OREGON STATE UNIVERSITY

2020 NASA SL TEAM

104 KERR ADMIN BLDG. # 1011

CORVALLIS, OR 97331

Preliminary Design Review

November 1, 2019

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Acronym Dictionary

ABS Acrylonitrile Butadiene Styrene - A Common Thermoplastic Polymer. 107, 119

AGL Above Ground Level. 47

AIAA American Institute of Aeronautics and Astronautics. 225

APCP Ammonium Perchlorate Composite Propellant. 182

ARLISS A Rocket Launch For International Student Satellites. 102

ATU Avionics Telemetry Unit. 29, 30, 33, 35, 36, 78, 79, 82, 84, 85

BEAVS Blade Extending Apogee Variance System. 4, 5, 9, 22–24, 28, 64, 65, 68, 69, 73, 74, 147, 163–165, 219, 231–233

BP Black Powder. 52, 53, 55–57, 59, 62, 63, 197

CAD Computer-Aided Design. 15, 62, 65

CDR Critical Design Review. 182, 186

CFD Computational Fluid Dynamics. 17

CG Center of Gravity. 7, 88

CNC Computer Numerical Control. 163

CO2 Carbon Dioxide. 9, 20, 46, 48, 52–59, 62, 63, 153, 156, 157, 160, 197, 235

CP Center of Pressure. 7, 88

DDM Design Decision Matrix. 4, 26, 27, 32, 33, 43, 45, 60, 61, 65, 69, 73, 74, 81, 82, 113, 114, 125

DUO Dynamic Ultra Once-over. 211

EHS Environmental Health & Safety. 173

ESD Electrostatic Discharge. 36, 85, 131

FAA Federal Aviation Administration. 146, 148, 149, 173, 207, 210

FMEA Failure Mode Effects Analysis. 5, 153, 156, 163, 166, 173

FN Foreign National. 177

FOD Foreign Object Debris. 171

FPV First-Person View. 8, 124–126

FRC FIRST Robotics Competition. 226

FRR Flight Readiness Review. 177, 186, 187, 189–191, 208

GLONASS Global Navigation Satellite System. 30, 79

GNSS Global Navigation Satellite System. 4, 6, 7, 30–32, 79–81

GPIO General Purpose Inputs and Outputs. 113, 114

GPS Global Positioning System. 30, 73, 79, 121, 125, 127, 128, 130, 131

GUI Graphical User Interface. 8, 32, 35, 36, 81, 84, 85, 128, 129, 131

HDPE High-density polyethylene. 105, 117–120

HPRSC High Power Rocket Safety Code. 144–147, 149

HPSC High Power Safety Code. 145

I/O Inputs and Outputs. 68

I2C Inter-Integrated Circuit. 35, 68, 84

ICE Innovative Composite Engineering. 225

IDE Integrated Development Environment. 34, 83

IMU Inertial Measurement Unit. 32, 81, 130, 131

IO Input/Output. 33, 82, 113

IoT Internet of Things. 34, 82

JHA Job Hazard Analysis. 173

LiPo Lithium Polymer. 104, 115

LoRa Long Range. 34, 37, 82, 83, 86

LPWAN Low-Power Wide-Area Network. 33, 34, 82, 83

LRR Launch Readiness Review. 208

MSDS Material Safety Data Sheet. 139, 141, 142, 173, 209

MSS Machine Shop Safety. 139, 140

NAR National Association of Rocketry. 137, 144–146, 148, 149, 173, 182, 207, 210

NASA National Aeronautics and Space Administration. 13, 14, 39, 49, 146, 148, 149, 173, 175, 177–179, 182, 189, 191, 192, 205, 210

NFPA National Fire Protection Agency. 137, 140, 173

OROC Oregon Rocketry. 44

OSGC Oregon Space Grant Consortium. 225

OSRT Oregon State Rocketry Team. 8, 13, 15–17, 22–26, 29, 35, 39, 42, 45, 46, 51–54, 57, 62–64, 67, 73–75, 78, 89, 92, 102, 103, 107, 110, 115, 117, 118, 123, 126, 137, 140, 154, 173, 174, 176, 177, 195, 209, 212–217, 219, 220, 223, 225, 226

OSU Oregon State University. 54, 139, 140, 173, 225, 226

PCB Printed Circuit Board. 68, 73, 74, 165, 169, 212

PDR Preliminary Design Review. 183

PID Proportional-Integral-Derivative. 74, 164

PPE Personal Protective Equipment. 173

PPP Point-to-Point Protocol. 34, 83

PVC Polyvinyl Chloride. 105

RAC Risk Assessment Code. 4, 137–151

RAM Random Access Memory. 68

RC Remote Control. 122, 123

RF Radio-Frequency. 17, 37, 86, 113, 126, 131

RPM Rotations per Minute. 108

RPN Risk Priority Number. 152–172

RSO Range Safety Officer. 142, 177, 182, 183, 189, 192, 196, 210

RTK Real-Time Kinetic. 32, 81

SD Secure Digital. 32, 35, 81, 84

SL Student Launch. 13

SLI Student Launch Initiative. 25

SNU Seol Nation University. 98

SO Safety Officer. 145, 192, 196, 197, 208–210

SPI Serial Peripheral Interface. 68

STEM Science, Technology, Engineering and Mathematics. 14, 177, 209, 225, 226

TI Texas Instruments. 33, 82

TRA Tripoli Rocketry Association, Inc.. 144–149, 173, 182, 210

TTL Through-the-Lens. 123

UART Universal Asynchronous Receiver-Transmitter. 123

UERE User Equivalent Range Error. 30, 79

US United States. 30, 79

USLI University Student Launch Initiative. 13, 14, 39, 50, 175, 176

UV Ultraviolet. 143

1 Summary of Report

1.1 Team Summary

Table 1: Team Summary Chart

Team Name:	Oregon State University 2020 National Aeronautics and Space Administration (NASA) Student Launch (SL) Team
Mailing Address:	104 Kerr Admin Bldg. #1011 Corvallis, OR 97331

Table 2: Team Mentor Summary Chart

Name of Mentor	Dr. Nancy Squires	Joe Bevier
Position within the OSRT	Team Advisor	Team Mentor
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TRA/NAR Number, Certification Level	TRA #15210 Level 3 NAR #97371 Level 3	TRA #12578 Level 3 NAR #87559 Level 3

1.2 Launch Vehicle Summary

The Oregon State Rocketry Team (OSRT) launch vehicle will be designed to safely deliver the payload to a target altitude and protect it during the recovery and landing process. It will complete this mission while maintaining reusability and the ability to be prepared for additional flights the same day. It will be reasonably user friendly while maintaining safe operations for the operators and bystanders. A successful launch vehicle will meet or exceed these criteria. The launch vehicle currently is 118 in. in length, and has an inner diameter of 6.25 in. with a wall thickness of 0.08 in. The launch vehicle's current projected weight is 60 lbs. The current motor selection is the AeroTech L2200, which was selected via various considerations including availability of motor components and current weight of the vehicle. The launch vehicles target altitude will be defined as 4000 ft apogee. The recovery system features one main and one drogue parachute. The drogue will deploy at apogee and the main will deploy at 550 ft above ground level. The primary ejection system is a CO_2 based ejection system, with black powder as a back up. The launch vehicle will have an active airbrakes system to successfully reach the projected altitude.

1.3 Payload Summary

The OSRT will be building a rover for the University Student Launch Initiative (USLI) payload. This rover will be retained during flight, ejected from the launch vehicle with a mechanical device, travel to the sample collection location, collect at least 10 mL of sample, and travel 10 linear feet away after collection. The final rover design is depicted in Figure 79. The design will include a lead screw motor for retention and ejection, expandable wheels, a carbon fiber tail for stability, and an auger-type collection system.

2 Changes Made Since Proposal

2.1 Vehicle Criteria Changes

Since proposal, the general layout and length of the launch vehicle has been changed as shown in Figure 1. The vehicle has been configured to descend on a single drogue and main parachute. Reducing the amount of systems in the over all design, reducing weight, and reducing the chance of Shock cord and parachute tangling during deployment. The airframe has also increased in length to 118 in. to accommodate a larger motor, and a larger payload bay. This is to ensure that the launch vehicle can fit a motor that is able to propel the payload to the desired altitude and increase the available space for the payload.

2.2 Payload Criteria Changes

Since proposal, improvements have been made to the payload drivetrain, structure, and ejection system. The wheel design has changed from a claw like design, shown in Figure 77, to an expandable wheel, shown in Figure 66. The expandable wheel will have lower torque requirements and better utilize the limited battery life. The rover structure has changed from a cylindrical design, shown in Figure 77, to a lighter weight design consisting of polymer sheet and carbon fiber struts, shown in Figure 91. The ejection and retention system has also been further refined.

2.3 Project Plan Changes

Since proposal, the main issue that has been faced in regard to the project plan is project slip. The Aerodynamics and Recovery and Structure and Propulsion subteams needed to push back their ordering to wait until the team was accepted to the NASA USLI competition, and then ordering has taken a decent amount of time. However, the project will be caught up Aerodynamics and Recovery and Structures and Propulsion-wise come the weekend of November 9th, 2019, when the subscale launch is scheduled to take place. Science, Technology, Engineering and Mathematics (STEM) engagement has also had some issues regarding project slip, and while the subteam has managed to reach out to a variety of schools and attend various events, the subteam is still working on scheduling times with schools to meet the minimum STEM Engagement requirements set by NASA.

3 Vehicle Criteria

3.1 Selection, Design and Rationale of Launch Vehicle

3.1.1 Launch Vehicle Layout Overview

OSRT has designed a launch vehicle to take a payload to a specified apogee and return it safely to the ground. The layout of the launch vehicle in its current configuration is noted below in Figure 1 and Figure 2 is a Computer-Aided Design (CAD) model of the leading launch vehicle design..

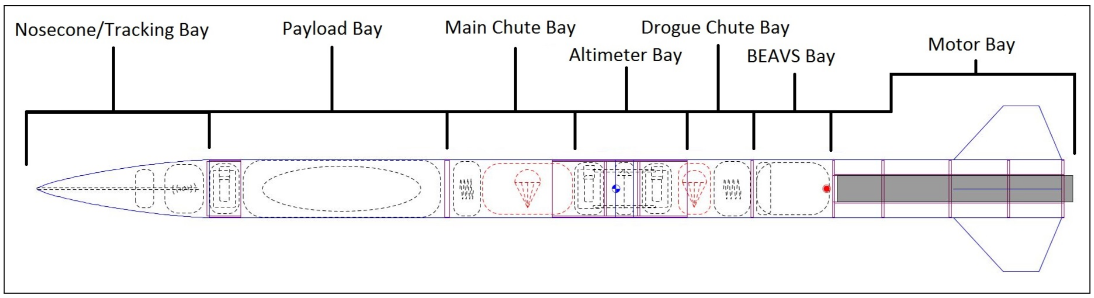

Figure 1: Launch Vehicle Layout

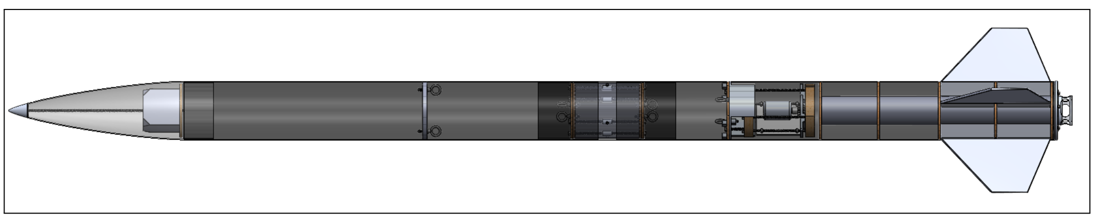

Figure 2: Launch Vehicle CAD Model

OSRT has created several different layouts for the launch vehicle aimed at improving aspects of the airframe and reducing complex systems as much as possible. The first layout had two dual deploy compartments and separate recovery sections, requiring independently tracked descents and recovery, but utilized an unconventional coupler design that allowed the aft and fore sections to slide into each other rather than use a coupler piece, enabling improved deployment and stability. This was determined to not be feasible given manufacturing constraints on airframe tubes. The second design was similar in layout but utilized the more feasible standard coupler design that complied with manufacturing constraints on the airframe size. The final design, shown in Figure 1, will utilize a much simpler layout, to reduce the weight and complexity of recovery and structural systems. This design will have independent drogue and main parachute bays separated by a coupler, and will descend as a single vehicle, eliminating the need for more than one independent tracking system. The fore section will be connected to the main parachute and the coupler, which will then be connected to the drogue chute and finally the aft section. This design results in fewer complex systems, which then have fewer points of possible failure leading to a larger chance of success and a higher level of safety.

3.1.2 Fore Section

The fore section of the launch vehicle houses the main chute bay and the payload bay. Since it houses the payload bay, its strength is vital to ensuring the payload is not damaged during flight. The fore section of the launch vehicle will be constructed out of filament wound carbon fiber tubing. Carbon fiber was chosen over fiberglass for structural components mainly for its desirable strength, ensuring that during flight and landing the rover would remain undamaged, and the launch vehicle would not break apart.

Table 3: Material Types

Design		Fiberglass		Carbon Fiber	
Requirements	Weight	Rating (1-5)	Score	Rating (1-5)	Score
Strength	10	3	30	5	50
Availability	5	4	20	3	15
Manufacturability	4	3	12	3	12
Total		62		77	

There is one bulkhead within the fore section of the launch vehicle; this is where the main parachute mounts on one side of the bulkhead with the payload retention system mounted to the other. The section will be mounted in place using G5000 RocketPoxy, a product that OSRT has used before with success. The weight is estimated to be 5 pounds for this section of the launch vehicle and a model of the fore section is noted below in Figure 3 with dimensions found in Figure 112 .

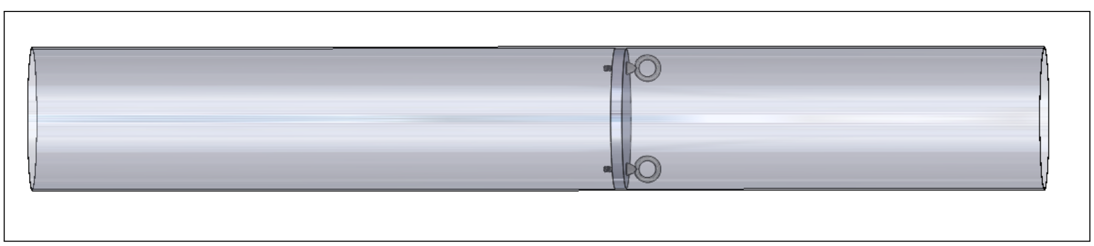

Figure 3: Launch Vehicle Layout

Previous iterations of this section were relatively similar. The first iteration included an avionics bay, which was needed because the aft section separated during decent, resulting in two separate sections that needed tracking. This also resulted in the section housing its own main and drogue parachutes. However, it was changed to the current design after the decision to recover the launch vehicle with one main and drogue parachute. Due to the new launch vehicle layout, the fore section no longer required an avionics bay as all deployment would be controlled via the avionics bay in the coupler. This greatly simplified its design to consist of a bulkhead with mounts for the drogue parachute, along with the payload bay, where the payload retention was mounted to that same bulkhead on the opposing side. The bulkhead had several different options for construction such as carbon fiber, aluminum and plywood however, plywood was chosen due to is ease of manufacture and relatively high strength to weight.

3.1.3 Nose Cone

The main factor in choosing the nose cone is the coefficient of friction. Open Rocket simulations suggest an average speed of Mach 0.5 for most of our simulations. Nose Cone design was chosen on optimal performance that includes low drag coefficient throughout the vehicle's launch, available internal volume for the avionics, and threaded rod to occupy space. Optimal nose cone design was based on Computational Fluid Dynamics (CFD) analysis on six typical designs that are used in low mach speeds that the aero/recovery subteam conducted. Nose cone designs were investigated in Table 4.

An ogive nose cone will be chosen for the OSRT launch vehicle with an estimated weight of 2.72 pounds and dimensions found in Figure 111. Even though there are comparable drag coefficients between the Von Karman, Parabolic, and ogive nose cones at subsonic speeds, the ogive nose cone is easier to purchase and manufacture. A manufacturing process to manipulate the size of the ogive nose cone will be carried out. Another factor to this consideration of choosing the ogive nose cone is that it is easier to manufacture out of all of the six nose cones with the purchase option of obtaining one with a suitable material that will benefit radio frequency transmissions.

