflation.

Table 29: Canopy Shape

Design		**Elliptical**		**Toroidal**		**Cruciform**		**Flat Sheet**		**Panel Style**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Coefficient of Drag	4	3	12	5	20	1	4	2	8	2	8
Cost	1	3	3	2	2	4	4	5	5	2	2
Performance	3	3	9	5	15	1	3	1	3	2	6
Entanglement Susceptibility	2	3	6	2	4	3	6	4	8	3	6
Total		**30**		**41**		**17**		**24**		**22**	

A toroidal parachute will be used for the aft and fore main parachute. The high coefficient of drag and performance means a smaller parachute can be used which will save weight and space in the launch vehicle.

Although these parachutes are more susceptible to entanglement than elliptical parachutes, the parachutes are inflated outside the launch vehicle utilizing a single compartment recovery, the risk for entanglement is low.

A cruciform parachute will be used for the aft and fore drogue parachute. These parachutes have a small coefficient of drag which will decrease the snatch load when the drogue is inflated. This lowers the risk of having a material failure. Cruciform parachutes also have good stability at any speed, making them an ideal drogue parachute.

3.3.1.4 Bridle Material and Shock Cord

The bridle is the line that tethers each separate section of the airframe to the parachutes. This cord must be able to withstand all recovery loads while maintain structural integrity when exposed to the heat of ejection. Nylon is a strong material with great elastic properties. This elasticity will decrease the snatch load experienced by the airframe and payload by increasing time taken to decelerate the launch vehicle. Nylon is a softer material that will help to prevent zippering of the airframe. Nylon has poor thermal characteristics and its properties can be compromised when exposed to high heat. Kevlar is a stronger material than nylon and has a higher strength to weight ratio. This reduces the diameter of cord needed, but will increase the chance of zippering. Kevlar is not as elastic as nylon, but has much better thermal characteristics. Another material being considered is spectra. Spectra is a very light material that is the strongest of the three materials. The increase in strength to weight ratio will increase the chance of zippering. Spectra is also the least elastic and thermally resistant. The engineering requirements for the bridle material are used in Table 30 and are as follows:

- Elasticity: A high elasticity will lower the snatch load the airframe and payload experience.
- Strength: A strong material that will not break is preferred.
- Thermal resistance: The material should be able to withstand the heat of ejection.
- Softness: The material cause minimal damage to the airframe during recovery.

Table 30: Bridle and Shock Cord Material

Design Material		**Nylon**		**Spectra**		**Kevlar**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Elasticity	5	3	15	1	5	1	5
Strength	2	3	6	5	10	4	8
Thermal Resistance	1	3	3	1	1	5	5
Softness	2	3	6	1	2	2	4
Total		**30**		**18**		**22**	

Nylon will be used for the bridle material because of its high elasticity and softness while still maintain decent strength. This will minimize the chance of the shock cord snapping, damaging the launch vehicle and payload due to a high snatch load, and damaging the airframe due to zippering. The chances of the nylon being damaged from heat will be minimized through the use of a Nomex blanket a Kevlar Sleeve. The length of the shock cords was determined from the advise of mentors and past OSRT teams. It was decided to use a shock cord for the drogue parachute that is 5 times the length of the specific launch vehicle body section it is attached to, and a shock chord length for the main parachute of 2 times the specific launch vehicle body section it is attached to.

3.3.1.5 Packing Method

The packing method is the way a parachute is stowed during flight. Different packing methods directly related the packing density in the airframe. The way a parachute is packed effects the success of recovery, as well as the ease of integration for recovery and structures.

Fold-and-wrap is the most common method used in recreational rocketry. This method consists of folding the parachute and wrapping the bridle lines around the parachute. A Nomex blanket will then by wrapped around the parachute for heat protection. The parachute is then placed into the airframe. This method has a packing density of 0.13 $^{oz}/_{cu}$ and is the simplest and fastest folding method to achieve. This method can increase the chances of entanglement and lower inflation control because the parachute will begin to inflate before the shroud lines are taught.

Deployment bags are another common packing method. This method can reach a packing density of 0.2 $^{oz}/_{cu}$ and consists of folding the parachute and placing it in the deployment bag. The bridle material can then be folded and placed inside an outer compartment. This method may be more complex than fold-and-wrap, but it can make transportation of parachutes simpler and will help protect the parachutes from the heat of separation. Using a deployment bag also allows for greater inflation and extraction control.

A pressure pack is a system consisting of a sealed canister, a pressurized ejection gas, and firing system. This system allows for the largest packing density of up to 0.28 $^{oz}/_{cu}$ and has a very reliable deployment with the ability to control the inflation of the parachute. This packing method is a very complex solution that would require specialized tools to execute and would take a greater amount of time to pack. The engineering requirements for the packing methods are used in Table 31 and are as follows:

- Packing Density: A smaller packing volume saves space in the airframe and is preferred.
- Simplicity: A packing method that can easily be executed in a short amount of time is preferred.
- Reliability: A consistent, successful parachute deployment that allows for inflation and extraction control is preferred.

Table 31: Packing Method

Design Material		**Fold-and-Wrap**		**Deployment Bag**		**Canister Pack**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Packing Density	2	3	6	4	8	4	8
Simplicity	5	3	15	3	15	1	5
Reliability	3	3	9	4	12	5	15
Total		**30**		**35**		**28**	

A deployment bag was chosen for both main parachutes because of its simplicity and reliability. The drogue parachutes will be packed using a fold-and-wrap method so they are easily pulled out and inflated at separation. Because of the size of the airframe, the packing radius using the correctly sized deployment bag is much smaller than that of the airframe. A Nomex blanket may be wrapped around the bag to prevent displacement of the parachute during flight. This will also help to protect the parachute from the heat of separation.

3.3.1.6 Altimeters

According to 2.3 in the USLI handbook, at least one barometric altimeter in the launch vehicle to record the apogee altitude during competition is required. With the current design, a minimum of two altimeters is needed. Currently, four altimeters that will ignite ejection charges will be contained in the launch vehicle: two in the fore section and two in the aft section. Three altimeters have been selected for comparison. The altimeters chosen include Missileworks Rocket Recovery Controller 3 (RRC3), Raven3, and StratoLoggerCF. The fifth altimeter mentioned which will not be lighting ejection charges is the Jolly Logic AltimeterThree.

Missileworks RRC3 has a proven track record at OSU. Many flights have been flown successfully with these altimeters. The RRC3 has many advantages. It provides three outputs, meaning it can ignite three ejection charges. Only two outputs are necessary for the launch vehicle, but in the case one output is malfunctioning, the third could be used instead of having to replace the altimeter. The altimeter provides noise reduction for more accurate pressure readings and can log 15 flights.

The Featherweight Raven3 altimeter has a dual altitude measurement system. It uses an accelerometer in conjunction with a barometric sensor; however, the altimeter is programmed to use the accelerometer first, and then the barometric sensor 1.5 seconds after apogee. This would need to be reversed if the altimeter was going to be used. Another problem is the altimeter is programmed to release the main parachute 700 ft AGL. This would also need to be reprogrammed with our current flight plan.

The PerfectFlite StratoLoggerCF altimeter has many great features, and it has been flown at OSU with success. The altimeter can have a delay set at apogee, making it perfect for a back-up altimeter, and the altitude the main parachute is released can be set in 1 foot increments.

Seen in Table 32 are the DDM for the three main choices for primary and secondary altimeters. The chosen requirements are listed below:

- Reliability: the launch vehicle needs to be recovered safely, requiring reliable altimeters.
- Accuracy: apogee and altitudes need to be sensed correctly for parachutes separation and release events to occur at the correct times.
- Programmability: some amount of programability is required to adjust the height the main parachutes are released.
- Cost: the altimeters need to be affordable.

Reliability was weighted the highest and cost was weighted the lowest. The altimeters are expected to function correctly every launch, and, while being cost effective is important, it will always be more important that the altimeters function reliably.

Table 32: Altimeters

Design		**Raven3**		**RRC3**		**StratoLoggerCF**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Cost	3	3	9	4	12	5	15
Reliability	10	3	30	5	50	4	40
Programmability	6	3	18	2	12	4	24
Accuracy	6	3	18	2	12	2	12
Total		**75**		**86**		**91**	

While the Raven3 has some advantages over the RRC3 altimeters, the cost and reliability of the RRC3 outweigh the programmability and accuracy provided byt the Raven3. Since the RRC3 has proven reliability with previous OSRT and is half the cost of the Raven3, the MissileWorks RRC3 altimeter will be one of the altimeters chosen.

The StratoLoggerCF is also superior to the Raven3 in almost every requirement. The StratoLoggerCF has previously proven its reliability with OSRT and it comes at a lower cost than the Raven3. The StratoLoggerCF will be the other choice for altimeters.

The StratoLoggerCF will be the primary altimeters while the RRC3 will be the secondary. The reason for this choice was the extra progammability the StratoLoggerCF provides. It can be programmed to ignite charges at intervals of one foot, while the RRC3 can be programmed in intervals of 100 ft. The StratoLoggerCF will be programmed to ignite the main parachute ejection charge 25 ft above the secondary charge, which will deploy at 500 ft AGL. This is to avoid any complications igniting both charges at the same time.

The Jolly Logic AltimeterThree is a newer altimeter capable of connecting with a tablet or a phone. The altimeter has many great functions. All of the data from the flight is sent directly to the paired device, showing a 2D plot of the flight trajectory labeled with burnout thrust time, maximum altitude and time to reach it, descent rates under main and drogue parachutes, and landing velocity. OSRT is in possession of one of these altimeters, and it will be strapped down in the fore ejection bay. The AltimeterThree provides no additional functionality, however it provides the team with the ability to rapidly view flight data after recovery.

3.3.1.7 Main Parachute Retainer

The methods compared for main parachute retention and/or ejection include Tender Descenders, Advanced Retention and Release Device (ARRD)s, Jolly Logic Chute Releases, and ejection charges. Two main overarching options were considered for main parachute retention and ejection:

- Drogue acting as a pilot chute after the main is released (all on one shock cord)
- Drogue and main ejected separately, two different shock cords

In the first case, the main parachute either has to be held inside the airframe or held shut outside of the airframe. ARRDs and Tender Descenders can be used in either way; however, they need to be connected to e-matches, so they are typically used for retention. A Jolly Logic Chute Release can be used to hold the parachute shut when it is outside of the airframe.

For the second case, a separate ejection charge ejects the main parachute. This can be accomplished in two ways: splitting the recovery bay into two compartments, one for the drogue and one for the main, and stacking the drogue on top of the main with two ejection charges between the parachutes, and two ejection charges beneath the main parachute.

Shown in Table 33 is the DDM. The requirements chosen for this matrix are listed below:

- Ease of integration: integration time will be minimized by choosing easily integrateable components.
- Likeliness of success: the launch vehicle needs to be recovered safely, requiring reliable altimeters.
- Simplicity of design: simple designs will be easier to integrate.

Likeliness of success is weighted the highest and simplicity of design is rated the lowest. Successfully recovering the launch vehicle is the main priority, while designing for ease of integration and simplicity come second and third.

Table 33: Main Parachute Extraction

Design		**Separate Ejection Charges**		**Pilot Parachute**		**Separate Ejection Charges with Separate Compartments**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Ease of integration	7	3	21	4	28	2	14
Likeliness of Success	10	3	30	4	40	4	40
Simplicity of Design	5	3	15	4	20	5	25
Total		**66**		**88**		**79**	

Shown in Table 33 is the decision matrix for main parachute extraction. The best alternative, which will be integrated into the airframe, is the pilot parachute extraction. Both other options have complications. When using separate ejection charges, there is a risk of breaking the tips of the e-matches. When the ejection charges separate the launch vehicle at apogee, the main parachute is compressed further, which could break the leads on the e-matches. To avoid this, the airframe can be separated into two compartments running the whole length of the recovery bay. This introduces manufacturing difficulties, complicating the design and increasing the difficulty of integration. When using the drogue parachute as a pilot, these complications are avoided, and having one shock cord minimizes any risk of tangling shock cords.

- Reliability: the launch vehicle needs to be recovered safely, requiring reliable altimeters.
- Ease of integration: integration time will be minimized by choosing easily integratable components.
- Modifications: some components will need to be modified to work in higher stress situations.

Table 34: Retention Method

Design		**ARRD**		**Tender Descenders**		**Jolly Logic Chute Release**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Reliability	8	3	24	4	32	2	16
Ease of Integration	3	3	9	4	12	1	3
Modifications	5	3	15	4	20	5	15
Total		**51**		**56**		**37**	

Previously, OSRT had several failures with Jolly Logic Chute Releases, which has been taken into consideration. Tender Descenders have scored the highest, so they will be the retention method utilized.

3.3.1.8 Reduction of Zippering Chance

Three main options for reducing the chance of zippering are listed below:

- Adding a tennis ball or other soft material to the shock cord, where the shock cord meets the end of the airframe/coupler
- Splitting the load into a minimum of two points (up to four or more points).
- Adding an artificial zipper, which is a variation of a frangible tie, to sections of a z-folded shock cord.

In issue 290 of the "Peak of Flight" newsletter, the first alternative listed is mentioned. It involves spreading the load out over a larger surface and softening the material impacting the airframe. The tennis ball's advantages over a larger piece of foam, for example, are its smaller volume. Since the shock cords are made of nylon, they will stretch during flight, and the amount of stretch is not reasonably predictable. The tennis ball will likely never end up exactly at the edge of the airframe, rendering it useless. The larger piece of foam will take up a larger space in the airframe but have a large margin of error for stretching of the shock cord. Using compressible foam could partially mitigate the larger space.

In issue 282 of the ”Peak of Flight” newsletter, the load splitting method is mentioned. Splitting the load into several points has several benefits. This method reduces the load felt by each connection point by $1/N$, where N is the number of connection points. Having more than one connection point will also split the force the harness/shock cord impacts on the edge of the airframe into N spots. This splitting up of the load will also likely ”whip” the airframe midair, instead of rotating, or zippering, it due to the force acting on the edge of the airframe. In theory, as one or two of the sections of the harness fold in on themselves, the opposite sections go into tension, causing the airframe to rotate. Since this is happening at high velocities, the whipping action will not occur immediately, but it will still drastically reduce the force impacted on the edge of the airframe.

In Rick Newlands ”Parachute Recovery System Design for Large Rockets,” he presents a method for reducing zippering called an artificial zipper. An artificial zipper, pictured in Figure 27, which is a variation of a frangible tie that uses tape instead of sewn sections of shock cord, will be incorporated into each section of shock cord to reduce the difference in relative velocity between the drogue parachute and the airframe. While z-folding the nylon shock cord, the folds can be taped together. When the ejection charges are fired, the tape will tear, dissipating some of the load, slowing the drogue parachute. This slowing of velocity lowers the snatch load experienced by all recovery components. Consequently, this lower velocity will decrease the force impacted on the edge of the airframe, decreasing the chance of zippering the airframe.

Figure 27: Artificial Zipper

Shown in Table 35 is the DDM for reducing the shock load. The main requirements for reducing the shock load are listed below:

- Ease of integration: integration time will be minimized by choosing easily integratable components.
- Predictability: the method needs to perform in a predictable manner.
- Space Efficiency: the method needs to be space efficient.

Table 35: Reducing Zippering Chance

Design		**Additional Material**		**Splitting Load**		**Artificial Zipper**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Ease of Integration	4	3	12	2	8	4	16
Predictability	7	3	21	4	28	5	35
Space Efficiency	6	3	18	4	24	5	30
Total		**51**		**60**		**81**	

An artificial zipper will be incorporated into each section of shock cord. If it is determined at a later date that the risk of zippering is too high, an additional decision matrix will be created comparing the two methods not chosen.

3.3.1.9 E-Matches

Every ejection charge will be ignited with an electric match, otherwise known as an e-match. The lead on the e-matches will be in the middle of all ejection charges, surrounded by black powder, and the opposite end is wired directly to the altimeters. Both the primary and secondary charges will be threaded through a hole drilled in the bulkhead of the altimeter bay.

The e-matches that will be used are the J-Tek igniters, which come in lengths up to 199 in. E-matches are regulated by the Alcohol, Tobacco, Firearms (ATF), so they will be purchased by the team mentor, Joe Bevier.

3.3.1.10 Arming Switch

When installing the ejection bays, there is a chance that static charges could build up and set off the black powder charges. To mitigate this risk, a single push double throw shunting switch grounds the leads of the e-match, preventing static charge from igniting an e-match. When the bays are integrated into the airframe, the switch can be safely turned on with the altimeters still unarmed. To arm the switches from the exterior of the launch vehicle, two allen wrench switches in each ejection bay sit just within the walls of the launch vehicle and bay walls. The holes have a 3/16 in. diameter and double as static port holes for the altimeters within. These rotary switches cannot be flipped through the forces of flight.

3.3.1.11 Ejection Charges

Two methods are commonly used as ejection charges: black powder and carbon dioxide canisters. Both methods have advantages and disadvantages.

Black powder is the most commonly used method for separating high powered rockets and launch vehicles. It is reliable and has been proven over and over to work. The powder and e-match will be contained in surgical tubing with santoprene plugs. Zip ties will be wrapped around the santoprene to ensure an airtight seal. The leading issues with black powder include high heat at ignition, corrosive gasses, and an incomplete ignition at high altitudes. Because of the high heat, shock cords and parachutes will need to be heat protected nearest the charges. Corrosive gasses present an issue to any sensitive electronics. To ensure no electronics are harmed, all will be in an air sealed container. At high altitudes, where there is less air to transfer heat through, black powder charges can ignite, if the charge is sealed correctly, but they are not guaranteed to burn completely. This could result in a failure to separate at apogee. However, the altitudes the launch vehicle is designed to reach will have a minimal impact on the charge performance. The benefits include minimal space taken up, simplicity of use, and reliability.

Carbon dioxide does not create any damaging or corrosive gasses like black powder, and it does not produce heat at the same scale as black powder charges. The CO_2 charges, however, are bulky and come in limited sizes, making successful separation less predictable. Seen in Table 36 is the decision matrix for ejection charges.

Shown in Table 36 is the DDM for ejection charges. The requirements are listed below:

- Cost: the budget will be considered before any components are purchased.
- Ease of integration: integration time will be minimized by choosing easily integrateable components.
- Space efficiency: the launch vehicle length should be limited, and shorter recovery bays require less black powder to cause separation.
- Reliability: the launch vehicle needs to be recovered safely, requiring reliable altimeters.

Table 36: Ejection Charges

Design		**CO_2**		**Black Powder**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score
Cost	4	3	12	4	16
Ease of Integration	7	3	21	4	28
Space Efficiency	6	3	18	4	24
Reliability	10	3	30	4	40
Total		**81**		**108**	

3.3.2 Recovery Sizing

3.3.2.1 Parachute Sizing

This section outlines the process followed to determine the appropriate size for all parachutes. The values used for the coefficient of drag for the toroidal and cruciform parachutes are listed in Table 37.

Table 37: Values for Coefficient of Drag

Variable	**Value**
$C_d, toroidal$	2.2
$C_d, cruciform$	0.98

FruityChutes toroidal parachutes have a coefficient of drag of at least 2.2, and in some cases over 3. OSRT decided to analyze the parachutes with a drag coefficient of 2.2, as it would give the safest results, keeping the kinetic energy upon landing within requirements.

Given that the density of air is a function of altitudes, it was calculated using the barometric formula for density. It was assumed that the standard temperature lapse rate, T_b, was not equal to zero. The equation used to solve for Density is as follows:

$$rho = rho_b[\frac{T_b}{T_b + L_b(h - h_0)}]^(1 + \frac{g_0 M}{R^* L_b}) \tag{3}$$

Where T is the standard temperature in K, L is the standard temperature lapse rate in K/ft, h is the height Above Sea Level (ASL) in ft, g_0 is gravitational acceleration in ft/s^2, M is the molar mass of air in kg/mol, R^* is the universal gas constant in ft^2/sK, and the subscript b corresponds to the layer of the atmosphere. The projected flight will not leave the first layer of the atmosphere ($b = 0, h < 36,089ft$).

$$KE = \frac{1}{2}mv^2 \tag{4}$$

KE is kinetic energy in $ft - lbf$, where m is mass in $slugs$, and v is velocity of the system in ft/s.

$$D = \frac{1}{2}C_d \rho_{air} v^2 A_r \tag{5}$$

D is drag around the parachute in lbf, where C_d is the coefficient of drag of the parachute, ρ_{air} is the density of air in $slugs/ft^3$, v is the velocity of the system in ft/s, A_r is the reference area of the parachute in ft^2, which is the cross sectional area, and W_{lv} is the weight of the launch vehicle in lbf.

$$D = W_l v = \frac{1}{2}C_d \rho_{air} v^2 A_r \tag{6}$$

$$A_r = \frac{1}{4}\pi(d_o^2 - d_i^2) \tag{7}$$

The reference area can be defined in terms of the outer and inner diameters of the toroidal parachute, where d_o is the outer diameter and d_i is the inner diameter. Typically, the ratio between outer and inner diameters for this shape of parachute is 5:1. Using this, the reference area equation becomes:

$$A_r = \frac{6}{25}\pi d_o^2 \tag{8}$$

Plugging equation 8 into equation 5 and solving for the outer diameter gives:

$$d_o = \sqrt{\frac{25W_{lv}}{3\pi\rho_{air}C_d v^2}} \tag{9}$$

To calculate the reference area of the drogue parachutes, Equation 10 was used.

$$A_{xform} = 2DW - W^2 \tag{10}$$

Width, w, is the length of the short edge and diameter, D, is the diameter of the drogue parachute.

Based on these calculations and the toroidal parachutes that are readily available online, it was decided to utilize a 10 ft compact toroidal parachute for the fore section, and an 8 ft compact toroidal parachute for the aft section. These sizes will ensure the landing kinetic energy of both sections is below 75 ft-lbf. Based on MATLAB and OpenRocket models these sizes also yield a descent time under 90 seconds and a drift radius under 2500 ft.

3.3.2.2 Ejection Size

The ejection charges are made of black powder and surgical tubing. One end of surgical tubing is sealed shut and filled halfway with black powder. The lighting end of an ematch is placed into the black powder and the rest of the surgical tubing is filled and sealed. To determine how large the charges should be, the following equations are used:

$$BlackPowder(g) = (CompartmentDiameter)^2 * CompartmentLength * 0.006 + 1 \tag{11}$$

$$BlackPowder(g) = (6.107in.)^2 * (5in.) * 0.006 + 1 = 2.119grams \tag{12}$$

The addition of one at the end of the equation is to be sure that the charges are strong enough and to account for any pressure loss due to faulty seals. The components have been engineered to handle more than what is necessary for the charges to deploy the parachutes so adding an extra gram of black powder will not hurt any components.

3.3.3 System Layout Alternatives

Two major designs were considered for the recovery layout. One of the design candidates has every component of the launch vehicle tethered together so that the vehicle descends as one body. This is the more common idea. The second design considered is less common; the launch vehicle separates into two independent sections at apogee with each section descending under their own parachutes. In both cases, the drogue parachutes are released at apogee and the main parachutes are released at 500 ft AGL. Below is a list of the requirements for the DDM. Shown in Table 38 is the DDM for recovery layout options.

- Descent time: how close to 90 seconds will either method be under allowable descent velocities.
- Minimum landing velocity: how fast the vehicle can descend while landing safely.
- Simplicity of integration: how many parachutes and recovery components are needed, including altimeters.
- Reliance on recovery systems: how many different wired recovery components are relied upon.
- Payload ejection: possible interference upon payload ejection.

Table 38: Recovery Layout Alternatives

Design		**One Tethered Body**		**Two Independent Tethered Body**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score
Descent Time	6	3	18	4	24
Minimum Landing Velocity	8	3	24	4	32
Simplicity of Integration	6	3	18	2	12
Simplicity of Recovery and Retrieval	6	3	18	2	12
Reliance on Systems	6	3	18	2	12
Payload Ejection	7	3	21	4	28
Total		**117**		**120**	

Both systems have their benefits and drawbacks. While recovering the whole launch vehicle in one tether is typical and convenient, it presents several issues. The main concerns include the packing of and unfurling a parachute that is as large as what would be needed and the ejection of the payload. Due to the high total weight for the projected launch vehicle, the final descent rate would be 11.36 ft/s under a 14 ft diameter toroidal parachute. The descent time would be 44.02 seconds under the main, giving a maximum of 45.98 seconds under drogue. The maximum cruciform drogue diameter needed to successfully land in under 90 seconds is 1.01 ft. These calculations, however, assume that the launch vehicle weight does not change after removing half of the recovery system. Incidentally, these calculations have been applied with a slight safety factor. The benefits of this system include recovering everything together, being able to further reduce the chance of zippering in the aft section, and less reliance on additional recovery methods.

Separating the launch vehicle into two independent sections provides a higher margin for error for descent time and landing kinetic energy. It also reduces the chance of payload ejection complications. The downsides include having to double on recovery components and having to rely on black powder and Tender Descenders.

Separating the launch vehicle into two independent sections was the primary method of choice. To mitigate the problems that may arise due to the reliance upon ejection charges and Tender Descenders, both systems have redundancy incorporated into them: primary and secondary charges and two Tender Descenders in series.

3.3.4 System Layout

The ejection events are pictured in Figure 28. At apogee, the launch vehicle will have two separation events that occur simultaneously as soon as the main altimeters sense apogee - assuming both main ejection charges fire and cause separation. No other separation events will occur, due to both dual deployment systems being contained in a single compartment. One separation event ejects the nosecone from the fore section, while the other separates the fore from the aft section. After separation, the fore and aft become independent sections, both descending under a drogue. At 500 ft AGL, both independent sections release a main parachute. The main parachutes are held in with two Tender Descenders in series connected to a loop in the shock cord. The charge inside each will be ignited at 500 ft AGL, allowing the drogue to be a pilot chute for the main parachutes The primary, Missilworks RRC3, and secondary, PerfectFlite StratoLoggerCF, altimeters are stored in altimeter bays located aft of the both recovery bays. These altimeters will be the ejection controllers and will be wired through the fore bulkheads of the altimeter bays to the ejection charges with J-Tek e-matches. All altimeters will be kept in separate compartments from other electronics and will be shielded from interference from other electronics.

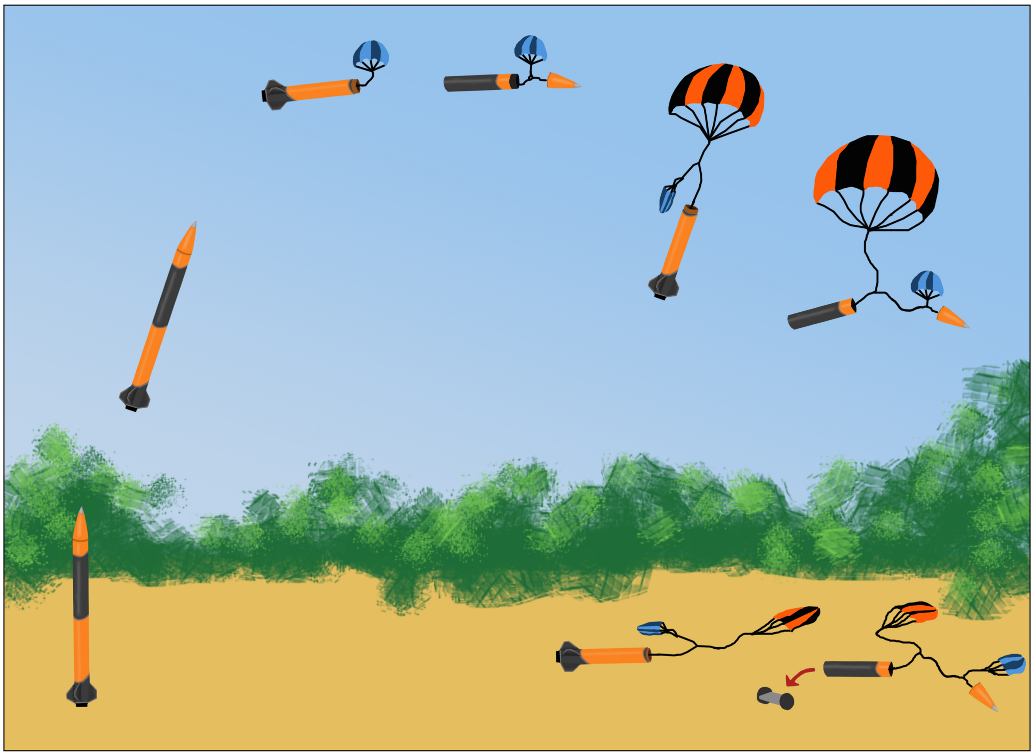

Figure 28: Recovery Separation Events (not to scale (NTS))

The aft section layout is shown in Figure 29. Two eye-bolts are attached to the aft altimeter bay with nylocs. The recovery harness consists of two sections of shock cord: one for each parachute. The main parachute is has two connection points: the shock cord is connected to one eye-bolt, and two Tender Descenders in series are connected to the other eye-bolt and a loop in the upper section of the shock cord. The other end is attached to the swivel attached to the main parachute. The main parachute is packed into a deployment bag and wrapped in a Nomex blanket threaded onto the shock cord. The bag is attached to the top of the main parachute and the drogue shock cord, which is attached to the swivel of the drogue parachute. All attachment points, excluding the swivels to the parachutes, have a quick link attachment. Extra wide mouth quick links will be used at all eye-bolt attachments.

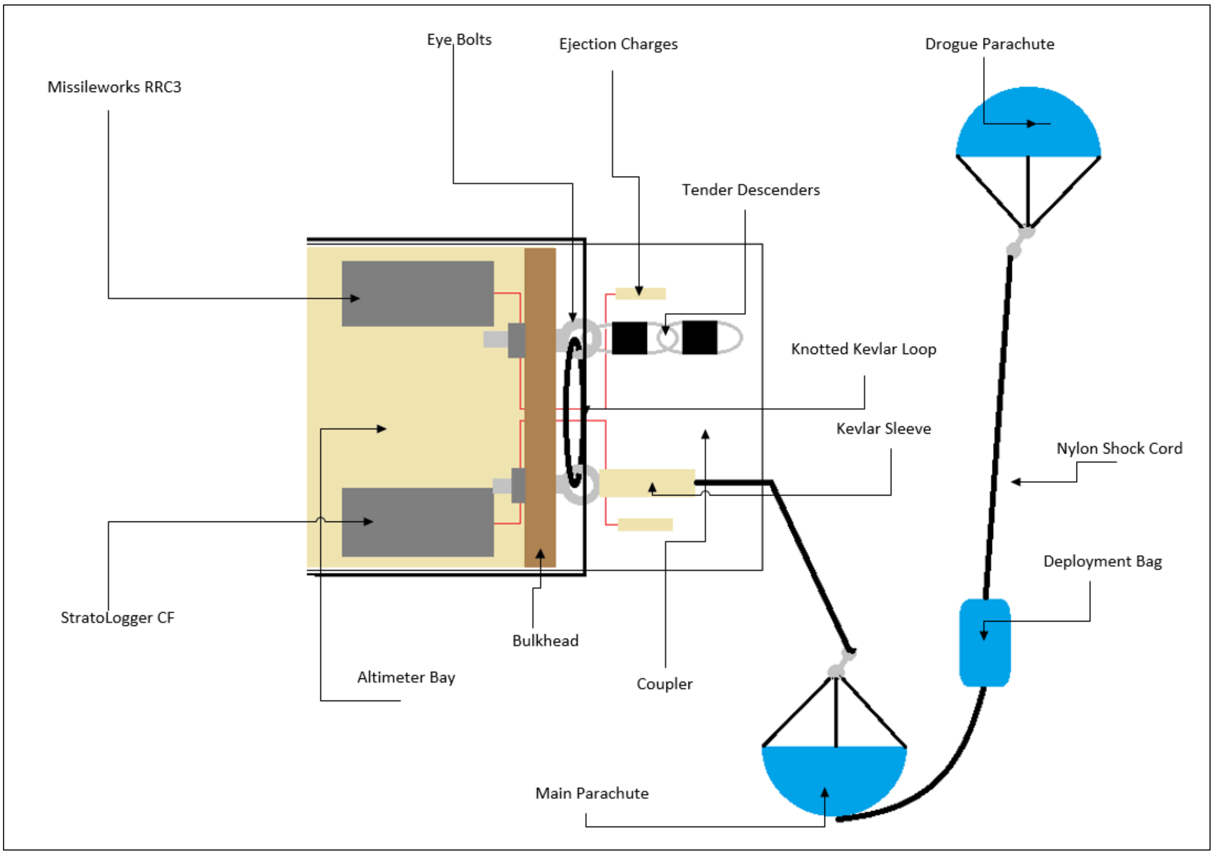

Figure 29: Aft Recovery Layout (NTS)

The fore recovery bay, which is shown in Figure 30, is nearly identical to the aft recovery bay in layout. The one exception is the drogue parachute shock cord. The drogue parachute is going to be placed 30% down the length of the shock cord from the nosecone, and it will be connected to a loop in the shock cord. The free end of the shock cord will be connected to the nosecone.

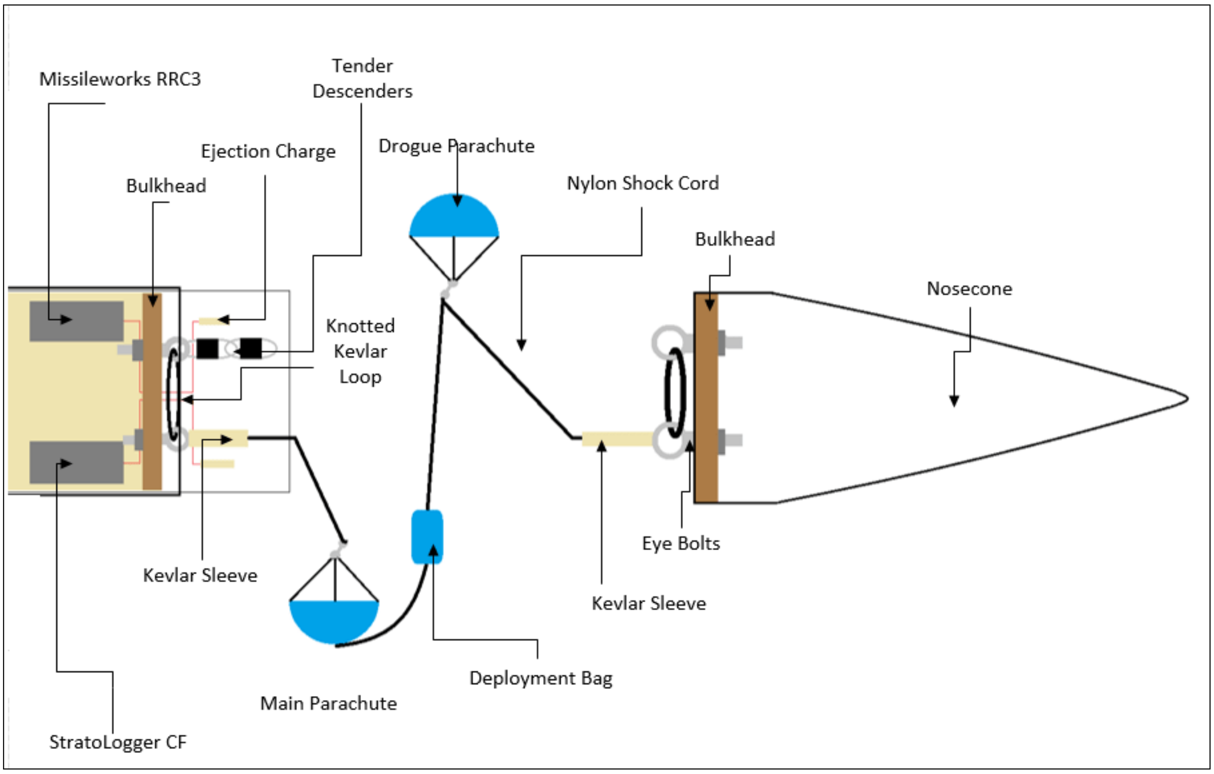

Figure 30: Fore Recovery Layout (NTS)

The eye-bolts at each bulkhead will be connected with a knotted piece of Kevlar. This will be done so neither eye-bolt can unscrew itself from the nyloc on the opposite end.

3.3.5 Redundancy

Several redundant components have been included in the launch vehicle design. The redundant components include altimeters, Tender Descenders, and ejection charges. In the fore and aft sections, primary ejection charges are set off by primary altimeters (RRC3) at apogee, and one second later the secondary charges are set off by the secondary altimeters (StratoLoggerCF). Similarly, two Tender Descenders linked in series are released by primary and secondary altimeters at 500 ft AGL.

Center of pressure simulations were performed using OpenRocket and RASAero II. The simulations calculated very similar center of pressure locations. OpenRocket's simulated center of pressure is 72.051 in. and RASAero's simulated center of pressure is 71.92 in. from the nosecone tip.

3.4 Mission Performance Predictions

3.4.1 Official Target Altitudes

The full scale launch vehicle will be designed by OSRT to reach an apogee altitude of 4,500 ft AGL.

3.4.2 Flight Profile Simulations

The software OpenRocket was used to simulate the launch vehicle's flight. Predicted apogee altitude and stability was simulated using the software. The OpenRocket model can be seen in Figure 31. All lengths and weights will be verified with the estimates presented in the simulation to ensure an accurate simulation prior to full scale launch. The expected velocity upon exiting a 12 ft rail is 83.8 ft/s.

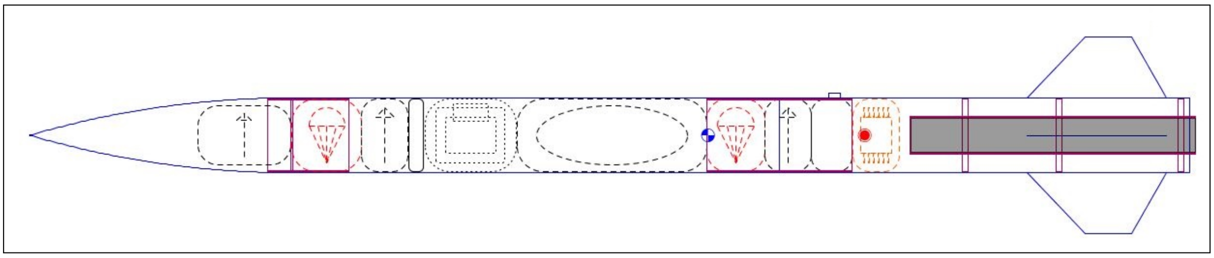

Figure 31: OpenRocket Simulation

The launch vehicle was simulated at varying horizontal wind speeds of 0 mph, 5 mph, 10 mph, 15 mph, and 20 mph. Multiple simulations with varying cross wind speeds gives the ability to refine a precise amount of ballast to use in the fore and aft ballast bays. For more on ballast requirements, reference Table 22. The simulation of 10 mph crosswinds can be seen in Figure 32.

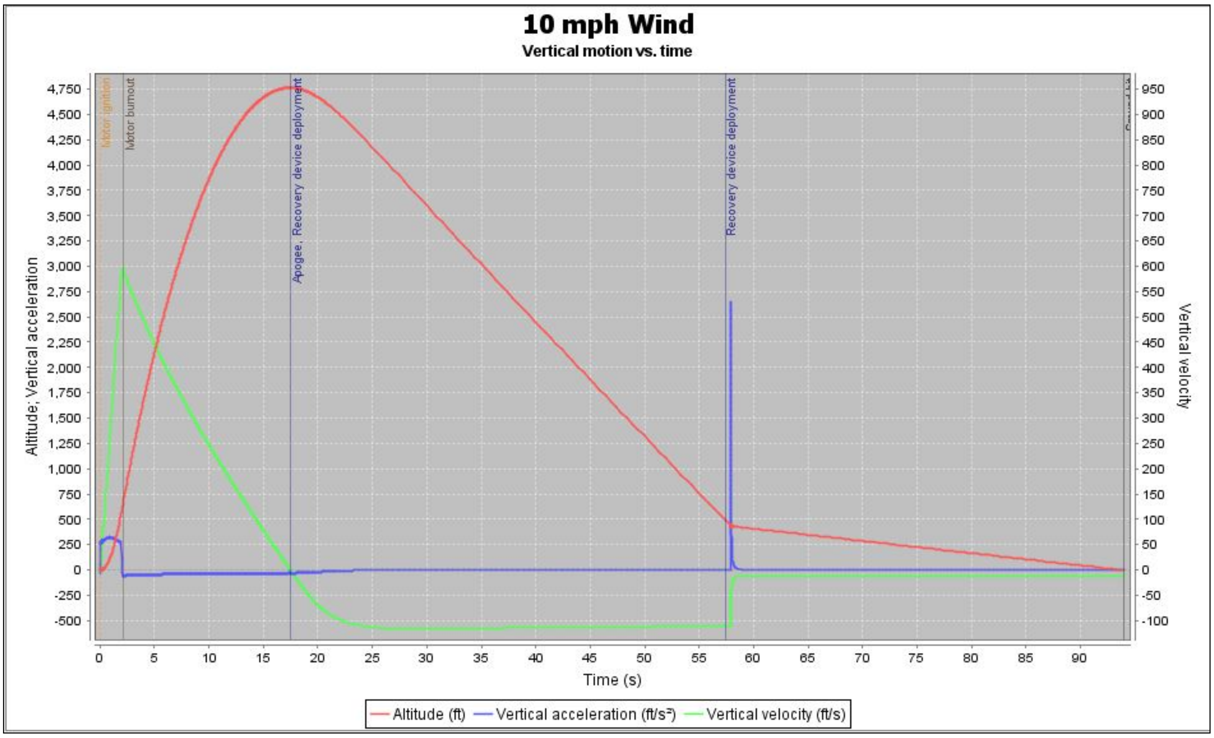

Figure 32: OpenRocket Simulation for 10 mph cross-winds

With 0 mph crosswinds, the launch vehicle is expected to reach an apogee altitude of 4797 ft. Apogee altitude decreases with the presence of horizontal crosswinds. The magnitude of this affect can be seen in Table 39. All of the predicted apogees in Table 39 are with no ballast added to the launch vehicle. For more on ballast requirements, reference Table 22 in Section 3.2.7.1. It would be preferable that the predicted altitude be higher than presented in Table 39, which would give future design work of internal components availability to increase in weight if required. All components in the launch vehicle will be critically investigated into decreasing packing length and weight.

Table 39: Projected Altitude with Cross-Winds

Wind Speed (mph)	Aft Ballast (lbf)	Fore Ballast (lbf)	Predicted Apogee (ft)
0	0.00	0.00	4797
5	0.00	0.00	4791
10	0.00	0.00	4775
15	0.00	0.00	4748
20	0.00	0.00	4713

3.4.3 Stability Margins

The stability of the launch vehicle was simulated using OpenRocket. After all components were input into the simulation, trapezoidal fins were designed to create a stability of 2.1 calibers. The center of gravity of the launch vehicle is simulated at 58.55 in. from the tip of the nosecone. The center of pressure is 72.05 in. from the tip of the nosecone. The center of gravity of the launch vehicle will be verified upon manufacturing of the launch vehicle. Ballast bays in the fore and aft will ensure the center of gravity is in the proper location to ensure a stability of 2.1 calibers with the center of pressure. The center of gravity and center of pressure locations from the OpenRocket simulation can be seen in Figure 31 in Section 3.4.2.

The software RASAero II was also used to verify the center of pressure of the launch vehicle. With the same dimensions, RASAero II simulated the center of pressure to be 71.92 in. from the tip of the nosecone, a difference of 0.131 in. from the OpenRocket simulation.

3.4.4 Landing Kinetic Energy

A MATLAB script was used to calculate the kinetic energies of the fore and aft sections upon landing. The weight, final landing velocity and landing kinetic energy are shown in Table 40 below. Both sections remain under the maximum kinetic energy at landing of 75 ft-lbf.