Materials will primarily be considered based on their versatility that factor in strength and attenuation avoidance. Fiberglass has the best combination of these qualities because of its Radio-Frequency (RF) transparency and high compression strength. Other engineering requirements for material selection were put into the following investigative variables:

- Strength: Necessary to maximize to reduce probability of nose cone 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 4: Nose Cone Designs

Design		**LD Haack**		**LV Haack**		**Power Series ($X^{0.5}$)**		**Parabolic**		**Ogive**		**Conical**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Drag Coefficient	10	3	30	3	30	3	30	3	30	3	30	2	20
Ease of Manufacturing	7	3	21	3	21	4	28	3	21	3	21	4	28
Ease of Purchasing	5	3	15	3	15	2	10	3	15	4	20	4	20
Total		71		71		68		66		71		68	

Table 5: Nose Cone Material

Design		**Fiberglass**		**Plastic**		**Carbon Fiber**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Strength	8	3	24	22	16	4	32
Ease of Manufacturing	9	3	27	2	18	3	27
RF Transparency	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**		**99**		**111**	

3.1.4 Main Coupler

The main coupler is central to the design of the launch vehicle, tethering the fore and aft sections together during flight and recovery. As the design of the launch vehicle layout has changed the coupler has changed to accommodate those designs as well. The first iteration actually did not have a coupler present in between the fore and aft sections of the launch vehicle due to the fact that the fore section was designed to slide back into the aft section to hold them together during flight; as shown in Figure ??.

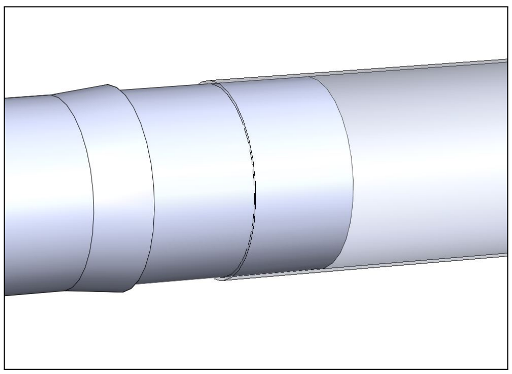

Figure 4: Coupler Gen One

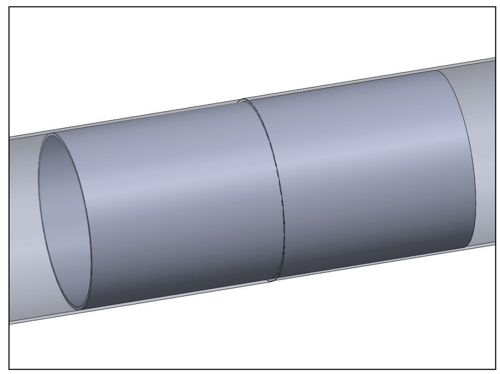

Figure 5: Coupler Gen Two

This was proven to be an unrealistic approach to the problem because of manufacturing constraints on the airframe diameters. Mandrels of correct sizes were not available and prohibitively expensive to manufacture. Generation Two of the coupler was a common design present in many high powered rocketry models. It connected the fore and aft sections with a 14 in. long coupler. This generation was phased out due to a redesign of the parachute compartment layout in the launch vehicle and a need to house avionics and deployment hardware. The next iteration, generation three, is shown in Figure ??.

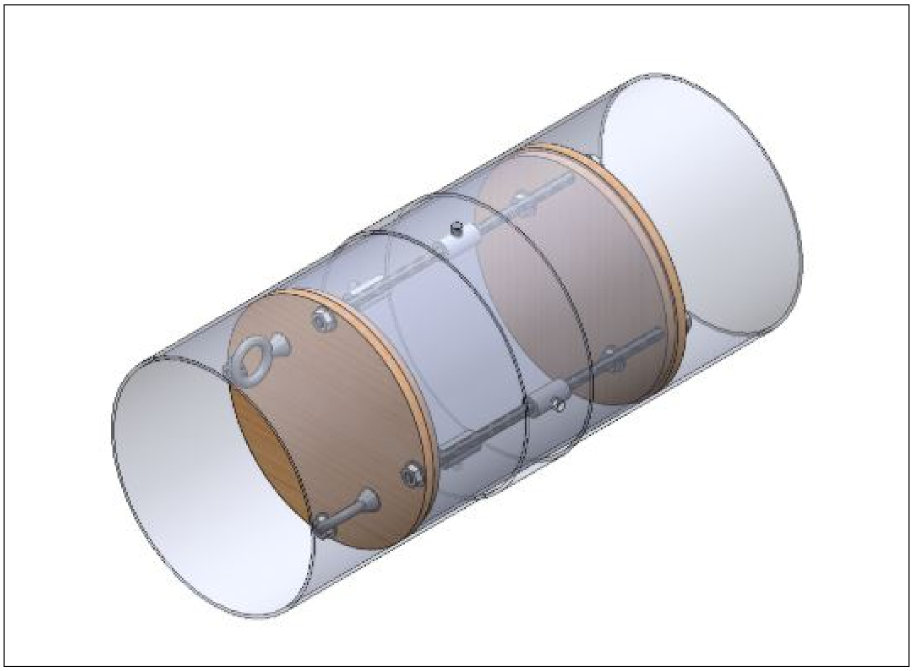

Figure 6: Coupler Generation Three

Generation three is the current general design for the coupler, enabling easier access to the altimeters and Carbon Dioxide (CO2) system, estimated to weigh 3.67 pounds, without electronics installed, with dimensions found in Figure 113. The design involves a larger 15 in. long coupler with a switch band in the center of it. This coupler is designed to allow the fore and aft sections of the launch vehicle to be ejected independently of each other off of the coupler. It also separates the drogue and main parachute compartments, in accordance with the new launch vehicle layout, reducing the chance that there will be tangling on the shock cords during the dual deployment of the recovery.

The center of the coupler houses the main altimeter bay, shown in Figure **??**, which is built as a separate unit from the structural coupler that houses it. It is held in place with four set screws. There is very little shear stress transferred through the set screws since the forces of launch travel through the outer coupler layer and the forces of recovery travel though the center of the coupler, only having to hold the altimeter bay in place, and not the rest of the launch vehicle. Because the coupler is crucial to the structural stability of the airframe during launch and descent, it will be largely made up of carbon fiber. The coupler itself will be laid up inside a section of tube manufactured on the same mandrel as the airframe; this will ensure a perfect fit for the nonstandard tube size. The bulkheads securing each end of the altimeter bay will be made of plywood due to the favorable strength to weight ratio, widespread availability, and ease of manufacture.

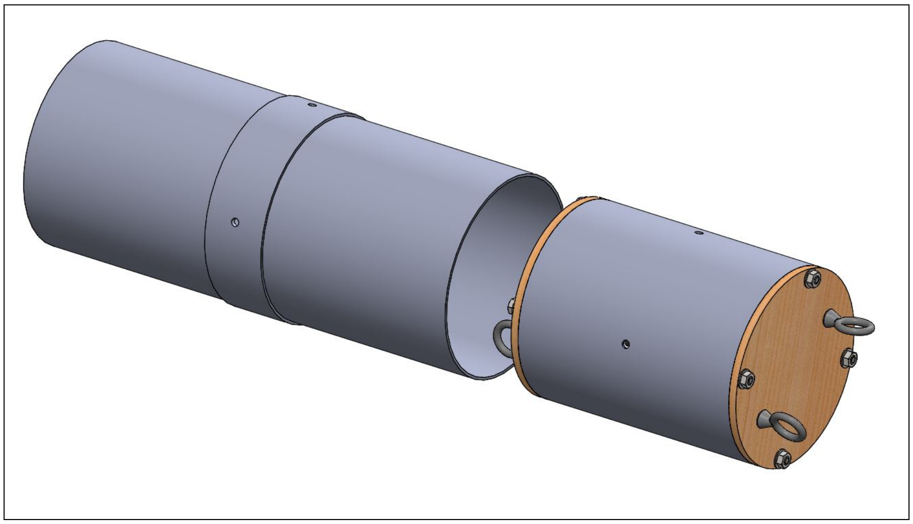

Figure 7: Main Altimeter Bay Placement

This design allows the altimeter bay to be removed and the electronics modified easily without having to reach into a tube. The bulkheads on either side are connected by 4 steel threaded rods that provide tensile strength to the unit for parachute deployment negating shear stresses from an epoxy joint and the outer tube. This structure also provides the linear compressive strength that is required during launch. The threaded rods also provide easy mounting for the battery and altimeter enclosures. Originally designed with six threaded rods, that was reduced to four in the interest of reducing weight after stress analysis determined the number was unnecessary even while maintaining a factor of safety due to the rod materials 150,000 psi of tensile strength. Software simulations will be performed to ensure the factor of safety of the coupler before manufacture and confirmed on simulations from the subscale launch with a scaled version of the coupler. Since the rods are 1/4 in. diameter, the max force they can hold can be calculated and show a factor of safety over the maximum possible calculated recovery forces seen through the shock cords right after main parachute deployment.

$$MaxExpectedStress = \frac{F}{A} = \frac{7321.6}{4 * \frac{\pi}{4}\frac{1}{4}^2} = 37,290psi$$

$$SafetyFactor = \frac{MaxTensileStrength}{MaxExpectedStress} = 4.02$$

This shows that the coupler is more then able to handle the maximum forces at main parachute opening. The maximum forces however will not be sustained through descent either as the launch vehicle slows to its final descent speed which will decrease the stress on the coupler significantly. The compressive launch forces will be passed through the switch band on the coupler, which will be cut from the airframe itself to ensure structural stability and a proper fit.

3.1.5 Aft Section

The aft section of the launch vehicle houses the drogue chute bay, Blade Extending Apogee Variance System (BEAVS) bay, and motor bay. The aft section is responsible for delivering the launch vehicle to the desired apogee, providing both the thrust and increase in drag with the BEAVS system to do so. Like the rest of the launch vehicle sections, the aft section has gone through several iterations before arriving at the current design. One option for the layout included its own independent altimeter bay, which was switched to being located in the coupler to reduce weight and complexity. The motor bay has changed as well, growing from a tube design for a 3 grain 2.95 in. L motor to handling motors up to 27 in. ling 2.95 in. in diameter L motors. This was to accommodate more powerful motors after the launch vehicle weight increase. Figure 8 is a cad model of the current design with dimension in Figure 114; without BEAVS or the motor, the aft sections weight is estimated to be 11.68 Pounds.

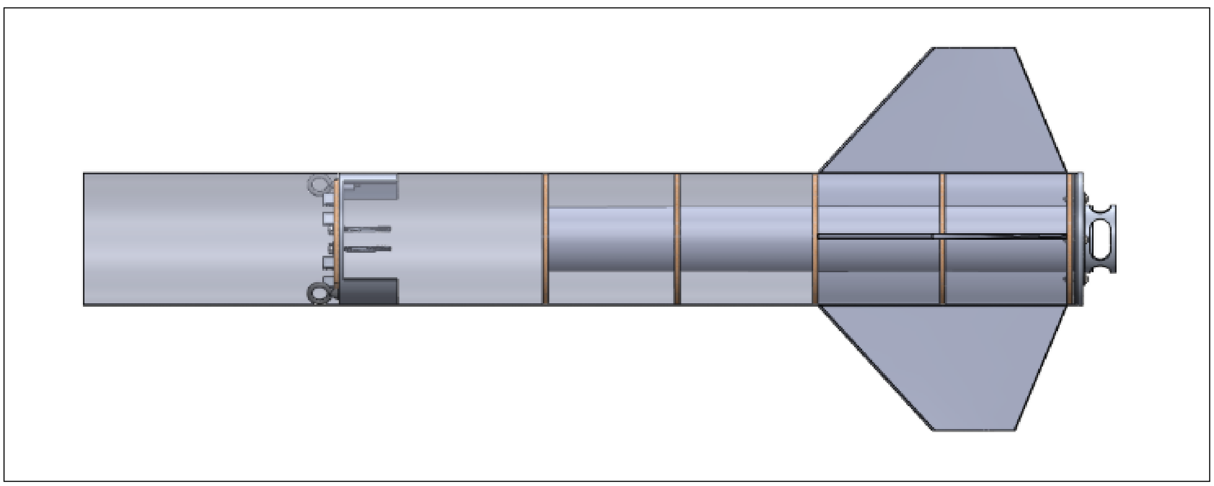

Figure 8: Aft Section

The aft section will be constructed out of filament wound carbon fiber, the same as the fore section, because of the strength needed to handle the forces of motor launch and landing. Each airframe will be manufactured with tapering, increased layering toward one end to add zippering protection from the shock cords deploying from the parachute bays during recovery. Fiberglass was considered but was just as accessible as carbon fiber from OSRT tube provider, so the stronger material was selected. The motor tube will be made of fiberglass as it only needs to keep the motor housed and positioned correctly in the tube and will not be handling much, if any of the thrust forces. The thrust forces will be transferred from the motor to the airframe using a thrust plate, in order to decease the shear stresses in epoxy joints in the motor bay. This design was chosen over having no thrust plate to increase the factor of safety by reducing failure points in the motor bay. The motor retainer and thrust plate, which can be seen below in Figure 9 and Figure 10 respectively, are being manufactured out of 1060 Aluminum by OSRT for cost reduction.

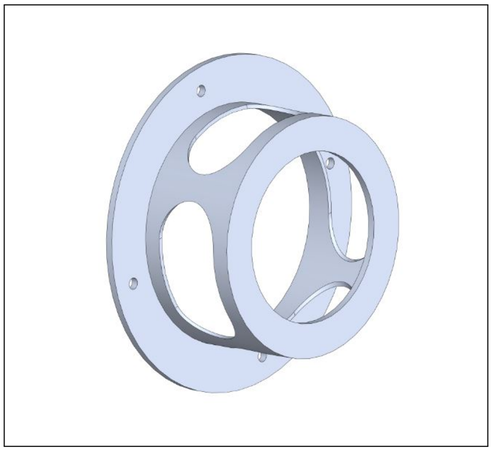

Figure 9: Motor Retainer

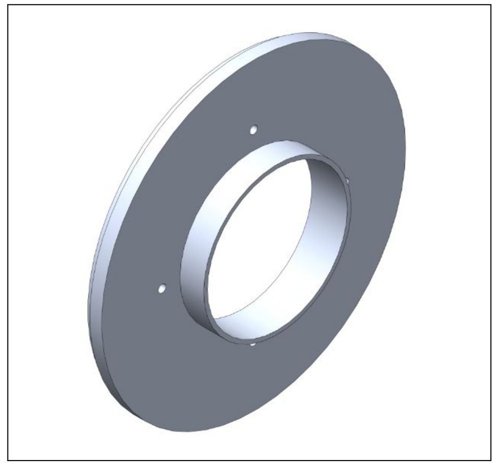

Figure 10: Thrust Plate

The BEAVS system will be mounted behind the aft parachute mount, and be able to be removed to modify as necessary to achieve the desired altitude. This creates the problem of needing solid parachute mounting while having enough room for the BEAVS system to fit through. Options were creating a maintenance hole in the airframe, and having a removable bulkhead that mounts to a solid bulkhead behind the BEAVS bay via threaded rods. The first idea compromised airframe integrity too much, and the second created issues with user friendliness and accessibility. The leading design was to manufacture an aluminum mount pictured in Figure 11 , which could be created smaller and stronger than wooden ones and have enlarged surface area to guarantee adequate bonding between the bulkhead and the airframe for the parachutes to mount to.

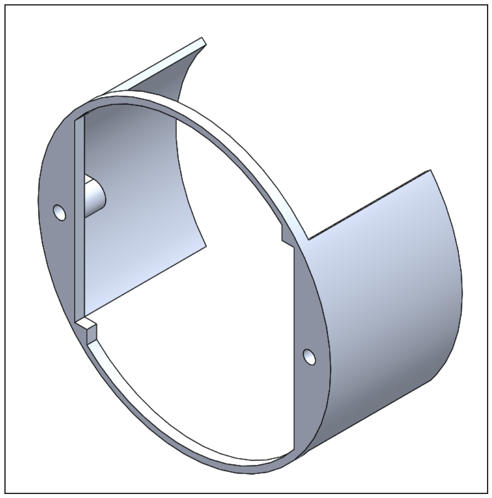

Figure 11: Aft Section

While this part is heavier, it ensures the parachutes will not break away and that the BEAVS system has a mount to seal against enabling the parachute bay to pressurize for ejection.

3.1.6 Motor Selection

The major technical considerations in launch vehicle motor selection were balancing thrust with burn time, and ensuring that the target apogee could be achieved given our current estimated vehicle weight. The project based considerations were availability, and cost. The current selection for OSRT is the AeroTech L2200G.