Table 40: Landing Kinetic Energy

Measurement	Fore Section	Aft Section	Nosecone
Weight (lbf)	26.8	20.1	2.2
Velocity with Main and Drogue Deployed(ft/s)	12.0	13.0	12.0
Kinetic Energy with Main and Drogue Deployed (ft-lbf)	60.2	52.4	4.9
Velocity with Only Drogue Deployed (ft/s)	111.0	105.0	111.0
Kinetic Energy with Only Drogue Deployed (ft-lbf)	5,137.5	3,444.5	419.3
Velocity with no Parachutes Deployed (ft/s)	115.0	112.0	115.0
Kinetic Energy with no Parachutes Deployed (ft-lbf)	5,514.4	3,919.1	450.1

3.4.5 Descent Time

A MATLAB script was written to calculate the descent times of the fore and aft section under the same 1.5 ft cruciform drogue parachute, but under a ranging size of main toroidal parachutes. This was plotted next to the maximum descent time allowed in the USLI Handbook. This plot can be seen in Figure 33.

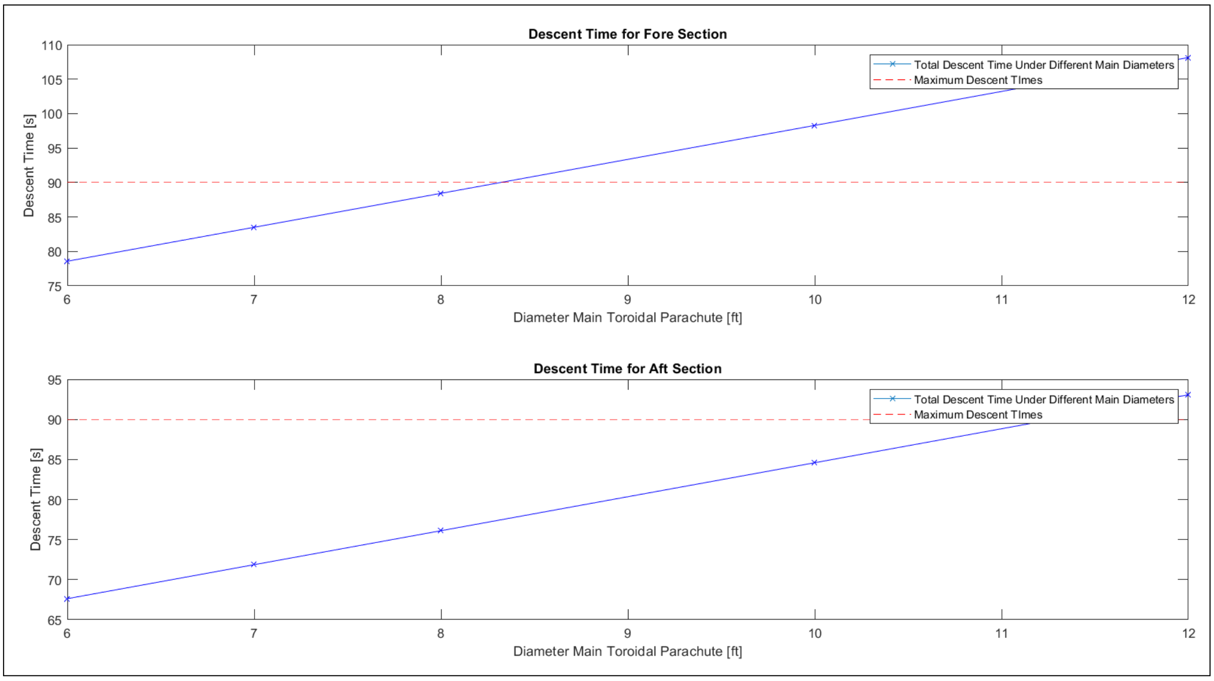

Figure 33: Descent Times for Ranging Parachute Sizes

The sizes for the fore and aft main parachutes was decided to be 10 and 8 feet, respectively. The trajectory of the descent for both sections were calculated on MATLAB and accounts for the changing air densities at different altitudes as well as the acceleration and deceleration of the launch vehicle at apogee and at deployment. This is shown in Figure 34. It can be seen that both sections land in under 90 s. The exact descent time can be seen in Table 41.

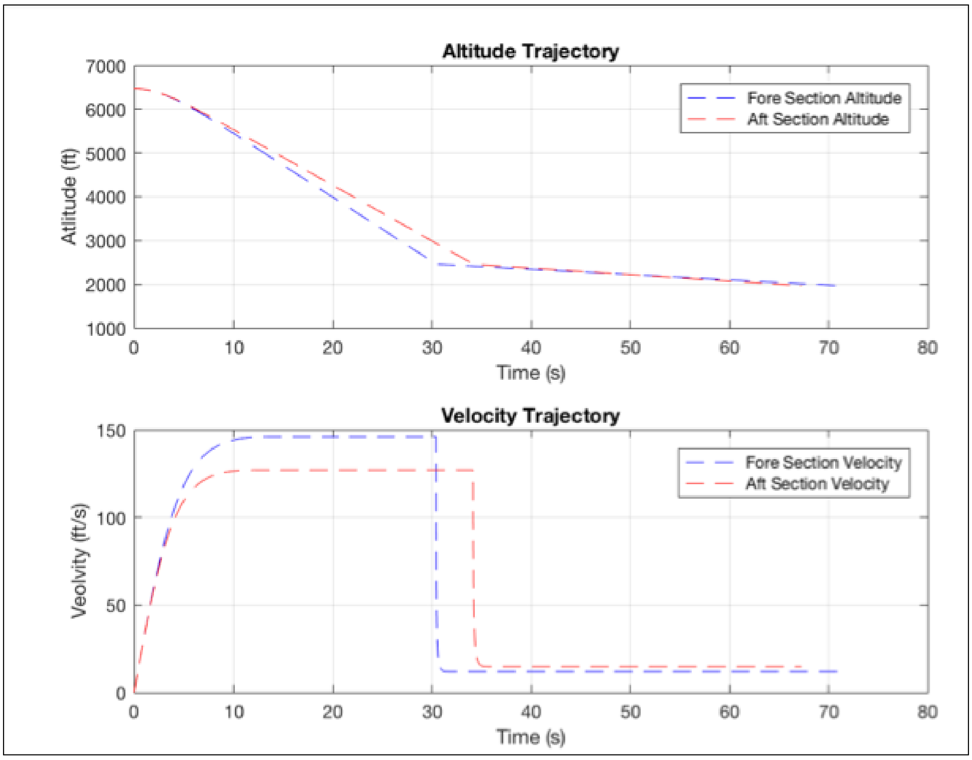

Figure 34: Descent Trajectory

3.4.6 Drift Calculations

Drift estimations were made using MATLAB. The wind speed was assumed to be a constant crosswind that does not impact the vertical trajectory. Weather cocking was not included in this calculation so it was assumed that apogee occurred directly above the launch pad. This is a conservative approach because weather cocking pushes the launch vehicle the opposite direction of the drift. These calculations are shown in Table 41. These are compared with drift calculations from OpenRocket.

Table 41: Drift and Descent Time

Wind Speed (mph)	**0**	**5**	**10**	**15**	**20**	**Descent Time (s)**
Drift of the Fore Section (ft)	0	523	1,046	1,596	2,092	71
Drift of the Aft Section (ft)	0	524	1,056	1,585	2,113	72
OpenRocket Simulation (ft)	8	248	521	856	1,069	76

All sections stay within the maximum drift radius of 2,500 ft

It can be seen that the drift calculations done on MATLAB and OpenRocket differ significantly, but all values stay under the maximum allowable drift radius of 2,500 ft The descent time for all sections also stays below the maximum allowable descent time of 90 s.

3.4.7 Redundant Simulations

The OSRT have utilized both MATLAB and OpenRocket to run simulations. Both programs were used to determine: descent time, landing velocity, landing kinetic energy, and drift radius. Some of these values differed from program to program, but all values meet the requirements stated in the USLI Handbook. Using different software will ensure that the numbers are correct. Additionally, center of pressure was validated in both RASAero II and OpenRocket.

4 Payload Criteria

4.1 Payload Objective

This year, the payload objective is to travel at least ten feet from the launch vehicle once it is ejected, collect a 10 mL soil sample, store the sample in an on-board containment unit, and possibly conduct a scientific experiment on the sample. To simulate a real life application, after the rover collects the soil sample it will drive to a base station where the sample will be deposited. The base station will then perform a scientific experiment on the soil sample. OSRT is looking into the feasibility of conducting an X-ray fluorescence experiment to determine the elements present in the sample. The function of the rover and base station is meant represent the collection and experimentation of a soil sample on a different celestial body such as Mars.

4.2 Design Justification Methodology

To make design decisions, the team used a Decision Matrix found in Table 42. Table 42 identifies the competition and team derived specifications for the specific system and then rated them on a 1 through 5 scale (5 as very important and 1 as not very important). Next, one system was given a baseline score of all 3's and the other systems were rated relative to the baseline. These ratings were also on a scale between 1 and 5, with 5 considered a better design and 1 considered a worse design. The two scores were then multiplied together to get a total weighted value. The sum of the weighted values indicated the total weight of the design choice. The system with the highest total rating was selected to be the leading design choice.

Table 42: Generic DDM

Design		**Idea 1**		**Idea 2**		**Idea 3**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Specification 1							
Specification 2							
Specification 3							
Specification 4							
Total							

4.3 Mechanical Sub-system Review

4.3.1 Chassis

The rover chassis has a few requirements for mission success. It must be easy to assemble and disassemble to allow the electrical engineering and computer science team to test different electrical systems. The chassis must fit within the diameter of the wheels while providing as much space for other components to fit within and around it. It must provide enough distance between the bottom of the chassis and the ground to prohibit the terrain from interfering with driving operations, such as running into larger rocks and preventing rover movement. The chassis must have sufficient strength to withstand all forces during mission operations, while conserving weight. The first major design decision was the overall chassis shape. Table 43 displays the four overall shapes considered, comparing the ease of assembly and disassembly, volume, ground clearance, and strength. The four designs consisted of a rectangular prism, truss frame, cylindrical frame, and triangular prism.

Table 43: Chassis Shape DDM

Design		**Rectangular Prism**		**Truss Frame**		**Cylindrical Frame**		**Triangular Prism**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Ease of Assembly/Disassembly	1	3	3	2	2	2	2	3	3
Volume	3	3	9	3	9	5	15	1	3
Ground Clearance	2	3	6	4	8	1	2	4	8
Strength	3	3	9	5	15	2	6	5	15
Total		**27**		**34**		**25**		**29**	

The leading design shape for the chassis is the truss frame. This design is harder to assemble than the box and triangular prism options, but excels in the strength category. The truss frame also allows for smaller components to be taken in and out of the frame. This also allows the SCAR to extend and retract from within the frame. This design also had moderate ground clearance.

The second major design decision compared the materials used to make the connection blocks and rods. The block and rod materials were analyzed in two separate tables. Table 44 compares the rod material and connection type. Ease of assembly and disassembly, weight, and strength were analyzed. The options compared were carbon fiber rods glued to the blocks, aluminum rods fastened with screws to the blocks, carbon fiber rods glued to the blocks with threaded aluminum rods for cross supports, and aluminum rods connected using set screws.

Table 44: Chassis Rod Material DDM

Design		**Carbon fiber rods, glued in place**		**Aluminum rods fastened using screws**		**Carbon Fiber rods glued to blocks with Aluminum rods for cross connections**		**Aluminum rods fastened with a set screw**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Ease of Assembly	1	3	3	5	5	5	5	3	3
Weight	3	3	9	1	3	2	6	3	9
Strength	2	3	6	4	8	5	10	4	8
Total		**18**		**16**		**21**		**20**	

The leading rod material and design was determined to be the carbon fiber rods glued to the blocks. Tapped aluminum rods will cross the truss, connecting one side of the frame to the other. The aluminum rods will be fastened to the blocks using screws. The carbon fiber rods provide a significant amount of strength while conserving weight.

Table 45 displays different alternatives for the chassis connection blocks. The options consisted of aluminum, 3D Printed materials, wood, and steel. The blocks must conserve weight, provide sufficient strength, and be durable.

Table 45: Chassis Connection Blocks DDM

Design		**Aluminum**		**3D Printed Material**		**Wood**		**Steel**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Weight	3	3	9	5	15	4	12	1	3
Strength	3	3	9	1	3	1	3	4	12
Durability	2	3	6	1	2	2	4	4	8
Total		**24**		**20**		**19**		**23**	

Aluminum blocks were determined to be the leading design. The aluminum blocks would conserve weight and are stronger and more durable than the other options besides steel. However, steel is much heavier and the additional strength and durability may not be needed. Testing will determine the strength requirements for the blocks.

The final chassis design will include carbon fiber rods glued into the connection blocks to form a truss frame. The blocks will be made from aluminum. Aluminum rods will be fastened between each side of the chassis, connection the frame together. A list of chassis components is found in Table 46.

Table 46: Chassis Assembly Weight

Component	Expected Unit Weight (lbf)	Quantity	Expected Total Weight (lbf)
Carbon Fiber Rods, 3.5 in. Long	0.006	16	0.096
Carbon Fiber Rods, 2.5 in. Long	0.005	14	0.070
Tapped Aluminum Bars	0.050	9	0.450
Aluminum Connection Blocks, A	0.060	4	0.240
Aluminum Connection Blocks, B	0.050	4	0.200
Aluminum Connection Blocks, C	0.040	10	0.400
Screws	0.005	18	0.090
Carbon Fiber tail	0.050	1	0.050
Rat Trap Spring	0.005	1	0.005
Metal Tail Connection Piece	0.100	1	0.100
Sub-system Total			**1.701 lbf**

4.3.2 Drivetrain

The rover's drivetrain is a sub-system integrated with the chassis which includes its two wheels, two drive shafts, two chassis mounting assemblies, and two drive motors. An exploded view of the drivetrain is shown in Figure 35.

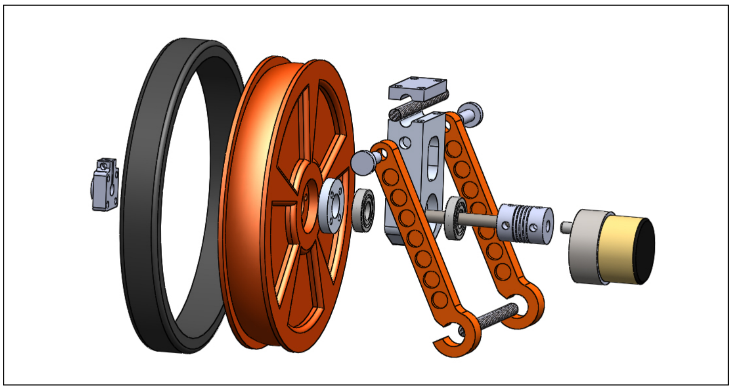

Figure 35: An exploded view of the drivetrain assembly.

The motors are the elements of the drivetrain most closely tied to the rover's success. The team is opting to buy commercial DC motors for simplicity's sake and since there are many small form factor motors available on the market with specifications in our range.

For torque requirement calculations, the rover's weight was assumed to be 10.23 lbf. This is the most recent approximation. The rover's mass is likely to change as the team finalizes designs and learns from payload testing, but this change will be a decrease, giving an excess of power. With a desired maximum speed of 1 ft/s, an acceleration time of 2 s, a maximum expected slope of 10°, and a 3 in. wheel radius, the required torque is 92.90 oz-in. Since the design has two independent motors, each is only required to output 46.45 oz-in of torque. The maximum speed and wheel radius also dictate a motor Rotations per Minute (RPM) of 38.2. Inefficiencies in the system will be present, so applying a safety factor of 2, the motor to be used should have a stall torque above 90 oz-in. However, on flat ground, the motors only require 3.82 oz-in each. With these numbers in mind, the options explored were:

- The **GHM-04 spur gear motor** (13.9 oz-in rated torque, 125 oz-in stall torque, 146 rpm, weighs 4.30 oz, 1.47 in. diameter, 1.65 in. length, $21.95)
- The **RS-775 motor** (10 oz-in rated torque, 125.69 oz-in stall torque, 6070 rpm, weighs 12.30 oz, 1.81 in. diameter, 2.62 in. length, $11.99)
- The **Pololu metal gearmotor** (170 oz-in stall torque at 12 V, weighs 7.94 oz, 1.46 in. diameter, 2.76 in. length, $39.95)

The scoring criteria include:

- **Cost**, since the rover will employ two of them and funding is limited
- **Diameter**, since the size of the rover's chassis interior is a hard limit
- **Length**, since extra length will unnecessarily take away from the space dedicated to other rover systems
- **Rated RPM**, since the wheels must spin at 38.2 rpm to meet the desired speed
- **Rated torque**, since the rover weighs 10.23 lbf and must be able to navigate gentle slopes
- **Weight**, since extra weight will affect the performance of both the rover and the launch vehicle

The results of the DDM are shown in Table 47. The GHM-04 is in the middle of the pack when it comes to cost, but appears to be the most appropriate choice for our application. The other two choices have high RPM ratings, but even the GHM-04 is adequate with regards to RPM as explained previously. All three motors also have enough torque, so the deciding metrics are size and weight. After all, a motor that exceeds the performance requirements can usually be replaced with a smaller one. This is the case here, because the RS-775 and Pololu are both heavier and longer than the GHM-04. It is clear that neither should be chosen instead of the GHM-04.

Wheel construction is another decision to be made. The options explored were:

- A **solid disk** (a circular wheel with a solid interior region)
- A **ring-shaped wheel** (a circular wheel with a spoked interior)
- A **solid noncircular wheel** (a wheel with a solid interior region but a noncircular profile)

Table 47: Drivetrain motor selection DDM

Design		GHM-04		RS-775		Pololu	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Cost	2	3	6	4	8	1	2
Diameter	1	3	3	3	3	3	3
Length	3	3	9	1	3	1	3
Rated RPM	2	3	6	5	10	4	8
Rated torque	4	3	12	3	12	4	16
Weight	3	3	9	1	3	2	6
Total		**45**		**39**		**38**	

- A **triangular conveyor** (a three-sided "wheel" with a moving track along its perimeter)

The scoring criteria include:

- **Effectiveness as a barrier to debris**, since foreign material should not interfere with the drivetrain mechanism
- **Ease of manufacturing**, since the team has constraints with respect to time and machining equipment
- **Relative torque required**, since some wheel types will require more torque to drive than equal-size disks
- **Weight**, since extra weight will affect the performance of both the rover and the launch vehicle

The results of the DDM are shown in Table 48. A ring-shaped wheel might offer a weight benefit, but would need to be made of a very stiff material to hold up against dynamic flight forces. It also introduces the problem of preventing foreign matter from the competition site from gumming up the interior of the rover.

A noncircular wheel might look like an octagon or an asterisk. It could provide traction, but a simple foam wrap does that equivalently with reduced manufacturing complexity and torque penalties. The last wheel type discussed was a triangular conveyor, which is a much more complex design which would indirectly add weight to the rover (by way of a more involved transmission design) without much tangible benefit.

The team's expected ejection method also influences this decision. The airframe is circular and the payload will be forced out of the fore section with a pressure ejection. A solid circular wheel with foam around its perimeter lends itself well to this system; choosing any of the other options would require significant modifications to the PEARS-rover interface. A solid disk is the leading design choice.

With a solid circular wheel chosen, the next decision concerns the material of the wheels. The options explored were:

Table 48: Wheel Type DDM

Design		Solid disk		Ring-shaped		Solid non-circular		Triangular conveyor	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Barrier to debris	2	3	6	1	2	3	6	2	4
Ease of manufacturing	3	3	9	2	6	2	6	1	3
Number of parts	3	3	9	2	6	3	9	1	3
Torque required	3	3	9	4	12	2	6	3	9
Weight	2	3	6	5	10	3	6	3	6
Total		**39**		**36**		**33**		**25**	

- **ABS**
- **Aluminum**
- **HDPE**
- **PLA**

The scoring criteria include:

- **Cost**, since the wheels require a lot of material and funding is limited
- **Machinability**, since modifications may need to be made depending on the manufacturing method
- **Rigidity**, since the wheels are the support structures of the rover
- **Suitability for complex shapes**, since the wheel design includes fillets and a groove along its perimeter
- **Weight**, since extra weight will affect the performance of both the rover and the launch vehicle

The results of the DDM are shown in Table 49. Even though aluminum is a good choice for lightweight metal components, it is still twice as dense as the three plastics the team is considering. The wheels are the largest components in the drivetrain, so using aluminum adds a significant weight penalty. Also, the recess cut into the perimeter of the wheel would need to be machined on a lathe. Manufacturing complexity is a significant barrier here for aluminum and HDPE. ABS and PLA, by contrast, can be bought as filament and used with additive manufacturing. These two thermoplastics are very similar by most measures including cost and density, but ABS has slightly better impact resistance. ABS is chosen as the wheel material, but PLA could be used if available in club resources.

The drive shaft material has a few common choices:

- **Aluminum**
- **Carbon fiber**
- **Carbon steel**

Table 49: Wheel material DDM

Design		ABS		Aluminum		HDPE		PLA	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Cost	2	3	6	2	4	3	6	3	6
Machinability	2	3	6	4	8	4	8	3	6
Rigidity	2	3	6	4	8	3	6	3	6
Suitability for complex shapes	2	3	6	1	2	1	2	3	6
Weight	3	3	9	2	6	3	9	3	9
Total		**33**		**28**		**31**		**33**	

The scoring criteria include:

- **Cost**, since funding is limited
- **Machinability**, since modifications may need to be made depending on the manufacturing method
- **Stiffness**, since the shaft will be subject to torsional and bending stresses;
- **Weight**, since extra weight will affect the performance of both the rover and the launch vehicle

The results of the DDM are shown in Table 50. The material initially proposed for the drive shaft was steel, but it will need work on the lathe to generate the proper step dimensions. Carbon steel is difficult to machine, while something like aluminum is more manufacturable. Carbon fiber and aluminum are much less dense than steel, and aluminum is cheaper than both. As long as further simulations and tests do not reveal any strength concerns, the drive shaft will be machined from aluminum.

Table 50: Drive shaft material selection DDM

Design		Carbon steel		Carbon fiber		Aluminum	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Cost	1	3	3	1	1	4	4
Machinability	3	3	9	2	6	4	12
Stiffness	3	3	9	3	9	2	6
Weight	2	3	6	5	10	5	10
Reliability	3	3	9	2	6	3	9
Total		**36**		**32**		**41**	

The initially proposed drivetrain-chassis interface simply involved one attachment point to an upper truss member, so the systems were susceptible to undesirable displacement. The team considered another attachment point for each half of the drivetrain on the lower member of each side of the chassis. Options for an attachment scheme were:

- **ABS connector with a padded hook**
- **Aluminum connector with a padded hook**
- **Elastic connector**
- **No connector**

The scoring criteria include:

- **Ease of manufacturing**, since the team has constraints with respect to time and machining equipment
- **Metal content**, since metal components pose a high safety risk in the event of a mishap
- **Support**, since the drivetrain-chassis assembly should be fixed
- **Suspension properties**, since the choice could mitigate vibrations in the chassis
- **Weight**, since extra weight will affect the performance of both the rover and the launch vehicle

The results of the DDM are shown in Table 51. An elastic connection is simple and would provide some suspension characteristics, but no support. A rigid connector of aluminum or ABS (Figure 36) gives support to the system and a hook-type end can have an elastomer bonded to it for a certain amount of give. ABS is also easily manufactured and relatively light. The drivetrain mechanism will go forward with an ABS connector rod.

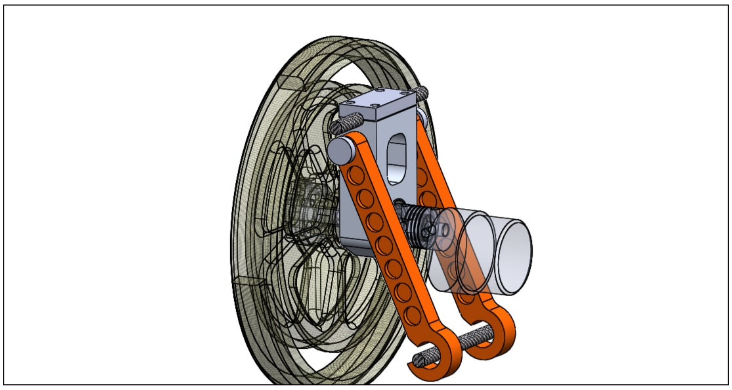

Figure 36: ABS rods connecting the mounting block to the lower truss member.

Another design choice to be made is the manner of attachment of each drive shaft to its wheel. This interface is critical in making sure as much as possible of the generated torque is transmitted to the wheels. The options explored for this interface were:

- **Clamping hub** (an aluminum component with a bore through the center which matches the drive shaft diameter when the hub is tightened with a hex screw; fixed to the far ends of the wheels with four 6-32 screws)
- **Keyed shaft** (a slot cut into the shaft accepts a parallel key which rotates along with it, transmitting torque to the wheel)
- **Fixed shaft** (the drive shaft is simply bonded to the center of the wheel using an epoxy)

Table 51: Drivetrain-chassis interface scheme selection DDM

Design		None		Aluminum connector with padded hook		Elastic connector		ABS connector with padded hook	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Ease of manufacturing	2	3	6	2	4	5	10	3	6
Metal content	1	3	3	1	1	5	5	5	5
Support	3	3	9	5	15	1	3	5	15
Suspension properties	2	3	6	4	8	4	8	4	8
Weight	1	3	3	1	1	5	5	2	2
Total		**27**		**29**		**31**		**36**	

The scoring criteria include:

- **Axial length**, since additional material beyond the far ends of the wheels increases the roverâĂŹs length and, ultimately, that necessary for the payload bay
- **Ease of disassembly**, since one of the major design goals is a more modular system
- **Ease of manufacturing**, since the team has constraints with respect to time and machining equipment
- **Number of parts**, since more parts generally increase complexity and maintenance time
- **Reliability**, since part failures will prevent the rover from carrying out its mission at competition

The results of the DDM are shown in Table 52. A fixed connection to the wheel is a simple design, but would rely purely on an adhesive to transmit torque. It risks shear failure. A keyed joint is effective and commonly used in shafts for torque transmission, but would require a keyway to be cut into the drive shaft and may not be disassembled easily. The clamping hub solution, with four fasteners going through the wheel (Figure 37, is simple to take apart and reassemble. Since modularity is one of the goals for the drivetrain assembly, the small axial length penalty is acceptable. The clamping hub is a simple, reliable option so it is considered to be the leading alternative.

The last drivetrain decision to be discussed is the type of foam tread to be applied to the wheels. The foam must be compressible so that the rover and its surrounding carbon fiber wrap will fit into the 6.250 in. airframe and offer a good pressure seal for PEARS. It also needs to expand significantly upon its ejection to maximize the rover's ground clearance. The team has reached out to companies like Rathbun and Rogers Corporation who provide expanding urethane foams with these qualities and is working to acquire test samples to compare foam choices.

Table 53 lists the drivetrain components and estimated masses for each. Component drawings can be found in Appendix A.

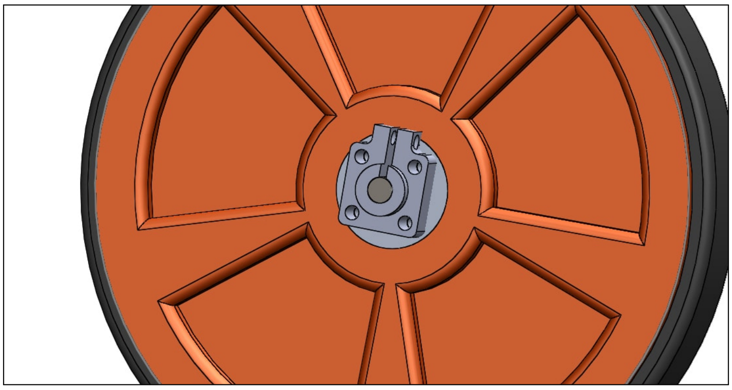

Figure 37: Clamping hub assembly.

Table 52: Wheel-shaft interface selection DDM

Design		Clamping hub		Fixed		Keyed joint	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Axial length	1	3	3	4	4	4	4
Ease of disassembly	3	3	9	1	3	2	6
Ease of manufacturing	2	3	6	5	10	1	2
Number of parts	1	3	3	5	5	2	2
Reliability	3	3	9	2	6	4	12
Total		**30**		**28**		**26**	

Table 53: Drivetrain Assembly Weight

Component	Expected Unit Weight (lbf)	Quantity	Expected Total Weight (lbf)
GHM-04 Spur Gear Motor	0.269	2	0.538
Shaft Coupling	0.034	2	0.068
Clamping Hub	0.016	2	0.032
Wheel	0.814	2	1.628
Urethane Foam Strip	0.096	2	0.192
Drive Shaft	0.015	2	0.030
Mounting Block Top	0.022	2	0.044
Mounting Block Bottom	0.194	2	0.388
Mounting Block Extension	0.006	4	0.024
Wheel Hub Plate	0.015	4	0.060
Connector Rod	0.030	4	0.120
Annular Ball Bearing	0.020	4	0.080
Misc. Fasteners	-	-	0.150
Sub-system Total		**3.354 lbf**	

4.3.3 Soil Collection and Retention System

The requirements for the SCAR are to autonomously collect and seal at least 10 mL of soil. There were three main designs for both the soil collection and soil retention methods. These designs were compared using a Decision Matrices, where the design that rated the highest was selected as the leading design choice.

4.3.3.1 Soil Collection

There are a few requirements and specifications for the soil collection system. It needs to be capable of collecting multiple soil types in different conditions,transporting the soil from the ground to the soil container, must be compact, and it must withstand an estimated 50G of force during flight operations. In Table 54, the design matrices for the Soil Collection method can be seen, indicating the auger as the leading design choice for the soil collection portion of the payload experiment.

Table 54: Soil Collection DDM

Design		**Auger**		**Shovel-like Contraption**		**Vacuum**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Able to collect all soil types in all conditions	5	3	15	2	10	1	5
Can transport soil from ground to container	5	3	15	2	10	5	25
Compact	3	3	9	2	6	3	9
Ability to withstand 50G of force	4	3	12	3	12	2	8
Total		**51**		**38**		**47**	

The shovel-like contraption would use a rotating shovel-head that collects soil when the head is in the downward position and rotate upward where it would drop off the soil into a container (similar in design to a water mill). Using a shovel-like contraption to collect the soil would allow for a significant amount of soil collected at once, however, that is not necessary. There is also a significant amount of research and pre-manufactured parts that can be purchased for this type of mechanism. This design allows for soil to easily be transported from the ground into a separate container, allowing for an additional scientific experiment to be performed upon the soil if desired. Another benefit with the shovel is the ability to collect many types of soils and in different conditions. However, this design would not be compact, as the full circular rotation of the shovel would require a lot of space.

The vacuum design would use a vacuum to collect the soil off the ground and store it in an internal storage compartment. This design could be easily bought from other manufacturers and doubles as a soil collection and a soil retention method. However, the vacuum would have problems collection soil that is either too heavy or too large to fit within the nozzle, such as mud or clay, two types of soil observed at competition last year. Also, this experiment would restrict the team from performing an additional scientific experiment, if desired.

The auger design was the leading design. It allows for soil collection to happen in a small space. This design is able to break apart large portions of soil by using a sharp edge at the base of the auger and can easily transport soil samples from the ground to the soil container by wrapping a flexible sheet around the auger, preventing the soil from falling out the sides. If needed, this design can also serve as a soil retention method by sealing the soil within the auger body. One downside of this design is that some parts may need to be 3D printed, which could result is structural strength problems.

4.3.3.2 Soil Retention

For soil retention, the three options were based upon the different door designs and their ability to seal the soil efficiently. The team had decided that it was going to use a box at the retainer. The three options are compared within the Decision Matrix in Table 55.

Table 55: Soil Retention DDM

Design		**"Center-pivot" Doors**		**"Edge-pivot" Doors**		**Spring-Assisted Doors**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Must seal the soil	5	3	15	1	5	3	15
Compact	3	3	9	4	12	2	6
Strength	4	3	12	2	8	3	12
Total		**36**		**25**		**33**	

Between the three design options, the center-pivot doors were the leading design choice as compared to the edge-pivot doors and the spring-assisted doors.

After the Preliminary Design Review (PDR) Question and Answer session, the team was able to calculate the weight of 10 mL of soil, which came out to about 0.07 lbf. For a spring-assisted door, this would require perfectly tuned springs that could not be affected by dynamic forces, which is not a realistic scenario. Any plastic deformation within the spring coils would hinder the spring constant and possible yield the spring useless and unable to seal the soil, as per the requirements. However, this design would negate the need for any motors, saving space and reducing the chance of electrical failures.

The remaining two designs are very similar and differ in the location of the motor shaft. These two designs can incorporate a top and bottom door, allowing for an exit space where the soil could be released from the container onto a platform for an additional scientific experiment. These designs are also not affected as much by dynamic forces during flight. A downside to these designs is the requirement to have one motor per door (two motors per design). This increases the chance of electrical problems and adds weight to the rover. The design with the doors pivoting from the edge of the container would require significantly more space than the center-pivot doors and would also require more torque to operate the doors as the moment arm is larger. However, the edge-pivot doors would be slightly more consistent in opening as soil could not prevent their rotation.

For the SCAR, the leading design choices are to use an auger to collect the soil and a center-pivot door to retain the soil.

4.3.3.3 Final Design

The completed SCAR assembly can be seen in Figure 38 and Figure 39.

Figure 38: Soil Collection Assembly

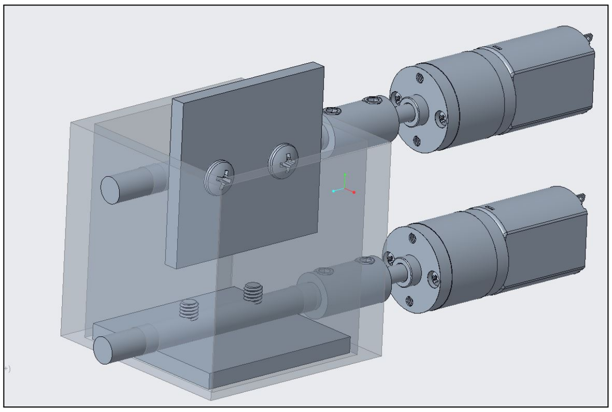

Figure 39: Soil Retention Assembly

The soil collection assembly will use an auger to drill into the ground and extract soil. In order to retain the soil from falling from the sides of the auger, a carbon fiber wrap will enclose the auger. Shown in Figure 40 are the auger, auger bar, coupler, and motor.

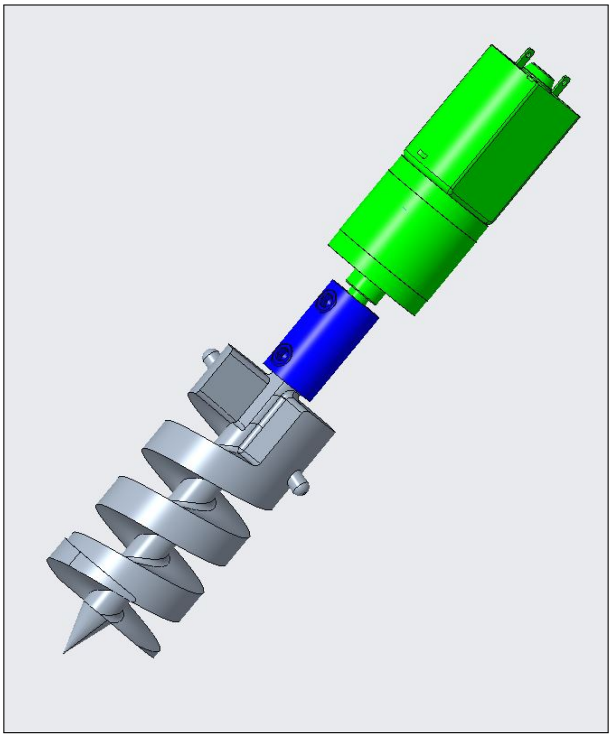

Figure 40: Soil Collection Assembly featuring the auger, auger bar, coupler, and motor

An enclosure for the motor was designed to allow for two screws to fit on each side. The motor enclosure is shown in Figure 41.

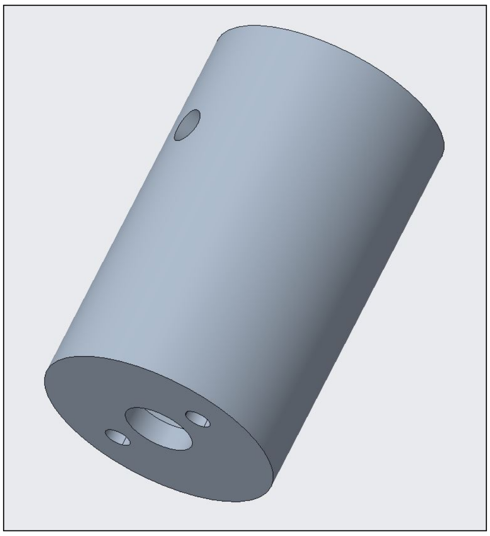

Figure 41: Motor enclosure with screw holes

The auger contains a hole near the top surface angled at a 15 degree angle. A bar will be placed through this hole as shown in Figure 40.

Using the two screws in the motor enclosure and the bar in the auger, a tube will be manufactured to feed the auger in and out of the ground. This design will allow the auger to rotate while the motor can only move in a linear path, causing the assembly to extend and retract from the rover while the motor is operated. Shown in Figure 42 is the Inner Auger Tube. In Figure 43, the Inner Auger Tube is displayed in blue surrounding the parts of the assembly inside it.

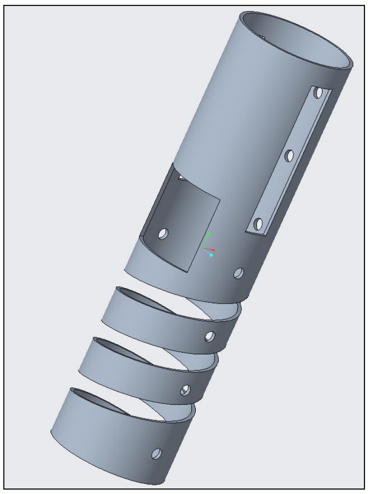

Figure 42: Inner Auger Tube (Empty)

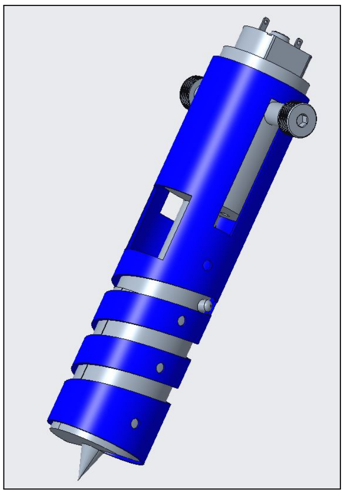

Figure 43: Inner Auger Tube (Assembly)

To provide structural support, a second tube will surround the Inner Auger Tube. Rubber washers will rest between the inner and outer tubes with screws entering from the exterior of the overall assembly. This should prevent the Inner Auger Tube from flexing and restricting the auger operation.

As shown in Figure 38, an opening was placed towards the centers of the tubes. This is the location where the soil will fall from the auger and onto the soil container.

The soil collection assembly will measure 4.86 in. in length with a 1.375 in. diameter (not including the external screw heads). The assembly will weigh 1.5 lbf.

The soil container shown in Figure 39 will have two doors, one on top and one on bottom, allowing for soil to be sealed and distributed. A motor will be used for each door. Not shown in the Figure **??**, screws will be through the sides of the container to act as hard stops for the doors, preventing them from over-rotating. Also, foam will be lining the edges of doors to prevent soil from falling through the cracks.

The soil retention container will be 2 in. x 2 in. x 2 in. with motors on the sides adding another 1.99 in. to the assembly. Overall, the soil retention assembly will weigh 1.45 lbf.

In total, the SCAR will weigh 2.95 lbf.

Individual component weights can be found in Table 56.

Table 56: SCAR Assembly Weight

Component	Expected Unit Weight (lbf)	Quantity	Expected Total Weight (lbf)
Auger	0.2	1	0.2
High-Strength 1045 Carbon Steel Rod	0.1	1	0.1
Carbon Fiber Auger Wrap	0.1	1	0.1
Motor Bar (motor enclosure)	0.2	1	0.2
Set Screw Shaft Coupler	0.1	3	0.3
26 RPM Mini Econ Gear Motor	0.2	3	0.6
Short-Thread Alloy Steel Shoulder Screw	0.025	2	0.05
Inner Tube	0.2	1	0.2
Outer Tube	0.2	1	0.2
Viton Fluoroelastomer Rubber Sealing Washer	0.0045	11	0.05
18-8 Stainless Steel Washer	0.0045	11	0.05
Black-Oxide Alloy Steel Socket Head Screw	0.0045	11	0.05
Low-Carbon Steel Sheet	0.7	1	0.7
Low-Carbon Steel Rod	0.05	2	0.1
Passivated 18-8 Stainless Steel Pan Head Phillips Screw	0.0045	11	0.05
Sub-system Total			**2.95 lbf**

4.3.4 Payload Ejection and Retention System

The requirements for the PEARS are to retain the payload safely inside the airframe during the entire flight of the vehicle until it is safely on the ground. Once given clearance, the system must be triggered to release the payload and eject it from the airframe. Additionally, the PEARS system must be able to armed once fully integrated into the airframe and ready to launch. The following sections detail the designs considered for each system requirement, their evaluations, and the final designs chosen.

The first decision needed in deciding the PEARS methods was the payload location within the airframe as well as whether the end of the airframe would be open after apogee separation. The payload was decided to be placed within the fore section of the airframe because placing it within the nosecone would create a very high center of gravity, and placing in the aft section along with the motor would make the launch vehicle very bottom heavy. Placing it in the fore section allows for optimal space for the payload as well as the PEARS without interfering with other internal components. While the payload would be more secured if the fore airframe was fully closed after apogee separation and until landing, the OSRT decided to have the aft end of the fore airframe open after apogee separation from the aft section. This allows the payload to be ejected right out of the airframe upon safe landing rather than having to first set of an additional ejection charge to open the airframe.

4.3.4.1 Retention Method

With the end of the airframe open after apogee separation, it is extremely important to have a robust retention system which securely retains the payload while still allowing for easy detachment upon landing. Two main systems were considered in depth and evaluated. The first system was implemented by the OSRT in last year's competition and consists of retention linking devices which can be released with a small black powder charge. These retention devices are most commonly seen in delayed parachute deployment. A Advanced Retention Release Device (AARD) and two L2 Tender Descenders were combined and are connected to a Kevlar harness which wraps around the payload. Upon safe landing, the retention devices can be remotely triggered to release the harness, and the payload will be able to be ejected easily. The second retention system investigated consists of a stepper motor retained bulkhead which separates the payload from the open end of the airframe. The stepper motor operates a high strength steel key which locks a loose bulkhead against the wheel of the payload. When given the remote signal upon landing, the motor will rotate the key, releasing the loose bulkhead. When the payload is ejected, it pushes the bulkhead out the end of the airframe along with the rover.

Once the designs of these systems were thoroughly conceptualized and iterated, they were compared against each other using the shown in Table 57. Requirements considered for retention are the ease of integration into the airframe, the safety of the system, the structural impact of the system, the reliability of the system to retain and release the payload when needed and the overall weight of the system. Since the AARD and Tender Descender system was implemented previously by OSRT, it was used as the baseline. While having the motor inside the airframe and having to lock in the bulkhead is not easy, wrapping the Kevlar harness around the payload, connecting it to the retention devices, and inserting the system into the airframe is more difficult. While there are many safety concerns when dealing with black powder, these retention devices are well tested and do not present much of a risk, as a result the motorized system, while safer, is only so by one point. Where the harness system beats the motorized system is in the structures, reliability, and weight added to the system. The structures needed to mount the motor in such a way that the payload can still pass by when ejected would create significant problems and add a lot of weight to the system. Additionally, the AARD and Tender Descenders are made for HPR and as long as the e-match connection is secure, are very reliable throughout a flight. A motorized system has more components which can be damaged during flight and recovery and as such is less reliable.

The results of the DDM indicate that the AARD and Tender Descender system is the better option, and as such was chosen for our preliminary design. The methods of how this system integrates with the overall PEARS will be discussed further after evaluating the other sub-system components.

Table 57: Payload Retention DDM

Design		AARD and Tender Descender with Kevlar Harness		Motor Retained Bulkhead	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score
Ease of Integration	5	3	15	4	20
Safety	6	3	18	4	24
Structural Impact	4	3	12	2	8
Reliability	6	3	18	2	12
Weight	3	3	9	1	3
Total		**72**		**67**	

4.3.4.2 Ejection Method

Upon retention system release, the payload will still be positioned freely inside the airframe, and needs to be ejected in order to complete the soil collection mission. Three designs were considered for the system: a black powder ejection, a spring ejection, and a lead screw ejection. These systems were evaluated against each other using the same metrics as the retention system: ease of integration, safety, structural impact, reliability, and weight. Because a black powder ejection system was used by the OSRT previously, it was chosen as the baseline to compare the other systems as can be seen in Table 58.

The black powder ejection charge will be placed between the payload and a pressure sealed bulkhead. Once released, the ejection charge will be triggered, pushing the payload out the open end of the airframe. This system is extremely simple and adds very little weight to the launch vehicle, it is very reliable and is easy to integrate into the system. It does present a safety risk as it is a small explosion, however with proper handling and protection it does not present too large of a risk. It does impact the structures of the system as a pressure seal is required to eject the payload.

The second system evaluated is a spring ejection. During airframe integration, a large spring will be compressed between the payload and a fixed bulkhead. During flight, the retention system will hold the spring in a compressed position. When given the signal to release, the spring expands to natural position accelerating the payload out of the end of the airframe and able to complete its mission. While this system is safer than the black powder ejection, it falls short in all other evaluated categories. The spring would require a high spring constant in order to eject the payload, and compressing it during integration would be difficult, as well as extra internal structures which would also increase the weight. Additionally, springs can be easily deformed especially if abnormal flight forces are experienced, and so the reliability of the system is lower than the very common black powder ejection.

The final system evaluated was a lead screw ejection. This system would consist of a long lead screw with a platform for the payload to sit against. Once the retentions are released, a motor would actuate the lead screw along a linear path, slowly pushing the payload to the end end of the airframe. This system is extremely safe as the are no components with chemical or potential energy like the other two options. However, having a very long lead screw causes issues in integration and structures as it must be able to pass through other components when the payload is fully integrated. The system is very reliable as the motor only has to rotate the lead screw. However, the weight this system would add is significant. The lead screw would need to be of a high strength material such as steel to withstand the flight forces and not get bent or have threads stripped. Adding an additional motor to the system as well as a way to retain the motor would further increase the weight.

The results of the DDM indicate that the black powder ejection system is the best option, and as such was chosen as our leading design. The methods of how this system integrates with the overall PEARS will be discussed further after evaluating the remaining sub-system component.

Table 58: Payload Ejection DDM

Design		**Black Powder Ejection**		**Spring Ejection**		**Lead Screw Ejection**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Ease of Integration	5	3	15	2	10	3	15
Safety	6	3	18	4	24	5	30
Structural Impact	4	3	12	2	8	1	4
Reliability	6	3	18	2	12	3	18
Weight	3	3	9	2	6	1	3
Total		**72**		**60**		**70**	

4.3.4.3 Arming Method

All of the ejection and retention systems described above would require a way to power on an off the PEARS once fully integrated into the airframe to save battery charge, it is a critical feature with the chosen retention and ejection systems. Because these systems involve a black powder charge, they cannot be armed before integration as that is a significant safety risk. Additionally, even if the system is not powered, an electric charge can be built up when integrating the system into the airframe from the friction between the PEARS and airframe. While not likely, this electric charge could ignite the retention or ejection charges which could cause serious injury. There were three arming methods considered and evaluated based on their safety, ease of integration, impact to the structures, and total weight added to the system.

The system used as the baseline for evaluation was the method implemented by the OSRT at last year's competition. This system used a switch which was permanently attached to the interior of the airframe and was accessible through a hole in the airframe. The switch had an electrical tether attached to PLEC, which kept the PEARS attached to the airframe making both the assembly of charges and payload difficult as well as the integration into the airframe. Additionally, the switch used did not account for the potential for charge build up, and so was not safe. The one benefit of this design is the pressure seal created with the permanent switch.

The first alternative evaluated was adding an access hatch in the airframe with a terminal block as can be seen in Figure 44. This access hatch makes the integration of the system much easier as it allows an additional access point for positioning and securing the PEARS into the airframe. It also allows for a simple terminal block in which the circuit can be fully disconnected until integrated. When ready to arm the system, the hatch can be removed, the circuit tested with a multimeter to determine if a charge was created, and once any charge is dissipated, the terminal can be connected and the hatch closed. This is a safe way of arming the system, however the hatch has a large structural impact both in manufacturing and aerodynamic performance. Finally, it also increases the weight significantly because of the need for an inner coupler to mount the hatch to. A disadvantage of this system is the need for an additional pressure seal elsewhere in the system to still allow the payload to be ejected using a black powder charge.

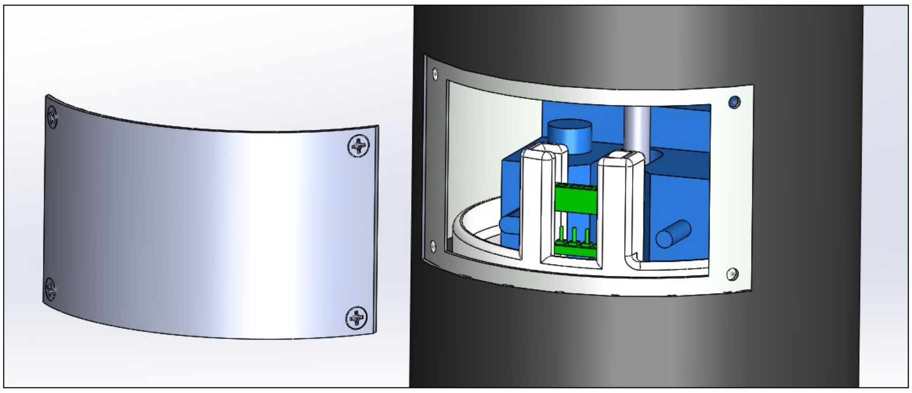

Figure 44: Arming Hatch in Fore Airframe for PLEC

The final arming system evaluated is a simple Single Pole Double Throw (SPDT) switch which is inserted and removed along with the PEARS as seen in Figure 45. The SPDT can shunt the ejection and retention circuit when in one throw position, dissipating any electrical charge build up during integration. This system is much safer since the charge build up in mitigated, and the weight remains very similar. Once downside to this system is the need for an additional pass-through bulkhead in order to maintain a pressure seal between the ejection charge and SPDT switch access hole in the airframe.

The results of the DDM indicate that the SPDT switch is the best option, and as such, was chosen for our preliminary design. The methods of how this system integrates with the overall PEARS will be discussed further after reviewing the PLEC in the following section.

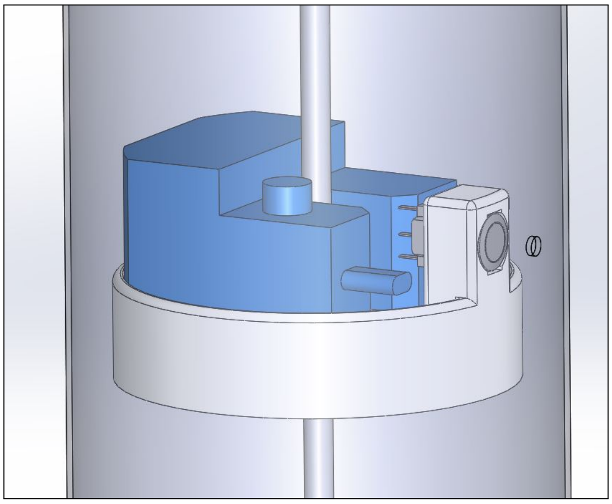

Figure 45: SPDT Switch in Fore Airframe for Arming the PLEC

Table 59: PEARS Arming DDM

Design		**Permanent Power Switch on Airframe**		**Hatch with Terminal Block**		**SPDT Switch with Access Hole**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Ease of Integration	5	3	15	4	20	3	15
Safety	6	3	18	4	24	4	24
Structural Impact	4	3	12	2	8	3	12
Weight	3	3	9	2	6	3	9
Total		**54**		**58**		**60**	

4.3.4.4 Payload Ejection Controller

In order to remotely activate the retention devices and ejection charge, the PEARS must include a controller, referred to as the PLEC. The PLEC will be nearly the same as was used by the OSRT last year. Instead of the simple open or closed switch, based on the decision matrix for the arming method discussed above, a SPDT switch will be added in addition to a simple shunt to dissipate any charge built up upon integration. The top level view of the PLEC can be seen in Figure 46.

The PLEC has a Teensy 3.6 hub which communicates with the RF ground station through an XBee Pro 900HP. The XBee transmits and received with a bandwidth of 26 MHz centered at 915 MHz. The Teensy has several digital outputs which control ignition of the retention and release devices discussed earlier.

A separate e-match is used to ignite each black powder charge. Therefore, a total of five ignition channels are needed on the PLEC: Three retention devices and two ejection devices. Because the ARRD is a backup device, and the two Tender Descenders are the primary devices, a more redundant circuit was used for the

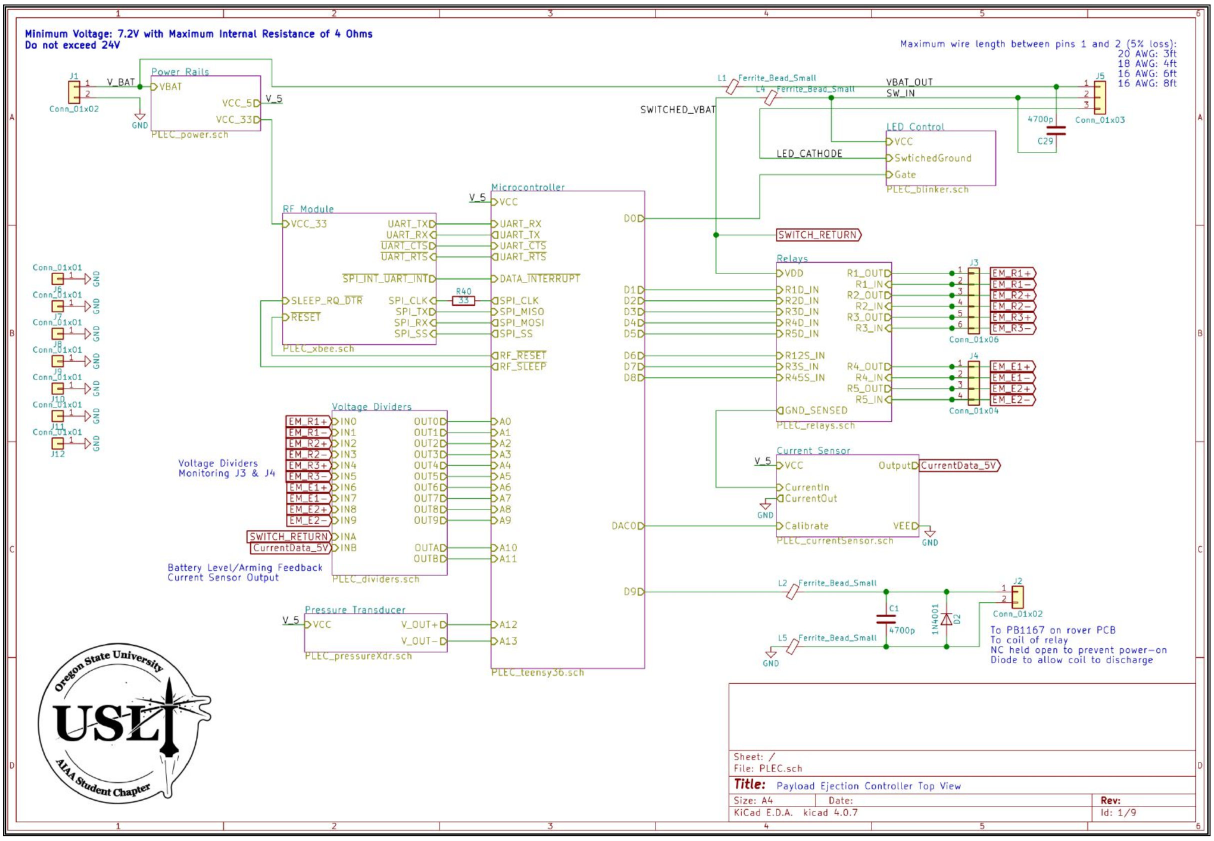

Figure 46: Top Level View of PLEC

ARRD due to an extra channel in the power relay. The Tender Descenders and ejection devices all rely on the same relay circuit. Figure 47 shows the difference in these two circuits. Channels 1 and 2 connect to the Tender Descender e-matches, and Channel 3 connects to the ARRD e-match. Channels 4 and 5 connect to the primary and backup ejection charges, and are functionally identical to Channels 1 and 2.

Each ignition channel requires two signals to arrive from the microcontroller in order to send current to its respective e-match: One to a PB1167 electromechanical relay, and one to a BDW42G NPN Darlington pair transistor chip. The Darlington pair prevents current from flowing to electrical ground, and the relay performs multiple functions. In the rest position, the PB1167 relay shorts the leads of each e-match together, so no voltage can be generated across it. When the relay is energized, one lead of the e-match is pulled to electrical high, and the other is left to float until the Darlington pair pulls it low. Channel 3 has an added safety feature, where even if one relay contact vibrates to a floating position, the second channel continues to short the leads together until both contacts are energized. A resistor-capacitor filter is also present at the terminal block of each channel to absorb Electromagnetic Interference (EMI) over long periods of time. Each electromechanical relay has a fly back diode to mitigate Counter Electromotive Force, AKA Back EMF (CEMF).

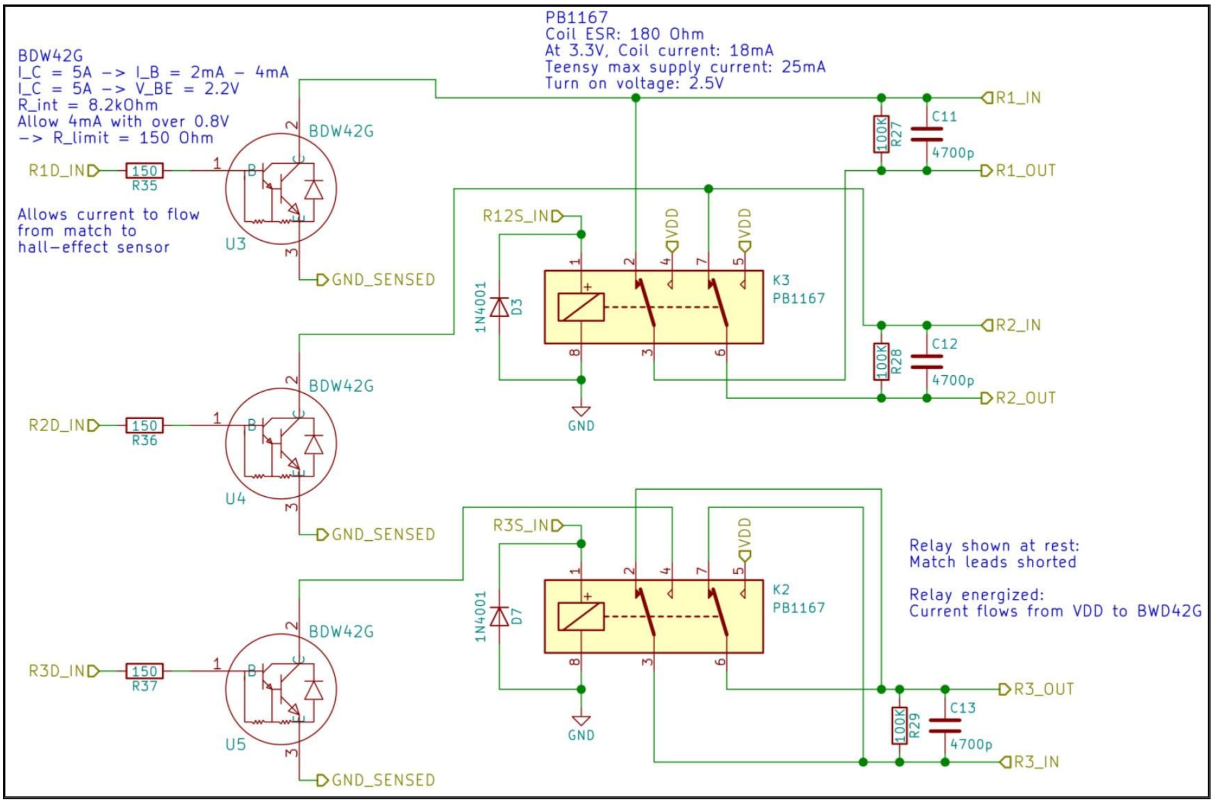

Figure 47: Three of the Five Relay Circuits (4 and 5 are Identical to 1 and 2)

Due to the mechanical design of the rover retention, it is extremely important that the charges are ignited in the correct order; simultaneous ignition would be a very dangerous event. If the retention devices are holding the rover in place when the ejection charges are activated, the rover will suffer catastrophic damage. Therefore, the microcontroller has an array of sensors to detect the ignition of each e-match:

- HXS 20-NP Current Sensor to Measure Current Through
- E-Match 13-bit Differential Voltage Sensors to Measure Voltage of Each Terminal
- Software Integration of Voltage Times Current Over Time to Calculate Energy to E-Match
- NBPMANN150 Pressure Sensor to Detect Rising Chamber Pressure

All of these devices are conditioned to output a 3 V analog signal to the microcontroller for software control. With these sensors, the microcontroller can calculate with high certainty whether the retention devices were activated properly, and proceed to eject the rover. These software safeguards are primarily for repetitive testing, but for safety at competition, all black powder will be burned by the controller after a timeout period. This will ensure no accidental ignition of ejection charges while approaching the forward section of the launch vehicle.

The PLEC communicates with the base station over a 918 MHz RF transmission through an XBee module.

The XBee transmits and receives over a 50 Ohm shielded transmission line to an antenna on the outside of the launch vehicle. The PLEC will be encased in a sheet aluminum Faraday cage, and the openings for other wires will be backfilled with grommets and putty to protect from black powder residue.

Figure 48 is a the printed circuit board (Pour planes hidden). The PCB will be fitted to the forward side of the ARRD bulkhead, and has rounded corners that have a 0.1 in. clearance from the inside of the body tube. A slot is removed from the center to accommodate the threaded rod and corresponding washer and nut that attach to the bulkhead.

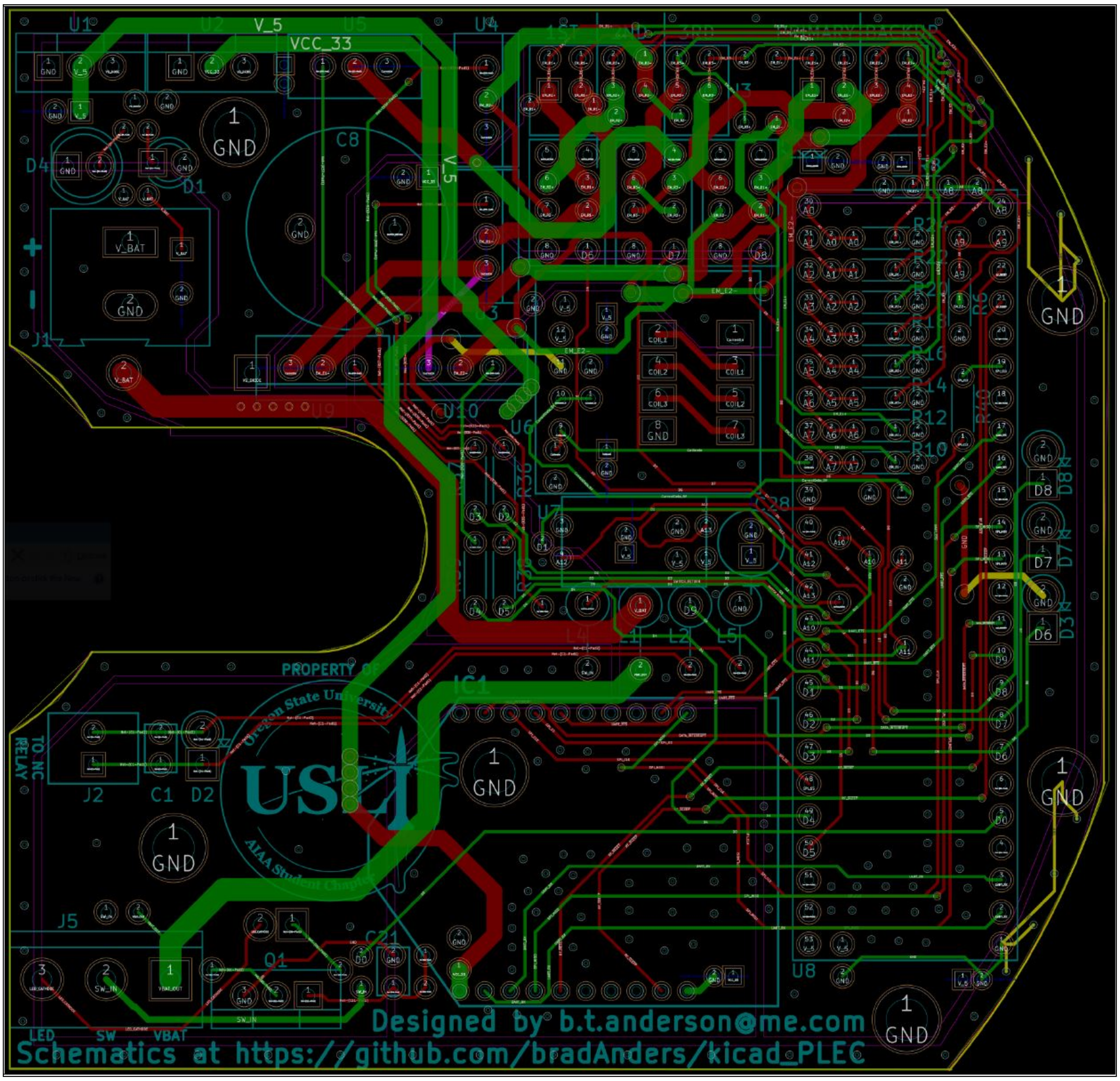

Figure 48: Ordered PLEC PCB

4.3.4.5 Final Design

The completed PEARS assembly can be seen in Figure 49. The payload is enclosed in a carbon fiber wrap before being attached to the assembly. The assembly process is as follows:

1) Check that SPDT switch is shunting ejection circuit
2) Pack and attach AARD and Tender Descenders to bulkhead
3) Wrap Kevlar harness around the payload and payload bulkheads
4) Enclose payload in carbon fiber wrap
5) Attached to wrapped payload to AARD and Tender Descenders
6) Integrate into fore airframe (see following section)

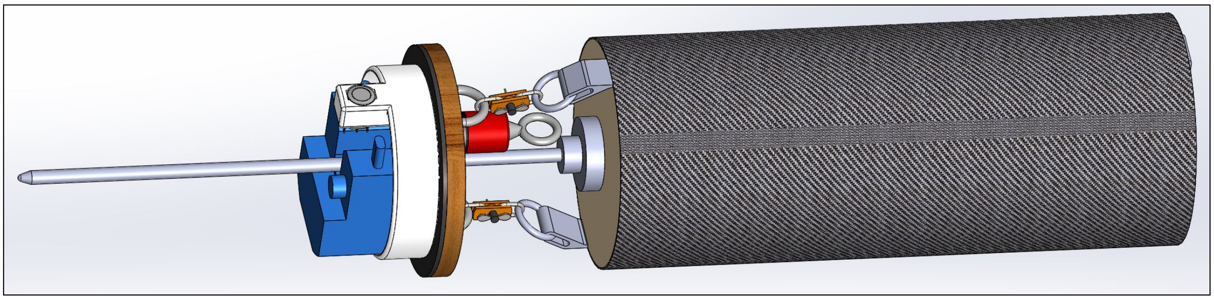

Figure 49: Assembled PEARS prior to airframe integration

The wrap and bulkheads both serve to protect the rover and to create a pressure seal on the inside of the airframe so the rover can be ejected easily. On the opposite side of the bulkhead from the ejection charges is a rubber ring which will also create a pressure seal when integrating into the airframe. The weight of the PEARS can be seen in Table 60.

4.3.4.6 Launch Vehicle Integration

Once assembled as shown in Figure 49, the PEARS will be integrated into the fore section of the airframe via the separation point between fore and aft sections. The Fore Hard Point (FHP) can be seen in Figure 50, and consists of three components. From aft to fore in the airframe the components are:

- Bulkhead with pass through for PLEC
- Additively manufactured funneled bulkhead to guide PEARS threaded rod
- Bulkhead with small hole for threaded rod to pass through and attach to from fore side

The fully integrated system can be seen in Figure 51. The threaded rod that runs through the PEARS passes through the FHP and is secured to the backside of the final bulkhead. The threaded rod continues beyond the bulkhead, and is what the fore ejection bay is mounted on and secured to. When fully integrated, the

Table 60: PEARS Assembly Weight

Component	Expected Unit Weight (lbf)	Quantity	Expected Total Weight (lbf)
Threaded Rod	0.49	1	0.49
PLEC	0.35	1	0.35
Aft PEARS Bulkhead - Loose	0.25	1	0.25
HDPE Rod Cap - Spacer	0.07	1	0.07
Aft Payload Bulkhead	0.25	1	0.25
Fore Payload Bulkhead	0.25	1	0.25
Kevlar Harness	0.025	1	0.025
Carbon Fiber Wrap	0.5	1	0.5
ARRD	0.25	1	0.25
Tender Decender	0.115	2	0.23
Ejection Charges	0.015	1	0.015
3D Printed PLEC Mount	0.31	1	0.31
SPDT Push Botton	0.01	1	0.01
1" Ring	0.029	2	0.058
Rubber Seal	0.052	1	0.052
Shunting Circuit	0.125	1	0.125
U Bolt	0.072	2	0.144
Nylon Locknut	0.015	2	0.03
Sealing Washer	0.01	2	0.02
Sub-system Total		**3.43 lbf**	

PLEC has passed through the aft bulkhead, and the rubber seal on the PEARS assembly creates a pressure seal between the ejection charges and PLEC. The airframe has a small hole through which the SPDT switch can be triggered to arm the ejection and retention charges when on the launch pad.

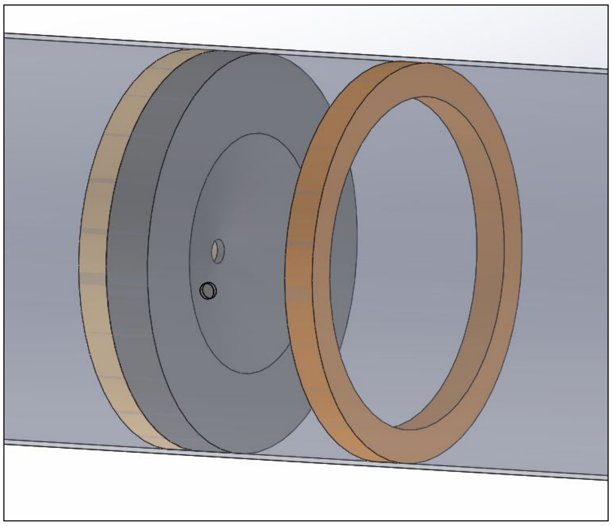

Figure 50: Fore hard point in fore section of airframe

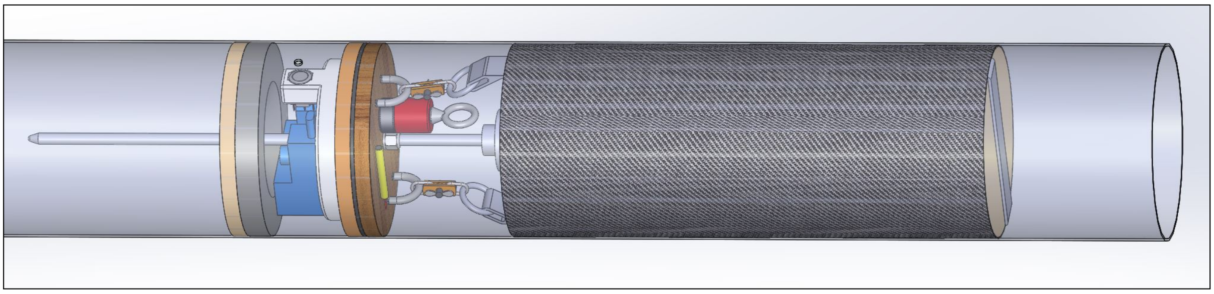

Figure 51: Fully integrated PEARS into fore airframe

4.4 Scientific Experiment Base Station

A scientific experiment is not required by USLI but is an additional project that would simulate a real-life mission. This also serves as a way to engage the students in aerospace and Science, Technology, Engineering and Mathematics (STEM) research. The team decided to choose an experiment that could be performed on another planet. The two leading options are planting a seed after soil collection and using x-ray spectrometry to analyze the elements found in the soil after collection. The team is planning to strive for the second option and will begin testing soon to determine the feasibility of the project. The seed planting option will be used as a backup, if needed. The final design and experiment decision may change as the project progresses and the feasibility of some options becomes impossible with the equipment and budget available. If detrimental to the overall mission success, the experiment will stop being pursued by OSRT.

The team has decided to build an external platform where the scientific experiment will be performed. This is to prevent expensive or hazardous scientific equipment and components from flying within the launch vehicle, placing people in danger. The team plans on performing the normal experiment as defined by the USLI Handbook and then, once the mission success has been confirmed by the Range Safety Officer (RSO), autonomously drive the rover to the base station (location determined by the RSO) to perform the experiment.

4.5 Payload Design with Leading Alternatives

The matrices in this section have helped the team make key design decisions. The result is a comprehensive payload design which is the combination of the leading alternatives for each decision. Table 61 is a summary of the team's choices which will be used moving forward into the critical design phase.

Table 61: Payload Design Decision Summary

Sub-system	Design Decision
Chassis	C1. The general form of the chassis will be a bridge style truss.
Chassis	C2. Corner connection blocks on the chassis will be made of aluminum.
Chassis	C3. The chassis will use glued carbon fiber rods for trusses and tapped aluminum rods for cross connections.
Drivetrain	D1. The rover will employ two independently-controlled GHM-04 DC brushed motors.
Drivetrain	D2. The wheels of the rover will be solid and circular.
Drivetrain	D3. The wheels of the rover will be 3D printed ABS plastic.
Drivetrain	D4. The rover's drive shaft will be machined from aluminum.
Drivetrain	D5. The drivetrain will employ 3D printed connector rods as second attachment points on the truss.
Drivetrain	D6. The wheel-drive shaft interface will be a clamping hub.
PEARS	P1. PEARS will be armed using a single-pole, double-throw switch.
PEARS	P2. PEARS will employ an ARRD and Tender Descender with a Kevlar harness.
PEARS	P3. The ejection method will utilize black powder.
SCAR	S1. The soil collection mechanism will be an auger.
SCAR	S2. The soil retention compartment will have two motor-driven doors which pivot at the top-center.

Figure 52 displays the leading alternatives for the rover. The orange indicates the Drivetrain, the blue indicates the Chassis, and the red indicates the SCAR. Fixturing and fasteners were not displayed in Figure 52.

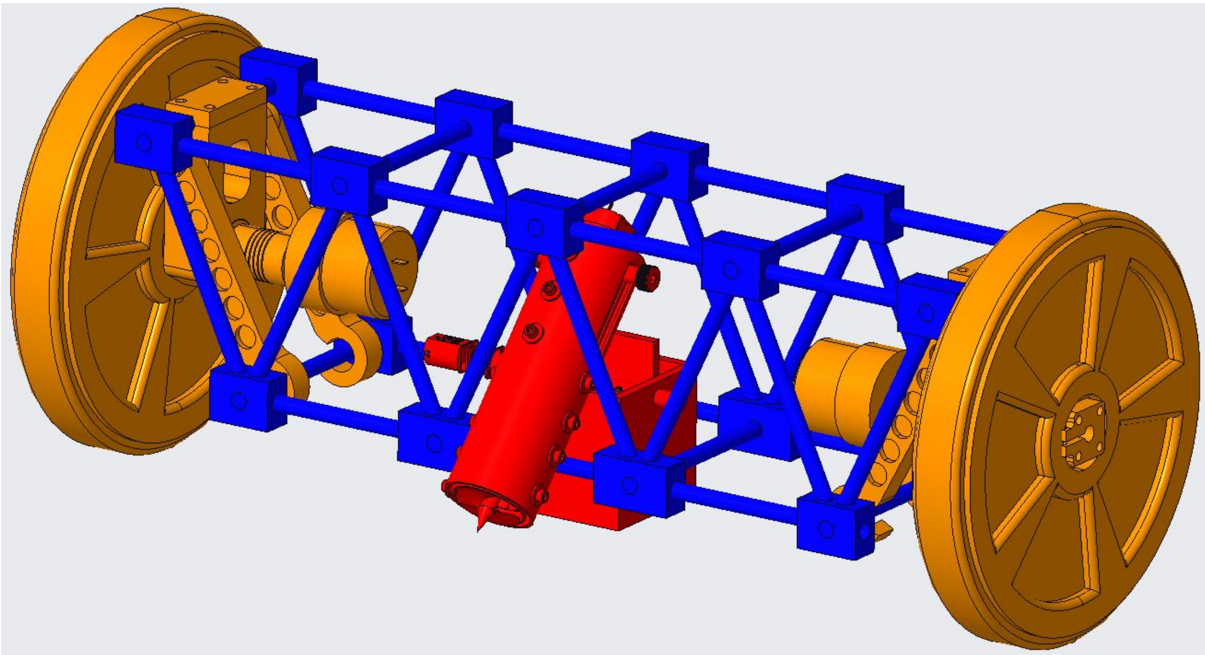

Figure 52: Assembly with the leading rover alternatives

4.6 Payload Electronics Sub-system Review

Figure 53 is a high-level block diagram of the integrated electronics on the rover. These electronics will be used to power and control the rover's motors, sensors, and on-board processing units. The components aboard the rover will be chosen to minimize power consumption in order to allow as much time as possible to complete the mission.

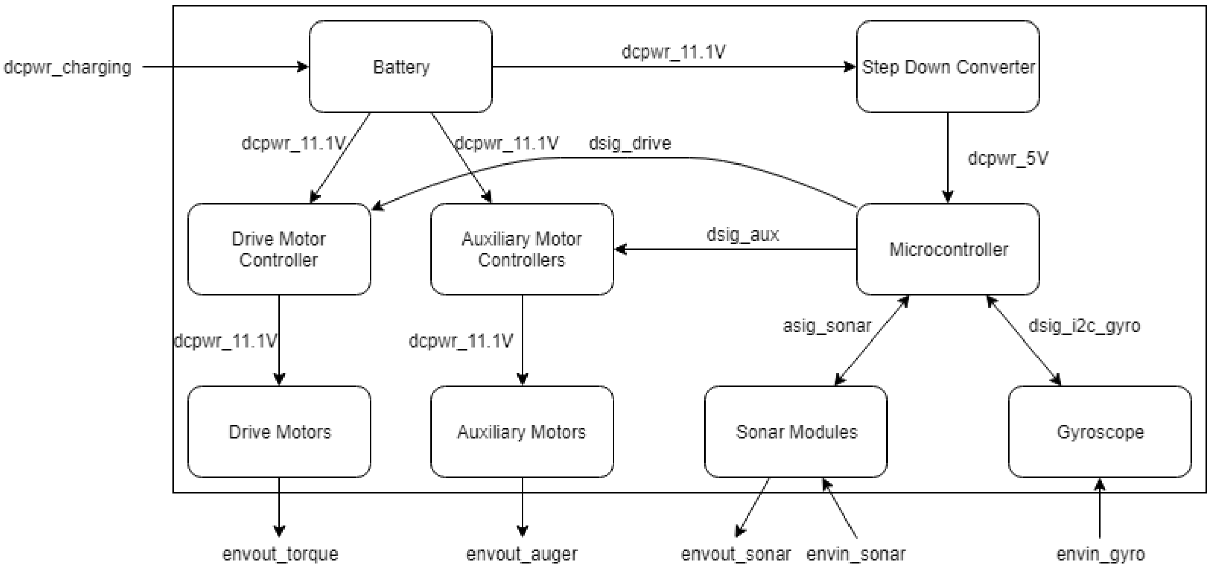

Figure 53: High Level Block Diagram

4.6.1 Batteries & Power Regulation

The rover will be powered by one battery or an array of lithium-polymer batteries, depending on the power budget of the rover. In series with the battery or battery bank, there will be two switches to keep the rover powered off when not in use. The first of these switches will be a physical switch, which will allow the user to turn the rover on and off. The second switch will be automatic and controlled by a quick-disconnect wire on the launch vehicle. When connected, a voltage-high signal from the launch vehicle will keep the circuit between the battery and the rover electronics disconnected, thereby saving the rover's energy while in flight or during testing. When released, the circuit will close and connect the battery to the on board electronics. Options for batteries are explored in Table 62.

Within the Battery Decision Matrix, all the batteries were considerably close with their total scores. The Turnigy Bolt 1000mAh battery resulted in the higher score mainly due to its small size. This increased the form factor score and weight score. The main drawback of this battery is that more than one of them would likely be used in the rover in order to have sufficient battery capacity. This creates more points of failure compared to the larger batteries. The Turnigy 2200mAh battery pack is the next highest rated pack.

Table 62: Battery Decision Matrix

Design		**Turnigy Graphene 1300mAh 45C LiPo**		**Turnigy 2200mAh 40C LiPo**		**Venom Fly 2200 mAh 30C LiPo**		**Turnigy Bolt 1000mAh 65C LiPo**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Form Factor	8	3	24	4	32	4	32	5	50
Weight	7	3	21	3	21	4	28	4	28
Continuous Current	7	3	21	4	28	4	28	4	28
Total		**66**		**81**		**81**		**106**	

It outranks the Turnigy Graphine 1300mAh pack in form factor which is the most important metric tested. It also outranks the Venom Fly 2200mAh battery pack in continuous current. While the margin is not wide, an instance where all components draw max current for an extended time would come close to the 30A continuous current rating of the Venom Fly.

To supply power for the rover's motors and other components, adjustable DC-to-DC buck converters will regulate voltage to levels usable by the microcontroller and sensors: The microcontrollers OSRT considered all use 5 Volts, which makes stepping down voltage a necessity. The proximity-detecting sonar sensors will use 3.3 V but will be powered by the microprocessor and therefore will not need their own step-down converters. There are several converters that are considered for the rover based on conversion efficiency, power consumption, and the ability to control output voltage. A comparison between the most effective three converters is given below in Table 63.

Table 63: DC/DC Buck Converter DDM

Design		**MP1584**		**TPS54383**		**LM2596**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Efficiency	7	3	21	4	28	3	21
Power Dissipation	6	3	18	5	30	4	24
Output Voltage Adjustability	8	3	24	3	24	3	24
Total		**63**		**82**		**66**	

Based on the decision matrix, the TPS54383 buck converter appears to be the best available option for the rover. The TPS54383 is rated at over 90% efficiency, as compared to the LM2596 at 73% efficiency and the MP1584 at 85% efficiency. This is an important metric for us due to limited battery life. The TPS4383 also has a switching frequency of 300 kHz, has two controllable output channels, and built-in overcurrent protection.

4.6.2 Microcontroller

A microcontroller will handle input data, output control signals, and navigation processing. It will receive analog and digital signals from the sonar sensors and output digital and PWM control signals to the drive motor controller. The autonomous driving system covered in Section 4.7.2 will be running for the duration of the rover's operation.

Table 64 compares possible microcontrollers and their viability in this project. The options are rated based on their physical form factor, I/O capacity, processor speed, and flash memory capacity.

Table 64: Microcontroller Decision Matrix

Design		**Teensy 3.2**		**Teensy 3.6**		**Arduino Pro Mini**		**Teensy 3.5**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Form Factor	6	3	18	3	18	4	24	3	18
I/O Capacity	5	3	15	5	25	2	10	4	20
Processor Speed	8	3	24	5	40	1	8	4	32
Flash Memory	7	3	21	4	28	1	7	3	21
Totals		**78**		**135**		**49**		**115**	

The Teensy 3.6 is the highest-rated microcontroller based on our criteria. Its processor speed, I/O capacity, and flash memory capacity are the best among the microcontrollers assessed. The Arduino Pro Mini and Teensy 3.2 are tied for the best form factor among these microcontrollers. This is an important metric, but their inferior processor speed and flash memory capacities make them non-viable for this application. The Teensy 3.5 is an older model of the 3.6 so they are similar, but the 3.6 is slightly better in all categories except form factor, which is the same as the 3.5.

4.6.3 Obstacle Detection

Based on power efficiency and the scope of the project requirements, sonar proximity sensing is the ideal method for obstacle detection, as sonar modules consume less battery power and processing power than alternative modules (i.e., computer vision, which requires an immense amount of processing power). The sonar modules draw under 4 mA of current at 5 V, which is negligible when compared to the power consumption of the motors and other components.

Given that the total scores of each sonar module within the matrix were so similar, any of them could be a viable option. With the sensors considered, the MB7060 has the greatest range at 300 in., which is ideal for long-distance navigation. The MB7360 has a smaller range at only 200 in.; however, it has a greater resolution, read rate, and consumes less power than the MB7060. Furthermore, out of all the units listed below, the MB7360 protects the best against all debris, like dirt, dust, and black powder.

Table 65: Sonar Module Decision Matrix

Design		**MB7139 XL-TrashSonar-WR**		**MB7360 HRXL-MaxSonar-WR**		**MB7060 XL-MaxSonar-Wr**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Range	6	3	18	4	24	4	24
Current Draw	7	3	21	3	21	4	28
Resolution	5	3	15	4	20	3	15
Read Rate	4	3	12	3	12	4	16
Durability	8	3	24	4	32	3	24
Total		**90**		**109**		**107**	

4.6.4 Soil Collection Electronics

The soil collection method involves one motor to control the auger and two motors to open and close the top and bottom of the soil collection container. Each of these motors will need a motor controller to deliver the proper power when necessary. Additionally, they will each need a feedback sensor to determine when they have reached the end of their intended travel. Biased switched connected back to the microcontroller will likely be used to determine when each component is at the end of its desired path. Section 4.7.3 details the circumstances under which each of these actions will take place.

4.7 Payload Software Review

The software used on the rover must provide the rover with the capabilities to autonomously perform a sequence of functions after being deployed from the launch vehicle. These functions consist of:

- Moving at least 10 feet from any part of the launch vehicle body.
- Avoiding obstacles that the rover cannot drive over or may become stuck on while navigating.
- Collecting a soil sample and containing it within an enclosed system.

The software on the control board must be sufficient to instruct the rover when and how to perform each of these tasks as they must be performed entirely autonomously after the payload has been ejected from the vehicle's body. In order to perform these tasks a certain language will be utilized to write embedded code that will run on a micro-controller. This will allow for interaction with all the input and output devices giving information about the real world to help make decisions on where to drill, paths to take, and any other choices the rover needs to make to autonomously fulfill its role.