Thrust VS. Burn Time

Thrust and motor burn time were the main attributes considered during motor research. Major requirements for the motor included thrust load, which could safely be distributed through the vehicle, while also allowing the post-burnout BEAVS system as much time as possible to accurately adjust apogee. For motors with short burn time (<2 seconds) and the necessary impulse needed to lift the vehicle accordingly, there is a significant increase in max thrust of the motor. Thrust forces generated by the motor propulsion need to be distributed through the launch vehicle and thus the launch vehicle must be able to sustain these forces. As such it was determined that the motor should induce loads which would be approximated at an absolute minimum safety factor of 2 for axial loading of the airframe. The axial strength of carbon fiber airframe should safely exceed the maximum thrust induced loading of any L-class rocket motor.

Burn time is a critical factor in motor selection given that the BEAVS system only operates after burnout. For motors with similar total impulse, a longer burn time leads to a greater percentage of the apogee height being covered prior to motor burn out. This reduces the degree to which the BEAVS system will be able to alter and achieve the goal apogee. Due to the aforementioned considerations, an upper limit of 3.5 seconds burn time was established for motor selection.

Table 6: Motor Options

Name	Diameter (in.)	Weight (lbf)	Max. Thrust (lbf)	Burn Time (sec)	Total Impulse (lbf-sec)	Cost ($)
Cesaroni L1395-BS	2.95	9.53	405	3.34	1100	$ 293.00
Cesaroni L1685-SS	2.95	13.3	517	3.03	1140	$ 355.00
AeroTech L2200G	2.95	10.5	697	2.27	1147	$ 280.00

Maximum Vehicle Weight

Given the motor size limitations set forth in the Student Launch Initiative (SLI) competition student handbook (no greater than L-class) and after selecting a fixed airframe size, it was helpful to determine a maximum vehicle weight such that, a feasible motor near maximum L-class impulse, would enable the launch vehicle to reasonably achieve apogee within the competition regulations. Using OpenRocket to test and simulate various engines with our current launch vehicle design, it was determined that a maximum weight of approximately 60 pounds at the launch rail would allow OSRT to reach an estimated 4500 foot apogee. This is considered a safe altitude given that the launch vehicle is not likely to extremely exceed the simulation estimates, and will more than likely under-perform relative to the simulation due to wind resistance, weather-cocking, and other factors. This allows 1000 feet of margin on either side of the predicted apogee to ensure that altitude points are earned. This also enables the apogee to be conservatively estimated in such a way that will allow the apogee variant system to be the determinant of the maximum altitude.

The AeroTech L2200G was selected because it meets each of the major technical components while also benefiting the project considerations as well. With a maximum thrust of lb-sec and total motor burn time of 2.32 seconds, this motor aligns safely within the technical performance requirements. Figure 12 [14] shows the simulated thrust curve of the AeroTech L2200.

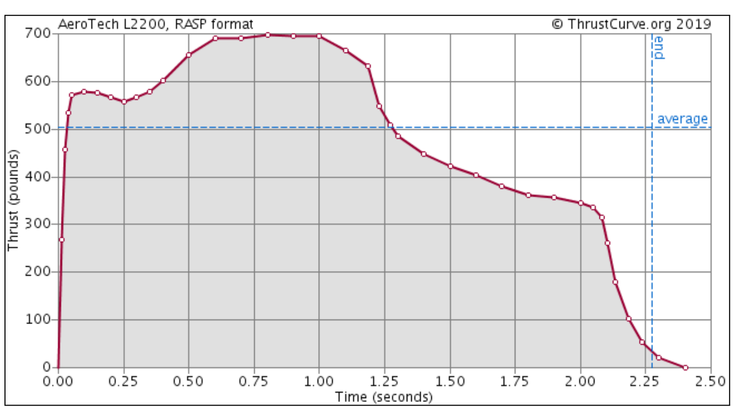

Figure 12: Motor Thrust Curve

Other benefits of selecting the L2200G are cost and availability. This motor propellant reload is cost effective and widely available through various rocketry supply vendors. Selecting a motor which produces total impulse near that of the competition maximum also means that launch vehicle payload and apogee control systems can have freedom to extra space and mass which may help to more successfully complete various other facets of the competition.

3.1.7 Fins

The fins on the launch vehicle have several options and need to adhere to criteria specified by the needs of the launch vehicle and OSRT. They fins need to be able to withstand launch forces; provide support to the motor tube; resist flutter and vibrations; provide stability to the launch vehicle during flight; and survive landing forces. Material options include wood, fiberglass, and carbon fiber.

The main deciding factors of fin shape and the number of fins came down to the need to provide stability while being able to survive landing forces. The fins are a trapezoidal shape with dimensions shown in Figure 114, in order to cut away the back of the fin so that the airframe has a lower chance of landing on a fin and breaking it and a higher chance of landing on the stronger main airframe tube. Because of angle propagation in manufacturing, four fins were chosen as opposed to three fins because manufacturing and fin alignment are crucial for a successful launch vehicle flight. The investigation for this decision is seen in Table 7.

Table 7: Number of Fins Design Decision Matrix (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	4	36	2	18
Weight	5	3	15	4	20
Total		**72**		**66**	

Wood was phased out of the material options due to the need for greater strength at a thinner profile than wood would allow. While Fiberglass is a much better alternative, carbon fiber offers higher strength with a similar manufacturing process. The need to mitigate flutter is relatively small given the speed of the launch vehicle, but strengthening the fins generally requires sandwiching materials, leading to a stiffer and stronger fin. Because of the complexity of composite layups, airfoil fins were determined to be unrealistic to make, causing the design to revert to a thinner flat profile with rounded edges as seen in Figure 13 and the the material types investigated are in table 8.

Table 8: Fin Shape DDM

Design		**Carbon Fiber Wrapped Fiberglass**		**Carbon Fiber**		**Wood**		**Fiberglass**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Strength	8	4	32	5	40	2	16	3	24
Manufacturability	6	1	6	2	12	5	30	3	18
Weight	7	4	28	5	35	2	14	3	21
Total		**66**		**87**		**60**		**63**	

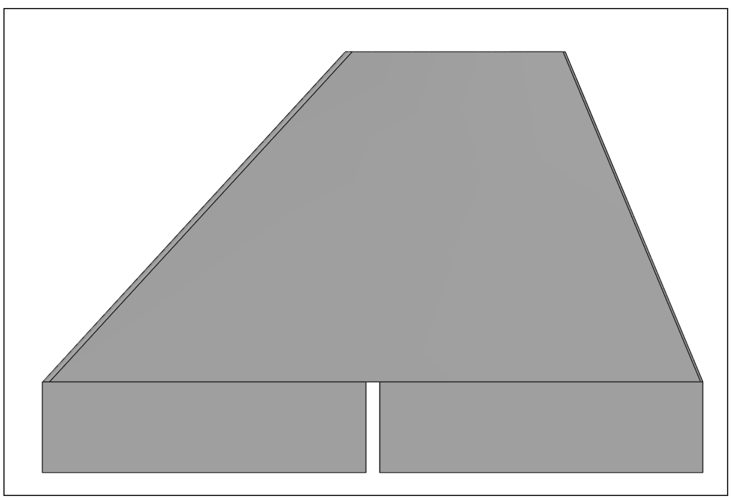

Figure 13: Aft Section

The fins currently are planned to be made out of three layers of carbon fiber sheets. The middle sheet will be cut out in a large honey comb design to reduce weight, with thinner sheets of carbon fiber bonded on either side. This will provide a very lightweight but strong fin, that is suited to the needs of the launch vehicle.

3.1.8 Avionics Bays

Several options for avionics bays were considered, such as a bay in both the fore and aft sections, along along with the nose cone tracking bay for each parachute deployment. This was determined to be unnecessary as all of the avionics could be housed in the coupler, separating the parachute bays. This has the advantage of only needing one bay in the center of the launch vehicle instead of two, decreasing complexity and weight. It also allowed easier access to the bay since it was not located at the bottom of a parachute compartment.

The two avionics bays are now located in the nose cone and in the main coupler of the launch vehicle. Both avionics bays will utilise 3d printing in order to create mounting points for the various altimeters, batteries, and tracking devices in order to reduce weight as much as possible. Batteries will be contained in a separate compartment within the avionics bays, to prevent damage in the case of a battery failure. There will be static port holes positioned such that a team member will be able to activate the avionics with a hex key; the port holes will also double as the venting in order to ensure the altimeters are at the same pressure as the outside atmosphere. The nose cone will house the tracking since it is RF transparent being made of fiberglass, while the couplers avionics bay will only house altimeters.

3.1.9 Subscale Launch Vehicle Design

The subscale launch vehicle will be designed with resembling the full scale as much as possible in mind. It will be built to as near a $^2/_3$ scale as able while still maintaining a reasonable level of cost and construction complexity. The inner diameter of the subscale airframe will be 4 in. instead of 4.17 in. because of the time frame, cost, and availability of material in the proper size. The payload and the BEAVS systems will both be simulated using weights, as subscales of those systems are unrealistic this early in their design stages given time constrains on the project. The subscale launches will incorporate different systems each launch, with the idea being to test each subsystem or component in launches as they are available or manufactured as some components have a much longer manufacture time frame due to training and certification of team members that are to perform the manufacturing.

3.2 Launch Vehicle Avionics

The OSRT Avionics Telemetry Unit (ATU) is designed to fulfill the requirements of tracking the launch vehicle in the air and transmit the data to the base station. In order to fulfill these requirements, the block diagram in Figure 48 was made. This block diagram shows the preliminary parts of fulfilling these requirements.

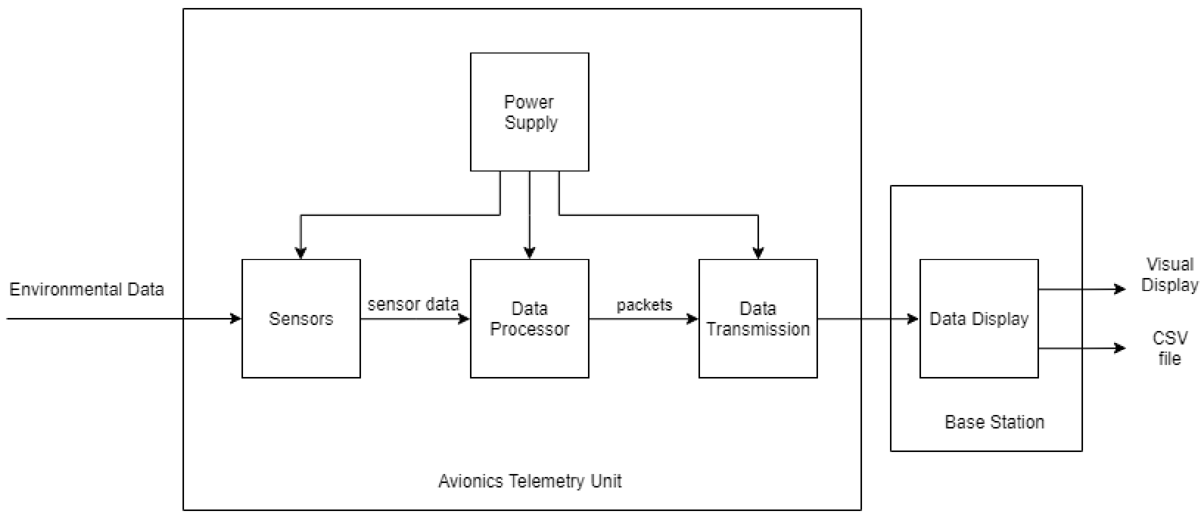

Figure 14: The Block Diagram for the Avionics System

3.2.1 Sensor Block

In order to meet the minimum requirements, location and altitude data must be gathered from the external environment. In addition to being able to track the launch vehicle's location upon landing, the ATU must also track the launch vehicle's route while airborne. As a team, it was decided to include an inertial measurement movement to gain information on the flight path and acceleration of the launch vehicle. This should help gain more information for analyzing the accuracy of predictions and give useful information for later analysis. There are many different possible options for each component, and different configurations of sensors that have been considered for the design.

A block diagram of the system is shown in Figure 49.

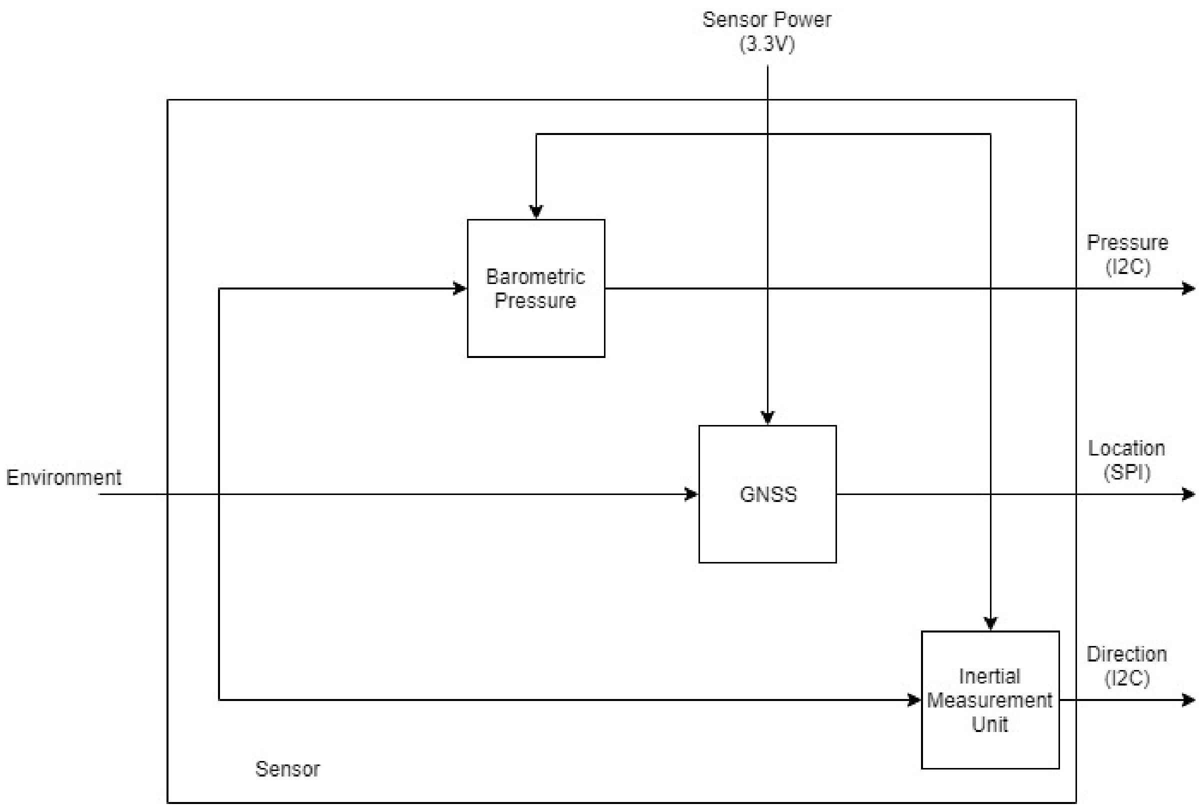

Figure 15: This shows an overview of the sensing operations that will be needed

3.2.1.1 Global Navigation Satellite System (GNSS) Modules

The avionics system will use a Global Navigation Satellite System (GNSS) module to collect location data from satellites. Global GNSS constellations include Global Positioning System (GPS), Global Navigation Satellite System (GLONASS), Galileo, and BeiDou. GPS is the most robust and accurate GNSS. For GPS, the United States (US) government guarantees a global average User Equivalent Range Error (UERE) of less than or equal to 25.6 ft[16]. with 95% probability. As of May 11, 2016, actual performance for GPS shows a global average UERE of less than or equal to 2.3 ft, 95% of the time[16]. GNSS units vary in accuracy, power consumption, and ability to withstand launch forces, which are all factors to compare in determining the best option for the ATU.

GNSS receivers must track at least four satellites to determine position through trilateration. A multi-constellation receiver uses a minimum of five satellites, four from one constellation and a fifth from another. By having access to multiple GNSS systems, the ATU's accuracy and reliability increase through redundancy.

GNSS satellites communicate using signals in the L band, ranging from 1 to 2 GHz. Most GNSS receivers use a single band, but multi-frequency solutions are becoming more prevalent. Using multiple bands provides additional data points from each satellite, increasing position and time accuracy. A significant source of error for GNSS comes from the delay experienced as signals travel through the ionosphere. This delay varies based on the frequency of the signal, so it may be resolved through observing multiple frequencies. Error incurred from environmental obstructions can also be mitigated as demonstrated in Figure 50.

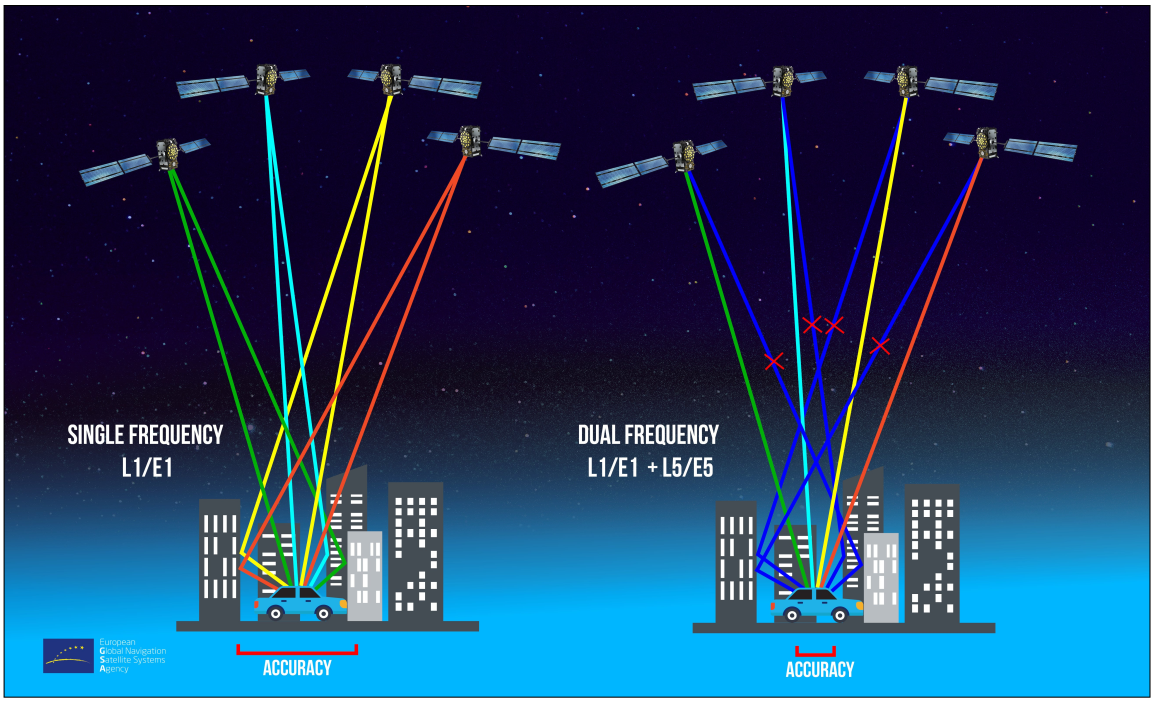

Figure 16: Dual-frequency GNSS obstruction avoidance ©European GNSS Agency

Each GNSS module has different power requirements depending on how it is operating. If a GNSS is completely without power and then turned on (a cold start), it will take longer and require more power to get a fixed signal since it must search for satellites. A warm start, or starting when there is still a small amount of power that has been continuously supplied requires less time and power since it has been able to keep the location of satellites stored.

There are also time and power consumption to consider when the GNSS is operating under normal conditions. These parameters define how the device will operate normally when left on for an extended period of time in optimal conditions. However, the conditions supplied are not optimal during flight, since temperature decreases as the altitude increases and there is significant force being applied during launch, parachute deployment, and landing. These conditions affect the operations of the GNSS units, and are important comparison measurements for making a decision.

A number of GNSS modules were considered. The three primary contenders, GlobalSat EM-506, U-Blox MAX-M8, and U-Blox ZED-F9P, are considered in Table 24.

Design		GlobalSat EM-506		U-Blox MAX-M8		U-Blox ZED-F9P	
Requirement	Weight	Rating	Score	Rating	Score	Rating	Score
Cost	2	3	6	4	8	1	2
Feature Inclusion	5	3	15	4	20	5	25
Position Accuracy	6	3	18	4	24	4	24
Power Consumption	3	3	9	4	12	2	6
Size	3	3	9	4	12	2	6
Total		57		76		63	

Table 9: GNSS Module DDM

After thorough consideration, the U-Blox MAX-M8 proved to be the best choice. The MAX-M8 features multi-constellation support and sports an attractively low price. While the U-Blox ZED-F9P has more features overall, one feature is unsuitable for this competition; that feature is Real-Time Kinetic (RTK) positioning, because using this feature requires sending location corrections to the avionics system from the base station. The ZED-F9P is also much more expensive, consumes more power, and is larger than its competitors.

3.2.1.2 Barometric Pressure

Barometric pressure is the best way to measure altitude; barometric pressure does change depending on weather conditions. Since barometric pressure is not always completely accurate, it becomes a design challenge on how to best minimize the error in data readings. This could be done by collecting more information in addition to the barometric pressure, or by taking into account the temperature at sea level. By taking temperature measurements progressively during flight, the altitude measurements can be more accurately calibrated. There are other factors that change the pressure at different altitudes, such as humidity. These are more difficult to measure accurately within the confined space of the launch vehicle.

3.2.1.3 Inertial Measurement Unit (IMU)

The Inertial Measurement Unit (IMU) should measure the forces on the device in and during flight. It will allow the development of a more robust data profile of the flight and enable the Graphical User Interface (GUI) to be more comprehensive. The IMU will act as an acceleration, magnetism, and orientation. Doing so eliminates the need of having an accelerometer, magnetometer, and gyroscope as separate sensors.

3.2.2 Data Processing

The data processing will be divided between on-board processing and external processing. The on-board processing will need to convert the signals from analog or digital signals into number packets that can be transmitted over the corresponding transceiver. The data it transmits will additionally be logged onto an on-board microSecure Digital (SD) card.

For the microcontroller, several options were considered. The top three are a Teensy 3.6, Cortex M4 TM4C123G by Texas Instruments (TI), and ESP32. These are compared in Table 25. For these the Teensy 3.6 was considered the baseline to compare the other microcontrollers since that was used in previous years, so it received 3's in all areas.

The criteria for comparison were:

- Memory - Sensors and data processing consume a fair amount of memory especially when considering processing data from three sensors and turning it into packets.
- Quality of Documentation - If the documentation is lacking it would cause unforeseen issues, drastically increasing time.
- Number of Input/Output (IO) - Since several sensors will be used, they all need to have an input source.
- Estimated Development Time - Since the system should be working in April, it is important that the project can be done in reasonable time.
- Size - Since the ATU must fit in a confined space; the microcontroller must leave room for the other devices.
- Power Consumption - The system must last while on the launch bay, while in flight, and while it is recovered.

Design		Teensy 3.6		Cortex M4 TM4C123		ESP32	
Requirement	Weight	Rating	Score	Rating	Score	Rating	Score
Memory	6	3	18	3	18	4	24
Quality of Documentation	5	3	15	4	20	2	10
Number of IO	6	3	18	2	12	1	6
Estimated Development Time	3	3	9	2	6	3	9
Size	3	3	9	5	15	4	12
Power Consumption	3	3	9	3	9	3	9
Total		78		80		70	

Table 10: Microcontroller DDM

It was decided to use a Teensy 3.6 and an MC4 as prototypes. They will be used for preliminary tests. Teensy 3.6 will be used for ease of development for the first subscale, and the MC4 will be tested in the following subscale.

3.2.3 Data Transmission

The ATU must be capable of transmitting sensor data to the base station wirelessly. Low-Power Wide-Area Network (LPWAN), WiFi, and cellular networks are three possible ways of transmitting and receiving data.

Popular with Internet of Things (IoT) applications, LPWAN allows for wireless communication with low power consumption. Long Range (LoRa) transceivers use LPWAN and offer great flexibility. LoRa's communication protocol can use a low-level, Point-to-Point Protocol (PPP) connection. The benefit of this is less overhead and the ability to define custom packets. Some reasons why custom packets may be desired are receipt acknowledgement and reducing interference impact. Its packets are limited to 255 bytes, but its power consumption is low. LoRa operates on lower frequencies, 915 MHz in North America, compared to 2.4 GHz or 5.8 GHz used by WiFi. By using lower frequencies, its range coverage is superior.

Pros:

- Long range
- Low power
- Receipt acknowledgement
- Reduced interference impact

Cons:

- Small packet size limit

Alternatively, the same LoRa transceivers are capable of using the LoRaWAN® network protocol, which expands LPWAN to using the internet as an intermediary to send data packets; however, this requires connecting to a network gateway. LoRaWAN® is not as reliable as LoRa because it does not guarantee that messages are received.

WiFi is well-suited for high bandwidth applications, such as video streaming, and less practical for small data packets. The disadvantage WiFi has is short range. Cellular networks can handle high speeds and long ranges, but the critical concern becomes coverage. Additionally, using 4G consumes power faster than WiFi.

3.2.4 Firmware

Many over the counter microcontrollers including the Teensy 3.6, allow for the rapid development of reliable firmware and software prototyping within the open source Arduino Integrated Development Environment (IDE). The Arduino IDE was chosen because it is compatible with a wide range of microcontrollers and simplifies the complexities of interfacing with hardware components within the avionics system. The Arduino IDE gives the flexibility of developing firmware using a variety of languages due to its vast library support; C, C++, and Java are only a few of many languages that can be utilized throughout the lifespan of the project, in conjunction with the Arduino IDE.

The avionics firmware will need to be able to aggregate, process, and update hardware data as quickly and reliably as possible. Several languages were considered for the development of the avionics firmware, however it was a clear choice to develop in C. The C language is fast, optimized, and portable when compared to more modern and bulky languages; it contains features from both low and high level programming languages, allowing for fast development and prototyping with both hardware and software. The language allows for quick, low-level access to memory; this is crucial for continuously updating avionics components and the software interface. Languages such as Python and Java were considered for the avionics firmware, however implementation of these languages would be more challenging and time consuming than utilizing C.

Moreover, other languages will be considered if C cannot perform within certain time and design specifications. At minimum, the firmware must update both the software and hardware peripherals ten times a second—this minimum standard is set to ensure that an accurate trajectory is followed, while providing the software with precise data artifacts for analysis. If the C language is simply incompatible with specific hardware peripherals or design, alternative languages will be implemented.

Regardless of which language is used, the design process of the avionics firmware will heavily focus on analyzing the time complexities of the code. Doing so will reduce the memory needed to store the code and decrease the amount of time needed to process input data from hardware peripherals.

3.2.5 Software

The avionics base station GUI will display transmitted data sent from the ATU on a GUI. Because the avionics base station GUI will not send data back to the ATU, control systems for this GUI are not needed; priority is placed on data visualization. The GUI will include a map plot to display the launch vehicle's location for recovery. Other graphical elements the OSRT are looking to include are flight instruments that can represent the data from the sensors in a user-friendly way. The base station will save received packets to check against the log saved on-board the ATU's microSD card for packet validation after launch.

The avionics GUI will be written in Python, resulting in a portable application that does not require installation. While C# is a popular language for GUI development, it is less compatible with operating system platforms compared to Python.

3.2.6 Schematics

From these design considerations, preliminary schematics were developed. In Figure 51, the connections selected were Inter-Integrated Circuit (I2C). The power supply was selected to supply the power to the Teensy 3.6. The power regulators on the Teensy 3.6 are being used to supply power to the sensors that require 3.3 V.

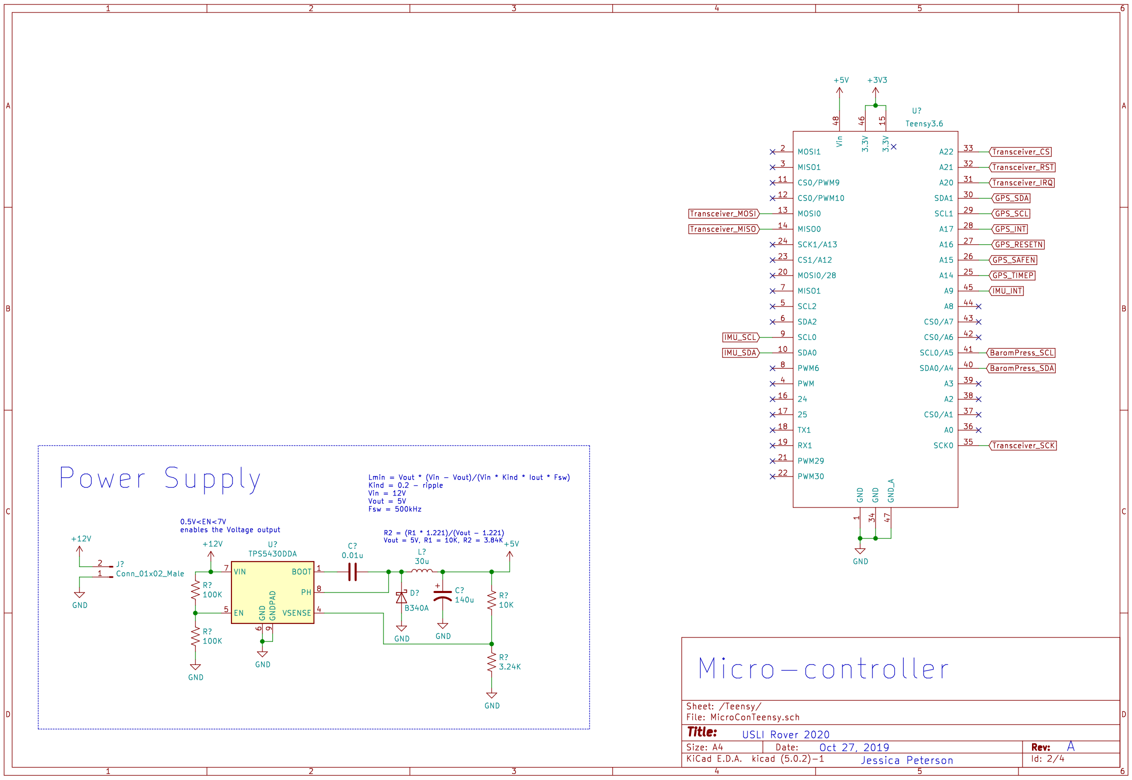

Figure 17: Avionics microcontroller and power supply

The sensors selected for the ATU, are displayed with their connections in Figure 52. These sensors are supplied with 3.3 V of power. These sensors were chosen to provide the best information to the GUI. They contain 22 ohm resistors on several of the lines to improve signal integrity, and protect from Electrostatic Discharge (ESD).

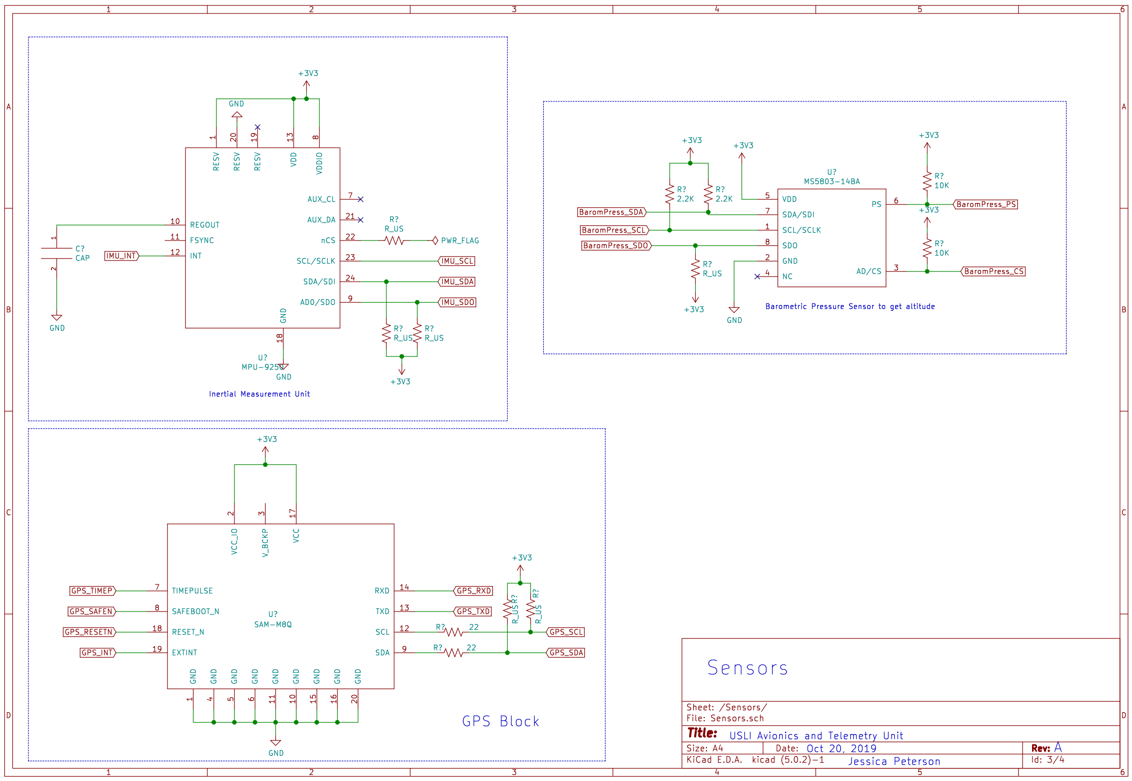

Figure 18: Avionics Sensors

The transceiver system schematic using the LoRa transceiver is shown in Figure 53. This demonstrates the system used to communicate on hardware. A capacitor is placed between ground and power to filter out noise from the power supply and keep the RF signals as isolated as possible.