4.7.1 Programming Language

While the programming language used is highly dependent on the chosen micro-controller, the language has a large role to play in smoothing out the development process and allowing for high performance and functionality. Commonly used languages for embedded systems consist of Assembly, C/C++, Python, and even Rust. While Assembly is fast and can cut down on resource usage through precision programming, it requires a lot of manual work and interfacing to other languages can be complicated. C/C++, along with Python and Rust, are higher-level languages which allow for a simpler approach and more powerful commands at the cost of memory consumption and speed. The desired features for the programming language are determined to be:

- Team Familiarity: Using a language that the CS and ECE team members are both well practiced in will help streamline the development process and allow for multiple members to work on each part of the code without a steep learning curve to get started.
- Memory Management: A language that is powerful enough to automatically clean up memory would help avoid many unnecessary run-time errors and memory leaks that can slow the process down or otherwise stop it from running entirely.
- Ease of Debugging: Given the scope of the project, we expect that extensive testing will take place over a lengthy testing and debugging stage. A language that is quick and easy to debug will allow for more rover features to be developed in the time given.
- Memory Overhead: Many micro-controllers have very little built in memory. While it is possible to extend that memory, extra resources are brought forward in order to allow for more space along with slower times to access the memory. Languages that make the most efficient use of the provided memory are more desirable.
- Performance: The operating speed of our rover is very important when it comes to carrying out tasks in real time. A language that can give us very fast reaction times for the rover will heavily increase its capabilities.

Table 66: Programming Language DDM

Design		**C/C++**		**Python**		**Rust**	
Team Familiarity	7	3	21	1	7	1	7
Memory Management	6	3	18	5	30	5	30
Ease of Debugging	7	3	21	5	35	5	35
Memory Overhead	5	3	15	1	5	1	5
Performance	7	3	21	1	7	2	14
Total		**96**		**84**		**91**	

4.7.2 Payload Obstacle Avoidance

To achieve its goal of navigating safely 10 feet away from any part of the launch vehicle body, the rover is going to need a protocol to follow in combination with the input from various chassis-mounted sensors. For this purpose we have determined that a sonar module is great for this task (see section 4.3.3). The rover is also likely to be equipped with a gyroscope to augment the data required to complete hard calculations for navigational purposes. The layout of the sensors is of utmost importance in the development of a navigational protocol. There are three main configurations for the sonar sensors to give the rover a clear reading for guidance.

- **Horizontally Inline Dual Sonar -** The Horizontally aligned sonar configuration consists of a sonar module on the right and a sonar module on the left with both forward facing and approximately 8 inches apart from one another. This configuration has a great horizontal area of detection but performs poorly when it comes to the third dimension in the vertical axis. The rover will have no way to differ an obstruction from a raise in elevation that can be easily climbed over. As a function of this, other unreliable methods of detecting and overcoming these obstacles will take place to compensate. This approach is not recommended.
- **Vertically Inline Dual Sonar -** This configuration will provide one sonar on top of the other sonar, both forward mounted, with approximately a 3 inch gap between the two vertically aligned sensors. This configuration, similarly to the Tri-mounted sonar, can calculate the incline of oncoming slopes but performs poorly in its ability to see peripheral objects along the path that can be hazardous to the roverś mission.
- **Tri-mounted Sonar -** This configuration consists of three sonars mounted on the front of the vehicle. One sonar on the right cocked slightly to the right by approximately 5 to 10 degrees. The second sonar will be on the left cocked slightly to the left similarly by 5 to 10 degrees. The third and final sonar is mounted on the middle of the front between and above the other sonars. This will be cocked upwards by approximately 5 degrees. While this configuration is the most expensive, it provides the largest area of coverage out of the three. The right and left sonar can detect upcoming obstacles that may obstruct the path of the rover. The third sensor is primarily for detecting changes in elevation on rolling hills where the left and right sensor will detect an object to be avoided. Where the Horizontally Inline Dual Sonar fails, this design succeeds. The rover will be able to measure the distance to the target on each sonar and calculate the slope of the obstacle reporting back if it is safe to climb or if corrective action should be taken. If the rover is on a slope, the use of a gyroscope within the rover will offset this issue. The Tri-mounted sonar is the only configuration of the three that offers a complete solution.

4.7.3 Soil Collection Routine

The task of collecting soil will be completed by an auger which is actuated by a soil collection protocol that initializes when the rover determines a safe and practical region from which to extract a soil sample.

Table 67: Sonar DDM

Design		Hoizontally Inline Dual Sonar		Vertically Inline Dual Sonar		Tri-Mounted Sonar	
Coverage	7	3	21	2	14	4	28
Cost	6	3	18	3	18	2	12
Ease of Implementation	4	3	12	3	12	2	8
Reliability	8	3	24	4	32	5	40
Total		**75**		**76**		**88**	

The ideal conditions consist of soft soil, and flat or slightly inclined ground. The rover will primarily use the gyroscope when looking for this spot, however, readings of the wheels output may be used to give indication of softer ground. When this location is found, the call to the collection protocol will activate the auger for drilling - but only when the proper conditions are met.

For safety, the auger will be disabled when the gyroscope is out of operable ranges. This will reduce the risk of unsafe drilling conditions. Additional safety protocols will be implemented for scenarios that could possibly indicate danger such as the auger becoming lodged into the ground, and the auger not finding purchase in soil.

4.7.4 Scientific Base Station

After collection of the soil sample the rover will navigate to a base station that will be set up by OSRT. The base station will be located within 300 feet of the launch vehicle's landing site. Two current methods will be used to navigate back to the base station after deployment:

- **GPS and Infrared Navigation-** With GPS and infrared navigation, instructing the rover where to go can be done on a large scale. This allows for distances up to a mile away and more with precision of +/- 5 meters. Once the payload is within 10 meters of the base station, we will switch to infrared. Infrared is cheap, reliable, and small enough to fit within the body of the rover. With this the rover should be able to dock and test the soil sample at the deployed lab using X-ray Fluorescence to find the composition of the soil, or performing another experiment still under consideration. The downsides of this system consists of the unreliability of GPS and costs of implementing hardware to carry out the task. Implementation of the task will also take a long amount of time.
- **Computer Vision Aided Navigation-** Computer vision can be implemented in order to help navigate the rover to its final destination; it will also assist in the process of docking. The micro-controller carrying out this task will be initialized with geometric objects of a certain color of which to follow. This geometric shape will be placed high on a pole that can be seen by the rover and followed until a certain distance is reached. As soon as this destination is reached, the rover will begin a new process to

seek out a different geometric shape and proceed to dock. Benefits to using computer vision would be the reliability and precision gained from the application of only one system. Alternatively, this system will have major power consumption's and only able to travel varying distances with the quality of camera and the size of the geometric object. This system would also require a high powered microcontroller in order to process so much information in a timely manner.

5 Safety

5.1 Risks & Delays

5.1.1 Safety Responsibilities

The Lead Safety Officer for the 2018-2019 OSRT is Jon Verbiest. He oversees the general safety for the overall project. Along with the Lead Safety Officer, there are two secondary Safety Officers for both the launch vehicle and payload. The Launch Vehicle Safety Officer is Al Lacey and the Payload Safety Officer is Jessica Jorgens. While everyone on the team is responsible for safety, it is the job of these three team members to regulate the safety of all project activities and ensure that all OSU, National Aeronautics and Space Administration (NASA), NAR, and National Fire Protection Agency (NFPA) safety rules and regulations are followed.

Team members are required to follow all safety rules and regulations during all team activities and this will be enforced by the Safety Officers. These team activities include but are not limited to: any launch activity, any machining process, any test activity, any time an individual is working in OSU machine shops, and any time an individual is handling hazardous materials. If safety rules and regulations are not followed, safety officers have authority on postponing or cancelling a team activity.

5.1.2 Safety Methods

Safety Officers will ensure safety through risk assessments and Failure Mode Effects Analysis (FMEA)s of all launch vehicle and payload components. The risk assessment is based off of a two-factor analysis and predicts the severity and likely hood of a failure. The template for the risk assessment codes is shown in Table 68. From the results the risk assessment analysis the team can put more focus to mitigating more hazardous tasks.

To prevent hazards, the Safety Officers will implement administrative controls and engineering controls when applicable. Engineering controls are favorable in all situations but harder to implement. For this reason, the team will use mostly administrative controls given the engineering and design limitations of this competition. Administrative controls include pre-launch checklists, final sign off sheets, Job Hazard Analysis (JHA) forms for all manufacturing processes, and a safety agreement among all team members. An example of a completed JHA form from subscale fin slot machining can be seen in Figure 54. A main engineering control that the team is implementing this year is the use of single pole double throw switches. These switches allow the circuit to be shunted so static charges cannot build up and set off any black powder charges unplanned.

Job Hazard Analysis

Job: Subscale Fin Slot Machining

Subteam: Structures

Task or Step	Hazards	Controls	Personal Protective Equipment (PPE)
fixture/machine set up	working around machines	Propper PPE and general machine shop caution	Safety glasses, close toe shoes
machining fin slots	moving machines, fiber glass particles and dust	Propper PPE, vacuum to reduce fiber glass dust	Safety glasses, respirator, closed toe shoes, gloves
Clean up	working around machines, fiber glass dust	Propper PPE, use vacuum on fiber glass dust	Safety glasses, respirator, close toe shoes, gloves

JHA by:	Jon Verbiest	Safety Officer Initials:	J.V.
Date:	10.15.18	Date:	10.15.18

Figure 54: Completed JHA Form

5.1.3 Delays

Many unforeseen events and scenarios can cause delays to the project. Some of these are at the fault of the team and some are uncontrollable. The greatest risk of delays that the team has identified are extreme weather on launch days and manufacturing delays. In the event that extreme weather occurs during a launch and the launch is deemed unsafe, it will be delayed or cancelled. This has serious implications on the team's performance in competition and in competition preparedness if either competition launches or test launches are delayed or cancelled. To mitigate this, the team will plan to launch

There are many reasons why manufacturing could be delayed. One of these reasons is to make sure that JHA forms are properly completed prior to manufacturing. This is an acceptable reason for delaying because safe manufacturing practices come before staying on schedule. This delay should not significantly impact the project as long as manufacturing is completed within a reasonable time window. Another reason why manufacturing could be delayed is due to waiting on parts. This should be avoided if at all possible, and mitigated through timely purchase orders. However, ultimately, some shipping factors could be out of the team's control.

5.2 Hazard Analysis

In Tables 69 - 75 are the Hazard Analyses which OSRT has developed for the mission. Table 68 displays the risk assessment template, where red tasks represent the highest risk items. The risk level is based off of severity of the hazard, and the probability of it happening. Each hazard is assessed for risk level, then a mitigation technique is presented to reduce the risk of the hazard. The hazard with the mitigation technique is ranked again. Ideally, after identifying mitigation of the hazard is implemented, the hazard will be

Table 68: Risk Assessment Code Template

Risk Assessment Codes (RAC)				
	Severity			
Probability	1 - Catastrophic	2 - Critical	3 - Marginal	4 - Negligible
A - Frequent	1A	2A	3A	4A
B - Probable	1B	2B	3B	4B
C - Occasional	1C	2C	3C	4C
D - Remote	1D	2D	3D	4D
E - Improbable	1E	2E	3E	4E

Table 69: Aerodynamics and Propulsion Hazard Analysis

Hazard	Cause	Effect	Pre-RAC	Mitigation	Verification	Post-RAC
Catastrophic Motor Failure	Manufacturing defects in motor or motor is damaged	Failure to reach target apogee, Major damage to all components, Possible injury	1D	Motors to be purchased from certified manufacturers and distributors. Careful handling of motors at all time by proper personnel.	Motor performance testing to be performed prior to full scale. Multiple motors will be purchased and tested.	1E
Motor Fails to Ignite	Manufacturing Defect in motor, or Ignition Failure	Motor does not start, launch vehicle fails to leave launch pad	2C	In the event of a motor failure, all NAR procedures will be followed to disarm the launch vehicle.	The OSRT is aware of all NAR safety procedures.	3D
Launch Vehicle Fails to Lift Off from Rail	Damage to the launch rail or launch lug	Mission failure if motor ignites without liftoff. Possible damage to components.	2D	Inspection of all launch components prior to set up. Assurance of a smooth fit and easy escape of launch vehicle.	Launch rail will be used many times prior to full scale launch. Advisers will inspect launch rails and verify prior to every launch.	2E
Early Motor Ignition	Motor ignites before ignition start	Mission failure if launch vehicle not ready on rail. Possible damage to components and injury to all personnel in area	2D	All NAR safety code to be followed during motor insertion and launch set up. Motors will be handled only by designated personnel qualified for L-class motor handling	Only qualified personnel will pack the motor in strict accordance to motor manufacturer.	2D

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Table 69 – continued from previous page

Hazard	Cause	Effect	Pre-RAC	Mitigation	Verification	Post-RAC
Lack of Stability	Center of Gravity differs from design, invalidating simulations	Launch vehicle does not follow anticipated path, possibly becoming ballistic and damaging components and injuring personnel in the area.	2C	Simulations will be performed and verified, adding ballast if needing to increase stability	OpenRocket and RASAero will be used to identify center of gravity and center of pressure to calculate stability. All components lengths and weights will be verified with simulations prior to launch.	2E
Damage to Airframe or Fins mid flight	Misassembly of launch vehicle, or the launch vehicle reaches unanticipated supersonic speeds	If the exterior of the launch vehicle is damaged in any way mid flight may cause the vehicle to go ballistic, causing damage to interior components upon landing and possible injury to personnel in the area	2D	The airframe will be constructed of carbon fiber and carefully inspected prior to every launch. Motor will be verified of thrust prior to full scale launch. Launch vehicle will be designed to stay in sub-sonic speeds.	Pre flight and post flight checks will be performed to ensure no damage is done to the airframe. In the event of fin breakage, detachable fins will be designed with replacement fins on site for easy replacement.	2E

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Table 70: BEAVS Hazard Analysis

Hazard	Cause	Effect	Pre-RAC	Mitigation	Verification	Post-RAC
Lack of stability in launch vehicle	Blade deployment by BEAVS during flight	Unpredictable and potentially dangerous flight path	1A	Place BEAVS aft of center of pressure to ensure stability cannot be decreased.	Simulations in OpenRocket and RasAero II to ensure center of pressure location is fore of BEAVS blade deployment location.	3E
Launch vehicle exceeds desired apogee altitude	Failure in active system of BEAVS	Potential to exceed waiver ceiling for maximum apogee altitude.	2C	Use the appropriate amount of ballast in the passive system of the BEAVS to control maximum apogee altitude	Simulate flight with OpenRocket and RasAero II to determine appropriate amount of ballast based on launch day conditions.	3D
Launch vehicle does not reach minimum apogee altitude	Failure in active system of BEAVS	Mission failure	3B	Repeated testing of all elements of the active system.	Tests conducted to ensure reliability of sensor data and filtering techniques. Testing conducted to ensure mechanical system performance prior to launch. Numerous test launches to determine system reliability.	3D

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Table 71: Recovery Hazard Analysis

Hazard	Cause	Effect	Pre-RAC	Mitigation	Verification	Post-RAC
Ejection charges fire early/late	Altimeters sense apogee incorrectly due to incorrect pressure sensing	Unpredictable ejection forces acting on launch vehicle and possible failure to separate. This could result in a tumbling or ballistic launch vehicle. Possible injury to inattentive observer.	2C	Ensure correct static port hole sizing and spacing, and minimize/eliminate delay on firing primary ejection charges	Correct static port hole sizing procedure is followed according to primary altimeters. Secondary altimeter procedure for sizing will be considered.	3E
Main parachutes do not deploy	Failure in fore and/or aft Tender Descenders	Tumbling launch vehicle resulting in damage to launch vehicle and payload. Possible injury to inattentive observer.	1D	Repeated testing of Tender Descenders	Test all Tender Descenders in high stress situations, and ensure black powder ejection charges consistently and successfully separate the the sections of the Tender Descenders.	3D
Shock cord snaps	Snatch load on shock cord from main parachute opening is too high, or unaccounted for snatch loads cause a failure in the nylon shock cord.	Launch vehicle section tumbles to the ground without a parachute. Possible injury to inattentive observer.	1D	Reduce shock load: main chutes in deployment bags, tape sections of shock cord together, place a steel slider ring around shroud lines..All components will have an appropriate safety factor for expected loads.	Test shock cords with a range of shock loads, past what is expected to be felt during launch, to ensure it will not fail.	1E

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Table 71 – continued from previous page

Hazard	Cause	Effect	Pre-RAC	Mitigation	Verification	Post-RAC
Eye bolt breaks	Unaccounted snatch load is higher than shock rating for eye bolts	Launch vehicle section tumbles to the ground without a parachute. Possible injury to inattentive observer.	1D	Reduce shock load: main chutes in deployment bags, tape sections of shock cord together, add a bungee to reduce velocity difference between separated sections	Test eye bolts with a range of shock loads, past what is expected to be felt during launch, to ensure it will not fail.	1E
Fire damage to parachutes	Improper packing methods were used	Parachutes will perform subotimally. Worst case scenario the launch vehicle tumbles to the ground. Possible injury to inattentive observer.	2C	Ensure all parachutes are packed appropriately.	Ensure no part of the parachute is exposed while loading into the launch vehicle.	1E
Launch vehicle falling on someone under a parachute	Wind causes launch vehicle to drift towards spectators	Possible injury to inattentive observer.	1D	Attentiveness to all descending launch vehicles, in order to move out of the way	Ensure launch vehicle is falling at a safe velocity so spectators can move out of the way	1D
Launch vehicle does not separate	Altimeters not connected properly to ejection charges or altimeters broken	Launch vehicle goes ballistic and is lost upon impact with the ground.	1C	Ensure all wires are connected correctly, all beep sequences are correct upon turning on altimeters, and ejection charges are made correctly.	Simulate flight with OpenRocket and RasAero II to determine appropriate amount of ballast based on launch day conditions.	1D

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Table 71 – continued from previous page

Hazard	Cause	Effect	Pre-RAC	Mitigation	Verification	Post-RAC
Main parachutes do not unfurl completely	Improper packing methods were used	Possible damage to launch vehicle and payload	3B	Repeated testing of packing method to ensure parachutes will unfurl completely	Tests conducted to ensure reliability of sensor data and filtering techniques. Testing conducted to ensure mechanical system performance prior to launch. Numerous test launches to determine system reliability.	2E

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Table 72: Ejection Bay Hazard Analysis

Hazard	Cause	Effect	Pre-RAC	Mitigation	Verification	Post-RAC
Altimeters do not communicate when to deploy parachutes	Battery power is low	The parachutes may not deploy	1B	Only new batteries will be used for each launch. An additional LED will be added to the circuit to easily tell if there is power.	Test each battery before using for launch.	2E
E-Match does not ignite	Poor connection to E-Match	Parachutes do not deploy	1D	Check E-Match and wiring thoroughly before integration and launch	Tests will be conducted to be sure that the system works properly and to lower the chance of a malfunction	1E

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Table 73: PEARS Hazard Analysis

Hazard	Cause	Effect	Pre-RAC	Mitigation	Verification	Post-RAC
Payload falls from airframe during flight	Retention devices release early	Payload Falls at apogee	2D	Redundant retention devices of differing manufacturers	Verify load rating of whole assembly through specifications or testing	2E
Ejection attempted with payload still retained	Retention devices do not release	Black power ejection blows out side airframe	1C	PLEC does not allow for ejection to be triggered if retention is not successfully triggered first	Rigorous ground testing of PLEC and PEARS assembly	1E

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Table 74: Drivetrain Hazard Analysis

Hazard	Cause	Effect	Pre-RAC	Mitigation	Verification	Post-RAC
Chemical hazard	Poor battery condition	Chemical burn	2D	Keep batteries in appropriate condition; inspect batteries with gloves when handling.	Enforce safety checklists and proper PPE.	3E
Electrical shock	Improper electrical design or installation	Injury to hands, face; damage to rover	2D	Power down rover during (dis)assembly and maintenance.	Enforce safety checklists and proper PPE.	2E
Injury from rotating components	Fingers, hair, or loose clothing is caught in the drive shaft mechanism.	Injury to scalp, hands; damage to rover	2C	Shield rotating components; power down rover during (dis)assembly and maintenance; tie back hair and wear appropriate clothing.	Enforce safety checklists and proper PPE.	3E
Motors overheat	Temperatures from launch and heavily torquing the motor.	Motor can catch on fire	2B	Use a motor that provides adequate characteristic for the project mission .	Motor supplier specification sheet verification. Payload ejection and rover mobility testing.	2E

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Table 75: SCAR Hazard Analysis

Hazard	Cause	Effect	Pre-RAC	Mitigation	Verification	Post-RAC
Auger breaks	Loads applied to the auger are more than the material can handle	Parts of the auger can be projected through the air.	3C	Use materials rated for high stresses.	Testing in the SCAR Test Bed. FEA and Force Analysis	3E
Motor overheats	Temperatures from launch and heavily torquing the motor.	Motor can catch on fire	2B	Use a motor has adequate characteristics for the mission operations.	FEA Analysis. Motor supplier specification sheet verification. Testing in the SCAR Test Bed.	2E
Threads are stripped	The threads are not rated for the desired forces.	SCAR can detach from the Chassis and damage the Rover.	2D	Use correctly rated components for the worst case scenario.	SCAR Test Bed. Component supplier spec sheet verification. FEA and Force Analysis.	2E

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5.3 Preliminary Failure Modes and Effects Analysis

5.3.1 Aerodynamics and Recovery FMEA

In Tables 76 and 77 are the Aerodynamics and Recovery Failure Mode Effects Analysis (FMEA) that OSRT has developed for the mission.

Table 76: Recovery Integration FMEA

Function	Failure Mode	Effects of Failure	Severity	Failure Causes	Occurrence	Detection	RPN	Mitigation
Ejection charges separate sections of the launch vehicle.	Primary ejection charges fizzle out	Secondary charges will be relied upon to separate the airframe sections. The launch vehicle could be destroyed if the vehicle goes ballistic. & 6 & Black powder charges are constructed improperly.	6	Black powder charges are constructed improperly.	2	1	12	Ensure all black powder charges are packed tightly and have no give, as well as electric matches being connected properly.
	All ejection charges fizzle out.	The launch vehicle goes ballistic, resulting in the destruction of the vehicle.	10	Black powder charges are constructed improperly.	2	1	20	Ensure all black powder charges are packed tightly and have no give, as well as electric matches being connected properly.
	Ejection charges ignite but do not separate the launch vehicle sections	The launch vehicle goes ballistic, resulting in the destruction of the vehicle.	10	Black powder charges are constructed improperly.	3	1	30	Ensure all black powder charges are properly sized and slightly oversized.
Altimeters sense apogee	Altimeters sense apogee & Altimeters sense apogee at different times	The recovery bay could be overpressurized if the charges go off at the same time, blowing out the side of the airframe.	6	The altimeters malfunction.	4	6	144	The altimeters will be tested at different altitudes to ensure they are sensing correctly.

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Table 76: Recovery Integration FMEA – continued from previous page

Function	Failure Mode	Effects of Failure	Severity	Failure Causes	Occurrence	Detection	RPN	Mitigation
Altimeters sense apogee	Altimeters sense apogee early	Unpredictable snatch loads will be experienced, possibly resulting in the failure of any given recovery component.	7	Static port holes were sized improperly, or the primary or secondary altimeters malfunction.	4	3	84	The altimeters will be tested at different altitudes to ensure they are sensing correctly.
	Altimeters sense apogee late	Unpredictable snatch loads will be experienced if the sections separate, and if the charges go off too late, the airframe sections will not separate.	8	Static port holes were sized improperly, or the primary or secondary altimeters malfunction.	4	3	96	The altimeters will be tested at different altitudes to ensure they are sensing correctly.
	Altimeters fail to sense apogee	The main parachutes are released too early, increasing the drift radius and descent time.	10	The altimeter's barometric sensor malfunctions.	3	6	180	Test altimeters at different altitude.
Altimeter senses main parachute deployment height, releasing Tender Descenders	Altimeter senses height too early	The main parachutes are released too early, increasing the drift radius and descent time.	2	The altimeter's barometric sensor malfunctions.	5	5	50	Test altimeters at several altitudes.
	Altimeters sense height too late	The launch vehicle possibly impacts the ground before the main parachute opens fully.	6	The altimeter's barometric sensor malfunctions.	3	6	108	The altimeters will be tested at different altitudes to ensure they are sensing correctly.

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Table 76: Recovery Integration FMEA – continued from previous page

Function	Failure Mode	Effects of Failure	Severity	Failure Causes	Occurrence	Detection	RPN	Mitigation
Altimeter senses main parachute deployment height, releasing Tender Descenders	Altimeters sense main parachute deployment height	The main parachutes are released at apogee. The drift radius increases drastically.	3	The release mechanism malfunctions.	6	6	108	Test Tender Descenders under high stress situations to ensure the release mechanism does not come undone.
Recovery system becomes taut as the main parachutes unfurl (includes unaccounted for snatch loads)	Shock cord snaps	The launch vehicle section tumbles to the ground.	8	High snatch loads and/or a weak point in the shock cord cause a failure.	1	3	24	Inspect all shock cords for any visible flaws to ensure the cords perform optimally.
	Shock cord tangled	Shock cord tangled & Unpredictable snatch forces will be experienced by the shock cord, and it is possible the parachute does not exit the launch vehicle if the tangling is drastic enough.	7	The shock cords are folded and packed improperly.	2	3	42	Ensure all parachutes are packed correctly, by following a list of instructions. Someone will watch over ensuring all steps are followed correctly.
	Quick link fails	The launch vehicle section tumbles to the ground.	8	High snatch loads and/or a weak point in the shock cord cause a failure.	2	4	64	Inspect all quick links for visible flaws.

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Table 76: Recovery Integration FMEA – continued from previous page

Function	Failure Mode	Effects of Failure	Severity	Failure Causes	Occurrence	Detection	RPN	Mitigation
Recovery system becomes taut as the main parachutes unfurl (includes unaccounted for snatch loads)	Eye bolt fails	he launch vehicle section tumbles to the ground.	8	High snatch loads and/or a weak point in the shock cord cause a failure.	2	4	64	Inspect all eye bolts for visible flaws.
	Bulkhead fails	Bulkhead slippage damages airframe	7	Epoxy fails to hold bulkhead in place.	2	3	42	Ensure all bulkheads held by epoxy have enough, if not an excess of, epoxy.
		The bulkhead slips out of airframe, causing the launch vehicle to tumble to the ground.	8	The epoxy fails to hold bulkhead in place.	2	3	48	Ensure all bulkheads held by epoxy have enough, if not an excess of, epoxy.
	The shear pins connecting the fore ejection bay to the fore airframe tear	The fore main parachute is released at apogee, increasing the drift radius drastically.	3	Unaccounted for snatch loads in the fore section when the drogue parachute is released	3	6	54	Test altimeters at various altitudes to ensure they are sensing correctly to limit the chances of unaccounted for snatch loads.
	Shroud lines snap	The launch vehicle section tumbles to the ground.	8	Unaccounted for snatch loads exceed the shroud lines maximum strength.	2	6	96	Test altimeters at various altitudes to ensure they are sensing correctly to limit the chances of unaccounted for snatch loads.
	Swivel breaks	The launch vehicle section tumbles to the ground.	8	Unaccounted for snatch loads exceed the swivels maximum strength.	2	6	96	Test altimeters at various altitudes to ensure they are sensing correctly to limit the chances of unaccounted for snatch loads.

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Table 76: Recovery Integration FMEA – continued from previous page

Function	Failure Mode	Effects of Failure	Severity	Failure Causes	Occurrence	Detection	RPN	Mitigation
Main parachute unfurls	Shroud lines tangle	The main parachute does not unfurl completely, causing increased kinetic energy upon landing.	7	Shroud lines and main parachute are packed improperly.	3	3	63	Ensure all parachutes and shroud lines are packed correctly, by following a list of instructions. Someone will watch over ensuring all steps are followed correctly.
	Parachute rips	he kinetic energy upon landing is increased, as the vehicle will be in a semi-tumble.	7	Unaccounted for snatch loads exceed the material strength of the main parachutes.	3	6	126	Inspect all parachutes for visible tears, and ensure all parachutes are packed correctly, by following a checklist of instructions. Someone will watch over ensuring all steps are followed correctly.
Recovery system becomes taut as the drogue parachutes unfurl (includes unaccounted for snatch loads)	Shock cord snaps	Increased velocity when the main parachute deploys and a less controlled ejection of the main parachute	6	High snatch loads and/or a weak point in the shock cord.	1	3	18	Inspect all shock cords for any visible flaws to ensure the cords perform optimally.
	Shock cord tangled	Increased velocity when the main parachute deploys.	5	Improper folding and packing of the shock cord.	2	3	30	Ensure all parachutes are packed correctly, by following a Inspectlist of instructions. Someone will watch over ensuring all steps are followed correctly.
	Bulkhead fails	Bulkhead slippage damages airframe	7	Epoxy fails to hold bulkhead in place.	1	2	14	Ensure all bulkheads held by epoxy have enough, if not an excess of, epoxy.

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Table 76: Recovery Integration FMEA – continued from previous page

Function	Failure Mode	Effects of Failure	Severity	Failure Causes	Occurrence	Detection	RPN	Mitigation
Recovery system becomes taut as the drogue parachutes unfurl (includes unaccounted for snatch loads)	Bulkhead Fails	Bulkhead slips out of airframe, causing the launch vehicle to tumble to the ground	8	Epoxy fails to hold bulkhead in place.	1	2	16	Ensure all bulkheads held by epoxy have enough, if not an excess of, epoxy.
	Shroud lines snap.	Increased velocity when the main parachute deploys and a less controlled ejection of the main parachute	6	Unaccounted for snatch loads exceed the shroud lines maximum strength.	1	6	36	Test altimeters at different altitudes to ensure they are sensing correctly to limit the chances of unaccounted for snatch loads.
Drogue parachute unfurls	Shroud lines tangle	The drogue parachute does not unfurl completely, causing increased velocity when the main parachute is deployed.	6	Improper packing of the shroud lines and drogue parachute.	2	3	36	Ensure all parachutes and shroud lines are packed correctly, by following a Inspectlist of instructions. Someone will watch over ensuring all steps are followed correctly.
	Parachute rips	Increased velocity when the main parachute deploys and a less controlled ejection of the main parachute	4	Unacounted for snatch loads exceed the material strength of the drogue parachutes.	3	6	72	Inspect all parachutes for visible tears, and ensure all parachutes are packed correctly, by following a checklist of instructions. Someone will watch over ensuring all steps are followed correctly.

Table 77: BEAVS FMEA

Function	**Failure Mode**	**Effects of Failure**	**Severity**	**Failure Causes**	**Occurrence**	**Detection**	**RPN**	**Mitigation**
Gear teeth mesh to actuate blades	Teeth slip	Blades remain extended in undesired position or do not actuate fully	3	Lack of precision tolerance during manufacturing. Flexibility of rack design.	3	1	9	Use CNC manufacturing to maintain precise tolerancing. Design rack to be rigid enough to avoid flexibility.
Blades deploy	Blades deploy during motor burn	Control system fails to accurately predict apogee altitude	5	Lack of precision in apogee altitude.	5	2	50	Repeated testing of system to ensure reliable blade deployment time
Blades deploy	Blades fail to deploy	Apogee altitude not within desired accuracy.	2	Lack of precision in apogee altitude.	5	4	40	Repeated testing of the system to ensure reliable blade deployment time. Passive ballast system used as backup to ensure mission performance despite failure.
Blades deploy	Friction in system exceeds motor capabilities to actuate blades	Blades do not actuate	2	Lack of precision in apogee altitude.	1	8	16	Linear bearings reduce friction in direction of actuation. Blades designed to actuate perpendicular to airflow to reduce required motor torque. Selected motor to be slightly oversized to ensure reliable performance.
Rack and pinion actuates blades	Blades over extend and come off central drive gear	Blade falls out of launch vehicle	5	Motor rotates further than desired	3	1	15	Implement a retention system on design to prevent blades from over extending.

5.3.2 Payload FMEAs

In Tables 78 and 81 are the Payload Failure Modes and Effects Analyses that OSRT has developed for the mission.

Table 78: PEARS FMEA

Function	Failure Mode	Effects of Failure	Severity	Failure Causes	Occurrence	Detection	RPN	Mitigation
Payload Retention	Failure to Retain Payload	Payload falls from airframe during flight	9	Retention devices fail in an open position from mechanical failure	3	1	27	Redundant retention devices to account for any abnormal flight forces
		Payload moves around within airframe	4	Improper harness length or integration into airframe	4	6	96	Testing of varying harness lengths as well as fatigue testing on Instron for harness elongation under loading
		Payload is damaged, unable to complete mission	7	Improper assembly with PEARS	3	3	63	Well documented and practiced process for integration. Only one correct way to fully insert assembly into airframe ensures an improper assembly is visually noticeable
	Failure to Release Payload	Payload stuck inside airframe, unable to complete mission	3	Retention devices fail in a closed position from electrical failure	2	1	6	Rigorous ground testing of all retention components, both individually and integrated into system
		Payload is damaged, unable to complete mission. Airframe damage sustained	9	Payload retained when ejection attempted	1	1	9	Ejection cannot be attempted unless successful release of retention devices is known
Payload Ejection	Failure to Eject payload	Payload is stuck inside airframe, unable to complete mission	3	Electrical failure PLEC	2	1	6	Ground testing of PLEC for reliable ejection ignition.

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Table 78: PEARS FMEA – continued from previous page

Function	Failure Mode	Effects of Failure	Severity	Failure Causes	Occurrence	Detection	RPN	Mitigation
Payload Ejection	Damage Payload on Ejection	Payload not able to complete mission	5	Object at end of airframe which payload ejects into	2	1	10	Object ejection avoidance not possible, additional padding on leading edge of payload wrap and bulkhead

Table 79: Chassis FMEA

Function	Failure Mode	Effects of Failure	Severity	Failure Causes	Occurrence	Detection	RPN	Mitigation
Tail	Tail does not touch the ground.	The chassis will rotate instead of the wheels, preventing movement.	8	The tail is of insufficient length.	1	1	8	Tail length will be determined by 3D models.
	Tail will not unwrap from rover.	The chassis will rotate instead of the wheel, preventing movement.	8	The spring breaks or critically deforms.	2	1	16	Spring force and strength will be determined based upon the expected tail angle of rotation.
Fixed Rods	The rods disconnect from the chassis blocks.	Rover disassembles and is unable to perform mission.	5	Insufficient or weak epoxy is used on the rods and blocks.	5	3	75	Use epoxy with sufficient characteristics for the forces experienced during all mission operations.
			5	Rods break on the rover.	4	2	40	Design the rod strength to withstand all mission operations.
Removable Rods		Chassis disassembles or becomes structurally compromised and is unable to perform mission.	7	The screw unfastens from the chassis.	6	4	168	Loctite will be applied to threads before assembly.
			7	Removable rod breaks during mission operations.	4	2	56	Design the rod strength to withstand all mission operations.
Chassis Blocks		Chassis disassembles or becomes structurally compromised and is unable to perform mission.	6	Chassis blocks fracture during mission operations.	2	6	72	Material choice will be determined upon the stresses expected during mission operations.

Table 80: Drivetrain FMEA

Function	Failure Mode	Effects of Failure	Severity	Failure Causes	Occurrence	Detection	RPN	Mitigation
Generate torque via DC motors	Motors fail to produce adequate torque	The rover is unable to move or unable to climb slopes.	8	The motor has a manufacturing defect.	1	1	8	The payload testing phase will verify motor functionality.
				Flight forces break an electrical connection.	4	3	96	Electronics will be protected within the rover's chassis and will be examined after payload ejection tests.
				The power supply drains prior to rover ejection.	2	3	48	The payload will be powered down until just before launch and power controls will be accessible from the outside of the airframe.
Transmit torque from motor to wheels	Motors fail to transmit adequate torque to wheels	The rover is unable to handle gentle slopes and may become stuck.	6	Drive shaft slips	2	3	36	Shaft coupling set screws ensure 1:1 rotation.
				The drive shaft assembly becomes misaligned.	4	2	48	The drive shaft assembly has multiple attachment points to truss to limit displacement.
				The drive shaft fails via torsion or bending.	1	1	6	Payload testing and stress simulations will prevent failure.

Table 81: SCAR FMEA

Function	Failure Mode	Effects of Failure	Severity	Failure Causes	Occurrence	Detection	RPN	Mitigation
Soil Collection Auger	Failure to collect at least 10 mL of soil	Auger cannot collect soil and transport it to the container	8	Insufficient Motor Torque	2	1	16	Perform force calculations and over-spec the motor. Testing.
		Auger does not rotate. Auger does not deploy into soil. Auger device detaches from the Chassis.	6	Threads become stripped on any component	2	5	60	FEA Analysis. Testing. Use favorable factors of safety.
		Auger does not collect soil. Auger does not transport soil to the container.	8	Auger breaks	5	2	80	Testing. FEA analysis. Use stronger materials.
		Auger cannot collection soil.	4	The sharp edges on the Auger become dull	5	7	140	Testing. Use a larger edge angle. Use stronger materials.
		Less than 10 mL of soil is collection	3	Automation falsely reads the amount of soil collected	5	8	120	Testing. Code the automation to collect more than 10 mL.
		Less than 10 mL of soil is collection	6	Batteries deplete all power prematurely	3	3	54	Testing. Make sure battery life will withstand all worst case scenarios
		Auger cannot be fed into the soil. Auger does not collect soil.	4	Corrosion	2	5	40	Use stainless steel parts when possible. Buy coated materials to resist corrosion. Seal from water contamination.

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Table 81: SCAR FMEA – continued from previous page

Function	Failure Mode	Effects of Failure	Severity	Failure Causes	Occurrence	Detection	RPN	Mitigation
Soil Collection auger	Failure to transport the soil to the container	Soil falls out of Auger before reaching the container	4	Auger Soil Containment Tube does not rest fully tangent against Auger	4	7	112	Test. Make sure the containment tube rests tangent to the auger.
		Auger does not rotate. Auger does not deploy into soil. Auger device detaches from the Chassis.	6	Threads become stripped on any component	2	5	60	FEA Analysis. Testing. Use favorable factors of safety.
		Soil cannot be collected.	8	Soil does not fall onto the container opening.	3	4	96	Testing. Use a funnel to bias the soil onto the doors.
		Auger cannot be fed into the soil. Auger does not collect soil.	4	Corrosion	2	5	40	Use stainless steel parts when possible. Buy coated materials to resist corrosion. Seal from water contamination.
Soil Collection Container	Failure to seal the soil	Soil is not sealed.	8	Soil prohibits the doors from closing	2	2	32	Design the container to be much larger than 10 mL.
		Container detaches from the Chassis. The auger does not drop soil onto the container doors. Doors do not open/close	6	Threads are stripped on any component	2	5	60	FEA Analysis. Testing. Use favorable factors of safety.

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Table 81: SCAR FMEA – continued from previous page

Function	Failure Mode	Effects of Failure	Severity	Failure Causes	Occurrence	Detection	RPN	Mitigation
Soil Collection Container	Failure to seal the soil	Gaps are present between the door and container. The doors are not rotated to the complete close position	7	Soil is not contained by the lower door	2	2	28	Make sure motor torque specifications are adequate. FEA Analysis. Testing. Foam around the edges of the doors.
		Doors do not open/close. Soil is not sealed.	4	Corrosion	2	5	40	Use stainless steel parts when possible. Buy coated materials to resist corrosion. Seal from water contamination.

5.3.3 Structures FMEA

In Tables 82 and 84 are the Structures Failure Modes and Effects Analyses which OSRT has developed for the mission.

Table 82: Structures FMEA

Function	Failure Mode	Effects of Failure	Severity	Failure Causes	Occurrence	Detection	RPN	Mitigation
Airframe	Airframe buckling from high loads at landing	Launch vehicle is not recoverable or reusable	8	Recovery system fails to deploy	1	7	56	Model forces acting on the eyenut and determine a factor of safety based on that force, adjusting system if necessary
	Temperatures cause delamination of airframe materials	Launch vehicle goes ballistic, resulting in the destruction of the vehicle.	8	Long term storage or use in high temperature conditions	1	5	40	Model forces and increase size of eye nut to account for forces in system based on calculation of minimum thread engagement
	Zippering along the edges of the tubes	Launch vehicle goes ballistic, resulting in the destruction of the vehicle	6	Recovery lines pulling across edge of tube	5	6	180	Ensuring all black powder charges are properly sized and slightly oversized
Nosecone	Non-Uniform, non-straight nosecone	Non-uniform flight path, increasing drag	5	Nosecone becomes damaged or deformed during testing or upon landing	5	4	100	Metal nosecone tip to absorb direct force, careful inspection before and after use
				Difficulty of getting a good nosecone with the resources available at Oregon State	2	3	30	Purchase nosecone and fit to airframe outer and inner diameter
	Nosecone detaches during flight	Flight failure, loss of nosecone, loss of avionics	8	Not correctly attached to recovery system or tip not properly secured to threaded rod	3	4	96	Follow installation checklists, do final inspection of launch vehicle before sealing

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Table 82: Structures FMEA – continued from previous page

Function	Failure Mode	Effects of Failure	Severity	Failure Causes	Occurrence	Detection	RPN	Mitigation
Nosecone	Nosecone detaches during flight	Flight failure, loss of nosecone, loss of avionics	8	Nosecone threads strip when recovery system forces are transferred to them	2	4	64	Analysis will be performed on required thread depth for tapped hole to prevent striping
Fins	Fins misaligned	Erratic flight profile, loss of launch vehicle	7	Damage to fins caused during transport of the launch vehicle or by hard landing, deformation of material based on atmospheric conditions	3	4	84	Handle fin section carefully, do not put weight on fins, inspect and repair fins after launch, manufacture fins with stiff materials
				Fins not inserted and epoxied at right angles to body tube, or epoxy fails to cure correctly	5	3	105	Build a fin jig for aligning the fins during the build, and allow epoxy to cure before moving
	Fins Fall off	Erratic flight profile, loss of launch vehicle, damage to surroundings	8	Forces at base of fin large enough to cause failure in epoxy	3	4	96	Inspect fins before and after launch, do not launch if fins can move
				Epoxy not allowed to cure properly, insufficient epoxy to maintain strength	1	9	72	Carefully apply epoxy and inspect after drying, properly fillet all edges

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Table 82: Structures FMEA – continued from previous page

Function	Failure Mode	Effects of Failure	Severity	Failure Causes	Occurrence	Detection	RPN	Mitigation
Coupler	Bending in the launch vehicle	Loss of launch vehicle, launch vehicle not reusable	8	Forces experienced by the launch vehicle create enough bending shear that the materials fail	4	3	96	Composites layed-up in directions of highest load with enough thickness to resist the loading
Bulkhead	Epoxy failure	Internal components damaged, launch vehicle unrecoverable, debris injuring spectators, debris littering property	8	Incorrect epoxy mixture used	2	2	32	Use scale to measure epoxy ratio, use unused mixing sticks and bowls for each batch
				Poor surface contact or dirty surface	2	8	128	Sand area where the epoxy will be applied, clean surface with acetone
				Insufficient amount of epoxy applied	1	4	32	Generously apply all epoxy with fillets
	Cross-grain failure	Internal components damaged, launch vehicle unrecoverable, debris injuring spectators, debris littering property	8	Plywood stored improperly	3	6	144	Store plywood in cool, dry areas and handle with care
				Low quality plywood	3	8	192	Select plywood from reputable sources, select best looking plywood
	Fracture or shatter	Internal components damaged, launch vehicle unrecoverable, debris injuring spectators, debris littering property	8	Threaded rod excess force	2	2	32	Use thick plywood and large washers to be able to withstand the force

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Table 82: Structures FMEA – continued from previous page

Function	Failure Mode	Effects of Failure	Severity	Failure Causes	Occurrence	Detection	RPN	Mitigation
Threaded Rod	Fail under tension	Loss of recovery system, loss of launch vehicle	9	Force experienced by the threaded rod during any point of the flight, with the primary concern being recovery, is greater than the strength of the rod	4	5	180	Model forces and increase size of threaded rod to have a safety factor of at least 2 for the entire system
	Strips hardpoints	Loss of recovery system, loss of launch vehicle	9	Force in the system is sufficient to pull the ends of the threaded rods through the threads of the hardpoint mountings	2	4	72	Design the hole depth in hardpoints to be higher than the minimum thread engagement length
Shear pins	Shear pins fail to break	Recovery system does not deploy, loss of launch vehicle, launch vehicle becomes projectile	10	Black power detonation creates insufficient pressure on shear pins	3	5	150	Have backup black powder ejection charges larger than primaries, test all charges with ejection test before launch

Table 83: Electronics Bays FMEA

Function	Failure Mode	Effects of Failure	Severity	Failure Causes	Occurrence	Detection	RPN	Mitigation
E-Match lights black powder	Poor connection to the E-Match	Launch vehicle doesn't separate and parachutes do not deploy	10	E-Match doesn't light	2	6	120	Check E-Match and wiring thoroughly before integration and launch
	E-Match malfunctions	Launch vehicle doesn't separate and parachutes do not deploy	10	Faulty E-Match	1	4	40	Inspect E-Matches before launch
	Poor pressure seal on electronic bay	Launch vehicle doesn't separate and parachutes do not deploy	10	Charges don't generate enough pressure to break shear pins	2	2	40	Test seals and inspect seals before every launch
Altimeters communi-cate when to deploy parachutes	Switches do not turn on properly	The parachutes deploy at incorrect times	8	Altimeters cannot give the signal	2	3	48	Add an LED to the circuit so that we can easily tell if the system is on. Be experienced with the system.
Altimeters communi-cate when to deploy parachutes	Battery power is low	The parachutes may have difficulty deploying	7	Altimeters do not have full power and neither to the E-Matches	3	3	63	Be sure to only use new batteries for each launch.
Provide a connection point for the parachutes	Electronic bay comes loose	The connection point is not strong enough to hold the parachutes in place	8	Bonding between the bay and the airframe is not strong enough	3	2	48	Run analysis on the best way to connect the bays to the airframe for optimum security

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Table 83: Electronics Bays FMEA – continued from previous page

Function	Failure Mode	Effects of Failure	Severity	Failure Causes	Occurrence	Detection	RPN	Mitigation
Provide a connection point for the parachutes	Eye bolt comes loose	The bulkhead/eye bolt cannot withstand the force of the parachute	8	The bulkhead comes loose and the parachutes detach from the the airframe	3	3	72	Run analysis on the bulkheads and eye bolts to ensure that they are strong enough to withstand the force of the parachutes

Table 84: 360 Camera FMEA

Function	**Failure Mode**	**Effects of Failure**	**Severity**	**Failure Causes**	**Occurrence**	**Detection**	**RPN**	**Mitigation**
Cameras film simultaneously	Trigger does not work on all shutter cables	Will not capture full 360 without all cameras on and running	7	Poor cable configuration	2	5	70	Testing of the module outside the launch vehicle before launch
Cameras film simultaneously	Not all cameras are turned on	Will not capture full 360 without all cameras on and running	7	Dead battery	2	2	28	Cameras will be charged before each flight and holes in airframe will show if camera is on or not

5.4 Environmental Concerns

Environmental concerns are very important to consider. Not only can the environment have impacts on the outcome of the project, but the project can have impacts on the condition of the environment. To guarantee mission success, the design of the project must take the varying conditions of the environment into account. This Environmental Hazard Analysis considers how a specific section of the launch vehicle or airframe may fail from environmental factors. The analysis does not include launch canceling environmental hazards such as extreme weather because the mitigation would be to cancel the launch and there is no effort the team can make to prevent this. The Environmental Hazard Analysis follows the same format as the Personal Hazard Analysis.

Table 85: Environmental Hazard on Mission Analysis

Hazard	Cause	Effect	Pre-RAC	Mitigation	Verification	Post-RAC
Launch vehicle flies beyond collection radius	High winds at launch site	Vehicle becomes difficult to recover because of distance traveled, violate the competition drift radius	1B	Adjust launch rail to counteract any wind conditions, reduce the time that the launch vehicle is under parachutes	The launch rail will be angled into the wind to counter act the wind effects, see {Launch Procedures}. The parachutes are designed to keep the drift launch vehicle in the drift radius during any launchable wind conditions, see {Parachute Sizing and Decent Rates}.	1E

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Table 85 – continued from previous page

Hazard	Cause	Effect	Pre-RAC	Mitigation	Verification	Post-RAC
Launch vehicle becomes water lodged	Launch vehicle being left in the rain or snow	On board electronics becoming damaged or destroyed	2C	Call launch in heavy precipitation, for light rain go and retrieve the launch vehicle immediately.	The launch vehicle will be recovered from any launch as soon as the range is deemed safe by the RSO to prevent any long term damage. See {Post Flight Inspection}.	2E
Sand or dirt filling sections of launch vehicle	Landing in loose soil, mud or sand	Pressure relief holes becoming filled, open ends of tubes filled with debris reducing mission performance on next launch	3B	Inspect all opening on Launch vehicle for foreign material and clean appropriately.	The launch vehicle will be inspected after each flight and any foreign material cleaned both internally and externally, see {Post-Flight Inspection}.	3D
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Table 85 – continued from previous page

Hazard	Cause	Effect	Pre-RAC	Mitigation	Verification	Post-RAC
Rover becomes stuck on rocks or mud	Rain creating muddy conditions, natural sections of a field	Rover becomes stuck and is unable to progress	2B	Design drivetrain to be able to drive over and through rough terrain.	The rover wheels and drivetrain were designed to create maximum ground clearance after exiting the launch vehicle, and the motor are designed with a large amount of additional torque to clear sticky terrain. See {Drivetrain}.	2D

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Table 85 – continued from previous page

Hazard	Cause	Effect	Pre-RAC	Mitigation	Verification	Post-RAC
Rover runs into large immobile obstacle	Large obstacles such as trees, shrubs or large rocks may be around the landing area	Rover unable to progress forward, rover damaged by collision	2B	Create a system that can detect and avoid any large objects in front of the rover, and adjust the rover's path accordingly.	An object avoidance system has been created and implemented in the rover using sonar sensors. This systems allows the rover to detect any objects and move around them or adjust course to avoid them. See {Algorithm}, {Sensors} and {Payload Object Detection Test}.	2E
Holes in airframe and electronics bays cause pressure gradient	High winds coming in through airframe holes	Accuracy of avionics is compromised	2C	Design airframe and electronic bay holes such that a pressure gradient is not possible.	Perform anaylysis on holes to make sure a pressure gradient does not occure	3E

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Table 85 – continued from previous page

Hazard	Cause	Effect	Pre-RAC	Mitigation	Verification	Post-RAC
Auger gets clogged with hard dirt or mud	Muddy or dry conditions	Auger will not be effective at collectiong a soil sample for the rover	2C	Design soil collection system to work with a wide range of soil conditions.	The auger and soil collection system as a whole will be tested in a wide range of soil collections and with different auger angles to optimize the system.	2D

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Table 86: Mission Hazard on Environment Analysis

Hazard	Cause	Effect	Pre-RAC	Mitigation	Verification	Post-RAC
Recovery systems fail to deploy	Black powder charges insufficient to cause separation of the launch vehicle, recovery harnesses fail	Entire launch vehicle or sections of it become raining debris, causing injury or damage on impact	1C	Calculate black powder charge sizes and test, design recovery system to take expected loads.	Black powder charge sizing calculations created and tested to achieve complete deployment of recovery system, see {Launch Vehicle Ejection System Test}. The Recovery system was designed to take all loads during parachute deployment, see {Harnesses} and {Attachment Hardware}.	1E
Wadding ejected from launch vehicle during recovery deployment	A wadding of some form is need to create the needed pressure for the charges to separate the launch vehicle	Wadding can become random trash or degrade poorly causing chemical leakage into the ground	3A	Choose a wadding that is biodegradable and collect all excess wadding during ground testing.	A biodegradable cellulose wadding was purchased. This wadding has minimal impact if left outside and will degrade over time; see {Final Assembly Checklist}.	4B

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Table 86 – continued from previous page

Hazard	Cause	Effect	Pre-RAC	Mitigation	Verification	Post-RAC
Faulty batteries	Faulty batteries leaking chemicals	Chemical components of the batteries can leak into the ground and water becoming hazardous to local flora and fauna	2B	Batteries will be inspected before use for any defects and properly disposed of if found to be faulty.	Batteries are inspected prior to use for any defects that would cause the to be faulty, see {Fore Ejection Unit} and {Aft Ejection Unit}.	2E
Fire from ignition	Motor ignition setting surrounding brush on fire	Creation of fire in scrub or grass leading to possible wide spread damage to plants and animals	1C	Only launch at certified sites under the supervision of a licensed RSO and follow all instruction given by the RSO.	For all launches a certified RSO will select the location for the launch pad and their instruction will be followed for launching, see {Launch Procedure}.	1E

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5.5 Associated Risks

All team members, volunteers, and spectators will be made aware of any associated risks prior to the task at hand. When manufacturing, all personnel will wear the proper Personal Protective Equipment (PPE) and take care of any environmental hazards that pertain to the task at hand. To ensure safety, only shop certified students will partake in the manufacturing process. During assembly, all personnel will be made aware of any risks that they may encounter, such as sharp edges or chemicals with toxic fumes. Launch days will have several associated risks, and all personnel will be made aware of these prior to the launch day.

6 Project Plan

6.1 Requirements Verification

All requirements are to be verified by one of the following methods: Test, Analysis, Demonstration, or Inspection. These are noted under the Verification Method column by their respective first letters (T, A, D, I). All requirements which are to be verified through testing will have accompanying test plans developed.

6.1.1 Competition Rules

Shown in Tables 88 through 91 is a breakdown of the competition rules. Included are the general, launch vehicle, recovery, payload, and safety verification matrices. Each matrix has a brief description of how OSRT is verifying these requirements will be completed, the current status of the verification implementation, and the verification method.

Table 87: Vehicle Rules Verification Matrix

Requirement	Verification Method	Verification Plan	Status
1.1 Students on the team will do 100% of the project, including design, construction, written reports, presentations, and flight preparation with the exception of assembling the motors and handling black powder or any variant of ejection charges, or preparing and installing electric matches (to be done by the team's mentor).	I	Students will do all the work of the project.	In progress - students have done all the work of the project so far, and will continue to do so for the remainder of the project life cycle.
1.2. The team will provide and maintain a project plan to include, but not limited to the following items: project milestones, budget and community support, checklists, personnel assignments, STEM engagement events, and risks and mitigations.	I	OSRT will provide and maintain a project plan throughout the project.	In progress - OSRT has provided and maintained a project plan so far. A project plan will be maintained throughout the remainder of the project life cycle.
1.3. Foreign National (FN) team members must be identified by PDR and may or may not have access to certain launch week activities during launch week due to security reasons. In addition, FNs may be separated from their team during certain activities.	I	OSRT will identify foreign exchange students by PDR.	Complete - no foreign exchange students are a part of the current OSRT roster.
1.3. The team must identify all team members attending launch week by the Critical Design Review (CDR).	I	OSRT will identify all members attending launch week by CDR.	Incomplete - team members attending launch week activities will be identified by CDR.
1.4.1. Students actively engaged in the project throughout the entire year.	I	OSRT will identify all members attending launch week by CDR.	Incomplete - team members attending launch week activities will be identified by CDR.

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Table 87 – continued from previous page

Requirement	Verification Method	Verification Plan	Status
1.4.2. One mentor (see requirement 1.13).	I	OSRT will identify all members attending launch week by CDR.	Incomplete - team members attending launch week activities will be identified by CDR.
1.4.3. No more than two adult educators.	I	OSRT will identify all members attending launch week by CDR.	Incomplete - team members attending launch week activities will be identified by CDR.
1.5. The team will engage a minimum of 200 participants in educational, hands-on science, technology, engineering, and mathematics STEM activities, as defined in the STEM Engagement Activity Report, by FRR. To satisfy this requirement, all events must occur between project acceptance and the FRR due date and the STEM Engagement Activity Report must be submitted via email within two weeks of the completion of the event.	D	OSRT will engage a minimum of 200 participants in STEM lessons before FRR.	Complete - OSRT has engaged 980 participants in STEM lessons.
1.6. The team will establish a social media presence to inform the public about team activities.	D	The team will establish social media presence on Facebook, Twitter, Instagram, and Snapchat.	Complete - social media presence has been established.
1.7. Teams will email all deliverables to the NASA project management team by the deadline specified in the handbook for each milestone. In the event that a deliverable is too large to attach in an email, inclusion of a link to download the file will be sufficient.	D	The team will submit all deliverables appropriately.	In progress - all documentation has been and will continue to be submitted appropriately.
1.8. All deliverables must be in PDF format.	I	The team will submit all deliverables appropriately.	In progress - all documentation has been and will continue to be submitted appropriately.

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Table 87 – continued from previous page

Requirement	Verification Method	Verification Plan	Status
1.9. In every report, teams will provide a table of contents including major sections and their respective sub-sections.	I	The team will include a table of contents with all reports.	In progress - a table of contents has been included on all documentation, and will be included for all future documentation.
1.10 In every report, the team will include the page number at the bottom of the page.	I	The team will include a page number on all pages.	In progress - page numbers have been on all documentation, and will continue to be included for all future documentation.
1.11 The team will provide any computer equipment necessary to perform a video teleconference with the review panel. This includes but is not limited to, a computer system, video camera, speaker telephone, and a sufficient internet connection. Cellular phones should be used for speakerphone capability only as a last resort.	I	The team will use a conference room with necessary capabilities.	Complete - two conference rooms have been selected for use in teleconferences. These rooms have all necessary equipment.
1.12 All teams will be required to use the launch pads provided by Student Launch's launch services provider. No custom pads will be permitted on the field. Eight foot 1010 rails and 12 foot 1515 rails will be provided. The launch rails will be canted 5 to 10 degrees away from the crowd on launch day. The exact cant will depend on wind conditions.	I	The team will make use of a 12 foot 1515 rail for all designs.	Complete - simulations and designs account for a 12 foot 1515 rail.
1.13 Each team must identify a mentor.	I	Team will identify a mentor.	Complete - Joe Bevier has been identified as the OSRT mentor.

Table 88: Vehicle Rules Verification Matrix

Requirement	Verification Method	Verification Plan	Status
2.1. The vehicle will deliver the payload to an apogee altitude between 4,000 and 5,500 ft AGL.	T	The motor selection is based on OpenRocket simulation to reach the required AGL range. This will be determined as the team refines the design and determines a definite weight.	In progress - launch vehicle has been designed to meet requirements.
2.2. Teams shall identify their target altitude goal at the PDR milestone.	A	The target AGL goal has been set during PDR.	Completed - launch vehicle has been designed to reach 4,500 ft.
2.3. The vehicle will carry one commercially available, barometric altimeter.	I	The launch vehicle will contain a commercially available barometric altimeter.	In progress - multiple commercially available altimeters have been selected in current design. Commercially available barometric altimeters will be used with any future design changes.
2.4. Each altimeter will be armed by a dedicated mechanical arming switch that is accessible from the exterior of the rocket airframe when the rocket is in the launch configuration on the launch pad.	I	The location of the altimeter housing will allow for each altimeter arming switch to be activated from the exterior of the launch vehicle.	In progress - arming switches will be accessible from exterior of launch vehicle. Externally armed mechanical switches will be present in all future designs.
2.5. Each altimeter will have a dedicated power supply.	I	All altimeters will have their own dedicated power supply.	In progress - each altimeter in design has dedicated power supply in design.
2.6. Each arming switch will be capable of being locked in the ON position for launch (i.e. cannot be disarmed due to flight forces).	D	All arming switches will have a mechanical locking system.	In progress - arming switches in all designs are armed through use of hex key which maintains ON position throughout flight.

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Table 88 – continued from previous page

Requirement	Verification Method	Verification Plan	Status
2.7. 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.	T	The launch vehicle will be designed to survive launch and recovery without needing repairs or modifications prior to an additional same day launch.	Completed - launch vehicle designs have been made to withstand all expected forces of launch and recovery. Any design changes will be capable of withstanding all expected forces of launch and recovery.
2.8. The launch vehicle will have a maximum of four (4) independent sections.	I	The launch vehicle will have no more than four independent sections.	Completed - launch vehicle has been designed to have three independent sections.
2.9. The launch vehicle will be limited to a single stage.	I	The propulsion system will consist of only one motor.	Completed - only one motor will be used.
2.10. The launch vehicle will be capable of being prepared for flight at the launch site within 2 hours of the time the FAA flight waiver opens.	D	The team will perform preparation drills to practice assembling and readying the launch vehicle within two hours.	In progress - design for assembly is being emphasized. Testing and practice will be implemented to ensure assembly process is less than two hours.
2.11. The launch vehicle will be capable of remaining in launch-ready configuration on the pad for a minimum of 2 hours without losing the functionality of any critical on-board components.	T	The team will perform testing for leakage current in order to optimize energy usage of all electrical systems.	In progress - all batteries were chosen to be capable of maintaining functionality for more than 2 hours. Testing will be conducted with all systems prior to launch.
2.12. 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 launch services provider.	T	The launch vehicle will have a separate launch system that is powered by an external 12-volt system.	Completed - current motor choice and all alternative motor choices are capable of being launched by standard 12-volt direct current firing system.

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Table 88 – continued from previous page

Requirement	Verification Method	Verification Plan	Status
2.13. The launch vehicle will require no external circuitry or special ground support equipment to initiate launch.	I	All electrical systems will run autonomously and wait for launch, internally. Acceleration sensors will inform the control systems of launch.	Completed - no external circuitry will be used.
2.14. The launch vehicle will use a commercially available solid motor propulsion system using ammonium perchlorate composite propellant Ammonium Perchlorate Composite Propellant (APCP) which is approved and certified by NAR, TRA, Canadian Association of Rocketry (CAR).	A	The launch vehicle will be designed to use a commercially available motor that is approved and certified by the NAR, TRA, and/or the CAR.	Completed - current motor choice and all motor alternative choices meet these requirements.
2.15. Pressure vessels on the vehicle will be approved by the RSO and will meet the provided criteria.	I	Pressure vessels will not be integrated into the launch vehicle.	Completed - no pressure vessels are used in current designs.
2.16. The total impulse provided by a College or University launch vehicle will not exceed 5,120 Newton-seconds (L-class).	I	The motor selection will be limited to using a L-class or lower as to not exceed 5,120 Newton-seconds of impulse.	Completed - current motor choice and all motor alternative choices meet these requirements.
2.17. The launch vehicle will have a minimum static stability margin of 2.0 at the point of rail exit.	A	Update weights and dimensions of in OpenRocket simulation for design changes. Modify fin shape to control stability until the time of manufacture. After manufacture, maintain any minor stability changes through adjusting ballast masses.	In progress - static stability has been determined and will be adjusted as design progresses. A minimum stability of 2.0 will be maintained.
2.18. The launch vehicle will accelerate to a minimum velocity of 52 fps at rail exit.	A	The selected motor will be simulated to achieve over 52 fps off of a 12 ft rail provided by USLI.	Completed - rail exit velocity is 83.8 fps with motor choice.
2.19. All teams will successfully launch and recover a subscale model of their launch vehicle prior to CDR. Subscales are not required to be high power rockets.	T	The team will successfully create, launch, and recover a subscale launch vehicle prior to submitting the CDR.	In Progress - manufacturing of a subscale launch vehicle has begun. The launch will be completed prior to CDR.

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Table 88 – continued from previous page

Requirement	Verification Method	Verification Plan	Status
2.20. All teams will complete demonstration flights.	T	The team will launch a subscale and full scale launch vehicle with retained payload included in the full scale, eliminating the need of a simulation mass. Both vehicles will be built with resources available at OSU, fully equipped with chutes and avionics. Information recovered from the flight will reported on the FRR. The launch vehicle will not be modified at this point. If re-flight is necessary, proper documentation will be filed for an extension which would be done before the FRR deadline.	In Progress - will be completed by each of the specified deadlines.
2.21. An FRR Addendum will be required for any team completing a Payload Demonstration Flight or NASA required Vehicle Demonstration Re-flight after the submission of the FRR.	D	If the team fails to complete a Payload Demonstration Flight prior to the FRR, the team will follow the proper procedure for re-launch.	Incomplete - will be completed by specified deadline if necessary.
2.22. Any structural protuberance on the launch vehicle will be located aft of the burnout center of gravity.	A	Any structural protuberances will be located behind the burnout center of gravity.	Completed - analyses have been conducted to show all protuberances in design are aft of center of gravity.
2.23. The team name and launch day contact information will be in or on the launch vehicle airframe as well as in or on any section of the vehicle that separates during flight and is not tethered to the main airframe.	I	There will be sufficient and obvious contact information on each section of the launch vehicle.	Incomplete - contact information will be labeled during manufacture of the launch vehicle.

Table 89: Recovery System Verification Matrix

Requirement	Verification Method	Verification Plan	Status
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.	T	At apogee, an ejection charge will separate the fore from the aft, and another will separate the fore from the nosecone. The ejection charge in the middle of the launch vehicle will separate the aft from the fore section, as well as push the drogue out of the aft section. An ejection charge located in the upper fore section will separate the nosecone from the fore section as well as push the drogue out of the fore section. At 500 feet AGL, the main fore parachute will be pulled out by the fore drogue parachute after a Tender Descender releases it, and the aft main parachute is pulled out by the drogue parachute after a Tender Descender releases it.	Incomplete - will be completed completed by FRR.
3.1.1 The main parachute shall be deployed no lower than 500 feet.	T	Barometric altimeters will sense the altitude AGL, and they will ignite the black powder in the Tender Descenders and the parachutes will deploy at 500 ft AGL.	Incomplete - will be tested on subscale and full scale flights.
3.1.2 The apogee event may contain a delay of no more than 2 seconds.	A	When the barometric altimeters sense apogee, the primary ejection charges will be set to 0 seconds, and the backup charges delay will be set one second later, less than two seconds after apogee is sensed.	Completed - primary charge will occur with no delay and secondary charge will occur with one second delay.
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.	T	Ground ejection tests will be performed prior to all launches, subscale and full scale, to ensure all parachutes are ejected properly during the launch.	Incomplete - will be completed prior to all launches.

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Table 89 – continued from previous page

Requirement	Verification Method	Verification Plan	Status
3.3 At landing, each independent section of the launch vehicle will have a maximum kinetic energy of 75 ft-lbf.	T	Appropriate main parachute sizes have been chosen in order to keep each independent section under 75 ft-lbf of kinetic energy upon landing.	In progress - analysis has been conducted to size all parachutes to meet kinetic energy requirements.
3.4 The recovery system electrical circuits will be completely independent of any payload electrical circuits.	I	All recovery and payload circuits will be independent of each other.	Completed - design has accounted for completely independent circuits for payload and recovery.
3.5 All recovery electronics will be powered by commercially available batteries.	I	All batteries will be purchased from a vendor determined to have batteries which meet all required needs.	In progress - all batteries will be purchased through reputable vendors.
3.6 The recovery system will contain redundant, commercially available altimeters. The term ”altimeters” includes both simple altimeters and more sophisticated flight computers.	I	The launch vehicle will contain four altimeters, two primary and two secondary. The primary will be PerfectFlite StratologgerCF, and the secondary will be MissileWorks RRC3.	Completed - redundant, commercially available altimeters have been selected.
3.7 Motor ejection is not a permissible form of primary or secondary deployment.	I	The motor will not be ejected from the launch vehicle during primary or secondary deployment.	Completed - has accounted for no motor ejection.
3.8 Removable shear pins will be used for both the main parachute compartment and the drogue parachute compartment.	I	Nylon shear pins will be used for all parachute compartments to ensure the launch vehicle’s sections are fixed together until ejection charges are fired.	Completed - all couplers include nylon shear pins in design.
3.9 Recovery area will be limited to a 2,500 ft radius from the launch pads.	T	Each independent section will fall quickly and controlled under drogue parachutes. The main parachutes will deploy, and the launch vehicle sections will fall as quickly as possible while staying under the kinetic energy requirement to limit drift.	In progress - multiple simulations have been performed which demonstrate recovery of all sections will fall within 2,500 ft radius.

Continued on next page

Table 89 – continued from previous page

Requirement	Verification Method	Verification Plan	Status
3.10 Descent time will be limited to 90 seconds (apogee to touch down).	T	To limit descent time spent under the main parachutes, the launch vehicle has been split into two independent sections. This allows for safer descent rates under the drogue parachutes while still staying under the required 90 seconds and meeting the kinetic energy requirement.	In progress - multiple simulations have been performed which demonstrate recovery of all sections within 90 seconds.
3.11 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.	I	The launch vehicle will contain two avionics units: one in the fore section and one in the aft section.	Completed - design has accounted for the inclusion of tracking systems.
3.11.1 Any launch vehicle section or payload component, which lands untethered to the launch vehicle, will contain an active electronic tracking device.	I	The fore and aft sections will land independently of each other. Both will contain avionics systems in their respective avionics bays.	Completed - design includes two avionics systems on both independently recovered sections of the launch vehicle.
3.11.2 The electronic tracking device(s) will be fully functional during the official flight on launch day.	T	All tracking systems will be tested on launch day to ensure they are working correctly.	Incomplete - will be completed during launches.
3.12 The recovery system electronics will not be adversely affected by any other on-board electronic devices during flight (from launch until landing).	I	All recovery systems needing protection will have proper protection, eliminating any adverse reactions.	Completed - design has accounted for appropriate shielding and protection of electronics.
3.12.1 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.	I	All electronics producing radio frequency or magnetic waves will be located in separate compartments from all altimeters and recovery electronics.	Completed - design has accounted for appropriate shielding and protection of electronics.
3.12.2 The recovery system electronics will be shielded from all on-board transmitting devices to avoid inadvertent excitation of the recovery system electronics.	I	All recovery system electronics will have proper protection, shielding from transmitting devices using conductive spray paint, ensuring charges are not ignited early.	Completed - design has accounted for appropriate shielding and protection of electronics.

Continued on next page

Table 89 – continued from previous page

Requirement	Verification Method	Verification Plan	Status
3.12.3 The recovery system electronics will be shielded from all on-board devices which may generate magnetic waves (such as generators, solenoid valves, and Tesla coils) to avoid inadvertent excitation of the recovery system.	I	The recovery system electronics will be shielded from all magnetic waves produced by any device onboard the launch vehicle using conductive spray paint to avoid inadvertent excitation of the recovery system.	Completed - design has accounted for appropriate shielding and protection of electronics.
3.12.4 The recovery system electronics will be shielded from any other onboard devices which may adversely affect the proper operation of the recovery system electronics.	I	The recovery system will be appropriately shielded from all devices which may have any adverse effect on the recovery system electronics using conductive spray paint.	Completed - design has accounted for appropriate shielding and protection of electronics.

Table 90: Payload Requirements

Requirement	Verification Method	Verification Plan	Status
4.3.1 The team's custom rover must deploy from the internal structure of its launch vehicle.	D	The rover will be situated within the fore section of the launch vehicle until landing.	Complete - Ejection and retention system will hold payload throughout launch and flight, and will deploy once safely landed.
4.3.2 The team's launch vehicle will feature a fail-safe active retention system to maintain control of the payload, even under atypical flight forces.	D	The retention system will feature an ARRD and two Tender Descenders for redundancy and retention under atypical conditions.	Complete - Retention devices to be used will still retain the payload even in electrical and mechanical failures.
4.3.3 Once on the ground, the team's rover must be deployed remotely.	T	The payload will be ejected by way of a properly sized black powder charge.	Incomplete - Charge size testing will occur after manufacture of payload.
4.3.4 The team's rover must travel at least 10 ft from its launch vehicle before collecting a soil sample.	T	The rover will drive itself away from its starting location for a duration long enough to guarantee more than 10 ft of covered ground.	Incomplete - Rover autonomous driving.
4.3.5 The collected sample must be greater than or equal to 10 mL in volume.	T	The sample collection mechanism will empty itself into a storage compartment several times to guarantee more than 10 mL of soil.	Incomplete - Auger and soil collection designs will be tested.
4.3.6 The soil sample must be stored in a compartment that can be closed to prevent contamination.	T	The storage compartment will feature a motor driven door.	Incomplete - Soil retention designs will be tested.
4.3.7 All rover batteries must be protected from impact with the ground.	I	Battery module will be contained within the chassis.	Complete - Chassis is designed to be very durable, and surrounds the battery module.
4.3.8 All rover batteries must be brightly colored, clearly marked as a fire hazard, and easily distinguishable from other rover parts.	I	Battery module(s) will be marked with colored electrical tape.	Incomplete - Batteries will be marked once purchased.

Table 91: Safety Requirements Verification Matrix

Requirement	Verification Method	Verification Plan	Status
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.	I	Each subteam will create a checklist of their required items. Checklists will be compiled and verified by the Safety Officer. All team members will verify checklists and comply with them at launch.	In progress - subscale checklists are under development. Full scale checklists will be under development as the team begins manufacture and assembly of full scale launch vehicle and payload.
5.2 Each team must identify a student safety officer who will be responsible I for all items in section 5.3.	I	The team Safety Officer has been selected.	Complete - the team Safety Officer is Jon Verbiest.
5.3 The role and responsibilities of each safety officer will include, but not limited to:	I	Safety Officer will manage all roles outlined within requirement section 5.3. There are two additional safety officers for the launch vehicle and payload. These two sub-team safety officers report to the Safety Officer and are responsible for maintaining safe practice in the case that the Safety Officer cannot be at the event	In progress - the safety officer is currently, and will continue to be responsible for team safety in all aspects.
5.3.1 Monitor team activities with an emphasis on Safety during:	I	Lead Safety Officer or sub-team Safety Officer will be present during all team activities which pose a safety risk.	In progress - Safety Officers will be present when necessary through all of manufacturing and testing phase.
5.3.1.1 Design of vehicle and payload	I	The Lead Safety Officer or one of the sub-team Safety Officers will be at all internal design reviews to make sure all design decisions follow all safety requirements. The Safety Officer has final say over a design when the safety of a design is in question.	In progress - Safety Officers are present at all design reviews to provide safety input when necessary.

Continued on next page

Table 91 – continued from previous page

Requirement	Verification Method	Verification Plan	Status
5.3.1.2 Construction of vehicle and payload	I	Before construction begins, the Safety Officer will inform all team members of potential hazards and mitigation plans. The Safety Officer will make sure that JHA forms are filled out prior to any manufacturing. One of the Safety Officers will stand by to assist in fulfilling safety protocols. Manufacturing safety rules are set in place by the OSU Machine Product and Realization Laboratory (MPRL)	Complete - Safety Officers have given safety briefings to team and taught a lesson on filling out JHA forms.
5.3.1.3 Assembly of vehicle and payload	I	Before assembly begins, Safety Officer will inform all team members of potential hazards and mitigation plans. One of the Safety Officers will stand by to provide assistance in fulfilling safety protocols. One of the safety Officers will verify checklists with their respective sub-team leads before assembly.	In progress - all checklists require a signature from safety officer to verify proper assembly procedures have occurred. Checklist development is still not complete, but all checklists will follow the same format.
5.3.1.4 Ground testing of vehicle and payload	I	Before ground testing begins the Safety Officer will inform all involved team members on potential hazards and mitigation plans. The lead Safety Officer or one of the sub-team Safety Officers will stand by to provide assistance in fulfilling safety protocols.	Incomplete - OSRT has not begun ground testing of vehicle or payload.
5.3.1.5 Subscale launch test(s)	I	During launch assembly, the Safety Officer will be responsible for monitoring checklists, PPE, and troubleshooting steps. The Officer ensured that the team was in compliance with safety restrictions set by the RSO.	Incomplete - OSRT has not launched the subscale launch vehicle.

Continued on next page

Table 91 – continued from previous page

Requirement	Verification Method	Verification Plan	Status
5.3.1.6 Full-scale launch test(s)	I	Before full-scale launch the Safety Officer will complete a checklist for launch with the help of the members taking part of the launch. The Safety Officer swill inform all members on the rules and regulation of the launch site and each members' role during the launch. A final check off of all components will then be carried out by the Safety Officer.	Incomplete - OSRT has not launched the full-scale launch vehicle.
5.3.1.7 Launch day	I	Before Launch Day, the Safety Officer will complete a checklist for launch with the help of the members taking part if the launch. The Safety Officer will inform all members on the rules and regulation of the launch site and each members role during the launch. A final check off of all components will then be carried out by the Safety Officer. Lead Safety Officer or one of the sub-team Safety Officers will stand by to make sure all safety regulations are followed throughout the duration of the launch activities.	Incomplete - no launch days have occurred yet.
5.3.1.8 Recovery activities	I	The Safety Officer will work closely with the appropriate range officers to determine the appropriate time to collect the launch vehicle. The Safety Officer will inform all team members of potential hazards and mitigation plans.	Incomplete - no recovery activities have occurred yet.
5.3.1.9 Educational Engagement Activities	I	Safety Officer will approve all engagement activities for safety.	In progress - Safety Officer has been present for engagement activities and approved lesson plans.

Continued on next page

Table 91 – continued from previous page

Requirement	Verification Method	Verification Plan	Status
5.3.2 Implement procedures developed by the team for construction, assembly, launch, and recovery activities.	I	The Safety Officer will verify all checklists and make sure all team members are informed of them. The Safety Officer will be in charge of making sure all checklists are followed.	In progress - procedures have been developed and will continue to be developed as necessary.
5.3.3 Manage and maintain current revisions of the team's hazard analyses, failure modes analyses, procedures, and Material Safety Data Sheet (MSDS)/chemical inventory data.	I	The Safety Officer will collect all required forms and analyses and make sure that they are available to all team members. New versions will replace older editions	In progress - Safety Officer is managing and maintaining current hazard analyses and FMEAs.
5.3.4 Assist in the writing and development of the team's hazard analyses, failure modes analyses, and procedures.	I	The Safety Officer will be in charge of collecting, compiling and reviewing all hazard analyses, failure mode analyses and procedures.	In progress - Safety Officer assists and reviews all hazard analyses and FMEAs.
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.	I	The team will communicate with the RSO for all test launches. The Safety Officer will work closely with the RSO and any concerns from the RSO will either be addressed before launch or the launch rescheduled to allow for more time to address them. The team understands that the decisions of the RSO are final and the RSO has the power to postpone or cancel any launch activities. NASA gives no authority regarding any test launches completed by the OSRT.	In progress - team has been briefed on OROC launch site rules. The team will work with the OROC RSO for all launches and review launch site rules prior to all launches.
5.5 Teams will abide by all rules set forth by the FAA.	I	The team has knowledge of all appropriate FAA regulations and will abide by them. The Safety Officer is responsible for verification that regulations are adhered to and will be assisted by the sub-team Safety Officers.	In progress - the team will abide by FAA rules for all launches.

6.1.2 Team Derived Rules

Shown in Tables 92 through 96 is a breakdown of the team derived requirements for the general, launch vehicle, recovery, payload, and safety sections. Included is a brief description of how OSRT is verifying these requirements, and the current status of the verification implementation.

Table 92: Team Derived General Verification Matrix

Requirement	Verification Method	Verification Plan	Status
All documentation will be created by OSRT and submitted by the OSRT competition deadlines.	I	The team will be made aware of all deadlines well in advance. All team members will contribute to have all documents ready for submission in advance of the deadlines.	In progress - to date, all OSRT documentation has been submitted on time.
The team will develop a unique mission patch which is representative of the 2019 OSRT competition that is specific to OSRT.	I	Team members will develop ideas and select the best of these ideas.	In progress - several rough draft mission patches have been submitted.OSRT will continue to refine and update these ideas.
The team will engage over 500 people from the community in STEM lessons and activities.	I	Conduct a variety of lesson plans targetted at all age groups, with the team's primary focus on students K-12.	Complete - more than 500 students have been taught already. Additional opportunities will still be pursued by the team to benefit the community and maintain a strong OSRT presence.
The STEM lesson plan ideas and outreach contact information will be saved in an appropriate location to provide the 2020 OSRT members with.	I	All of the pertinent information will be stored on the OSRT Google Drive.	In progress - all current information is being managed and stored on the OSRT Google Drive.
Documentation of available OSRT resources and materials will be saved in an appropriate location to provide the 2020 OSRT members with.	I	All of the pertinent information will be stored on the OSRT Google Drive.	In progress - all current information is being managed and stored on the OSRT Google Drive.

Continued on next page

Table 92 – continued from previous page

Requirement	Verification Method	Verification Plan	Status
The team will appropriately display logos of all sponsor logos and information.	I	The finance team will request high resolution logos from all sponsors when they decide to sponsor OSRT.	In progress - all sponsors who have committed to sponsoring the team have high resolution logos which have been obtained.
At least 25% of the team members will receive a Level 1 HPR Certification from NAR or TRA.	I	Build certification rockets and provide the team with opportunities at all OSRT launch opportunities.	In progress - 16 team members have purchased or built a Level 1 HPR Certification rocket. Kits will be flown when opportunities are available.
The team will get volunteer members involved with the project.	I	Develop a list of projects which are capable of being completed by engineers of varying levels of expertise, providing support for these projects throughout their life cycle.	Complete - to date, 39 volunteers have attended OSRT meetings. These students are working on a wide array of volunteer lead team projects, which are each supported by a mentor who is completing the OSRT competition for course credit as a part of their Capstone class for OSRT. The OSRT will continue to involve volunteers and engage students in the project for the duration of the school year.
All senior Mechanical Engineers will obtain certifications to work in the at OSRT.	I	Complete the ME 250 course offered by OSRT to receive safety training and learn about manufacturing processes on a variety of equipment present at OSRT.	Complete - All senior Mechanical Engineers are certified to work in the .

Table 93: Team Derived Vehicle Verification Matrix

Requirement	Verification Method	Verification Plan	Status
All components will be able to withstand the heat and pressure from ejection charges.	I	All components will be inspected after ground tests are completed for damage or burning. Designs will be changed if they do not pass.	Incomplete - testing will be conducted on subscale and full scale launch vehicles.
Launch vehicle will not be over stable or susceptible to weather cocking.	A	Stability will be limited to maximum of 3.5 at rail exit.	Completed - OpenRocket simulations have shown stability to be 2.1. The simulations will be updated to account for future design changes.
Launch vehicle will be able to be stowed in a 4 ft x 4 ft x 2 ft container for shipping.	I	Launch vehicle will be able to disassemble into sections no longer than 4 ft and no wider than 2 ft to fit into container.	Completed - sections will fit within the container.
The launch vehicle will be recoverable and reusable.	I	The launch vehicle will be using the recovery system in place to minimize any kinetic energy upon landing. Once the launch vehicle is recovered, an inspection will be done to determine its structural integrity. The inspector will be looking for any signs of tearing, deformation, and any other visual indicators of wear. If there are any visual indicators of damage, then appropriate actions will be taken to repair the body. If there are no indicators of damage to the body, then the launch vehicle will be deemed capable of launching again.	Incomplete - will be completed after successful recovery of full scale launch vehicle.
Motor will provide required thrust to launch vehicle.	A	Using OpenRocket simulations, OSRT will be able to test the thrust output of the motor acting on the launch vehicle.	Completed - L2375-WT has been selected based on OpenRocket simulations.

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Table 93 – continued from previous page

Requirement	Verification Method	Verification Plan	Status
The launch vehicle will accommodate the payload and avionics systems.	D	Launch vehicle will have specific areas to accommodate the payload and avionics system in the payload bay and avionics bay, respectively. Both the payload and avionics system will be designed within the constraints of the interior dimensions or their respective bay. The payload and avionics system will be verified upon assembly of full scale launch vehicle.	Complete - payload and avionics will fit in the airframe.
The launch vehicle will be able to be rapidly integrated for launch.	D	Launch vehicle assembly integration will be practiced to minimize assembly time.	In progress - design for assembly is being emphasized throughout the design process.
MATLAB scripts will be used in conjunction with all OpenRocket simulations.	A	Descent velocities, descent trajectory, landing energy, and an estimated apogee will be calculated using MATLAB scripts. All simulations will be checked to ensure no values disagree by more than 15%.	In progress - MATLAB code has been developed for necessary calculations thus far.
Bulkheads will have a factor of safety of 2.0 with respect to the maximum pressure forces experienced during separation.	T	Maximum pressure forces and stress on the bulkheads and at the bulkhead/airframe bond will be calculated and compared to the epoxy's bond strength.	Incomplete - will be accounted for and completed in final design.
Threaded rods will have a minimum tensile safety factor of 5.0 during recovery.	A	Simulations will determine maximum forces on the recovery harness, which will be compared to the tensile strength of selected threaded rod.	Complete - threaded rods have been sized appropriately to provide a safety factor above 5.0.
All threaded attachments have a length greater than 1.5 times the minimum shear length of the selected threads.	D	All purchased threaded components will be compliant, all manufactured components will be compliant.	Incomplete - will be accounted for and completed in final design.

Table 94: Team Derived Recovery Verification Matrix

Requirement	Verification Method	Verification Plan	Status
Separation ejection charges will eject parachutes a minimum of five consecutive times during testing	T	Separation ejection charges will be sized based on initial calculations and ground tested for reliable parachute ejection. If a ground test fails, charge size will be increased or charge construction reevaluated. Testing will take place prior to subscale and full scale flights.	Incomplete - will be tested after complete manufacture of subscale and full scale airframes.
Recovery system will allow for the payload to be deployed from the airframe.	T, D	Designs will allow for an open end of the launch vehicle for the payload to deploy from.	Completed - recovery system design allows for the payload to be deployed from the aft end of the fore airframe.
Avionics will be able to track the the launch vehicle during and after the flight, broadcasting GPS information to a ground station.	T, D	An avionics board has been developed by OSRT which is capable of tracking the launch vehicle throughout flight. This will be tested on all flights, including subscale.	Incomplete - the avionics system will be tested on subscale and full scale launches.
Avionics will have enough transmission power to communicate with the ground station through the entirety of the flight.	T	The avionics system will be sufficiently ground tested.	Incomplete - the avionics system will undergo testing.
Avionics system will have a minimum of four hours of battery life to transmit to the ground station after being armed.	T	The battery life of the system will be tested.	Incomplete - the avionics system will undergo testing.
Recovery system will have appropriately sized static ports.	A, D	Calculations will be performed to determine the correct sizing and number of static port holes.	Completed - the port hole calculations have been performed.

Table 95: Team Derived Payload Requirements

Requirement	Verification Method	Verification Plan	Status
Payload will be an adequate weight for both flight trajectory and payload mission.	T	Design weight will be based upon ideal flight simulations. Testing will occur during test launches and payload test missions	In Progress - Initial designs have been modeled. Additional design changes will involve reducing weight when feasible. Manufacturing will begin soon.
Payload will remain launch ready on the rail for an extended period of time.	T	Testing will occur during test launches and payload test missions.	In Progress - Electrical components and motors are specified. A test-rover will be manufactured soon to test all battery related components.
Payload will be integrated into the airframe quickly.	D	Payload will be designed for quick integration and practiced until the process is streamlined.	Completed - Battery sizing and circuit designs account for being left on the rail for an extended period of time.
The rover and retention system will be able to withstand all potential forces acting on them.	T	Finite Element Analysis will be performed within 3D modeling programs. Testing will occur during test launches and payload test missions.	Incomplete - Rover retention systems will be tested on full scale flights.
All components will withstand heat and pressure from ejection charges.	T	Tests will be performed during test launches and ejection ground testing. Finite Element Analysis will be performed within 3D modeling programs.	Complete - all components will be able to withstand the pressure of the ejection charges.