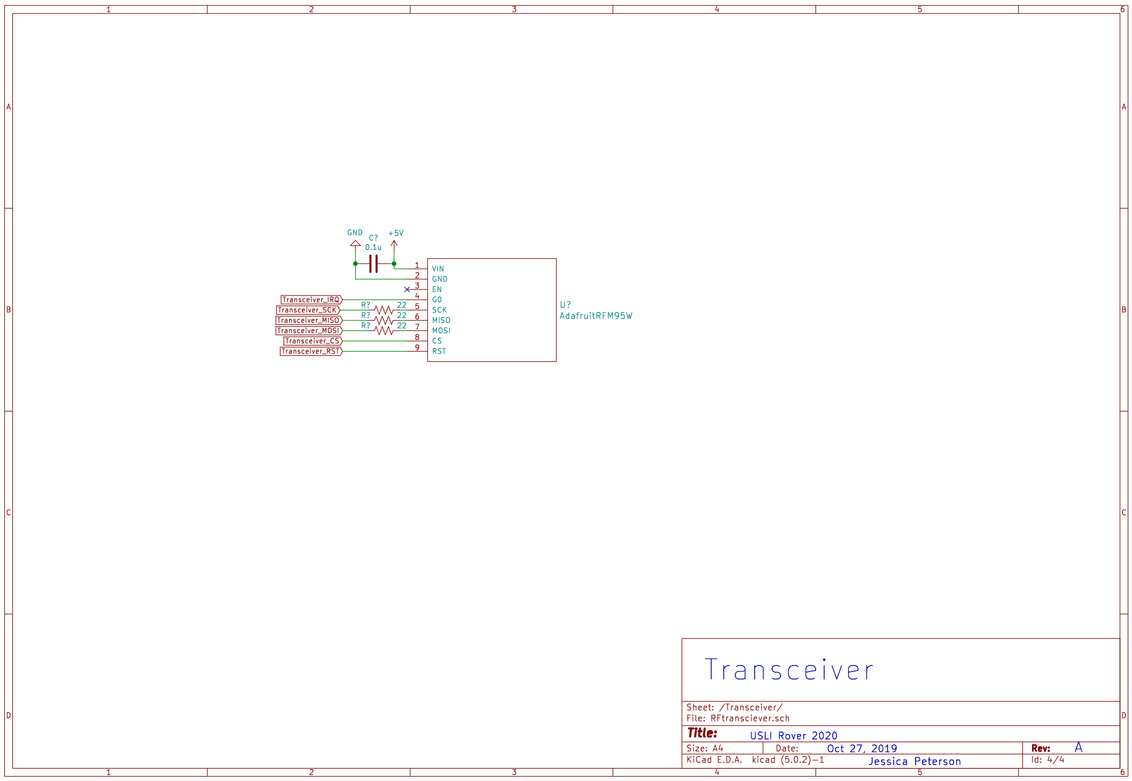

Figure 19: Avionics Transmission

3.3 Recovery Subsystem

3.3.1 Component Analysis

3.3.1.1 Recovery Section Selection

The OSRT explored two recovery system styles/layouts for the 2019-2020 launch vehicle: The first option being a multiple section, single compartment, system and the second option being a dual-compartment, dual-recovery system.

The multiple section system would require the fore and aft sections of the launch vehicle to split, and each section would contain its own recovery system. This is unfavorable because each section must have a drogue, main parachute, avionics, and tracking devices. This design option is inefficient because it requires compact parachute packing and unreliable deployment due to multiple moving parts in one small compartment. The benefit of this system is that the parachutes can be smaller for each recovery section due to there being less weight for each parachute to hold. The lower weight also decreases vehicle drift, which allows for all of the sections to land within the NASA USLI guidelines provided.

A single section recovery system would involve the entire launch vehicle to be tethered together, descending by means of one large main parachute, and one drogue. The single compartment section is heavier, therefore requiring a larger parachute to meet the kinetic energy landing requirements. With a larger parachute, the packing density becomes a larger concern as it requires more power to eject. This design, however, is less complex and more reliable than the multiple recovery section. Being tethered together, there is only need for one compartment for avionics, a single tracking device in the launch vehicle, a single drogue, a single main parachute, and fewer ejection systems. The engineering specifications in Table 11 were used to help guide the decision in OSRT's recovery system.

- Landing Kinetic Energy: Dependent upon parachute size and weight
- Recovery Radius: How far the launch vehicle drifts from its launch position
- Ease of integration (recovery): How easy the system is to integrate with the recovery system
- Ease of integration (payload): How easy the system is to integrate with the payload system
- Ease of integration (structures): Effort required to integrate the system with the structures system

Table 11: Number of Recovery Sections

Design		Two Sections		One Section	
Requirement	Weight	Rating	Score	Rating	Score
Landing Kinetic Energy	3	2	6	4	12
Recovery Radius	2	4	8	2	4
Ease of integration Recovery	2	1	2	3	6
Ease of integration Payload	1	2	2	2	2
Ease of integration Structures	2	2	4	3	6
Total		22		30	

From this data in Table 11 it was determined that a single section recovery system would be the most practical. This system is overall less complex than a multi-section system and has less failure points. The recovery radius will be discussed further below.

3.3.2 Parachute Canopy

The shape of the parachute, known as the parachute canopy, is chosen based on desired coefficient of drag. There are numerous canopies for varying types of parachute descent rates and uses. The parachutes that were under consideration during the decision process were: Toroidal, Elliptical, Cruciform, Helicopter style, and Flat sheet.

A Toroidal parachute is a complex parachute with a spill hole in the middle of the canopy and shroud lines connecting on the spill hole and the perimeter of the chute[5]. This design works differently than any other parachute by flattening the canopy and as a result increasing the drag coefficient to 2.2. While this higher drag means it can handle more weight, it also is more expensive and prone to tangling with its numerous shroud lines. A Toroidal chute is shown in Figure 20.

Figure 20: Toroidal Parachute

An Elliptical parachute is similar to a Toroidal parachute, except it does not have the shroud lines connecting to the spill hole[11]. Due to these missing lines, the parachute is more rounded and has a lower drag coefficient of 1.6. This types of chute come in all sizes and is fairly cost effective. An Elliptical parachute is shown in Figure 21.

Figure 21: Elliptical Parachute

A Cruciform parachute is shaped much like a square crown with holes in the corners[2]. It has a drag coefficient ranging from 1.10-1.3. These parachutes have very little shroud lines and can be packed quite compact. However, they are more expensive and more difficult to find. A cruciform parachute is shown in Figure 22.

Figure 22: Cruciform Parachute

The Helicopter-style parachute is an extremely complex parachute consisting of 4 small "canopies" all connected to a singular point[19]. The drag coefficient on this parachute is dependent upon its overall surface area. The calculated drag based off OSRT's launch vehicles requirements was 1.8. This type of parachute can be packed smaller than most, but has the downside of becoming tangled when released at altitude. A helicopter style parachute is shown in Figure 23.

Figure 23: Helicopter Parachute

Lastly there is the Flat sheet style parachute, usually these types of parachutes are handmade out of a flat sheet of material. These parachutes are extremely cheap, but not reliable at high speeds or weights. The typical drag coefficient seen is 0.7. The parachute is shown in Figure 24.

Figure 24: Flat Parachute

For these parachutes the engineering requirements are the following:

- Cost: Reduce cost
- Coefficient of Drag: As the coefficient increases, the size of parachute that can be used decreases
- Risk of Entanglement: The parachute should have resistance to entanglement
- Performance: The parachute descent compared to its weighted descent

Table 12: Parachute DDM

Design		Toroidal		Elliptical		Cruciform		Helicopter		Flat	
Requirement	Weight	Rating	Score	Rating	Score	Rating	Score	Rating	Score	rating	score
Cost	1	2	2	3	3	3	3	4	4	5	5
Coefficient of Drag	4	5	20	3	12	3	12	4	16	2	8
Risk of Entanglement	2	1	2	1	2	4	8	2	4	5	10
Performance	3	5	15	4	12	3	9	3	9	2	6
Total		39		29		32		33		29	

From the results in Table 12, a Toroidal parachute will be selected as the main deployment parachute in the launch vehicle. The high performance and coefficient of drag will make it the ideal main parachute for a lower altitude deployment. While this parachute has a higher risk of entanglement, the parachute will be located and deployed from a deployment bag outside of the vehicle, and a swivel will be utilized to prevent tangling during the recovery phase. Thus, the risk of entanglement can be combated.

The cruciform parachute will be best used as a drogue parachute. Its desirable performance at high speeds will allow it to deploy at apogee and slow the vehicle enough for the main parachute to deploy with a safe level of impulse to the vehicle.

3.3.3 Shock Cord Material

The shock cord is the cord that connects the sections of the vehicle together as it descends. It is also the material that connects the parachutes to the bulkheads. This cord must be able to withstand all recovery forces that will act upon it, including the impulse of the parachute deployment, and holding the vehicle together as it descends. There are three main types of shock cord material available that were considered for the recovery system. They include nylon, Kevlar, and spectra. Each have their own benefits as well as adverse effects. Nylon has great elasticity, which reduces shock load, but has poor thermal resistance and can be burned easily. Kevlar is extremely strong and durable against heat and high forces, but can zipper and damage the vehicle upon descent. Lastly, spectra is stronger than all the materials, but it has a higher chance of zippering, is weak to heat, and has low elasticity.[12]

Listed are the engineering requirements used to compare these materials:

- Elasticity: The higher the elasticity, the lower the initial shock force is
- Strength: The higher the strength of a material, the more force it can take
- Thermal Resistance: The more Resistance to heat a cord can take, the less packing is needed
- Material texture: A softer material is less likely to damage the airframe

Table 13: Shock Cord Materials

Design		Nylon		Kevlar		Spectra	
Requirement	Weight	Rating	Score	Rating	Score	Rating	Score
Elasticity	4	4	16	2	6	2	6
Strength	3	3	9	4	12	5	15
Thermal Resistance	1	2	2	3	3	2	2
Material Texture	2	4	8	2	4	3	6
Total		35		25		29	

Table 13 concludes that a nylon shock cord material will be used to tether the vehicle together and connect all of the parachutes. The nylon is less likely to zipper and damage the vehicle. Nylon also has higher elasticity to reduce the shock force as the parachute deploys. The nylon does have a low heat resistance, but that can be mitigated using Nomex blankets and Kevlar sleeves to reduce the heat exposure, where excessive heat is expected to be experienced on the material. Furthermore, after consultation with Oregon

Rocketry (OROC) and OSRT, the main parachute shock cord will be 2x the length of the body, and 5x the length for the drogue parachute cord.

3.3.4 Packing Method

A packing method is described as a way of folding or preparing a parachute for its storage and deployment before a launch. Different methods have different rates of success and deployment. Choosing a reliable and repeatable packing method is crucial to mission success as it is a high point of failure in the recovery system.

The two main methods seen in high powered rocketry are (1) deployment bags and (2) the fold and wrap.

The first method of the deployment bag uses a fire resistant bag for the parachute to fold into. The bridal cord and shock cord wrap in between the elastic allowing for less susceptibility to tangling and a higher chance of deployment. The packing density on the deployment bags is around 0.2 oz/cu [4]. Overall, these deployment bags have a higher rate of inflating and extracting from the vehicle after ejection.

The second method, the fold and wrap does not use a bag for protection. This method consists of folding the parachute and then wrapping the bridal and shock cords loosely around the folded chute. Then, a Nomex blanket is wrapped around the parachute and usually packing material (such as dog barf) is added to avoid melting or burning. This method, while not as reliable, has a slightly better packing density at 0.13 oz/cu [6]. This method also has a higher probability of tangling after ejection from the vehicle.

The engineering specifications used to determine these methods are the following:

- Packing density: The smaller the packing density the more parachute can fit in the vehicle
- Reliability: Will the parachute deploy more than 7 out of 10 times?
- Time to pack/simplicity: How long or complex is this packing method?

Table 14: Packing Methods DDM

Method		Deployment bag		Fold and Wrap	
Requirement	Weight	Rating	Score	Rating	Score
Time to pack	2	3	6	4	8
Reliability	4	4	16	2	8
Packing density	4	3	12	4	16
Total		34		32	

From the engineering specifications in Table 14, a deployment bag will be used as packing method for the main parachute. The current design allows for leeway in the airframe allowing for the larger packing density of the deployment bag. The bag is also useful as it acts as a heat resistant blanket to protect the parachute from melting and burning.

3.3.5 Redundancy and Retention

Redundancy and retention will exist in the system primarily with the use of tender descenders, e-matches, quick links, and multiple ejection charges.

The drogue parachute will be housed in the forward-most location of the aft section. Once released at apogee, it will apply load to the shear pins holding the coupler. If these shear pins break from the initial release then the main parachute will be pulled out at apogee. To avoid this, tender descenders will be used to retain the parachute inside the housing. This will involve two quick links in series with two tender descenders, and then two more quick links to retain the parachute inside the bay.

For parachute deployment, the redundancy portion revolves around using a primary ejection charge of CO2, with a back up black powder charge at 1.5x the needed amount. This allows for a second shot at deployment and an overall higher success rate as far as parachute deployments. Safety is OSRT's top priority. Having a launch vehicle come down ballistic is unacceptable by the teams standards and will be avoided at all costs.

3.3.6 Load and Impulse Reduction Methods

A higher load and impulse can result in a higher chance of zippering. Zippering occures when the shock cord cuts into and damages part of the airframe.

There are a few methods to reduce zippering and overall load to the walls of the launch vehicle. They include:

- The tennis ball method: Adding a tennis ball or soft material between the shock cord and the area where the shock cord touches the airframe.
- Artificial Zipper: Using tape and weak material to hold together the shock cord until parachute deployment.
- Multiple connection points: Splitting the shock cord and attachments so the entire load is not felt in one place.

The first tennis ball method is mentioned in the article, Peak Flight: How to avoid zippering body tubes[8]. This article mentions the use of a tennis ball or soft malleable material to reduce shock and avoid zippering as it will not cut through the airframe. Unfortunately, the tennis ball or soft material may not have the strength to hold up to the forces felt from the parachutes and would easily break. Furthermore, the ball or foam placement would take up valuable room.

The second method, also the most feasible, uses information from, Aspire Space Technical papers: Parachute recovery system for large launch vehicles[21]. The article states that using a simple thread and sewing parts of the shock cord together will reduce load when these threads rip and keep the launch vehicle from zippering. An alternative, that is readily available and requires less prep time, uses masking tape to hold together the shock cord to reduce the overall forces. In order to successfully enact this, the masking tape must make the shock cord into a Z-shape and when the parachute is released, these will break, but will help to reduce the load and velocity on the bulkhead.

The last method, splitting the shock cord to different connection points, uses information from Peak Flight: Make your own anti-Zipper harness [8]. This article goes into depth about how splitting up the shock cord into different connections will reduce the overall load felt upon the vehicle as it descends. The method involves adding more connections at the bulkhead of the vehicle to reduce the chance of the shock cord sliding and zippering into the wall. However, this design poses the problem of whipping the launch vehicle straight up, as the fixed points make the cord more rigid and more likely to break or fail.

The following engineering requirements were used to determine which system to use:

- Space: How much space does this system take up?
- Force reduction: A quantifiable scale to reduce forces and overall zippering
- Ease of integration: How long and how easy is this to prepare on launch day?

Table 15: Methods to Reduce Zippering

Design		Tennis ball		Artificial Zipper		Multiple attachment points	
Requirement	Weight	Rating	Score	Rating	Score	Rating	Score
Space	3	4	12	3	9	3	9
Force reduction	4	4	16	4	16	3	12
Ease of integration	3	2	6	5	15	4	12
Total		34		40		33	

Using Table 15 and the engineering requirements, the team has decided to move forward with an artificial zipper method. This uses Z-formation layering of the shock cord secured with masking tape to reduce the initial forces felt as the parachutes unfurl.