Table 96: Team Derived Safety Verification Matrix

Requirement	Verification Method	Verification Plan	Status
All team members that use the manufacturing and machining facilities at OSU will have appropriate certification.	I	All team members who need to use the OSU, MPRL, the woodshop, or the composites manufacturing lab will get appropriate certification from the administrator of said lab before use.	In Progress - Necessary certifications will be obtained as they are required.
Additional team members to assist the SO in explicitly promoting team safety and the preparation of safety documents.	I	Two additional safety officers, a Launch Vehicle Safety Officer and a Payload Safety Officer will assist the Team Safety Officer. JHA form developed for internal use when completing hazardous tasks.	Completed - Two team members volunteered for Launch Vehicle Safety Officer and Payload Safety Officer.
The team will secure all hazardous material so only certified personnel can access them.	I	Hazardous materials will be kept in a separate area of the team workspace secured with a lock. Only team leaders, Safety Officer (SO), and team mentors will have access to the hazardous materials.	Completed - Hazardous materials have been locked away in cabinets.
The team will follow all safety rules and guidelines set by the NAR, TRA and OSU.	D	The SO will understand both NAR/TRA safety regulations, OSU safety codes and will ensure team members abide by all rules. Team members are also expected to be familiar with all safety regulations.	In Progress - All team members have followed all safety regulations so far.
The team will have written checklists with instructions on how to safely assemble the rover, recovery systems, and launch vehicle.	I	Each team member or sub-team responsible for designing a part on any assembly pertaining to the launch vehicle or payload will write a formal checklist to ensure that any team member can assemble the part without the presence of the designer of the part. All checklists will be verified by assembler, inspector, and safety officer.	Incomplete - Will be implemented when assembly processes are developed.
The team will create a comprehensive list of FMEAs for each subsystem of the project, to mitigate as many of the failure modes as the team can.	I	Each team member will write a FMEA for each and every part of the project they are working on. These will be organized by sub-team and subsystem so they can be easily referenced.	Complete - FMEAs for all parts have been created

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Table 96 – continued from previous page

Requirement	Verification Method	Verification Plan	Status
The team will charge all LiPo batteries with a smart charger to prevent nonuniform or over charging of the batteries	D	The team will agree to only buy smart chargers, so that non-smart chargers are never used for charging the batteries.	Incomplete - Will be implemented when team buys chargers.
The team will not short circuit any of the batteries while installing them into systems requiring batteries.	D	A formal procedure will be written with instructions explaining how to safely install the batteries into the rover.	Incomplete - Will be implemented when team reaches that point.
The team will use appropriate PPE when handling and machining composite materials.	D	Safety briefings will be conducted, JHAs will be filled out as necessary.	In Progress - appropriate PPE has been used so far
There will be no sharp edges on the payload.	I	Any sharp edges on payload will be machined off or will be completely encased in a safe container. See section 4.3.3.3 for auger encasing.	Incomplete - Will be implemented when team reaches that point.
There will be a light to indicate the payload ejection charges are armed.	I	Blinking Light Emitting Diode (LED) indicator will be installed, connected to payload ejection controller. LED blinking will be visible upon arming vehicle.	Incomplete - Will be implemented when team reaches that point.
The payload ejection controller will have an arming switch.	I	Turn the switch to verify the LED is blinking to indicate armed status.	Incomplete - Will be implemented when team reaches that point.
The Mentor and Educational Advisor will have a final say in safety decisions on all activities and designs.	I	If the safety of an event or activity is disputed, whatever the Mentor or Advisor decides will be the final decision.	Complete - All team members have agreed to follow this rule should this issue arise.
All team members must remain attentive and at safe distances from the launch area during subscale and full scale launches.	I	Each team member will be responsible for having awareness of their surroundings during all launch related activities. All three safety officers will oversee team members' safety.	Incomplete - Will be implemented at first launch event and maintained through all future launches.

6.2 Budget

The team's detailed bill of materials for its leading design is shown in Tables 97, 98, and 99. Since the proposal, the item identifying scheme was changed to a simpler XX-YYY format where XX is the launch vehicle section code and YYY is a number given to each item. Items that will no longer be needed are still included in the bill of materials; However, they're marked as "former" and do not affect the final cost. Finally, the costs denoted with "n/a" are associated with items that do not require being purchased, due to the items already being in possession.

Budget: Structure Section Codes

- 01: Motor
- 02: Blade Extending Apogee Variance System (BEAVS)
- 03: Aft Avionics/Ejection Bay
- 04: Aft Chutes
- 05: Payload
- 06: Fore Hard Point
- 07: Fore Avionics Bay
- 08: Fore Chutes
- 09: Fore Ejections Bay
- 10: Nosecone
- 11: Airframe

Table 97: Structures Budget

Section	Assembly	Identifier	Description	Quantity	Unit Cost	Cost	Vendor/ Source	SKU	Status
01	Aero/Recovery	01-001	Propellant	1	$ 331.99	$ 331.99	Wildman	PR75-4G-WT	CURRENT
01	Aero/Recovery	01-002	Motor Casing	1	$ 415.22	$ 415.22	Apogee Rockets	71043	CURRENT
01	Aero/Recovery	01-003	75 mm Aft Closure	1	$ 70.99	$ 70.99	Wildman	P75-CL	CURRENT
01	Aero/Recovery	01-004	75 mm Retainer	1	$ 48.89	$ 48.89	Apogee Rockets	24054	CURRENT
01	Aero/Recovery	01-005	Centering Ring	3	$ -	$ -		-	CURRENT
01	Aero/Recovery	01-006	Trapezoidal Fins	4	$ -	$ -		-	CURRENT
01	Aero/Recovery	01-007	Fiberglass Motor Tube	1	$ -	$ -		-	CURRENT
03	Aero/Recovery	03-001	MissileWorks RRC3 Sport Altimeter	2	$ 69.95	$ 139.90	MissileWorks	90905	CURRENT
03	Aero/Recovery	03-002	PerfectFlite StrattoLoggerCF	2	$ 57.50	$ 115.00	PerfectFlite	-	CURRENT
03	Aero/Recovery	03-003	4f Black Powder	1	$ 36.54	$ 36.54	Graf and Sons	SC4FG	CURRENT
03		03-004	Santoprene		$ -	$ -		-	CURRENT
03		03-005	Surgical Tubing		$ -	$ -		88210	CURRENT
03	Aero/Recovery	03-006	E-matches	20	$ 2.00	$ 40.00	Australian Rocketry	-	CURRENT
03	Structures	03-007	Battery Enclosure	2	$ 1.86	$ 3.72	LEDSupply	BH-9V-SWITCH	CURRENT
03	Structures	03-008	Spacer	8	$ 0.07	$ 0.56	Alliedelec	901-605	CURRENT
03	Structures	03-009	9V Battery (4-pack)	1	$ 12.96	$ 12.96	Walmart	MN16RT4Z	CURRENT
03	Structures	03-010	Board Screws (100 pack)	1	$ 5.29	$ 5.29	McMaster-Carr	91772A081	CURRENT
03	Structures	03-011	Battery Screws (100 pack)	1	$ 5.82	$ 5.82	McMaster-Carr	90471A215	CURRENT
03	Structures	03-012	Terminal Block	8	$ 1.12	$ 8.96	McMaster-Carr	8076K12	CURRENT
03	Structures	03-013	Switch Mounting Bolt (100 pack)	1	$ 8.05	$ 8.05	McMaster-Carr	91772A059	CURRENT
03	Structures	03-014	Switch Mounting Nut (100 pack)	1	$ 5.02	$ 5.02	McMaster-Carr	90480A001	CURRENT
03	Structures	03-015	Eye Bolt	1	$ 6.65	$ 6.65	Grainger	U16010.037.0300	CURRENT
03	Structures	03-016	Sealing Washer (10 pack)	1	$ 8.58	$ 8.58	McMaster-Carr	93303A102	CURRENT
03	Structures	03-017	Seal Washer Bolt (50 pack)	1	$ 8.52	$ 8.52	McMaster-Carr	92620A564	CURRENT
03	Structures	03-018	Seal Washer Nut	1	$ 2.14	$ 2.14	McMaster-Carr	94815A007	CURRENT
03	Structures	03-019	Long Eye Bolt	1	$ 8.67	$ 8.67	McMaster-Carr	3018T26	CURRENT

Continued on next page

Table 97 – continued from previous page

Section	Assembly	Identifier	Description	Quantity	Unit Cost	Cost	Vendor/ Source	SKU	Status
07	Aero/Recovery	07-001	MissileWorks RRC3 Sport Altimeter	1	$ 69.95	$ 69.95	MissileWorks	90905	CURRENT
07	Aero/Recovery	07-002	PerfectFlite StrattoLoggerCF	1	$ 57.50	$ 57.50	PerfectFlite	-	CURRENT
07	Structures	07-003	Battery Enclosure	2	$ 1.86	$ 3.72	LEDSupply	BH-9V-SWITCH	CURRENT
07	Structures	07-004	Spacer	8	$ 0.07	$ 0.56	Alliedelec	901-605	CURRENT
07	Structures	07-005	9V Battery (4-pack)	1	$ -	$ -	Walmart	MN16RT4Z	CURRENT
07	Structures	07-006	Board Screws (100 pack)	1	$ -	$ -	McMaster-Carr	91772A081	CURRENT
07	Structures	07-007	Battery Screws (100 pack)	1	$ -	$ -	McMaster-Carr	90471A215	CURRENT
07	Structures	07-008	Terminal Block	8	$ 1.12	$ 8.96	McMaster-Carr	8076K12	CURRENT
07	Structures	07-009	Switch Mounting Bolt (100 pack)	1	$ -	$ -	McMaster-Carr	91772A059	CURRENT
07	Structures	07-010	Switch Mounting Nut (100 pack)	1	$ -	$ -	McMaster-Carr	90480A001	CURRENT
07	Structures	07-011	Eye Bolt	1	$ 6.65	$ 6.65	Grainger	U16010.037.0300	CURRENT
07	Structures	07-012	Sealing Washer (10 pack)	1	$ -	$ -	McMaster-Carr	93303A102	CURRENT
07	Structures	07-013	Seal Washer Bolt (50 pack)	1	$ -	$ -	McMaster-Carr	92620A564	CURRENT
07	Structures	07-014	Seal Washer Nut	1	$ 2.14	$ 2.14	McMaster-Carr	94815A007	CURRENT
07	Structures	07-015	Long Eye Bolt	1	$ 8.67	$ 8.67	McMaster-Carr	3018T26	CURRENT
09	Aero/Recovery	09-001	MissileWorks RRC3 Sport Altimeter	1	$ 69.95	$ 69.95	MissileWorks	90905	CURRENT
09	Aero/Recovery	09-002	PerfectFlite StrattoLoggerCF	1	$ 57.50	$ 57.50	PerfectFlite	-	CURRENT
10	Structures	10-001	Nose Cone	1	$ 169.95	$ 169.95	Madcow Rocketry	-	CURRENT
11	Structures	11-001	Aft Airframe	1	$ 3,000.00	$ 3,000.00	-		CURRENT
11	Structures	11-002	Fin Stock	1	$ 159.99	$ 159.99	RockWest Composites		CURRENT
11	Structures	11-003	RocketPoxy (2 pints)	1	$ 43.75	$ 43.75	RocketPoxy		CURRENT
11	Structures	11-004	Bulk Heads		$ -			-	CURRENT
11	Structures	11-005	Motor Tube	1	$ 94.05	$ 94.05			CURRENT
			Structures Subtotal			$ 5,036.81			
			10 % Contingency			$ 503.68			
			STRUCTURES TOTAL			**$ 5,540.49**			

Table 98: Aerodynamics and Recovery Budget

Section	Assembly	Identifier	Description	Quantity	Unit Cost	Cost	Vendor/ Source	SKU	Status
02	Mechanical BEAVS	02-001	1/8″ aluminum plate (2″ x 24″ Bar)	1	$ 11.08	$ 11.08	McMaster-Carr		CURRENT
02	Mechanical BEAVS	02-002	1/4-20 Fasteners	4	$ 0.19	$ 0.78	McMaster-Carr		CURRENT
02	Mechanical BEAVS	02-003	8-32 Threaded Rod (4 in)	4	$ 1.67	$ 6.68	McMaster-Carr		CURRENT
02	Mechanical BEAVS	02-004	1/2″ Aerospace Grade Plywood Bulkhead	21.24	$ 0.003	$ 0.06	Wicks		CURRENT
02	Mechanical BEAVS	02-005	PLA 3D Printer Filament (1 kg)	1	$ 19.99	$ 19.99	Amazon		CURRENT
02	Mechanical BEAVS	02-006	M2 Fasteners (Qty 100)	1	$ 13.26	$ 13.26	McMaster-Carr		CURRENT
02	Mechanical BEAVS	02-007	7mm Linear Guide Block	4	$ 65.47	$ 261.88	McMaster-Carr		CURRENT
02	Mechanical BEAVS	02-008	7mm Linear Rail (172 mm)	4	$ 21.06	$ 84.24	McMaster-Carr		CURRENT
02	Mechanical BEAVS	02-009	10 GA Steel Plate	10	$ 2.37	$ 23.70	JCI		CURRENT
02	Electrical BEAVS	02-010	SparkFun Venus GPS	1	$ 49.95	$ 49.95	SparkFun		CURRENT
02	Electrical BEAVS	02-010	Teensy 3.6	1	$ 31.25	$ 31.25	DigiKey		CURRENT
02	Electrical BEAVS	02-011	MPL3115 Barometer	1	$ 4.87	$ 4.87	Mouser		CURRENT
02	Electrical BEAVS	02-012	BNO055 9DOF IMU	1	$ 34.95	$ 34.95	Adafruit		CURRENT
02	Electrical BEAVS	02-013	Turnigy 2200mah LiPo	1	$ 10.99	$ 10.99	HobbyKing		CURRENT
02	Electrical BEAVS	02-014	OSRT Designed PCB	1	$ 92.90	$ 92.90	DFRobot		CURRENT
02	Electrical BEAVS	02-015	Xbee Pro 900hp	1	$ 39.00	$ 39.00	DigiKey		CURRENT
02	Electrical BEAVS	02-016	7 in RPSMA whip antenna	1	$ 4.29	$ 4.29	Amazon		CURRENT
02	Mechanical BEAVS	02-016	Retention Ring	1	$ 0.10	$ 0.10	3D Printed		CURRENT

Continued on next page

Table 98 – continued from previous page

Section	Assembly	Identifier	Description	Quantity	Unit Cost	Cost	Vendor/ Source	SKU	Status
02	Mechanical BEAVS	02-017	Retention Link	4	$ 0.10	$ 0.40	3D Printed		CURRENT
02	Mechanical BEAVS	02-018	4-40 Bolt 3/8″ long	8	$ 0.30	$ 2.37	McMaster-Carr		CURRENT
04	Aero/Recovery	04-001	Main Parachute	1	$ 265.00	$ 265.00	FruityChutes	IFC-72-S	CURRENT
04	Aero/Recovery	04-002	Drogue parachute	1	$ 19.99	$ 19.99	Top Flight Recovery	XTPAR-18	CURRENT
04	Aero/Recovery	04-003	Eye Bolts	2	$ 6.65	$ 13.30	Grainger - Ken Forging	35Z511	CURRENT
04	Aero/Recovery	04-004	Nyloc	2	$ 5.67	$ 11.34	Zoro	G5360591	CURRENT
04	Aero/Recovery	04-005	Nylon shock cord	1	$ 29.00	$ 29.00	FruityChutes	SCN-1000	CURRENT
04	Aero/Recovery	04-006	Quick links - Standard	2	$ 4.10	$ 8.20	Apogee Rockets	29621	CURRENT
04	Aero/Recovery	04-007	Swivel	2	$ 9.00	$ 18.00	FruityChutes	SWIV-3000	CURRENT
04	Aero/Recovery	04-008	Nylon shock cord	1	$ 24.80	$ 24.80	FruityChutes	SCN-1000	CURRENT
04	Aero/Recovery	04-009	Kevlar sleeve	2	$ 28.75	$ 57.50	BlackCat Rocketry	HK-S-250	CURRENT
04	Aero/Recovery	04-010	Deployment bag		$ 45.00	$ -	FruityChutes	-	CURRENT
04	Aero/Recovery	04-011	Nomex blanket	1	$ 13.00	$ 13.00	FruityChutes	NB-9	CURRENT
04	Aero/Recovery	04-012	Tender Descenders	2	$ 129.00	$ 258.00	TinderRocketry		CURRENT
04	Aero/Recovery	04-013	Wide Mouth Quick Link	2	$ 4.76	$ 9.52	McMaster-Carr	3711T23	CURRENT
04	Aero/Recovery	04-014	Shear Pins	3	$ 3.22	$ 9.66	Apogee Rockets	29615	CURRENT
04	Aero/Recovery	04-015	Stainless Steel Slider Ring	1	$ 4.00	$ 4.00	FruityChutes	RING-1375	CURRENT
08	Aero/Recovery	08-001	Main Parachute	1	$ 345.00	$ 345.00	FruityChutes	IFC-84-S	CURRENT
08	Aero/Recovery	08-002	Eye Bolts	4	$ 6.65	$ 26.60	Grainger - Ken Forging	35Z511	CURRENT
08	Aero/Recovery	08-003	Nyloc	4	$ -	$ -	Zoro	G5360591	CURRENT
08	Aero/Recovery	08-004	Nylon shock cord	1	$ 29.00	$ 29.00	FruityChutes	SCN-1000	CURRENT
08	Aero/Recovery	08-005	Quick links - Standard	2	$ 4.10	$ 8.20	Apogee Rockets	29621	CURRENT
08	Aero/Recovery	08-006	Nylon shock cord	1	$ 24.80	$ 24.80	FruityChutes	SCN-1000	CURRENT
08	Aero/Recovery	08-007	Kevlar Sleeve	2	$ 28.75	$ 57.50	BlackCat Rocketry	HK-S-250	CURRENT

Continued on next page

Table 98 – continued from previous page

Section	Assembly	Identifier	Description	Quantity	Unit Cost	Cost	Vendor/ Source	SKU	Status
08	Aero/Recovery	08-009	Deployment bag		$ 45.00	$ -	FruityChutes		CURRENT
08	Aero/Recovery	08-010	Nomex blanket	1	$ 13.00	$ 13.00	FruityChutes	NB-9	CURRENT
08	Aero/Recovery	08-011	Stainless Steel Slider Ring	1	$ 4.00	$ 4.00	FruityChutes	RING-1375	CURRENT
08	Aero/Recovery	08-012	Wide Mouth Quick Link	2	$ 4.76	$ 9.52	McMaster-Carr	3711T23	CURRENT
08	Aero/Recovery	08-013	Shear Pins	6	$ -	$ -	Apogee Rockets	29615	CURRENT
08	Aero/Recovery	08-014	Drogue parachute	1	$ 19.99	$ 19.99	Top Flight Recovery	XTPAR-18	CURRENT
08	Aero/Recovery	08-015	Swivel	2	$ 9.00	$ 18.00	FruityChutes	SWIV-3000	CURRENT
08	Aero/Recovery	08-017	Tender Descenders	2	$ 129.00	$ 258.00	TinderRocketry		CURRENT
			Aero/ Recovery Subtotal		$ 2,247.66				
			10 % Contingency		$ 224.77				
			AERO/ RECOVERY TOTAL		**$ 2,472.43**				

Table 99: Payload Bill of Materials

Section	Subsystem	Identifier	Description	Quantity	Unit Cost	Cost	Vendor/Source	SKU	Value	Status
05	PEARS	05-001	HDPE Rod Tip - fore	1	$ -	$ -	McMaster-Carr			former
05	PEARS	05-002	Threaded Rod	1	$ 5.56	$ 5.56	McMaster-Carr			CURRENT
05	PEARS	05-003	Fore EARS Bulkhead - Loose	1	$ -	$ -	Home Depot			former
05	PEARS	05-004	PLEC	1	n/a	n/a	Club Resources		$ 35.00	CURRENT
05	PEARS	05-005	Aft PEARS Bulkhead - Loose	1	n/a	n/a	Club Resources		$ 0.35	CURRENT
05	PEARS	05-006	HDPE Rod Cap - Spacer	1	$ 3.54	$ 3.54	McMaster-Carr			CURRENT

Continued on next page

Table 99 – continued from previous page

Section	Subsystem	Identifier	Description	Quantity	Unit Cost	Cost	Vendor/Source	SKU	Value	Status
05	WRAP	05-007	Aft Payload Bulkhead	1	n/a	n/a	Club Resources		$ 0.35	CURRENT
05	WRAP	05-008	Fore Payload Bulkhead	1	n/a	n/a	Club Resources		$ 0.35	CURRENT
05	WRAP	05-009	Kevlar Harness	1	$ 10.00	$ 10.00	Dutchware Gear			CURRENT
05	WRAP	05-010	Carbon Fiber Wrap	1	$ 43.63	$ 43.63	Fibre Glast			CURRENT
05	PEARS	05-011	AARD	1	n/a	n/a	Club Resources		$ 119.00	CURRENT
05	PEARS	05-012	Tender Decender	2	n/a	n/a	Club Resources		$ 158.00	CURRENT
05	PEARS	05-013	Misc Hardware	1	$ -	$ -	Homedepot			former
05	Soil Collection	05-014	3D Printed Auger	1	$ 5.00	$ 5.00	3D Print (OSU)			CURRENT
05	Soil Collection	05-015	Auger Circular Tube	1	$ -	$ -	Possibly 3D Print (OSU)			former
05	Soil Collection	05-016	Planetary Gear Motor PGHM-03	2	$ -	$ -	Various vendors (Robotshop.com)			former
05	Soil Collection	05-017	Metal Gear - 14-1/2 Degree Pressure Angle	1	$ -	$ -	McMaster-Carr			former
05	Soil Collection	05-018	Roller Track Frame - 3D print	1	$ -	$ -	3D Print (OSU)			former
05	Soil Collection	05-019	Threaded Track Rollers	4	$ -	$ -	McMaster-Carr / Home Depot			former
05	Electronics	05-020	Raspberry Pi	1	$ -	$ -	Digikey			former
05	Electronics	05-021	MB7360 HRXL-MaxSonar-WR sensors	2	$ 99.95	$ 199.90	Karlsson Robots			CURRENT
05	Soil Collection	05-022	Misc Hardware	1	$ -	$ -	Mcmaster-Carr			former
05	Drivetrain	05-023	GHM-04 Spur Gear Motor	4	$ 21.95	$ 87.80	RobotShop			CURRENT
05	Drivetrain	05-024	6x12x0.75" HDPE Plate	2	$ -	$ -	McMaster-Carr			former

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Table 99 – continued from previous page

Section	Subsystem	Identifier	Description	Quantity	Unit Cost	Cost	Vendor/Source	SKU	Value	Status
05	Drivetrain	05-025	24" 4140 Chamfered Steel Rod	1	$ -	$ -	McMaster-Carr			former
05	Drivetrain	05-026	0.5x1x12" 6061 Aluminum Bar	1	$ -	$ -	McMaster-Carr			former
05	Drivetrain	05-027	1x1x12" 6061 Aluminum Bar	1	$ -	$ -	McMaster-Carr			former
05	Drivetrain	05-028	1x1.5x12" 6061 Aluminum Bar	1	$ -	$ -	McMaster-Carr	8975K518		former
05	Drivetrain	05-029	6-32 1.5" Steel Round Head Screws, 100 ct.	1	$ -	$ -	McMaster-Carr	90276A157		former
05	Drivetrain	05-030	6mm Shaft Coupling	2	$ 1.82	$ 3.64	Banggood	994356		CURRENT
05	Drivetrain	05-031	0.25" Bore 0.770" Pattern Clamping Hub	2	$ 5.99	$ 11.98	ServoCity			CURRENT
05	Drivetrain	05-032	6-32 Brass Hex Nuts, 100 ct.	1	$ -	$ -	Bolt Depot			former
05	Electronics	05-033	Turnigy Graphene 950 mAh LiPo Battery Pack	9	$ -	$ -	Newegg			former
05	PEARS	05-034	Ejection Charge	1	n/a	n/a	Club Resources	-	$ 2.00	CURRENT
05	PEARS	05-035	Snap Button	1	$ -	$ -	ezup.com			former
05	Soil Collection	05-036	High-Strength 1045 Carbon Steel Rod	1	$ 6.60	$ 6.60	McMaster-Carr	8279T16		CURRENT
05	Soil Collection	05-037	Carbon Fiber Auger Wrap	1	$ 10.00	$ 10.00				CURRENT
05	Soil Collection	05-038	Motor Bar (motor frame)	1	$ 5.00	$ 5.00				CURRENT

Continued on next page

Table 99 – continued from previous page

Section	Subsystem	Identifier	Description	Quantity	Unit Cost	Cost	Vendor/Source	SKU	Value	Status
05	SCAR	05-039	Set Screw Shaft Coupler	3	$ 4.99	$ 14.97	ServoCity	625118		CURRENT
05	SCAR	05-040	26 RPM Mini Econ Gear Motor	3	$ 9.99	$ 29.97	ServoCity	638830		CURRENT
05	Soil Collection	05-041	Short-Thread Alloy Steel Shoulder Screw	2	$ 4.10	$ 8.20	McMaster-Carr	94361A112		CURRENT
05	Soil Collection	05-042	Feeding Tube	1	$ 11.82	$ 11.82	McMaster-Carr	89955K919		CURRENT
05	Soil Collection	05-043	Outer Tube	1	$ 15.04	$ 15.04	McMaster-Carr	89955K989		CURRENT
05	Soil Collection	05-044	Fluoroelastomer Rubber Sealing Washer (pk.)	1	$ 11.98	$ 11.98	McMaster-Carr	93412A401		CURRENT
05	Soil Collection	05-045	18-8 Stainless Steel Washer (pk.)	1	$ 1.40	$ 1.40	McMaster-Carr	92141A005		CURRENT
05	Soil Collection	05-046	Black-Oxide Alloy Steel Socket Head Screw (pk.)	1	$ 8.52	$ 8.52	McMaster-Carr	91251A105		CURRENT
05	SR	05-047	Low-Carbon Steel Sheet	1	$ 23.50	$ 23.50	McMaster-Carr	6544K71		CURRENT
05	SR	05-048	Low-Carbon Steel Rod	2	$ 1.20	$ 2.40	McMaster-Carr	8920K115		CURRENT
05	SR	05-049	Passivated 18-8 SS Pan Head Phillips Screw (pk.)	1	$ 7.54	$ 7.54	McMaster-Carr	91772A194		CURRENT
05	Chassis	05-050	Carbon fiber rod, .25in	16	$ 0.44	$ 7.04				CURRENT
05	Chassis	05-051	Carbon fiber rod, .25in	14	$ 0.32	$ 4.48				CURRENT
05	Chassis	05-052	Aluminium rod, tapped on both ends, .25in	9	$ 0.47	$ 4.23				CURRENT

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Table 99 – continued from previous page

Section	Subsystem	Identifier	Description	Quantity	Unit Cost	Cost	Vendor/Source	SKU	Value	Status
05	Chassis	05-053	Aluminium connection blocks, .5-.75-1.25	4	$ 0.79	$ 3.16				CURRENT
05	Chassis	05-054	Aluminium connection blocks, .5-.75-1.0	4	$ 0.75	$ 3.00				CURRENT
05	Chassis	05-055	Aluminium connection blocks, .5-.75-.75	10	$ 0.70	$ 7.00				CURRENT
05	Chassis	05-056	Screws	18	$ 0.25	$ 4.50	Homedepot			CURRENT
05	Chassis	05-057	Carbon Fiber tail	1	$ 1.60	$ 1.60				CURRENT
05	Chassis	05-058	Rat Trap Spring	1	$ 2.50	$ 2.50	Homedepot			CURRENT
05	Chassis	05-059	Metal tail connection piece	1	$ 1.25	$ 1.25				CURRENT
05	Drivetrain	05-060	Elastic Bands	1	$ 1.99	$ 1.99				former
05	Drivetrain	05-061	Misc. Fasteners	1	$ 10.00	$ 10.00	McMaster-Carr			CURRENT
05	Drivetrain	05-062	3D Printed Wheel	2	n/a	n/a	Club Resources	-	$ 20.66	CURRENT
05	Drivetrain	05-063	Urethane Foam Strip	2	$ 10.00	$ 20.00	TBD			CURRENT
05	Drivetrain	05-064	.5 in. dia, 1 ft 6061 Al Rod	1	$ 2.66	$ 2.66	McMaster-Carr	8974K28		CURRENT
05	Drivetrain	05-065	.25x.75x12 in. 6061 Al Bar	1	$ 2.14	$ 2.14	McMaster-Carr	8975K594		CURRENT
05	Drivetrain	05-066	.75x1.5x12 in. 6061 Al Bar	1	$ 11.42	$ 11.42	McMaster-Carr	8975K45		CURRENT
05	Drivetrain	05-067	.5 in. dia, 1 ft 6061 Al Rod	1	$ 2.66	$ 2.66	McMaster-Carr	8974K28		CURRENT
05	Drivetrain	05-068	1 1/8 in. dia, 6 in. Al Rod	1	$ 5.86	$ 5.86	McMaster-Carr	8974K15		CURRENT

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Table 99 – continued from previous page

Section	Subsystem	Identifier	Description	Quantity	Unit Cost	Cost	Vendor/Source	SKU	Value	Status
05	Drivetrain	05-069	3D Printed Connector Rod	4	n/a	n/a	Club Resources	-	$ 1.52	CURRENT
05	Drivetrain	05-070	Annular Ball Bearing	4	$ 5.78	$ 23.12	McMaster-Carr	60355K504		CURRENT
05	PEARS	05-071	3D Printed PLEC Mount	1	n/a	n/a	Club Resources	-	$ 4.00	CURRENT
05	PEARS	05-072	SPDT Push Botton	1	$ 14.27	$ 14.27	McMaster-Carr			CURRENT
05	PEARS	05-073	1" Ring	2	$ 1.20	$ 2.40	McMaster-Carr			CURRENT
05	PEARS	05-074	Rubber Seal	1	$ 4.00	$ 4.00	McMaster-Carr			CURRENT
05	PEARS	05-075	Shunting Circuit	1	n/a	n/a	Club Resources		$ -	CURRENT
05	PEARS	05-076	U Bolt	2	$ 0.95	$ 1.90	McMaster-Carr			CURRENT
05	PEARS	05-077	Nylon Locknut	2	$ 0.16	$ 0.32	McMaster-Carr			CURRENT
05	PEARS	05-078	Sealing Washer	2	$ 0.86	$ 1.72	McMaster-Carr			CURRENT
05	Electrical	05-079	Teensy 3.6 Development Board	1	$ 29.25	$ 29.25	SparkFun			CURRENT
05	Electrical	05-080	Solid-Core Wire Spool	3	$ 2.95	$ 8.85	Adafruit			CURRENT
06	Fore Hard Point	06-001	Fore plywood bulkhead	1	n/a	n/a	Club Resources		$ 0.35	CURRENT
06	Fore Hard Point	06-002	Fore 3D printed funnel	1	n/a	n/a	Club Resources		$ 5.00	CURRENT
06	Fore Hard Point	06-003	Alf FWD Hard Point	1	n/a	n/a	Club Resources		$ 0.35	CURRENT
			Payload Subtotal		$ 907.03					
			10 % Contingency		$ 90.70					
			PAYLOAD TOTAL		**$ 997.73**					

6.2.1 Financial Plan

The largest amount of funding is coming from OSGC, which offers up to $12,000 at 1.5:1 matching. This means that for every one dollar that OSGC donates, the OSRT will have to spend $1.50 of either the student's money or sponsorship donations. The specific grant that the OSRT is attempting to get is the OSGC Undergraduate Team Experience Award 2018-2019. Obtaining this funding would be a huge benefit to the program and students. This program has funded many student designs,and build programs. They also sponsored OSRT last year. The program requires a diverse team and provides many students with great opportunities to advance their knowledge and interest in aerospace and rocketry.

To be able to meet the match requirements, the OSRT, specifically the budget and finance team, will reach out to many possible sponsors and donors. There are several other space and aeronautics teams at OSU so, for sponsors that multiple teams will be reaching out to, OSU will reach out as a whole school. This helps eliminate asking companies and donors for funding several times, instead of having only one point of contact for all of OSU AIAA. All sponsors will be featured on the OSRT website under a sponsors page and also will have a high resolution logo placed onto the body of the launch vehicle. Additionally, at all student outreach events, a billboard with a list of all the sponsors will be present.

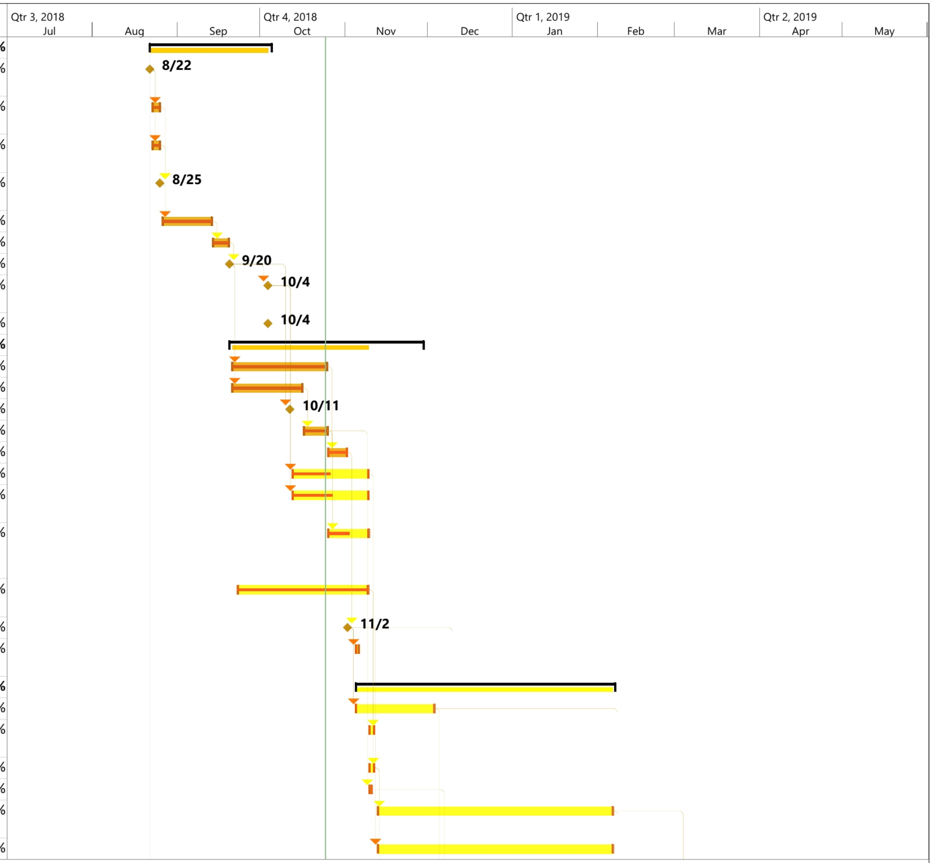
Last year, the OSRT had a budget of $26,000 with $10,400 of that coming from OSGC. The remaining $15,600 came mostly through sponsorship, however, students covered the cost of their plane tickets. The budget and finance team has a total goal of $30,000 split between the OSGC grant and other donations. This goal can be obtained through deliberate outreach to donors and sponsors.

Figure 55: OSGC Logo

6.3 Timeline

Currently, OSRT is nearing the end of the Preliminary Design Phase and beginning to transition to Critical Design Phase. Displayed in Figures 56-57 is the current project schedule.

ID	Task Name	Duration	Start	Finish	% Complete
1	**Proposal Phase**	**45 days**	**Wed 8/22/18**	**Fri 10/5/18**	**100%**
2	Competition Rules Released	0 days	Wed 8/22/18	Wed 8/22/18	100%
3	Launch Vehicle Concept Development	3 days	Thu 8/23/18	Sat 8/25/18	100%
4	Payload Concept Development	3 days	Thu 8/23/18	Sat 8/25/18	100%
5	Team Decisions and Kickoff Meeting	0 days	Sat 8/25/18	Sat 8/25/18	100%
6	Compile Proposal	18 days	Sun 8/26/18	Thu 9/13/18	100%
7	Edit Proposal	6 days	Fri 9/14/18	Wed 9/19/18	100%
8	Submit Proposal	0 days	Thu 9/20/18	Thu 9/20/18	100%
9	Proposal Admittance Announced	0 days	Thu 10/4/18	Thu 10/4/18	100%
10	AIAA Presentation	0 days	Thu 10/4/18	Thu 10/4/18	100%
11	**Preliminary Design Phase**	**72 days**	**Thu 9/20/18**	**Thu 11/29/18**	**82%**
12	Compile PDR	35 days	Fri 9/21/18	Thu 10/25/18	100%
13	Research Design Choices	26 days	Fri 9/21/18	Tue 10/16/18	100%
14	Kickoff and PDR Q&A	0 days	Thu 10/11/18	Thu 10/11/18	100%
15	Complete CAD Models	9 days	Wed 10/17/18	Thu 10/25/18	100%
16	Edit PDR	7 days	Fri 10/26/18	Thu 11/1/18	100%
17	Manufacture Subscale	28 days	Sat 10/13/18	Fri 11/9/18	50%
18	Ground Test Subscale Systems	28 days	Sat 10/13/18	Fri 11/9/18	51%
19	Manufacture BEAVS Prototype and Test Airframe Modifications	15 days	Fri 10/26/18	Fri 11/9/18	51%
20	Build L1/L2 Certification Rockets	48 days	Sun 9/23/18	Fri 11/9/18	100%
21	Submit PDR	0 days	Fri 11/2/18	Fri 11/2/18	100%
22	PDR Video Teleconferences	1 day	Mon 11/5/18	Mon 11/5/18	0%
23	**Critical Design Phase**	**96 days**	**Mon 11/5/18**	**Thu 2/7/19**	**0%**
24	Compile CDR	29 days	Mon 11/5/18	Mon 12/3/18	0%
25	Certification Rocket Launch Day	2 days	Sat 11/10/18	Sun 11/11/18	0%
26	Subscale Launch	2 days	Sat 11/10/18	Sun 11/11/18	0%
27	BEAVS Test Launch	1 day	Sat 11/10/18	Sat 11/10/18	0%
28	Manufacture Full Scale Launch Vehicle	87 days	Tue 11/13/18	Wed 2/6/19	0%
29	Integration Testing	87 days	Tue 11/13/18	Wed 2/6/19	0%



Project: PDR Schedule
Date: Thu 10/25/18

Task	Project Summary	Manual Task	Start-only	Deadline
Split	Inactive Task	Duration-only	Finish-only	Progress
Milestone	Inactive Milestone	Manual Summary Rollup	External Tasks	Manual Progress
Summary	Inactive Summary	Manual Summary	External Milestone	

Page 1

Figure 56: Oregon State Rocketry Team Project Schedule.

ID	Task Name	Duration	Start	Finish	% Complete
30	Ground Test Full Scale Systems	87 days	Tue 11/13/18	Wed 2/6/19	0%
31	Edit CDR	7 days	Tue 12/4/18	Mon 12/10/18	0%
32	BEAVS Test Launch	1 day	Sat 12/8/18	Sat 12/8/18	0%
33	BEAVS Test Launch	1 day	Sat 12/15/18	Sat 12/15/18	0%
34	Motor Performance Testing	25 days	Tue 12/11/18	Fri 1/4/19	0%
35	Submit CDR	0 days	Fri 1/4/19	Fri 1/4/19	0%
36	CDR Video Teleconferences	16 days	Mon 1/7/19	Tue 1/22/19	0%
37	**Final Readiness Phase**	**75 days**	**Mon 1/7/19**	**Thu 3/21/19**	**0%**
38	Compile FRR	50 days	Mon 1/7/19	Mon 2/25/19	0%
39	Full Scale Launch	1 day	Sat 2/9/19	Sat 2/9/19	0%
40	Design Updates and Testing	6 days	Sun 2/10/19	Fri 2/15/19	0%
41	Full Scale Launch	1 day	Sat 2/16/19	Sat 2/16/19	0%
42	Design Updates and Testing	6 days	Sun 2/17/19	Fri 2/22/19	0%
43	Full Scale Launch	1 day	Sat 2/23/19	Sat 2/23/19	0%
44	Edit FRR	6 days	Tue 2/26/19	Sun 3/3/19	0%
45	Submit FRR	0 days	Mon 3/4/19	Mon 3/4/19	0%
46	FRR Video Teleconferences	14 days	Fri 3/8/19	Thu 3/21/19	0%
47	**Competition Phase**	**37 days**	**Tue 3/5/19**	**Wed 4/10/19**	**0%**
48	Paint Rocket	22 days	Tue 3/5/19	Tue 3/26/19	0%
49	Develop Displays	22 days	Tue 3/5/19	Tue 3/26/19	0%
50	Motorpool Departure	6 days	Wed 3/27/19	Mon 4/1/19	0%
51	Arrive in Huntsville	0 days	Tue 4/2/19	Tue 4/2/19	0%
52	LRR	1 day	Wed 4/3/19	Wed 4/3/19	0%
53	Launch Day	1 day	Sat 4/6/19	Sat 4/6/19	0%
54	Backup Launch Day	1 day	Sun 4/7/19	Sun 4/7/19	0%
55	Depart from Huntsville	0 days	Tue 4/9/19	Tue 4/9/19	0%
56	**Post Launch Phase**	**18 days**	**Wed 4/10/19**	**Sat 4/27/19**	**0%**
57	Compile PLAR	12 days	Wed 4/10/19	Sun 4/21/19	0%
58	Edit PLAR	4 days	Mon 4/22/19	Thu 4/25/19	0%
59	Submit PLAR	0 days	Fri 4/26/19	Fri 4/26/19	0%

Figure 57: Oregon State Rocketry Team Project Schedule.

6.4 STEM Engagement

OSRT has completed two STEM engagement activities by the time of PDR completion. On Friday October 26th, 2018 OSRT completed a design review with a group of students from Yamhill-Carlton High School about building a level one high power rocketry certification rocket. OSRT asked questions about the designs and provided feedback on improvements. Members of OSRT then presented the 2018-2019 launch vehicle to the group of high school students.

The other event that OSRT has completed is Discovery Days on the campus of Oregon State University. Over two days, 950 elementary age students cycled through different STEM related booths. At the OSRT booth, members of the team discussed the goals of this year's rocket in the NASA Student Launch competition. After the quick discussion, students were tasked with making individual straw rockets. Each student received a straw about six in. long. Duct tape fins were then given to each student for them to attach near the end of their straw or rocket body. It was discussed that the fins on their rocket and our large rocket would enable them both to fly straight. After the fins were attached to the rocket bodies, a square piece of duct tape was handed to each student as the nosecone. The students were told to seal the other end of their rocket body to ensure no air can escape from that end. The last step is inserting the student made rockets onto a small diameter straw to act as the launcher. The students then blew into the launching straw, sending their rockets a few feet into the common area. A photo of OSRT setting up for the students can be seen in Figure 58.

Figure 58: OSRT setting up for Discovery Days

OSRT has a goal to engage at least 3000 students in STEM related activities. Many events have been planned, and more are in the process of being planned. The activities will aim to engage students to be creative and complete scientific processes. Lesson plans are intended to tie into teachers existing units and provide a more in-depth discussion of the STEM related topic.