3.3.7 System Layout

The recovery of the launch vehicle starts at apogee. When the altimeter detects apogee, the fore coupler and aft section will decouple and release a drogue parachute. The drogue parachute is held in with a shock cord and ejected with a /glsCO2 charge. At 525 ft Above Ground Level (AGL), the fore section will decouple from the coupler and the main parachute will release from its bay. The main parachute is retained and held in place with tender descenders and quick links in series. The vehicle will still be tethered together from fore to coupler to aft section as it descends to the ground. This is the setup of the dual-deployment, dual-compartment method. The coupler will be tethered between the fore and aft section, containing the altimeters, electronics, and e-matches that will control the ejection and descent of the vehicle.

Figure 25 shows a modeled depiction of the current parachute and recovery system that will retain and eject the drogue parachute in the fore section. Starting on the right side of the picture, the system starts with an aluminum bulkhead with two eye bolts that are attached with Nyloc nuts. The system then uses three quick links, right next to CO_2 charges, and attached to the shock cord which tethers the coupler to the main body. About 3/4 of the way to the coupler, a butterfly knot is used to hold the second shock cord, which then attaches to the drogue parachute. The drogue parachute is attached to the shock cord via a swivel (unpictured). This parachute is packed in the body tube with a Nomex blanket to protect it from heat and friction when decoupling occurs.

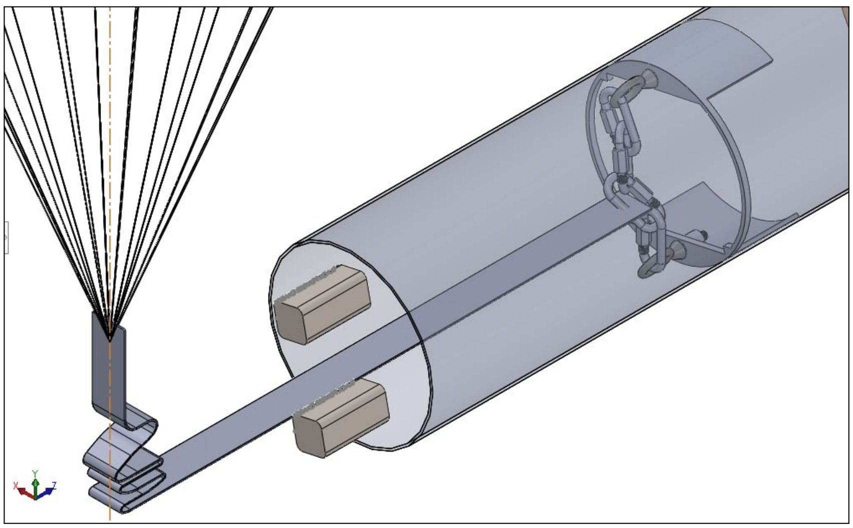

Figure 25: Fore Recovery Drogue Parachute

Figure 26 shows the layout of the aft recovery section containing the main parachute. The system begins on the right with two eye bolts attached with Nyloc nuts to the bulkhead as an anchor. The system also uses three quick links attached to the bolts to attach the shock cord. There are two quick links in series and two tender descenders in series also attached to the eye bolt. These tender descenders are then attached to the parachute deployment bag (unpictured) and retained from ejecting prematurely. There is then a charge, a Nomex blanket, and a parachute bag directly above the eye bolts that will eject the parachute at appropriate altitude. The parachute is attached 3/4 of the way down the tethered cord to the coupler. The main parachute will be held inside the deployment bag and attached to the shock cord with a swivel(unpictured).

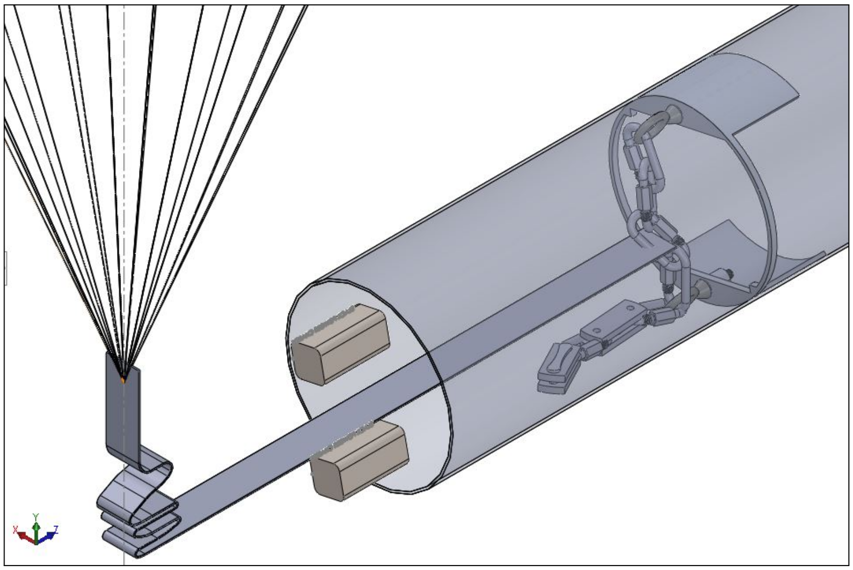

Figure 26: Main Parachute Recovery Layout

3.3.8 Parachute Sizing and Kinetic Energy

The initial weight of the vehicle will be 60 lbf, however the heaviest tethered section will be the fore section loaded with the payload at 20.75 lbf. Once the fuel is burned and the vehicle reaches apogee, the launch vehicle will seperate and the drogue parachute will deploy. The speeds of each section of the vehicle were calculated to ensure that each section of the vehicle lands below the 75 ft-lbf of impact energy. In Equation 1, KE is kinetic energy in ft-lbf, m is in slugs, and v is in ft/s.

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

From the NASA website [3], drag can be calculated using Equation 2. In this equation, D is drag of the parachute in lbf, C_d is the coefficient of drag on the parachute dependent upon its shape. These calculations assume the use of Toroidal parachutes for the main and cruciform chutes for the drogue. ρ_{air} is the density of the air in Huntsville, Alabama, v is the velocity in ft/s, A_r is the area of the parachute canopy in ft^2, and W_{lv} is the weight of the launch vehicle in lbf.

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

Expanding Equation 1 into Equation 2 results in

$$D = W_{lv} = \frac{1}{2}C_d\rho_{air}v^2A_r \tag{3}$$

A_r can be calculated from Equation 4

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

For a Toroidal parachute, there is an inner and outer diameter defined. Using the Fruity Chutes website, the ratio between the two is 5:1 [5]. Equation 5 the simplified equation of area with this ratio.

$$A_r = \frac{6}{25}\pi d_o^2 \tag{5}$$

Combining Equation 3 with Equation 5 results in Equation 6

$$d_o = \sqrt{\frac{25W_{lv}}{3\pi C_d \rho_{air} v^2}} \tag{6}$$

With the current weight, the parachute diameter required for the launch vehicle is 12 ft. This will result in the descent speeds of vehicle being 14.7 ft/s. The impact force of the main vehicle is 63.1 ft-lbf. The drogue chutes from the below calculations result in 4 ft diameter parachutes. With a coefficient of drag of 0.9, the speed from the drogue is 66.5 ft/s.

Using the above equations and MATLAB code the impact speeds and kinetic impact energies found are:

Table 16: Vehicle Descent Rates and Energies

Measurements	Tethered body sections
Dry Weight	49.2 lbf
Max Impact Velocity w/ Parachutes	9.69 ft/s
Max Impact Energy w/ Parachutes	60.45ft lbf

3.3.9 Descent Time and Drift Calculations

A OpenRocket simulation was used to calculate the descent time of the launch vehicle under a 12 ft main parachute and a single 4 ft drogue parachute. This was plotted 90 s max time that the USLI handbook laid out.

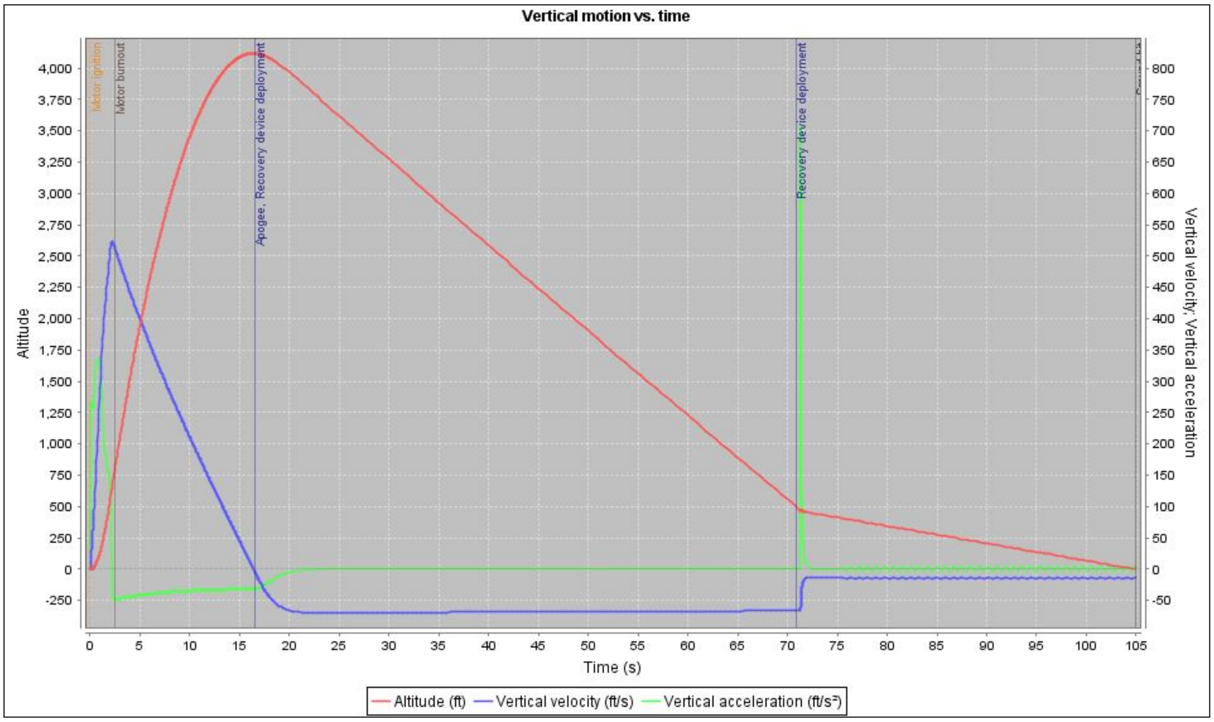

Figure 27: Flight Simulation

The size of the main parachute for the 60 lbf launch vehicle is 12 ft. The drogue parachute is a 4 ft elliptical. Using these parachutes and another script in OpenRocket, the drift from launch was estimated. The wind speed was variable from 0 mph to 20 mph using a crosswind simulation. It was assumed that apogee occurred directly over the launch pad. This is a conservative estimate as the launch vehicle usually drifts from the launch pad after motor burn.

Table 17: Drift Calculations

Wind Speed (mph)	0	5	10	15	20	Descent time (s)
Drift MATLAB (ft)	0	459	1100	1682	2100	63.2
Drift OpenRocket (ft)	6.5	412.5	875	1250	1750	88.8

The difference in these calculations is likely due to the fact that the launch vehicle is tethered together, but can only be simulated in MATLAB as either one body piece, or three separate members with separate parachutes. Both scenarios were tested in MATLAB, but ultimately the launch vehicle simulated as one body under one drogue and one main resulted in closer estimates to the OpenRocket simulation. OSRT decided that OpenRocket is a more reliable source in the context of drift calculations.

3.4 CO2 Parachute Ejection Charges

Historically, OSRT has used Black Powder (BP) ejection charges to deploy the parachutes of the fore and aft sections of the launch vehicle at their respective altitudes. While this section of the recovery system has worked reliably over the last two years, BP is an inherently dangerous substance that also has a large set of challenges, both technical and safety-related, to overcome. On the technical side, BP needs oxygen to fully burn, which, at higher altitudes, can be problematic, as the oxygen content decreases at higher altitudes. If the BP charge does not fully burn, it may not be able to deploy the parachutes at the needed times during flight, resulting in the launch vehicle going ballistic and potentially destroying itself upon impact with the ground, or, even worse, injuring a bystander. This issue of the BP charges needing oxygen can be mitigated by ensuring that the area in which the charge will be contained is air-tight once integrated into the launch vehicle, and therefore, will not be venting oxygen to the thinner atmosphere. Having an air-tight seal is also extremely important because BP, once ignited, will leave behind a corrosive residue that can be extremely dangerous to sensitive electronics, so it is critical that the BP charge is isolated from the electronics that it is connected to.

Continuing the technical perspective, BP absorbs around 1.5% of its weight in moisture when exposed to 75% relative humidity at 75 degrees Fahrenheit over 24 hours, which results in slower burn rates for the BP [24]. This slower burn rate can hinder the success of the recovery system because slower burn rates take a little longer to reach the peak pressure, meaning that the parachutes might deploy later than expected. Given that the average humidity in Alabama in April is 70%, and the average high and lows for temperatures are 72.6 degrees Fahrenheit and 49.3 degrees Fahrenheit, respectively, if the BP that is used for the final ejection charges is stored outside in Huntsville for any extended period of time, that can compromise the quality of the BP and lower the chances of a successful recovery [15].

From the more explicitly safety-oriented side, for example, one major aspect to consider while working with BP is that it is moderately easy to ignite. While BP itself needs a heat source to be ignited, and therefore cannot be ignited directly with electrical current like static electricity, it still can be ignited via static electricity if the static electricity provides enough power to set off the electrical match. To make matters even worse, static electricity can be created by sliding the ejection charges any distance into the airframe. This increases the potential of a premature detonation occurring during the integration phase, when individuals might need to do things such as reach into the airframe to place a charge. If the charge detonates at this time, injuries up to and including the installer's arm being burned down to the bone could be sustained.

Because of these issues with BP, OSRT has decided to develop a BP-free CO2 ejection system that uses CO2 canisters to deploy the parachutes at the predetermined altitudes. In order to design their own system, OSRT started by looking to see what systems are currently available on the market. As CO2 systems go in general, there are three different styles that are available. The first, sold by Tinder Rocketry, is called the

Peregrine. This one can handle eight to twelve-gram canisters of CO_2, and punctures the canisters by using a small amount of BP to drive a pin into the canister, which is subsequently released when the CO_2 pushes the pin out and vents around it out into the parachute bay. While this system can handle CO_2, its downfalls are that, first, it requires BP in order to puncture the CO_2 canisters, and in order to eject anything with these canisters, one needs to find the amount of BP needed to eject the item in question, and then multiply it by a factor of five. Because of this, with OSRT's recovery system last year where the smallest BP charge was four grams, meaning that, at the very least, OSRT would have needed a 20-gram canister of CO_2 to do the same amount of work as the four-gram BP charge. For this reason, the Peregrine would not be able to handle this system.

The next system that is available is also sold by Tinder Rocketry, and is called the Raptor. The Raptor is very similar to the Peregrine, but it can handle 20 to 85-gram CO_2 canisters. This is also a suitable system, and would meet all of OSRT's requirements, however, since it requires the use of BP to propel a pin forward into the CO_2 canister, it does not meet OSRT's requirements.

The third option that is available for sale is called The Hawk and is sold by FruityChutes. The Hawk has the same capabilities as the Raptor, where it can handle 16 to 45-gram CO_2 canisters. However, it uses mechanical systems to puncture the CO_2 canister, and therefore, does not require any sort of pyro-consumables like BP. This would be the system that OSRT would use given no budget constraint. Unfortunately, it costs $465 per system before shipping and purchasing any of the canisters, and OSRT can design and build a system for a much lower price.

Because these three systems were sub-optimal for OSRT, the team designed eight different potential custom, fully-mechanical CO_2 ejection systems that could be manufactured and installed into the launch vehicle to eject the parachutes. The eight designs are detailed below, starting with Figure 28, which is a design based off of the Peregrine System. What this system does is replace the need for BP with a combination of a servo with a sliding panel and three springs. The system is installed so that the large spring on the left is initially depressed and held against the interface plate with the servo and the sliding panel and the sliding opening pin are placed next to the sliding panel on its right side. When it is time to release the parachutes, the servo pulls the panel up and out of the way of the spring, which forces the opening pin into the CO_2 canister. The venting gas, in combination with the two smaller springs on the right side of the opening pin, then push back on the opening pin, allowing for the canister to vent.

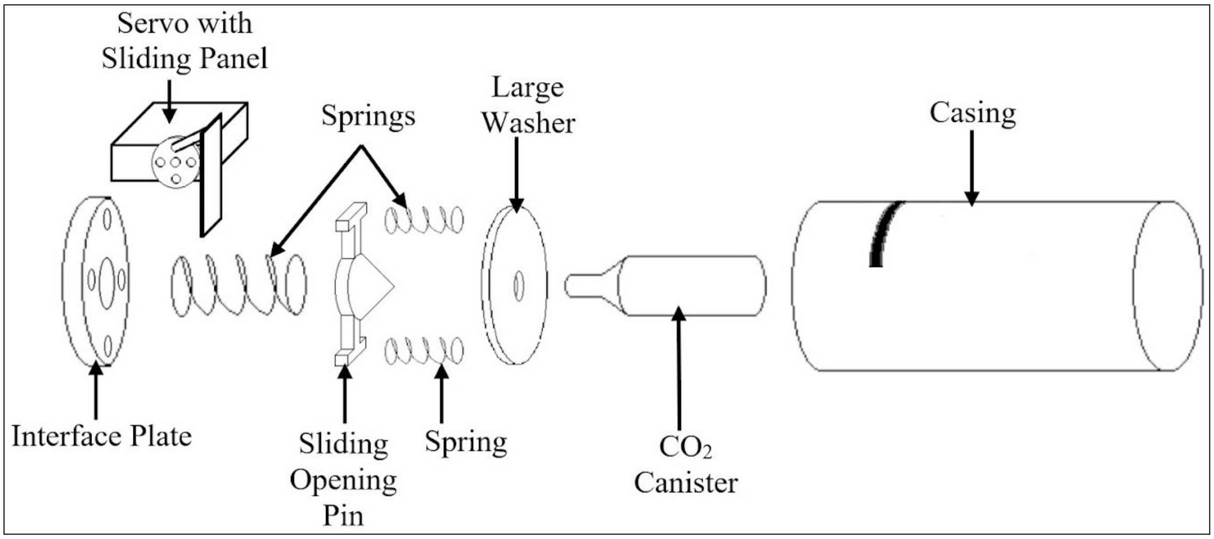

Figure 28: Peregrine-Based Design

The pros in this design are that it does not need black powder to puncture the CO2 canister, and would be relatively low cost to OSRT. The springs could be sourced for free from Oregon State University (OSU), and the casing, interface plate, and opening pin would simply be made from aluminum. However, particularly when it comes to the opening pin, this will be a challenge to manufacture, and will be difficult to integrate into the launch vehicle.

The second design is similar to that of the Peregrine-based design, however, it is based off of the way a BB gun works, and is shown in Figure 29. In this design, a spring on the left and the servo with the sliding panel function exactly like in the Peregrine-based design. Instead of an opening pin, it has an opening needle that is hollow and has an opening below the point at the top of the pin and right above the base at the bottom of the pin so that when the needle punctures the canister, the pin can remain in the canister and the gas can be vented through the needle and out of the casing.

While this design does well in the sense that it does not require black powder to vent the CO2 from the system, it is also quite difficult to manufacture as the opening needle would need to be precisely milled and turned on a lathe.

The third design, shown in Figure 30, is based off of a video that OSRT found online where an individual was testing various CO2 canisters by drilling into them in order to vent the canisters. The way that this design would work is a spring would sit behind the CO2 canister, between the canister and the back wall of the casing, and push the canister into the drill bit. When the parachutes need to be ejected, the motor is actuated so that it continuously turns and drills into the canister, which will vent through the flutes of the drill bit and around the motor out into the parachute bay.

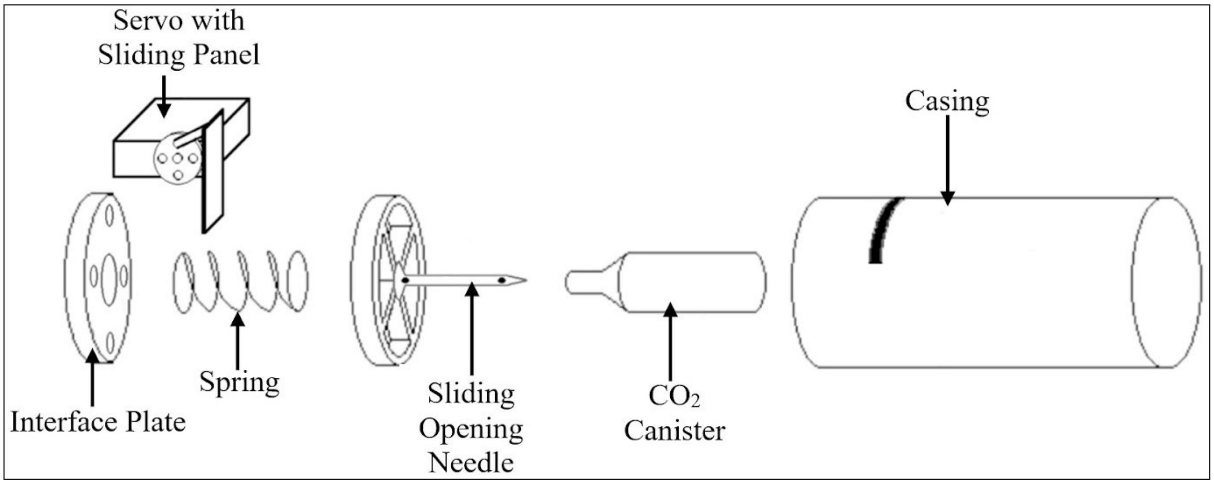

Figure 29: BB Gun-Based Design

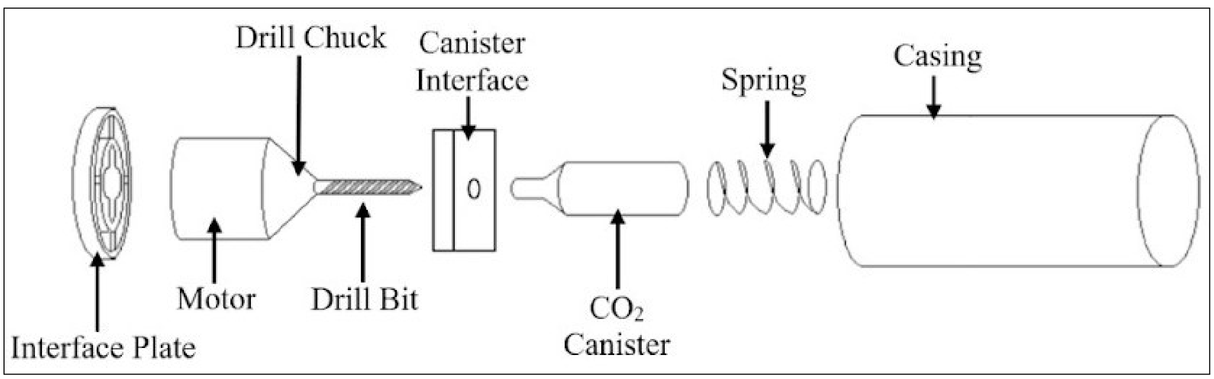

Figure 30: Drill-Based Design

Much like the first two designs, what is great about this one is that it does not require BP to vent the CO2 canister, however, it has potential to be heavier than other designs because of the motor and the drill chuck that can be used.

Fourth is the Sliding Pin Design, where a sliding solid pin is attached to an arm, which is attached to a servo as seen in Figure 31. This servo either pushes the pin into the canister when the launch vehicle reaches the correct altitude, and then pulls it out rapidly to allow the canister to vent, releasing the CO2 into the parachute bay.

While this does meet the requirement of not using any BP, this is another design that would be difficult to manufacture and integrate so that it works reliably, particularly given how the servo is attached to the arm, which is attached to the pin, and how the pin needs to be removed before the CO2 can vents.

Design number five is the Wired BB Gun Design, which is very similar to the first BB Gun-Based Design,

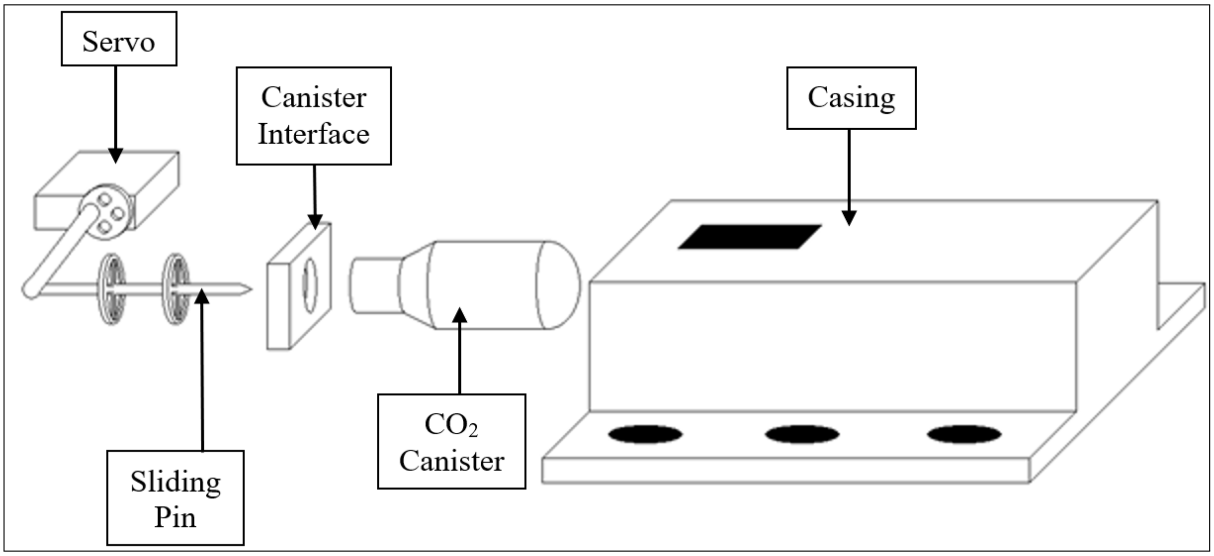

Figure 31: Sliding Pin Design

however, instead of a series of springs like the first design had, the needle is pulled back into the CO2 canister via two wires, a motor, and two pulleys, as seen in Figure 32, like a winch.

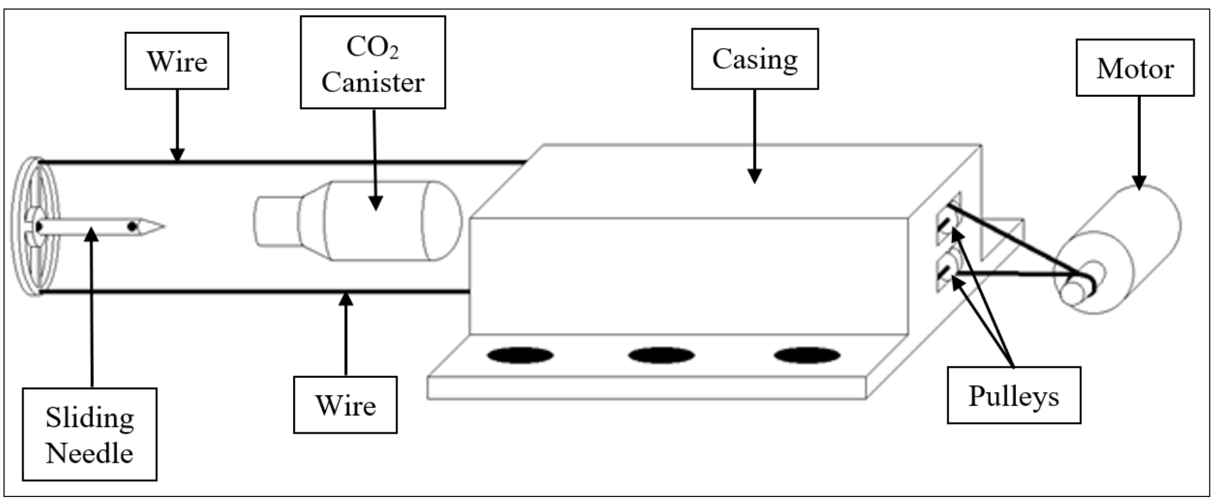

Figure 32: Wired BB Gun-Based Design

While this design also avoids the use of BP, it runs the risk of being very difficult to manufacture, particularly when it comes to pulley installation. There is also an issue in the mechanism being too heavy, especially since the motor is needed, and being difficult to integrate since the motor would need to be aligned perfectly to make sure that the wires do not become out of sync and cause the needle to jam in the casing at an odd angle.

Figure 33 shows the sixth design which uses four electronic magnets to draw the sliding hollow needle into the CO2 canister and holds it there as the canister vents.

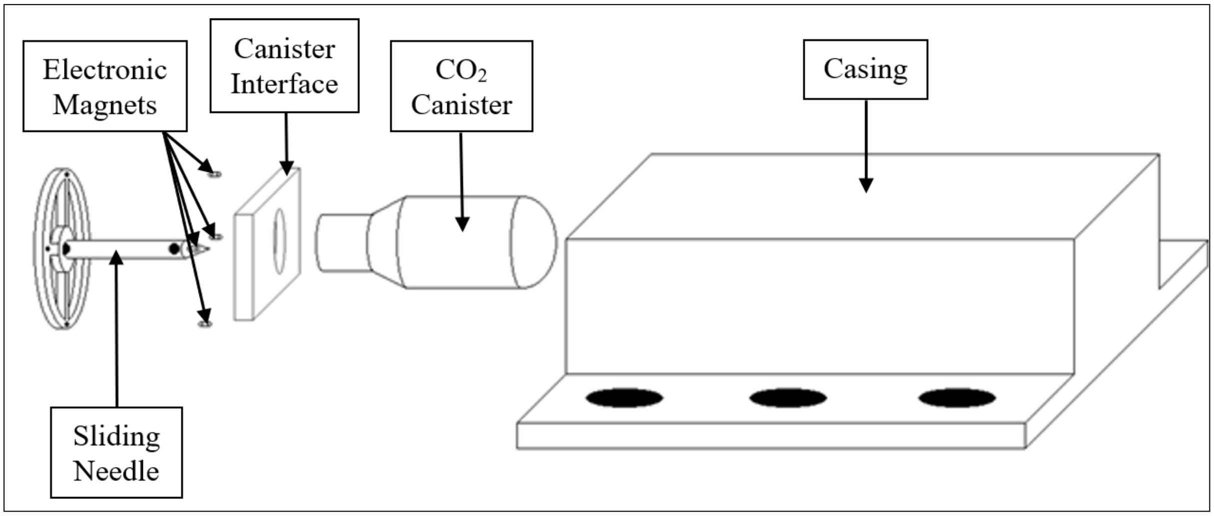

Figure 33: Electronic Magnet Design

This, much like the previous designs, does not use BP, and this design in particular has fewer components and is one of the lightest weight designs in this selection. However, the components that do exist in this design would be difficult to manufacture, particularly when it comes to the hollow needle, and it would be difficult to properly embed the electronic magnets into the appropriate places so that they do not fall out or become dislodged and prevent the needle from accidentally becoming misaligned.

The seventh design is the Solenoid Design, which uses a solenoid as a linear actuator to push a customized pin, which is attached to the end of the solenoid via a set screw, into the CO2 canister to puncture it, and then pulls the pin back out to allow the CO2 canister to vent into the parachute bay. This design is shown in Figure 34.

The great thing about this design, besides the fact that it does not need to use BP, is that it is relatively easy to manufacture, even with the more complicated pieces like the opening pin, and since it does not have that many parts, it is not that difficult to access the CO2 canister to switch it out, and when the CO2 canister is being replaced, there are not many pieces to lose. The downside about this design is that it is a little heavier than some of the other designs, and OSRT would be required to order portions of the design, as they are not kept in the machine shop's stock.