APPENDIX A

DRAWINGS AND SCHEMATICS

A.1 Structures

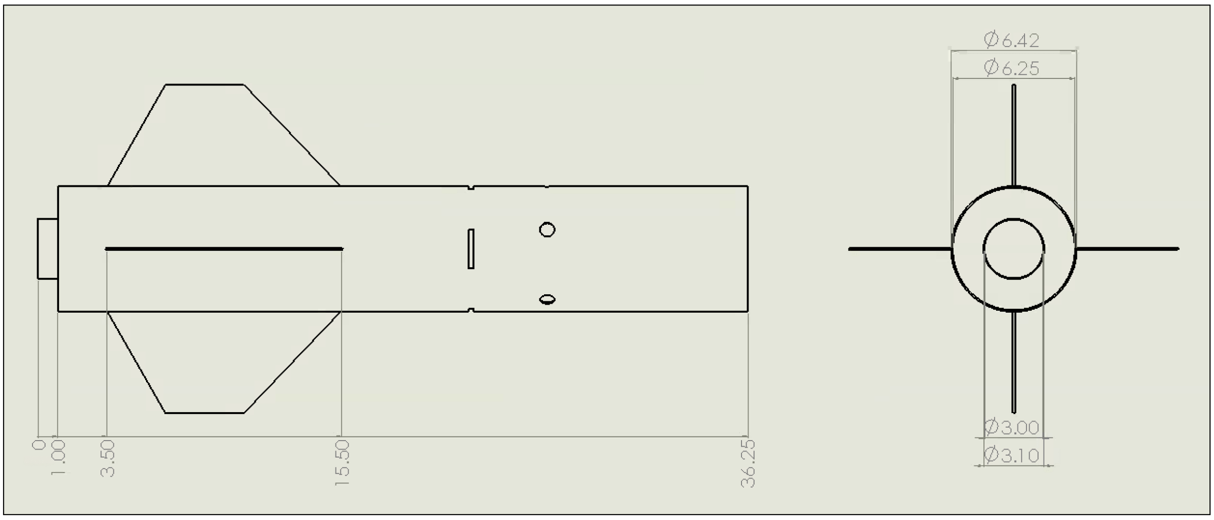


Figure 59: Aft Airframe Drawing

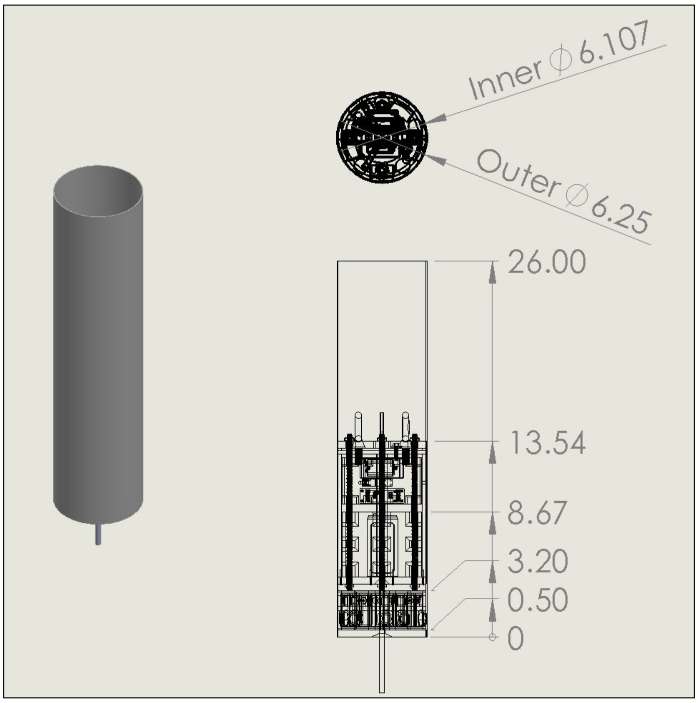


Figure 60: Aft Canister Drawing

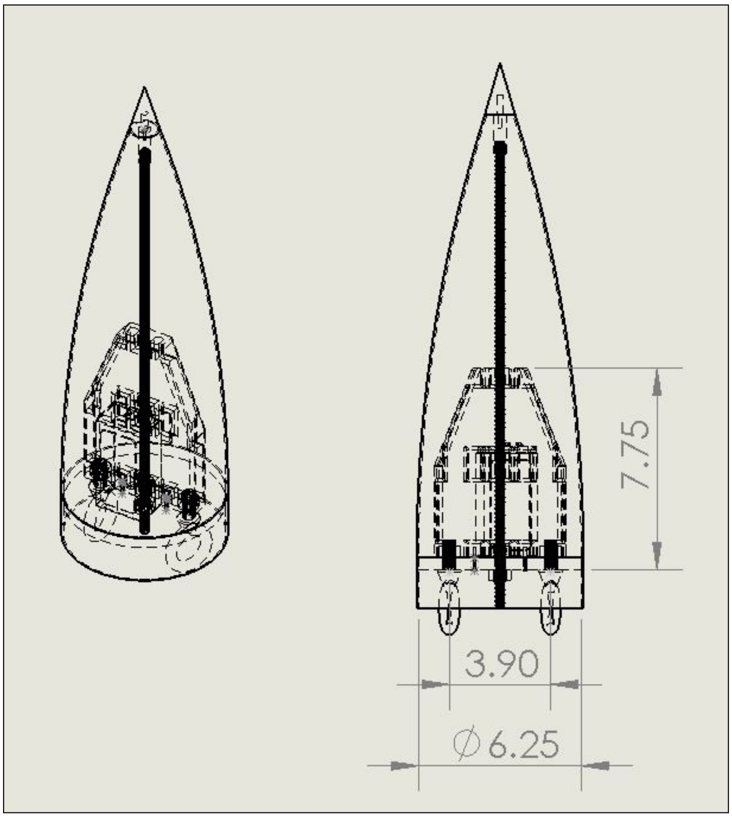


Figure 61: Fore Avionics Drawing

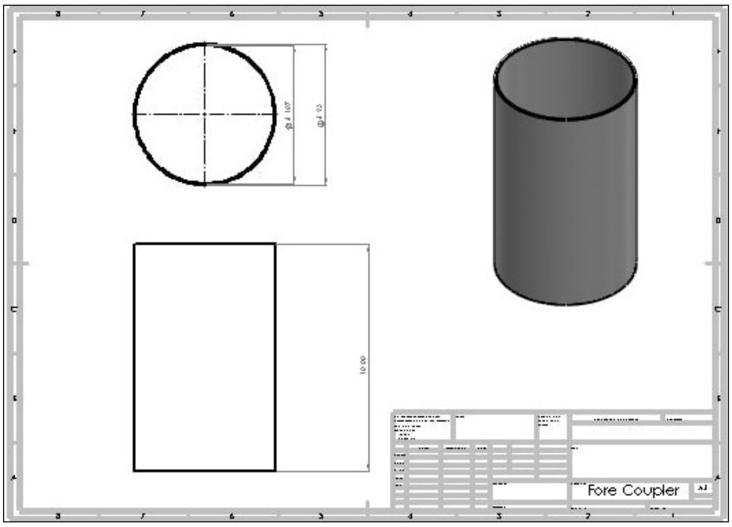


Figure 62: Fore Coupler

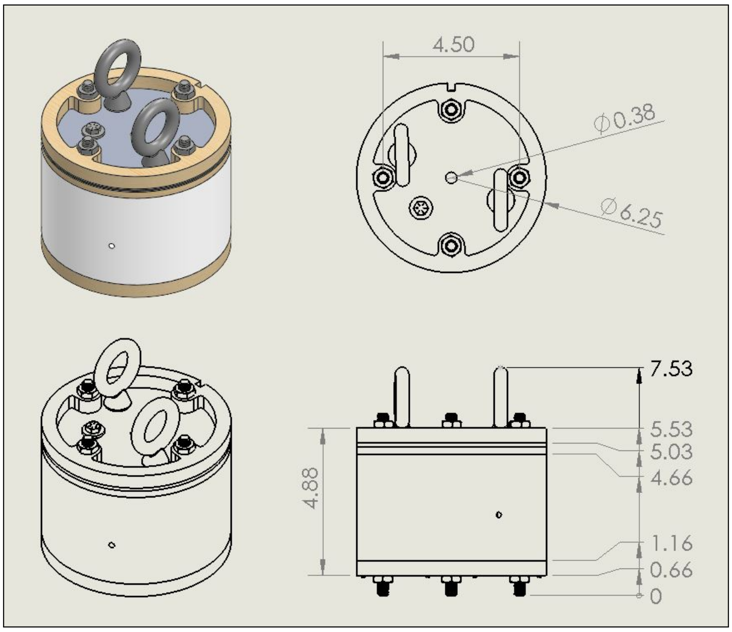


Figure 63: Fore Ejection Bay Drawing

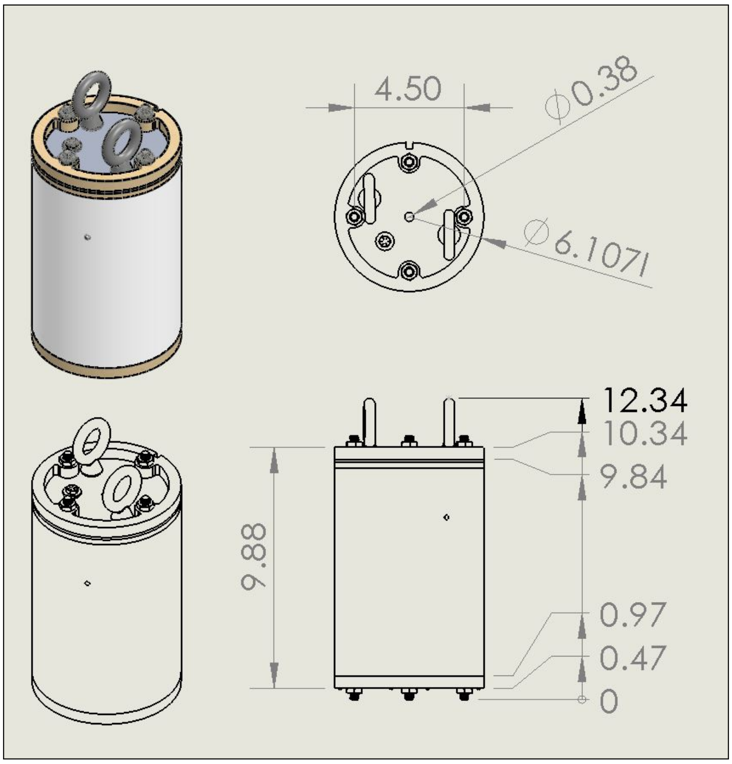


Figure 64: Aft Ejection Bay Drawing

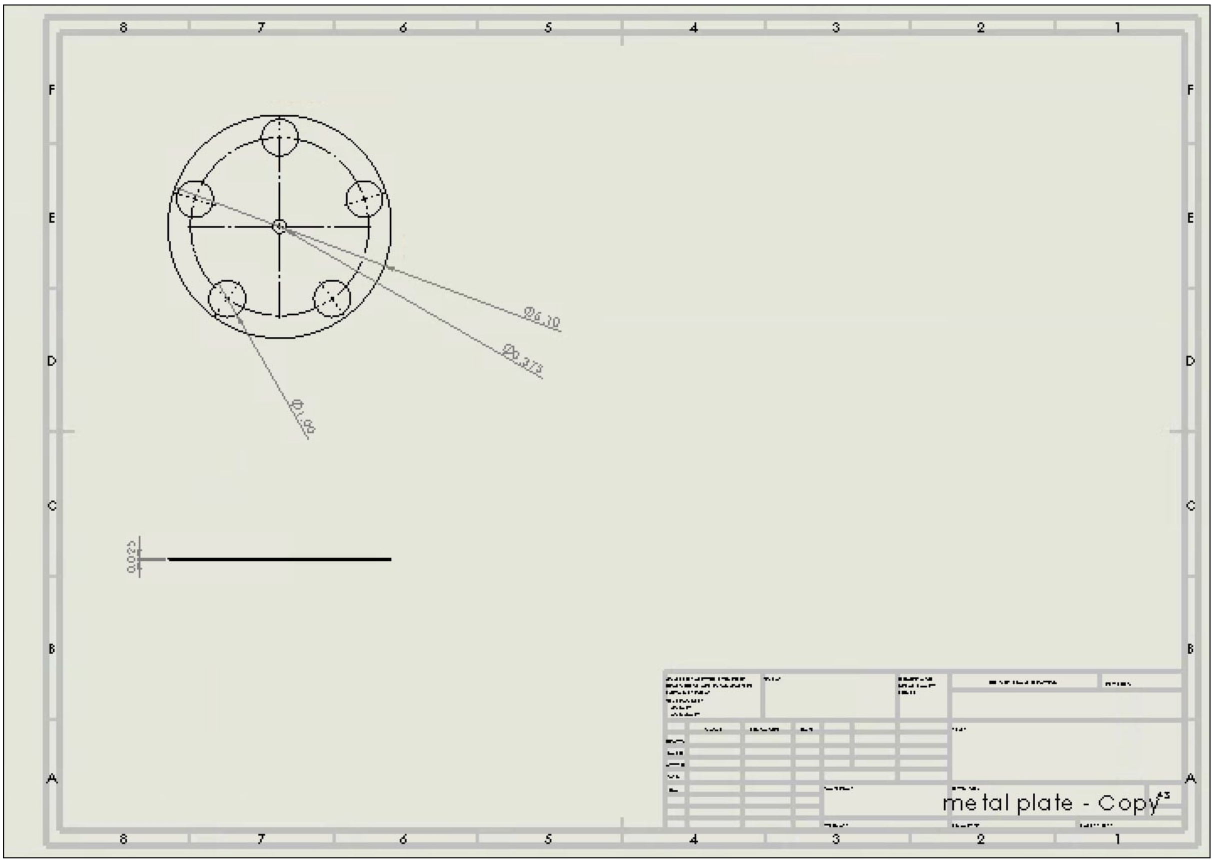


Figure 65: 360° Camera Metal Plate

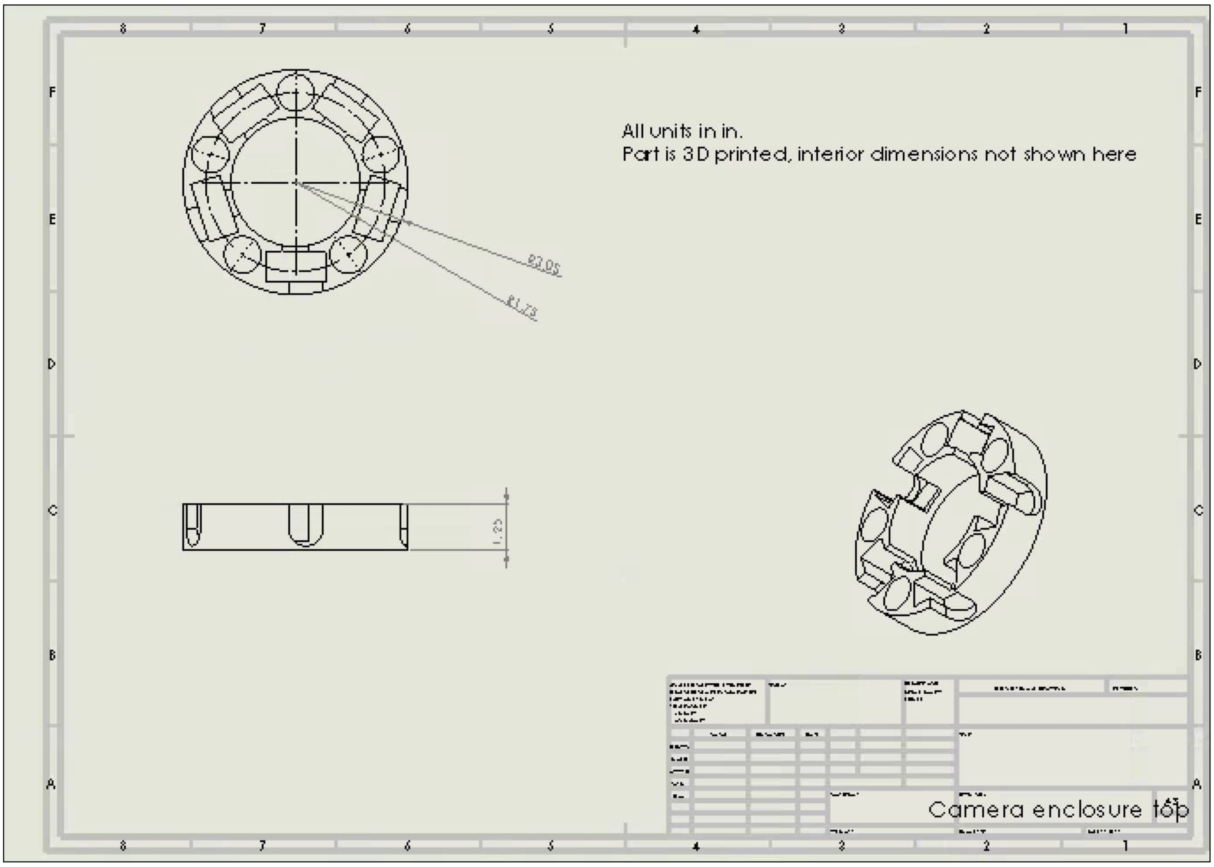


Figure 66: 360° Camera Top Enclosure

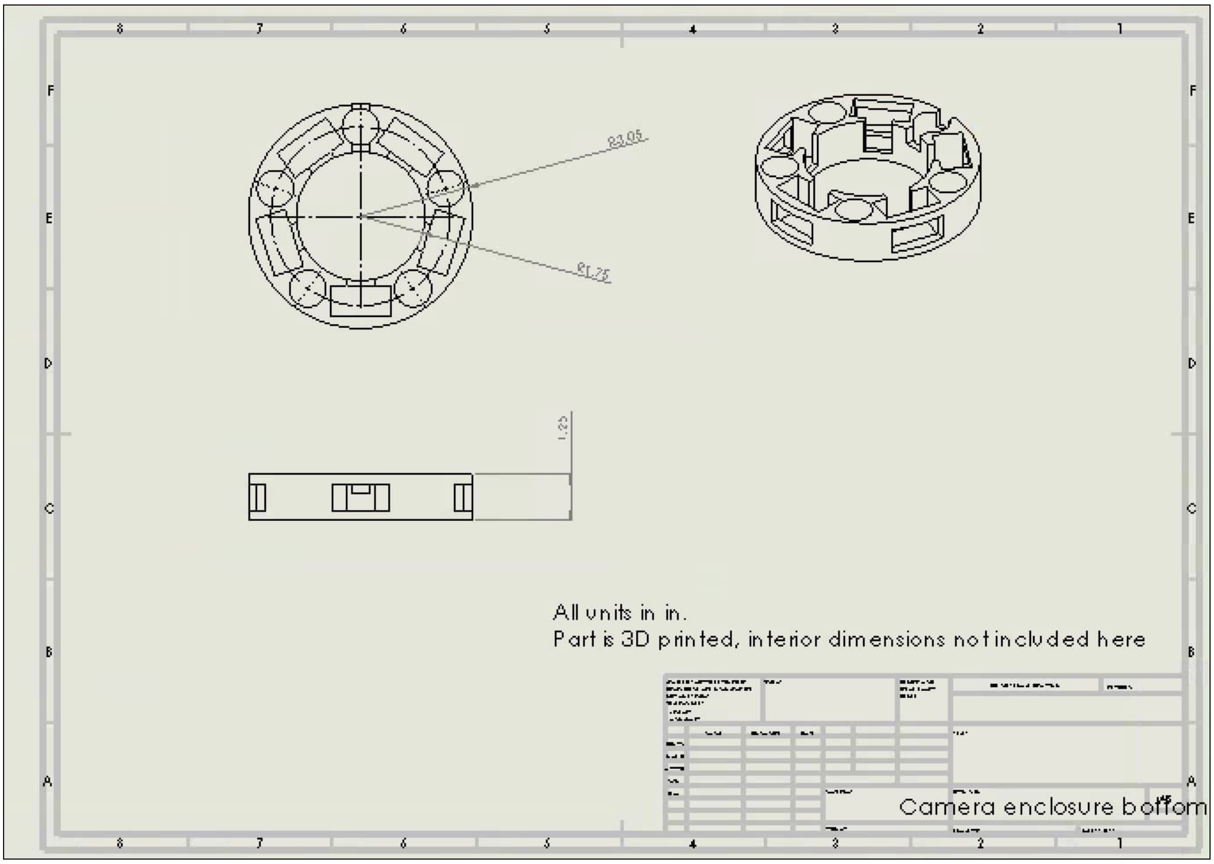


Figure 67: 360° Camera Bottom Enclosure

A.2 Aerodynamics & Recovery

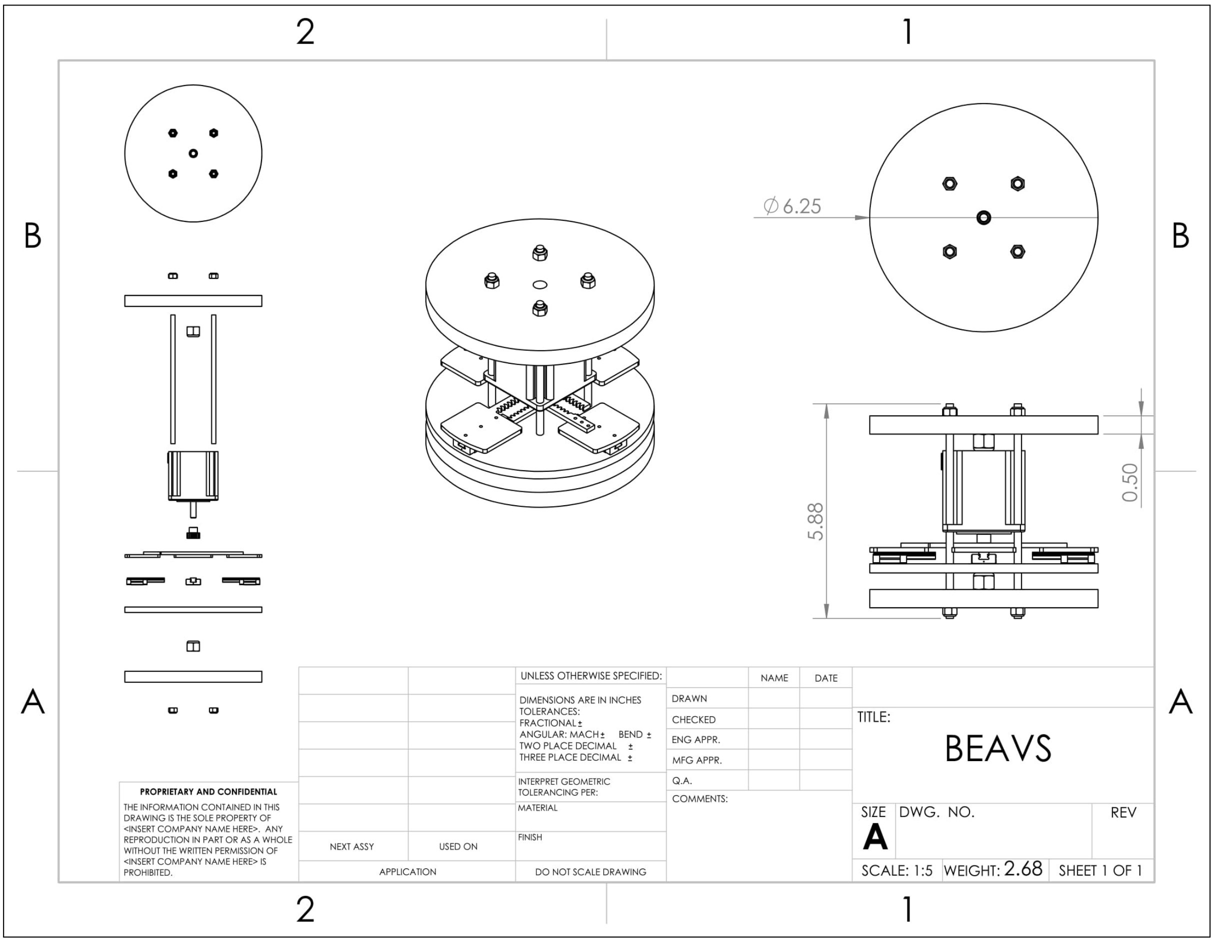


Figure 68: BEAVS - Full Assmebly Drawing

A.3 Payload

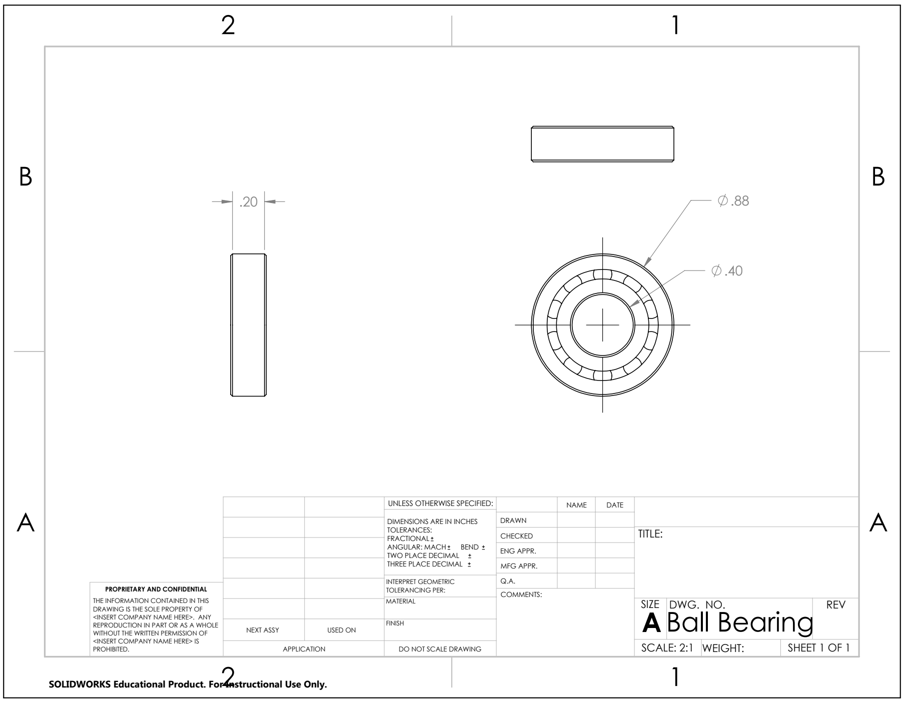


Figure 69: Drivetrain - Annular Ball Bearing

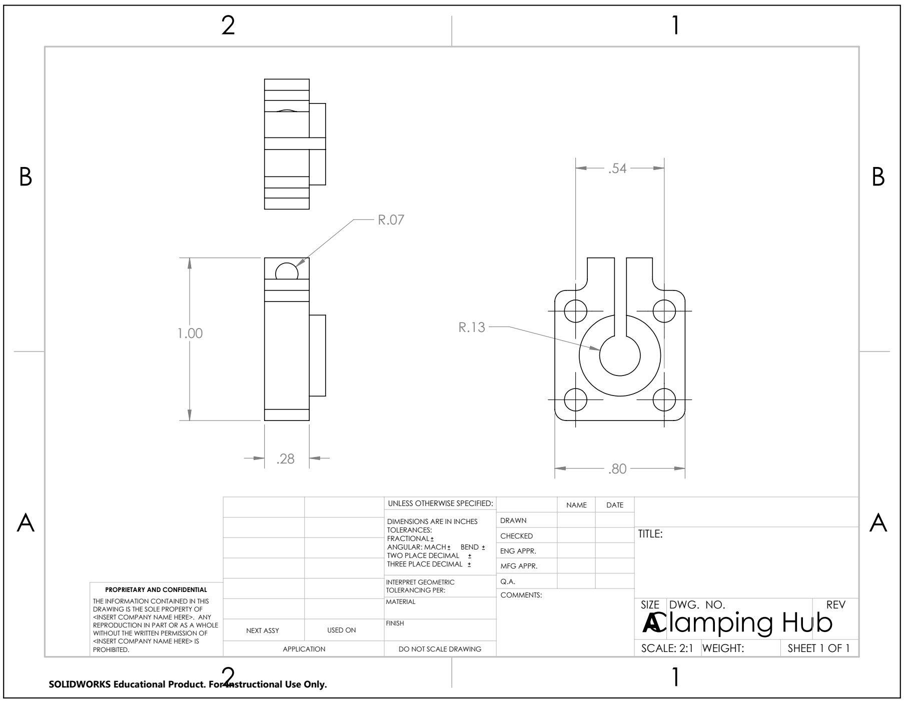


Figure 70: Drivetrain - Clamping Hub

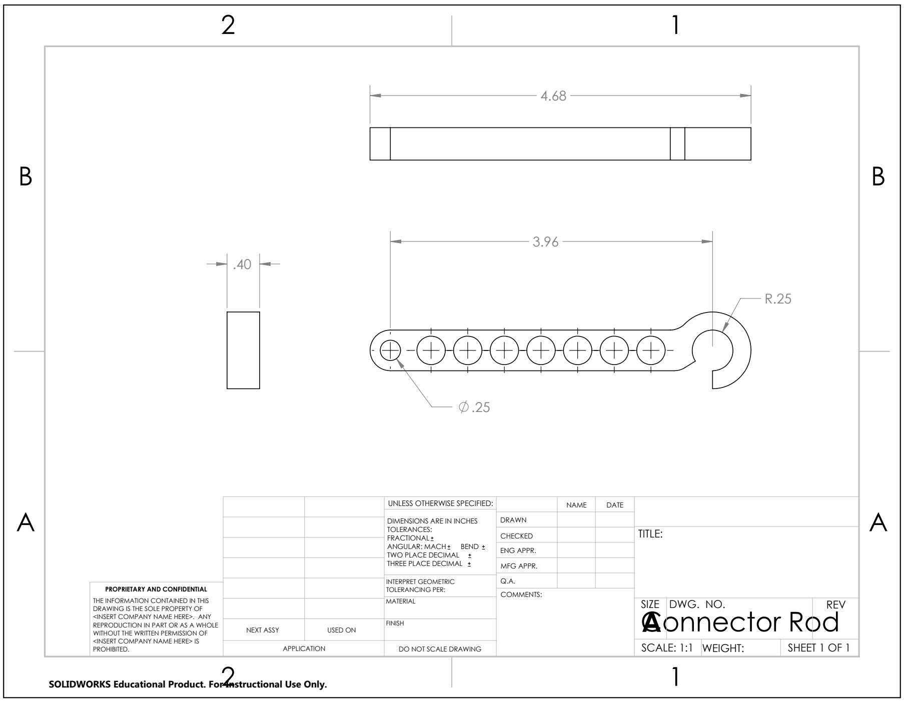


Figure 71: Drivetrain - Connector Rod

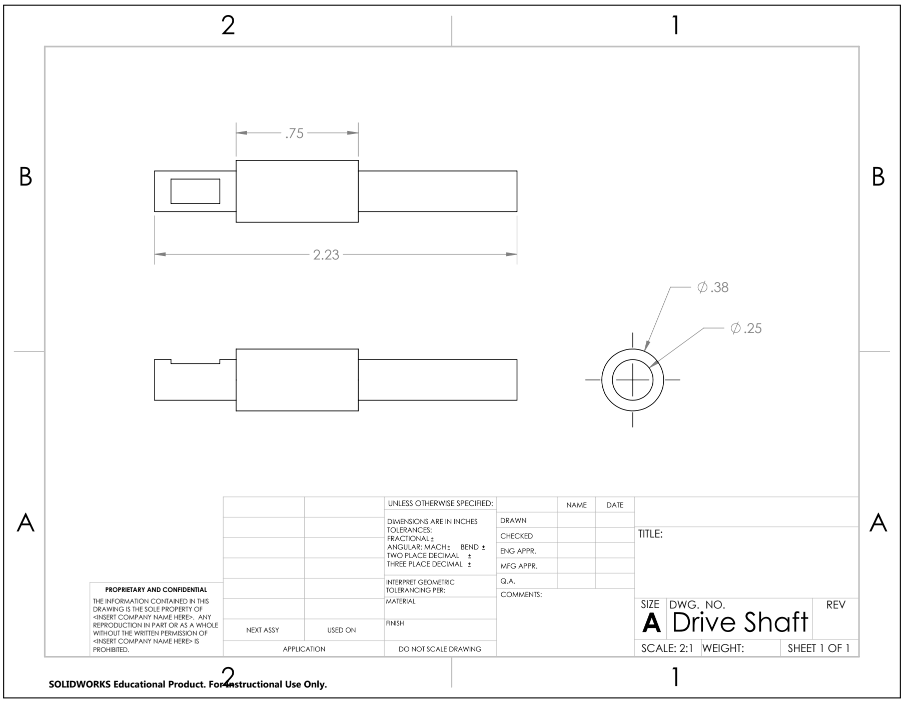


Figure 72: Drivetrain - Drive Shaft

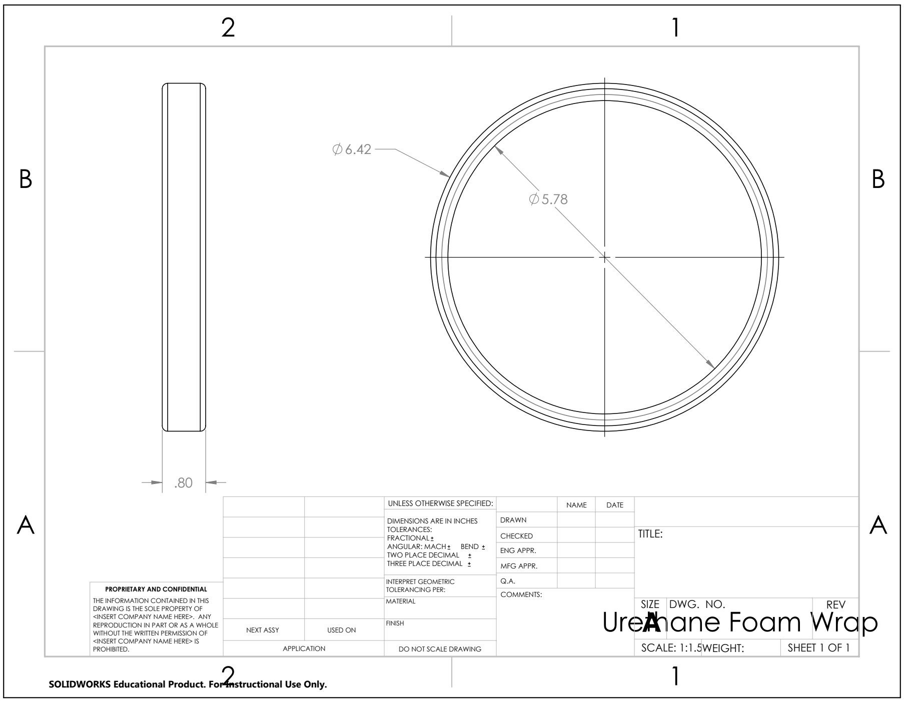


Figure 73: Drivetrain - Urethane Foam Wrap

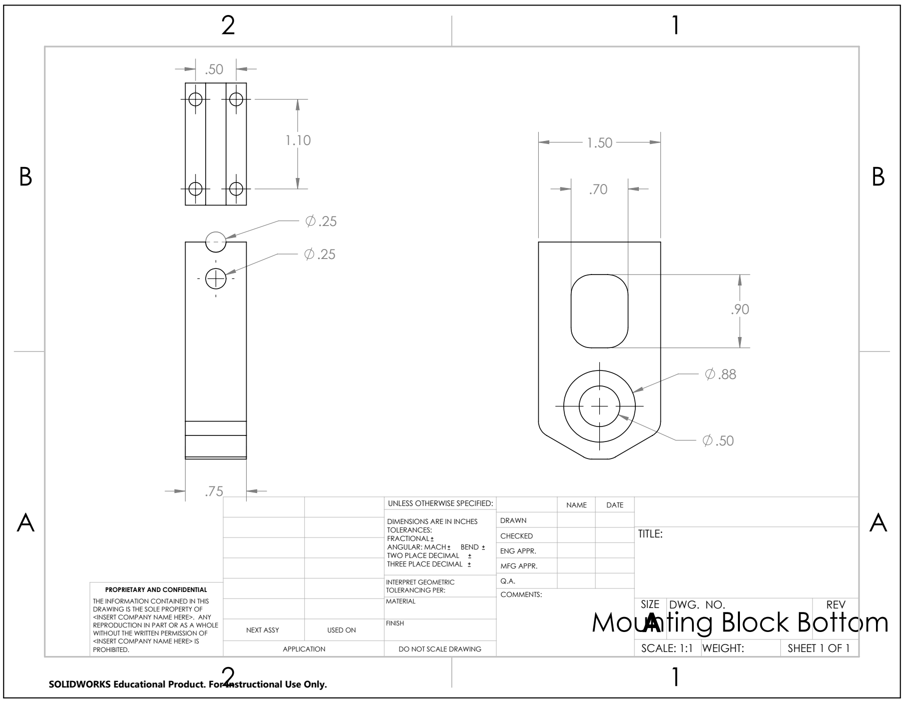


Figure 74: Drivetrain - Mounting Block Bottom

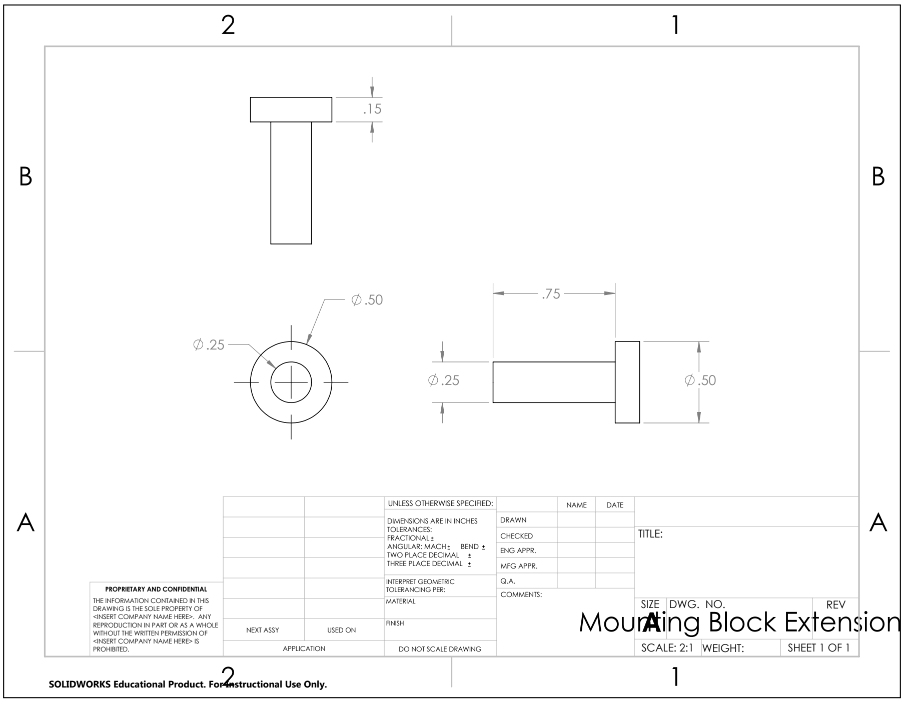


Figure 75: Drivetrain - Mounting Block Extension

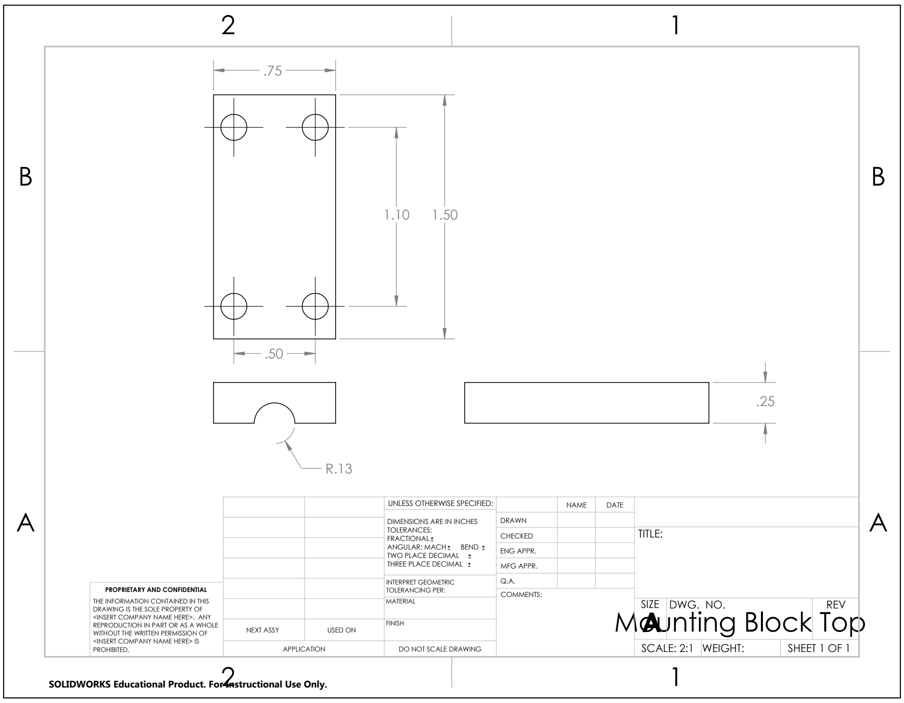


Figure 76: Drivetrain - Mounting Block Top

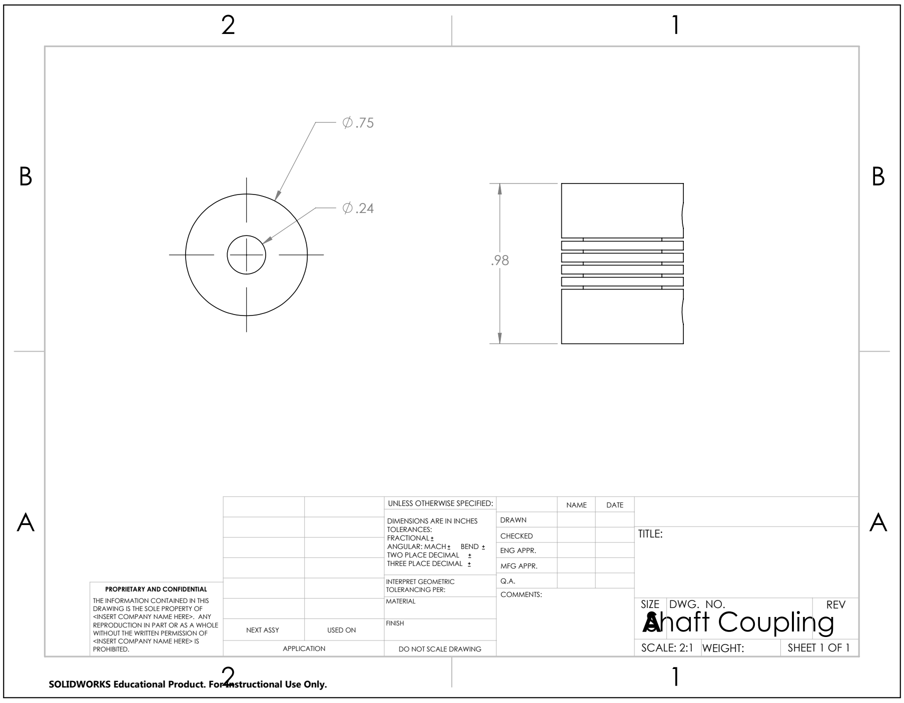


Figure 77: Drivetrain - Shaft Coupling

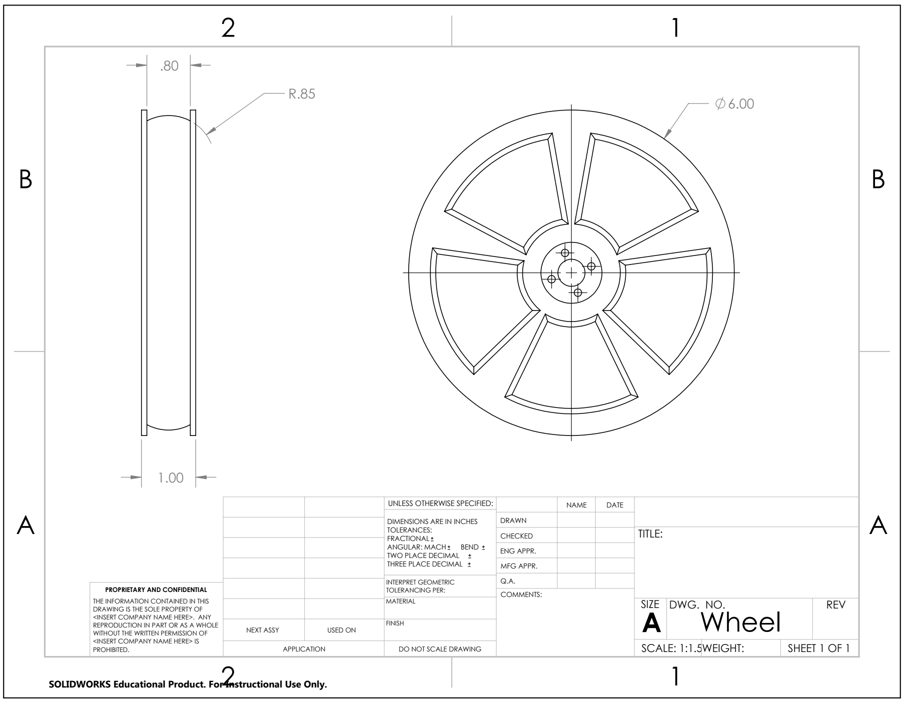


Figure 78: Drivetrain - Wheel

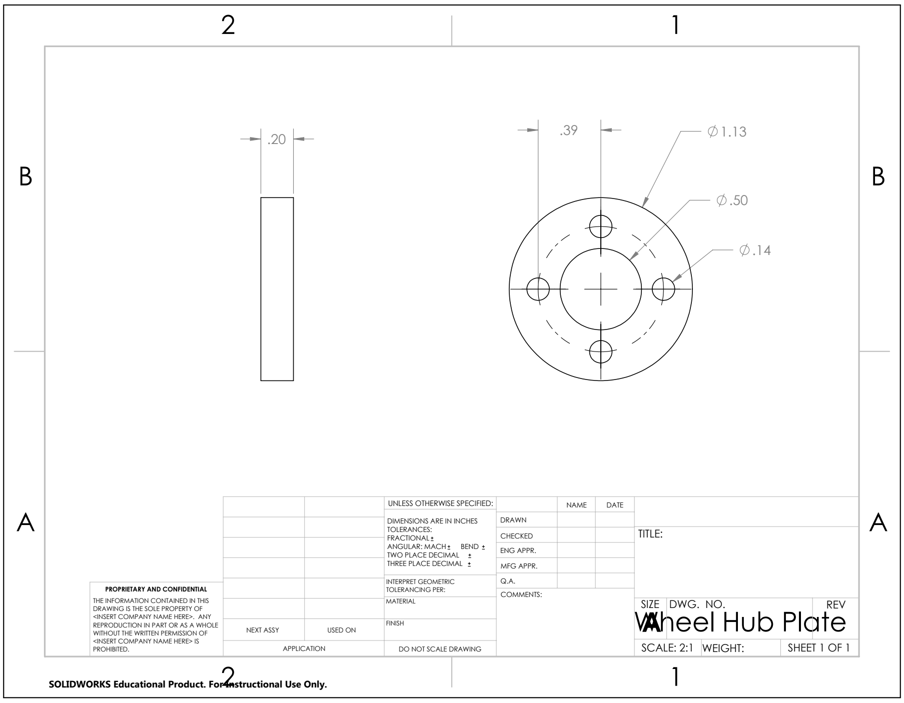


Figure 79: Drivetrain - Wheel Hub Plate

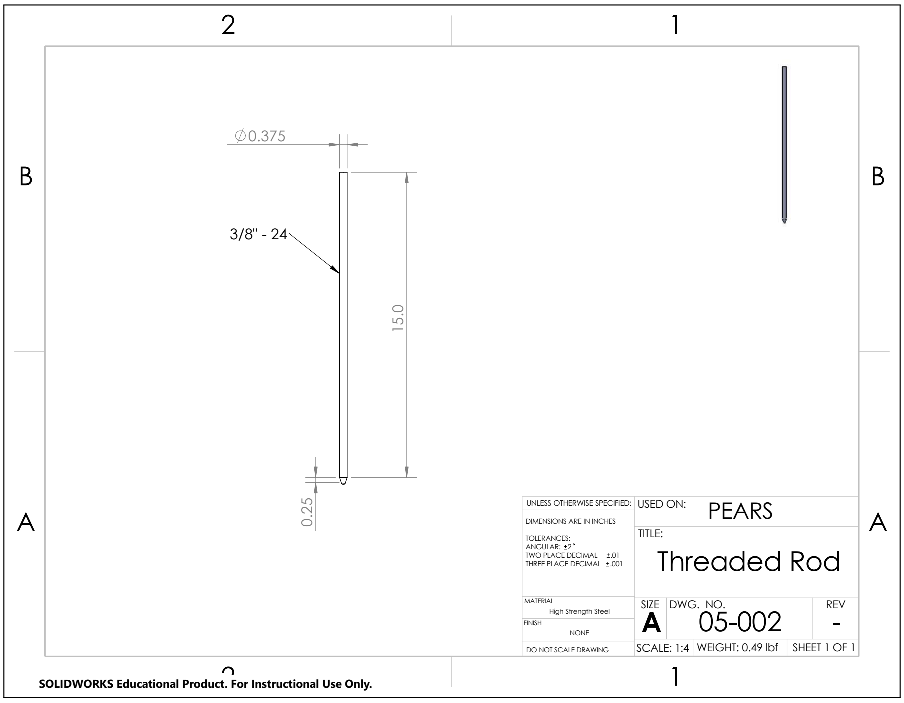


Figure 80: PEARS - Threaded Rod

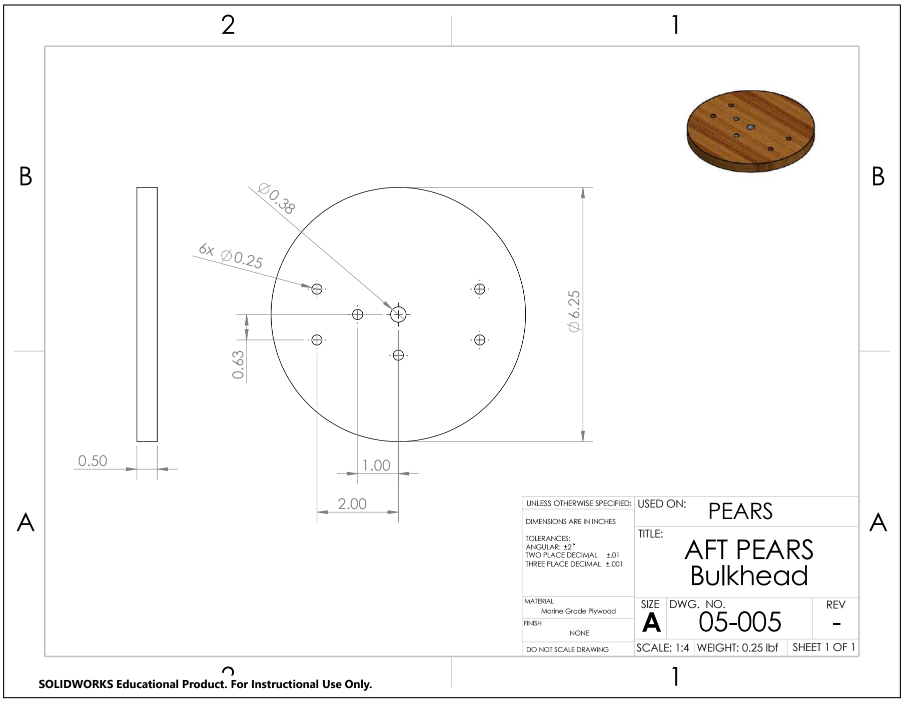


Figure 81: PEARS - Aft Bulkhead

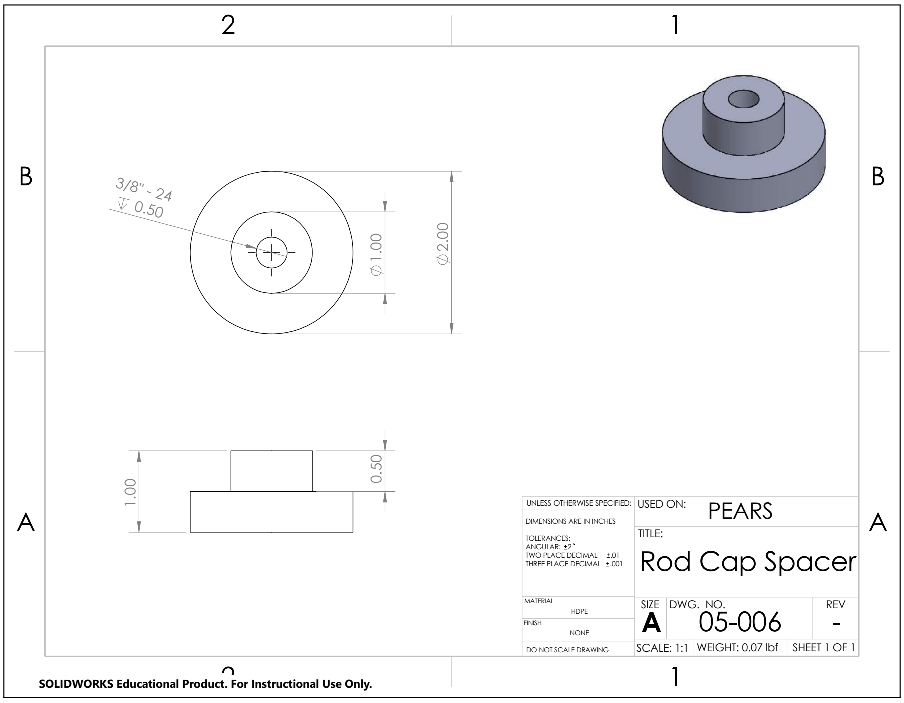


Figure 82: PEARS - Rod Cap Spacer

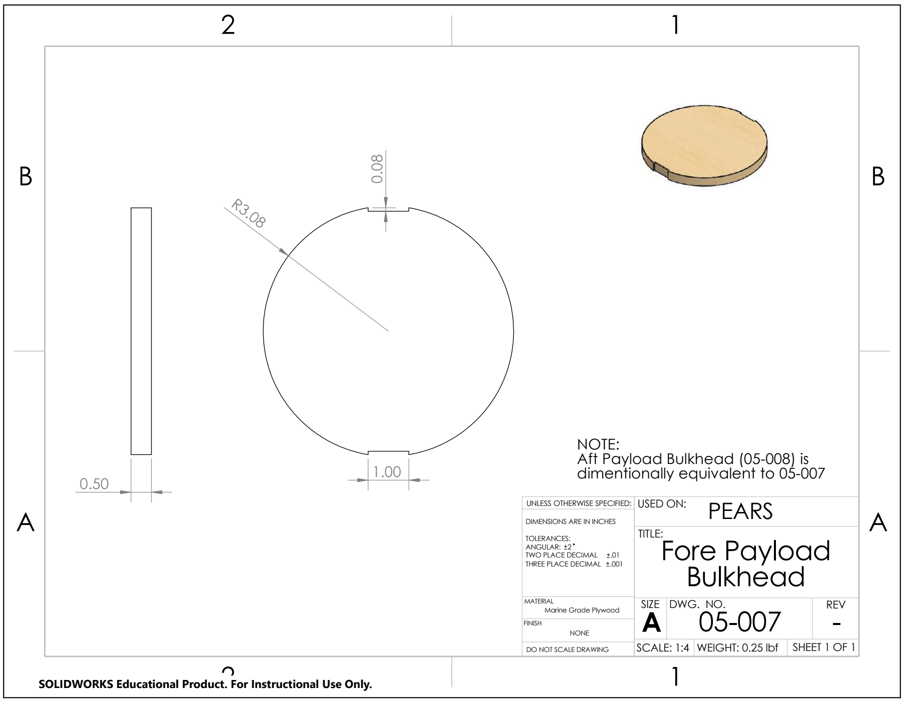


Figure 83: PEARS - Fore Payload Bulkhead

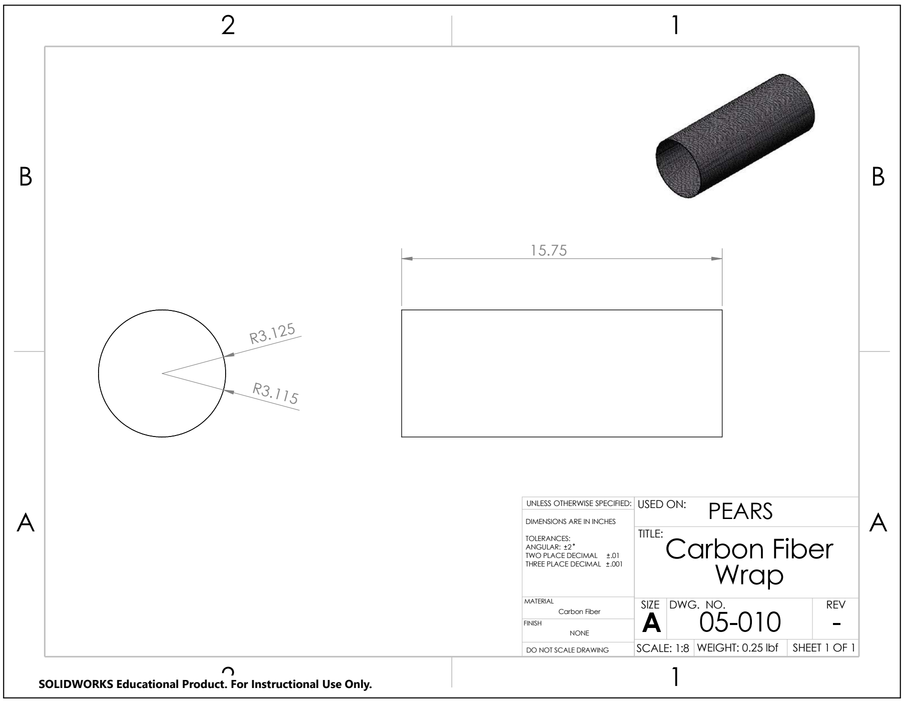


Figure 84: PEARS - Carbon Fiber Wrap

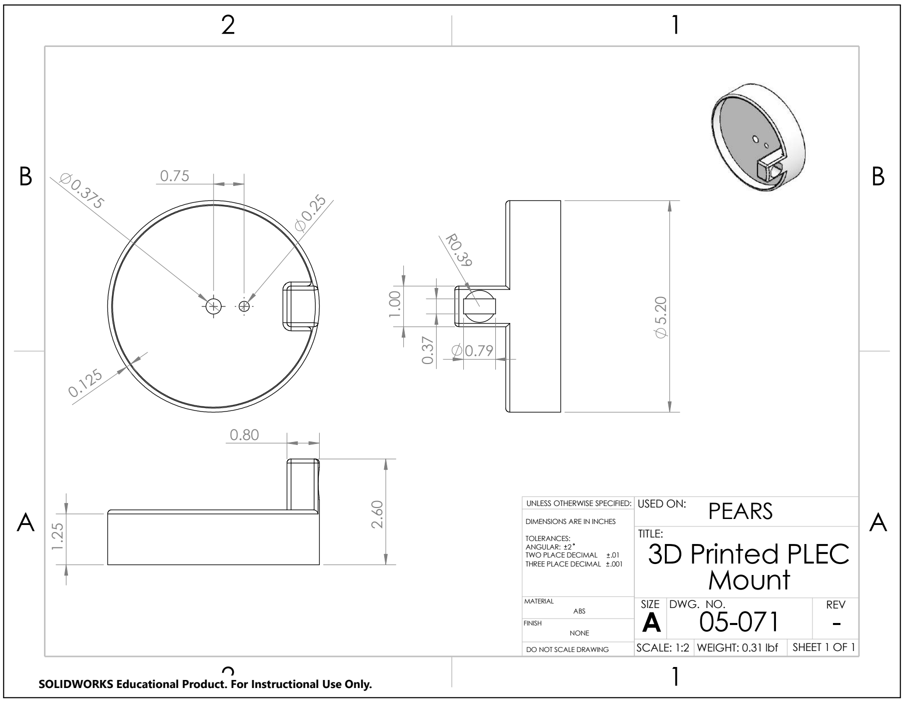


Figure 85: PEARS - 3D Printed PLEC Mount

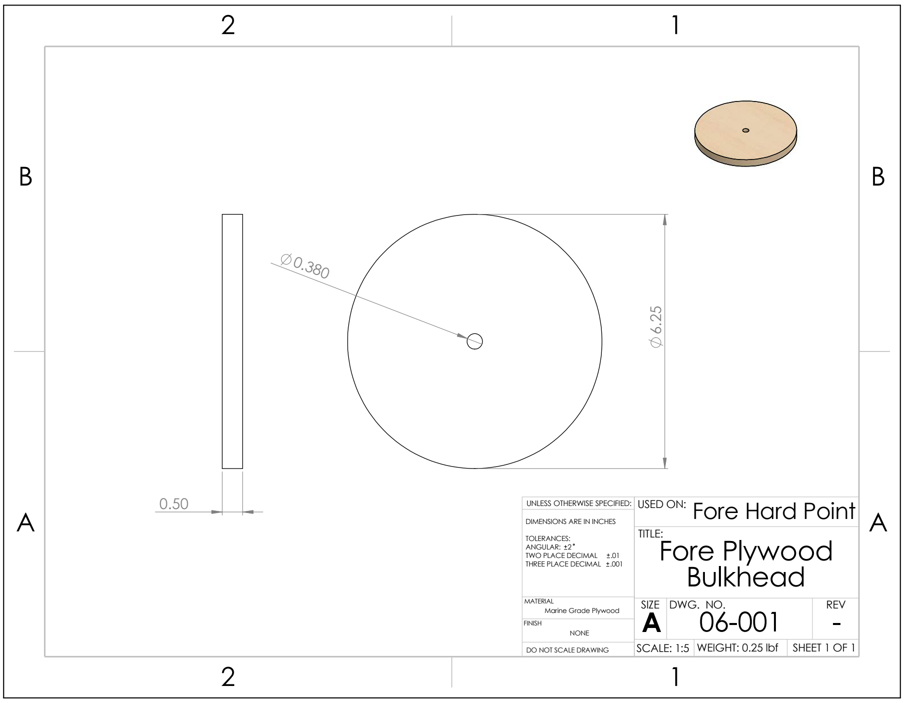


Figure 86: FHP - Fore Plywood Bulkhead

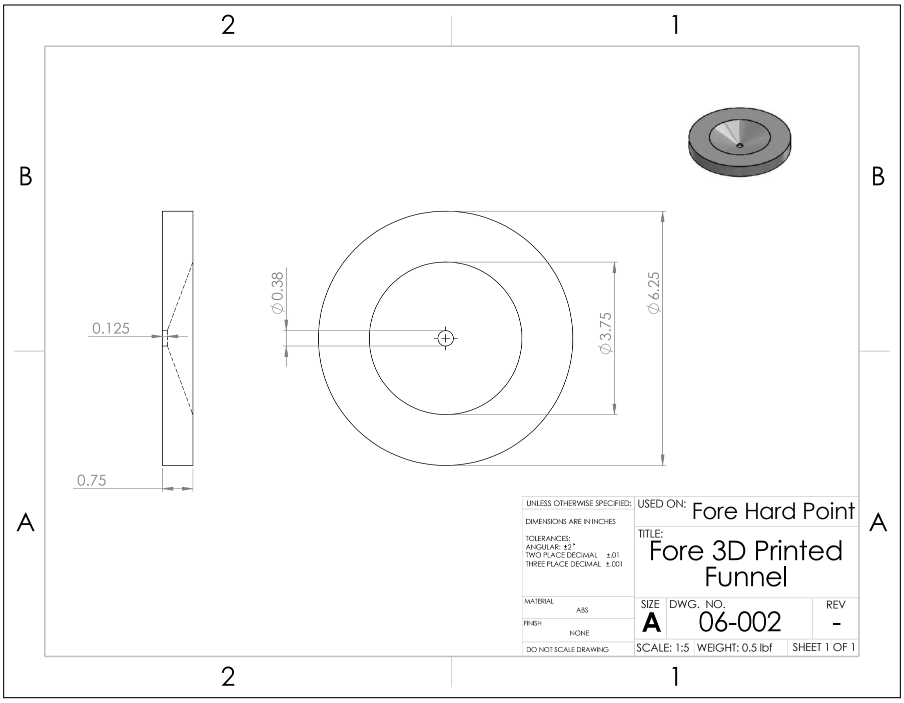


Figure 87: FHP - Fore 3D Printed Funnel

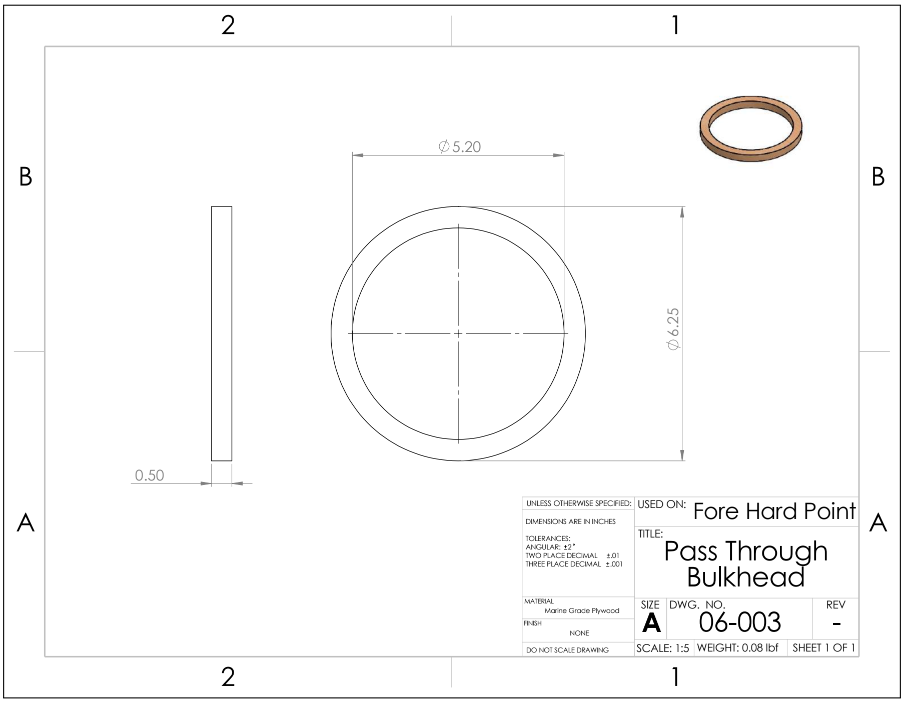


Figure 88: FHP - Pass Through Bulkhead

Figure 89: SCAR - Shaft for Container

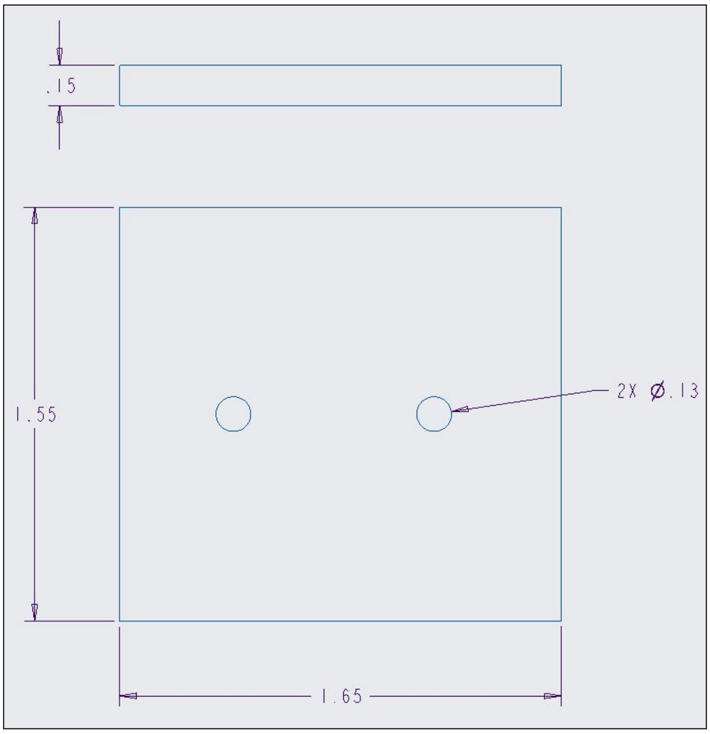


Figure 90: SCAR - Doors for Container

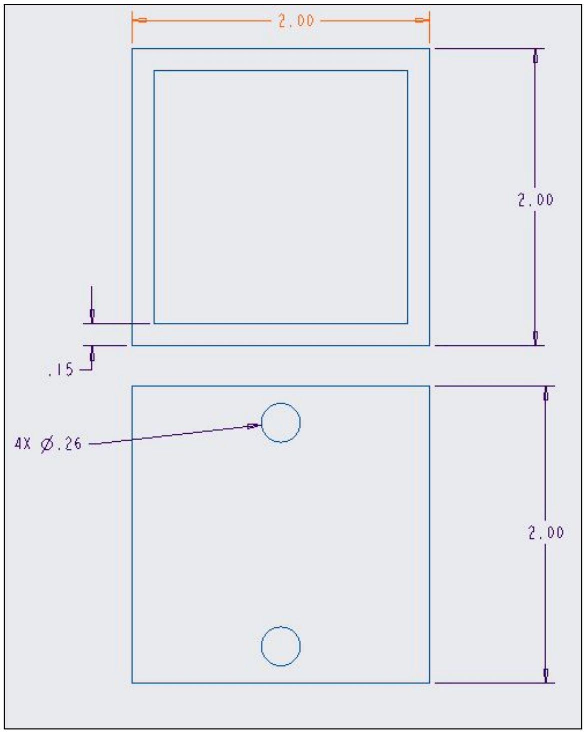


Figure 91: SCAR - Container

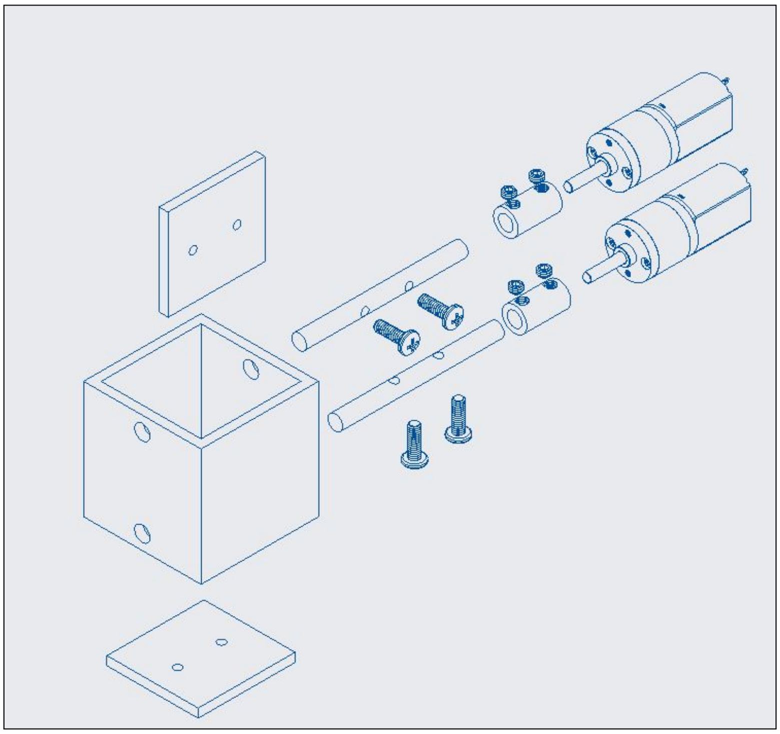


Figure 92: SCAR - Assembly Drawing for Container

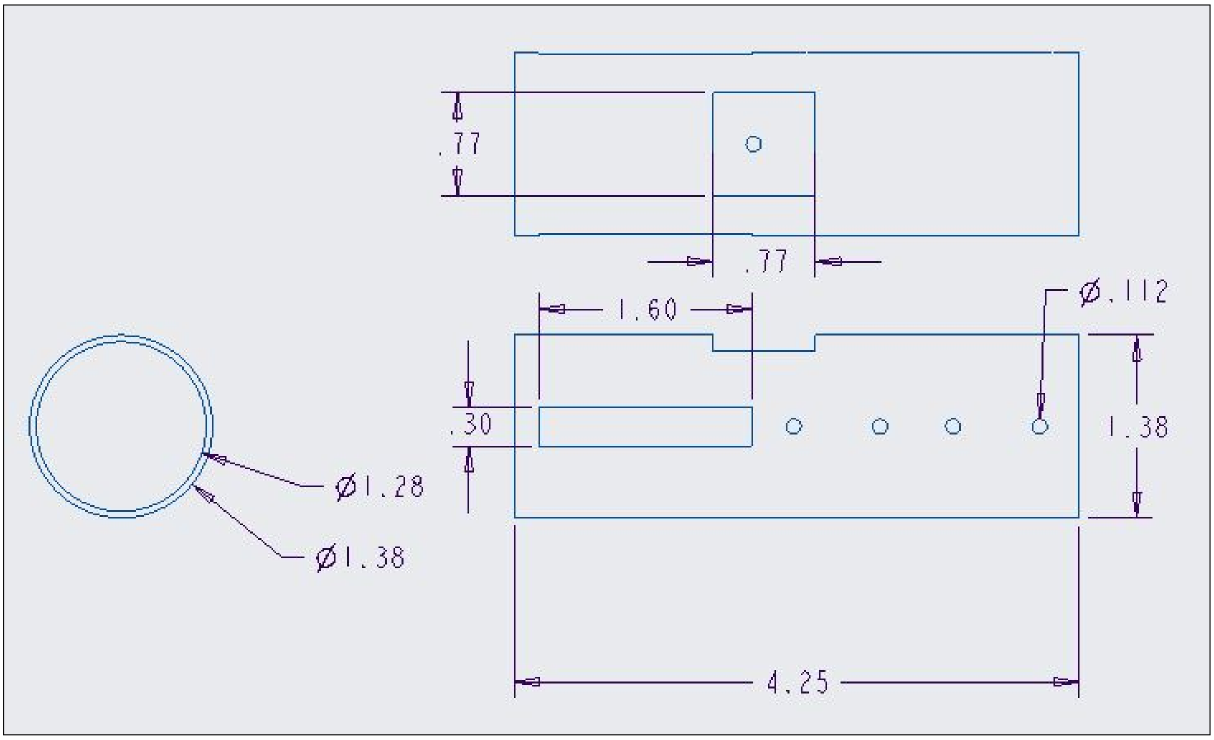


Figure 93: SCAR - Outer Auger Tube

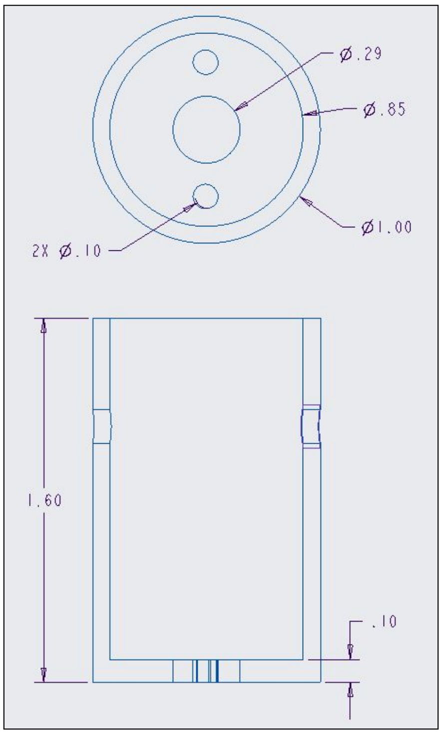


Figure 94: SCAR - Motor Enclosure

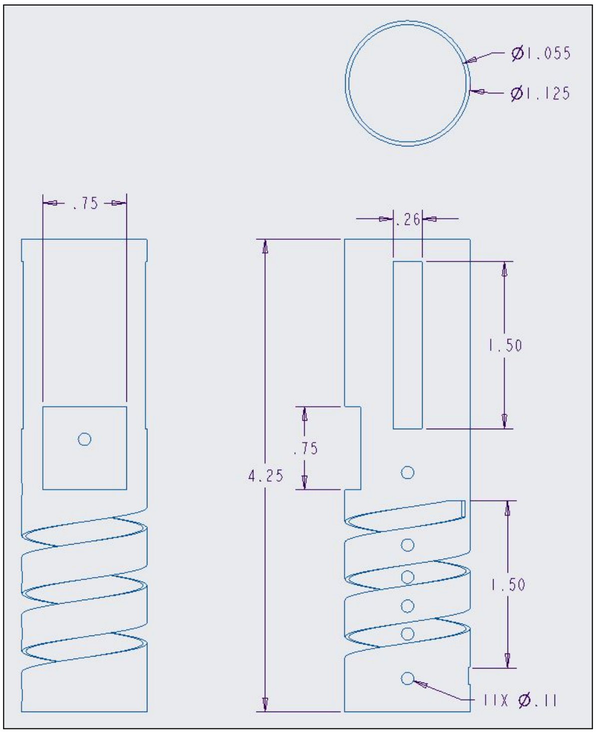


Figure 95: SCAR - Inner Auger Tube

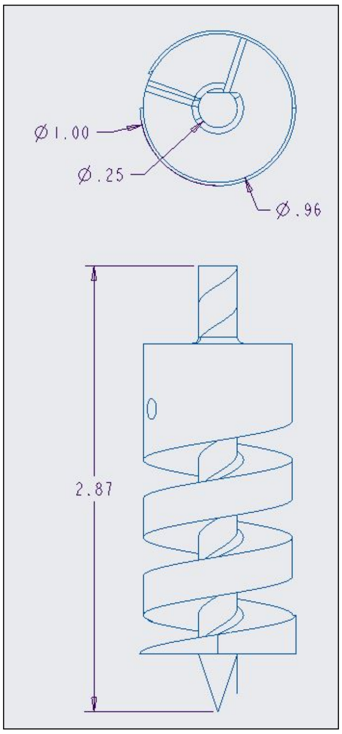


Figure 96: SCAR - Auger

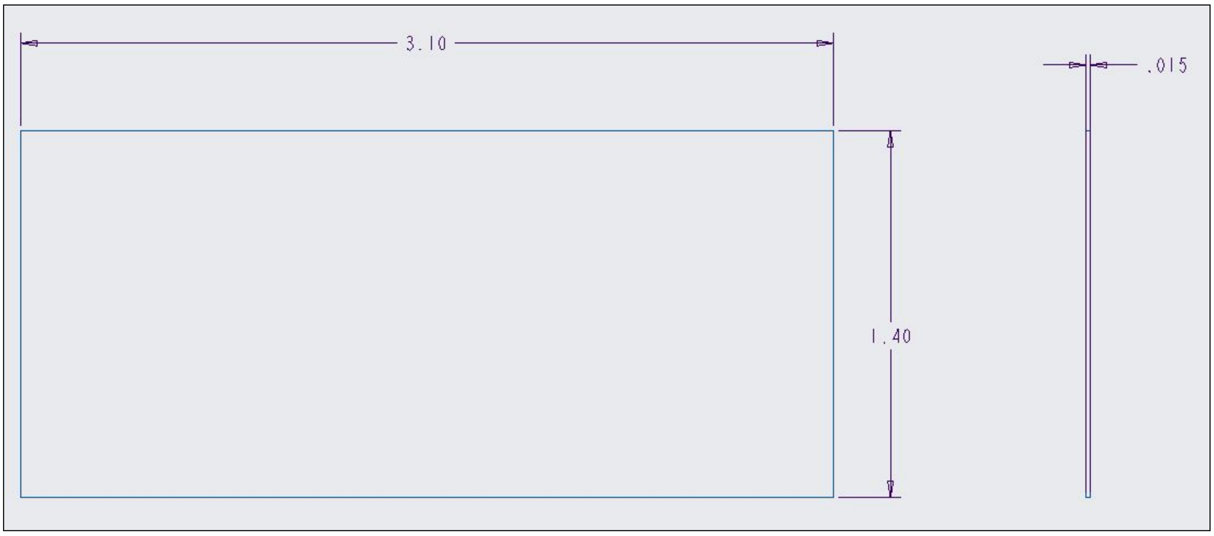


Figure 97: SCAR - Unwrapped Auger Wrap

Figure 98: SCAR - Auger Bar

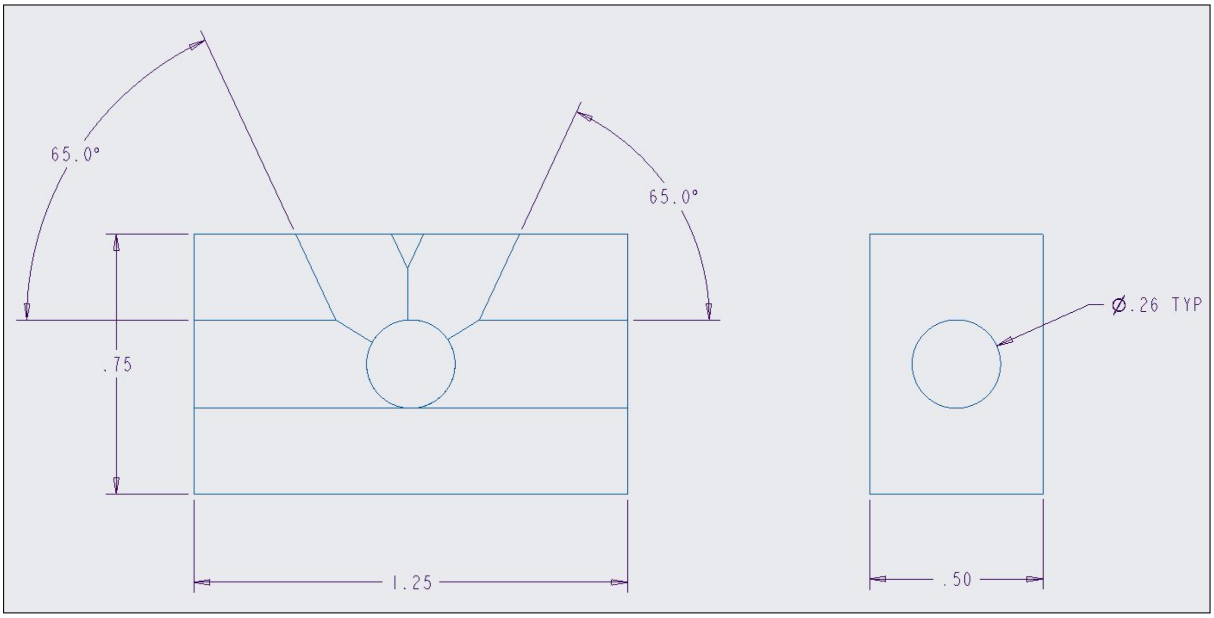


Figure 99: Chassis - Connection Block

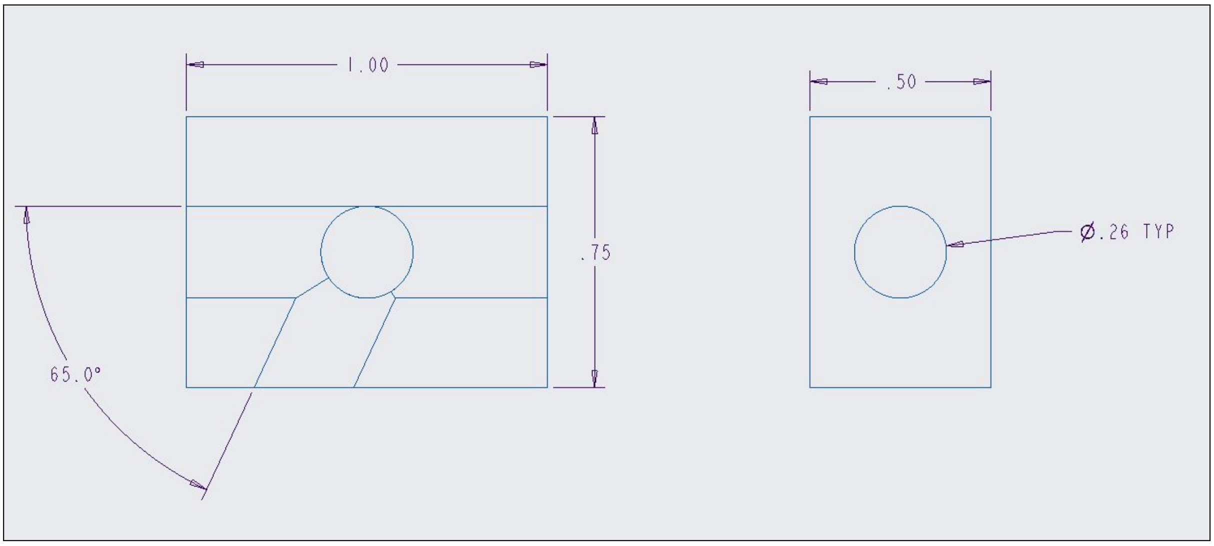


Figure 100: Chassis - Corner Connection Block

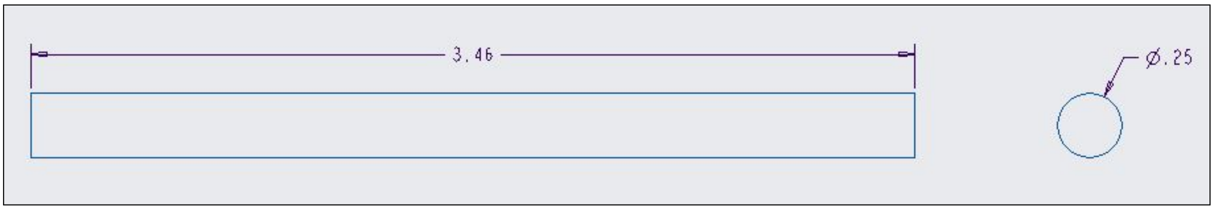


Figure 101: Chassis - Long Truss Member

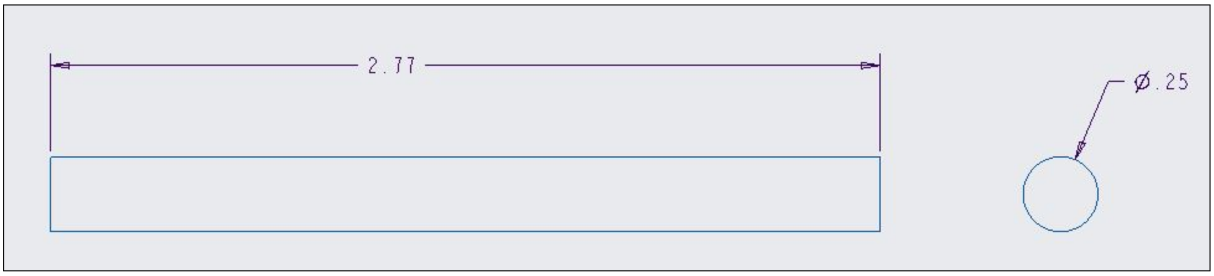


Figure 102: Chassis - Short Truss Member

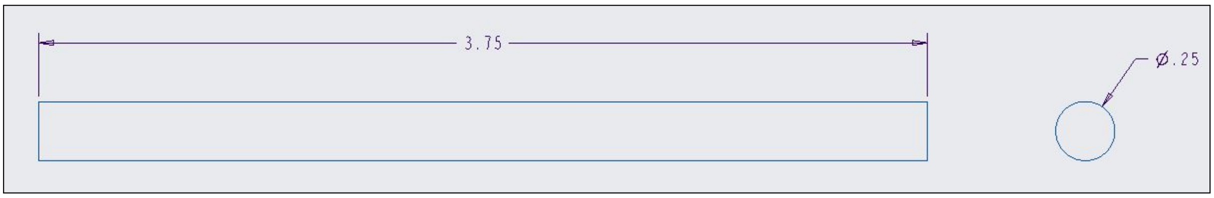


Figure 103: Chassis - Cross Member