The final design, shown in Figure 35, is very similar to that of the Peregrine design, and the first completely CO2 design mentioned here, but instead of needing to puncture the CO2 canister at the correct altitude, the canister would be installed into the launch vehicle already punctured, and the opening pin would simply need to be released from the punctured area to vent the gas into the parachute bay. Therefore, the opening

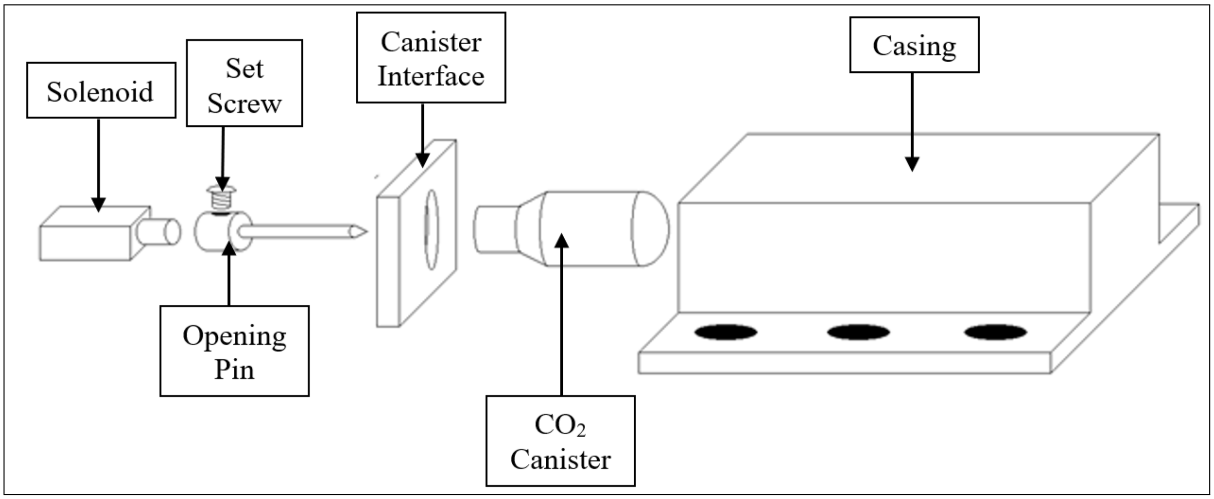

Figure 34: Solenoid Design

pin would be driven into the CO2 canister on the ground and integrated into the launch vehicle so that an actuated servo holds the pin in place, pushed into the canister and into springs on the top and bottom of the device, which will assist in removing the opening pin once the servo releases the pin. When the servo lifts the metal pin that prevents the opening pin from moving, the pressure from the CO2 canister and the stored energy from the springs will push the opening pin away from the canister and allow the canister to vent.

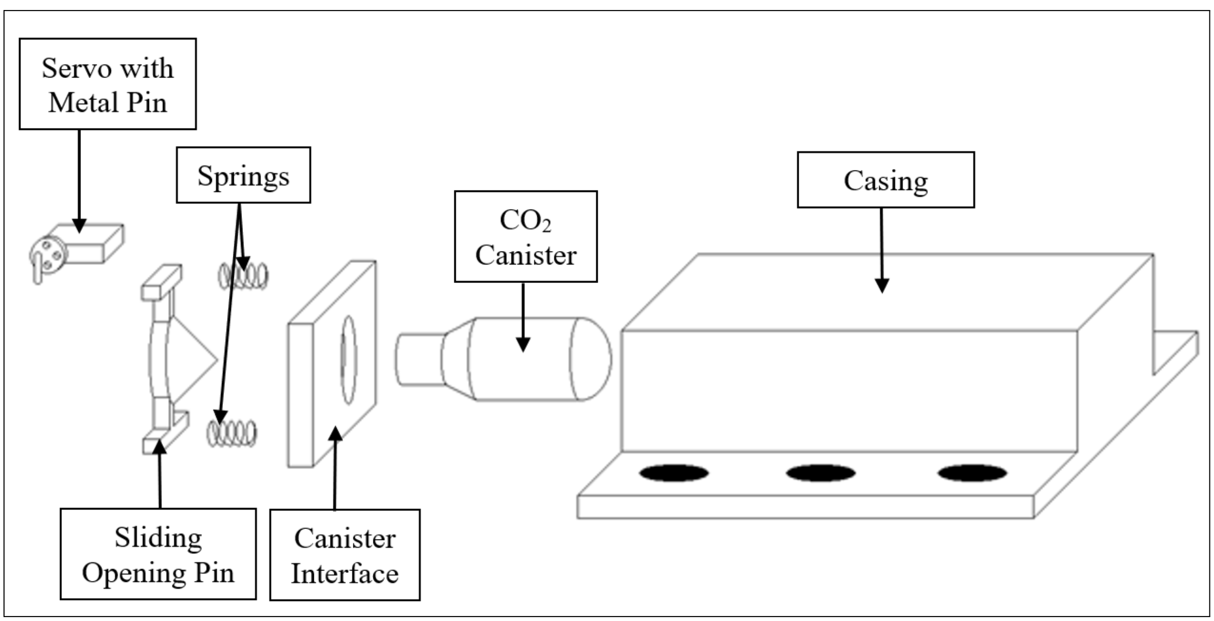

Figure 35: Pre-Puncture Design

The great thing about this design, besides the fact that it does not require the use of BP to be successful, is that it is really easy to source materials, as most of this design would be made out of aluminum. The major downside about this design is that integrating it into the launch vehicle would be extremely dangerous, as the operator would be installing an already-punctured CO2 canister that is only plugged with an aluminum opening pin, and the chances of that pin either falling out of the canister or, worse, being shot out of the canister, is relatively high.

From all of these designs, a decision matrix was created, as shown below in Table 18. This matrix not only includes the eight designs detailed here, but includes the Peregrine, Raptor, and Hawk ejection systems, as well as the standard BP system, which would be constructed from rubber surgical tubing, two rubber stoppers, two cable ties, an electrical match, and BP.

Table 18: Ejection DDM, Part 1

		Option 1		Option 2		Option 3		Option 4		Option 5		Option 6	
		Black Powder		**CO2 Drill**		**CO2 Solenoid**		**CO2 Pre-Puncture**		**CO2 BB Gun Spring**		**CO2 Wired BB Gun**	
Criteria	Weight	Rating	Score	Rating	Score	Rating	Score	Rating	Score	Rating	Score	Rating	Score
Reliable Ejection	2	5	10	3	6	3	6	3	6	3	6	3	6
Easy to Manufacture	1.5	5	7.5	3	4.5	5	7.5	3	4.5	1	1.5	1	1.5
Lightweight	1	5	5	1	1	3	3	5	5	3	3	1	1
Easy to Integrate	0.5	5	2.5	3	1.5	3	1.5	1	0.5	3	1.5	1	0.5
Safe to Handle	3	1	3	5	15	5	15	1	3	5	15	5	15
Low Cost	1	5	5	3	3	3	3	5	5	5	5	3	3
Easy to Reuse/Replace	0.5	5	2.5	3	1.5	5	2.5	3	1.5	3	1.5	3	1.5
Easy to Obtain Parts/Materials For	0.5	3	1.5	3	1.5	3	1.5	5	2.5	5	2.5	3	1.5
Total:	**10**		**37**		**34**		**40**		**28**		**36**		**30**

Table 19: Ejection DDM, Part 2

		Option 7		**Option 8**		**Option**		**Option 10**		**Option 11**		**Option 12**	
		CO2 E-Magnets		**CO2 Sliding Pin**		**CO2 Peregrine-Based**		**Peregrine**		**Raptor**		**Hawk**	
Criteria	Weight	Rating	Score	Rating	Score	Rating	Score	Rating	Score	Rating	Score	Rating	Score
Reliable Ejection	2	3	6	3	6	3	6	3	6	5	10	5	10
Easy to Manufacture	1.5	1	1.5	1	1.5	1	1.5	5	7.5	5	7.5	5	7.5
Lightweight	1	5	5	3	3	3	3	5	5	3	3	3	3
Easy to Integrate	0.5	1	0.5	3	1.5	3	1.5	3	1.5	3	1.5	3	1.5
Safe to Handle	3	5	15	5	15	5	15	1	3	1	3	5	15
Low Cost	1	1	1	3	3	5	5	3	3	3	3	1	1
Easy to Reuse/Replace	0.5	3	1.5	3	1.5	3	1.5	5	2.5	5	2.5	3	1.5
Easy to Obtain Parts/Materials For	0.5	3	1.5	3	1.5	5	2.5	1	0.5	1	0.5	1	0.5
Total:	**10**		**32**		**33**		**36**		**29**		**31**		**40**

Given this matrix, one can see that there are two options that are tied as the best option: the CO2 system that uses a solenoid and the Hawk. Given that the Hawk costs $465 before buying a canister, and OSRT can make a similar system from a solenoid that costs about $11 from Amazon, and a collection of stock aluminum that can be sourced from the scrap barrels in the machine shop for free. The screws can also be sourced from the machine shop for free. A CAD of the Solenoid Design is shown in Figure 34, and a drawing of the casing is shown in the Appendix, in Figure 123

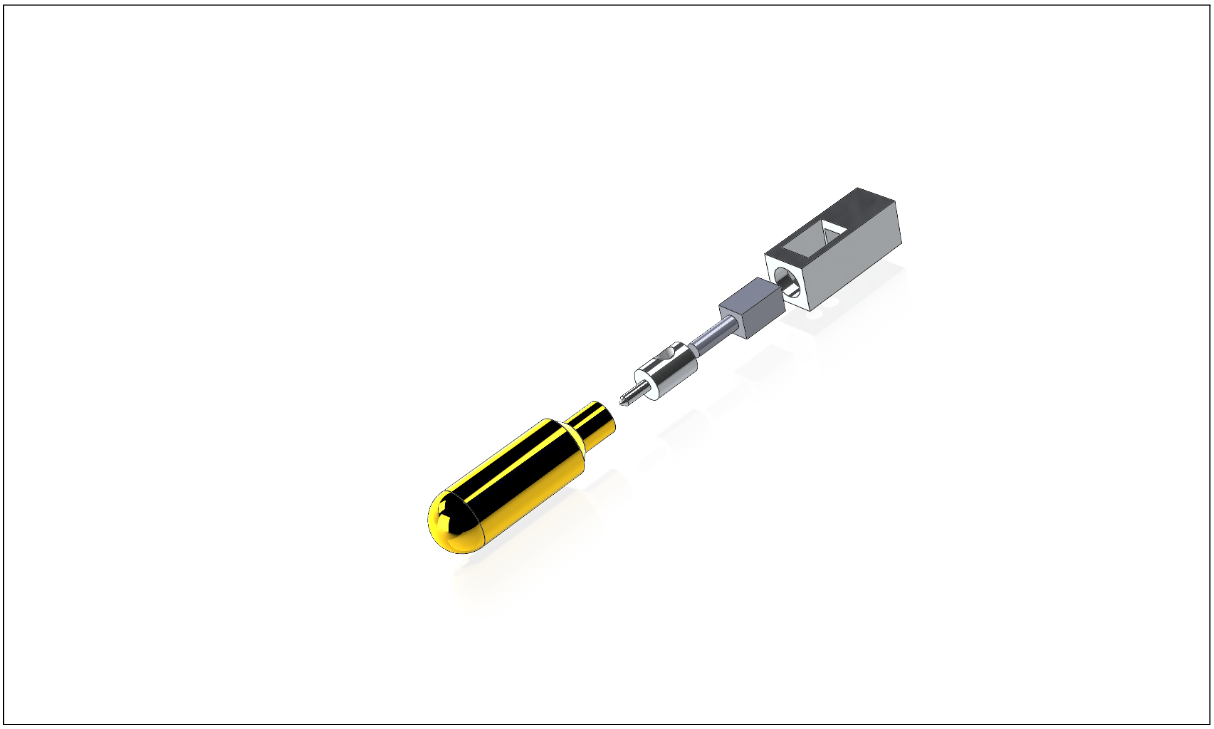

Figure 36: Peregrine-Based Design

For sizing the CO2 canister, as mentioned before, the equation was:

$$CO2 = grams_{BP} * 5 \tag{7}$$

It will be necessary to first figure out how much BP is needed to eject these parachutes via both mathematics and ejection tests in order to figure out how much CO2 is necessary. First off, the necessary amount of BP will be calculated with the following equation:

$$BP = 0.006 * l_{compartment} d^2_{compartment} \tag{8}$$

This equation was used to figure out the amount of BP necessary to separate the nose cone from the fore section of the OSRT's L2 launch vehicles. In this instance, the launch vehicle's parachute bay had a diameter of 4 in. and a length of 18 in., so the necessary amount of BP to eject the parachute from this parachute bay was calculated to be 1.72 grams. Upon manufacturing and testing, the scale used to measure the black powder was a Hornady G2-1500 Digital Powder Scale, which can only accurately measure out to one-tenth of a gram. Therefore, 1.7 grams of BP was used in this ejection test. The rest of the charge was constructed by plugging 1.5 in. of one end of 0.5 in. inner diameter surgical tubing with a circular rubber plug, and tightening a cable tie around the outside of the surgical tubing where the plug is located. Then, the 1.7 grams of BP was poured into the surgical tubing, and the electrical match was inserted into the BP so that the red cover and the rest of the wire hung out the top of the surgical tubing. Another rubber plug was inserted in the top of the surgical tubing and a cable tie was tightened around the rubber stopper to ensure that no BP leaked out. When it was properly installed, the 1.7 grams of BP ejected the parachutes really well, showing that this is a good equation to use for the initial calculations to gain a starting point for testing the BP.

3.5 Back Up Charges

Since safety, particularly that of the bystander, is extremely important to OSRT, a back-up BP charge will also be implemented in the ejection bay to ensure that, regardless of if the CO2 system works, the parachutes will be ejected from the parachute bay and given every opportunity to inflate and slow the launch vehicle down to below the landing kinetic energy.

3.6 BEAVS

The BEAVS 2.0 is an airbrakes system which will be utilized to hit the team's projected apogee altitude of 4,000 ft. OSRT has dedicated one mechanical engineer, one electrical engineer, and one computer science student to the design, manufacturing, electrical system, and controls of BEAVS 2.0. Various mechanical, electrical, and controls systems have been considered for the optimal performance of the airbrakes. Last year, OSRT achieved an apogee altitude just forty-eight ft above the target altitude, and winning third place in the altitude challenge. This was achieved with motor selection and ballasts in the fore and aft sections of the launch vehicle. While this was a great approach to tackling the altitude challenge, an active system will allow the launch vehicle to take in-flight data and adjust its altitude by producing drag with the airbrakes. Last year, an airbrakes system was designed but was not able to be implemented into the launch vehicle before competition. This year, OSRT is determined to integrate an active airbrakes system to hit the desired altitude with as much precision as possible.

In order for the active airbrakes system to work, the motor selected must simulate higher apogee altitudes than desired, since the airbrakes can only actively reduce the altitude of the launch vehicle. The dual-ballast system will be implemented again this year, and BEAVS 2.0 will be utilized to fine-tune the altitude to the 4,000 ft mark. Launch day conditions are variable, but the ballasts and airbrakes are capable of adjusting on launch day to fit these conditions. While this is a complex system, OSRT has a dedicated team and previous year's knowledge to help it succeed.

BEAVS 2.0 will be located just fore of the motor, below the center of pressure to ensure stability is not compromised when activated during flight.

3.6.1 BEAVS 2.0 Mechanical System

Per the mechanical system, three designs were considered:

1) Rack and Pinion – This is the design which was presented in the OSRT Proposal, and also similar to the design tested last year [22]. Two blades mounted onto racks are housed inside the airframe and extend perpendicular to the airframe once the pinion is activated by the motor. The pros are that it is a reliable method of actuation, cheap to manufacture, and lightweight. The cons are that the rack and pinion must fit together with tight tolerances to reduce frictional losses or damage to the teeth. If the pinion cannot be manufactured to an acceptable tolerance, one will be sourced from online.
2) Sliding Pin/Scotch Yoke – This was inspired by the classic scotch yoke design that was encountered when researching for rotational to linear motion actuation methods[1]. The classic design drives one rod on a rotating gear with a rotating pin that absorbs the motion in one axial direction from the rotational movement, leaving the other axis of motion to be actuated. The idea was that two pin and rod assemblies could be assembled onto one gear. The pros are that this design is easy to manufacture,

affordable, and reliable. The cons are that it would create an uneven drag profile on the launch vehicle, and that it is a bit of a complex design in terms of defining forces on each member.

3) Pulley – A pulley design was tested and successfully implemented by Iowa State University in the 2017-2018 year during test launches[23]. A similar design was considered for this year. However, when comparing the pros and cons, the system had way more cons than pros in implementation. The pros are that it is a lightweight system. The cons are that having external members may interfere with the drag on the aft section of the launch vehicle body during recovery, it's a complex system exposed to higher flight forces, and overall more expensive than the alternate designs.

Both the "Rack and Pinion" and "Sliding Pin/Scotch Yoke" designs were selected to move on in the design process, and SolidWorks CAD models are pictured below in their respective orders.

The DDM shown in Table 20 assisted in the process of narrowing down the designs.

Table 20: BEAVS 2.0 Mechanical DDM

Design		Rack & Pinion		Sliding Pin		Pulley	
Requirement	Weight	Rating	Score	Rating	Score	Rating	Score
Weight	2	3	6	3	6	4	8
Easy to Manufacture	4	2	8	2	8	3	12
Low Cost	2	4	8	3	6	2	4
Max Stress Observed	3	4	12	3	9	1	3
Reliable	4	5	20	4	16	5	20
Increase in Cross-Sectional Area	5	5	25	5	25	5	25
Total		79		70		72	

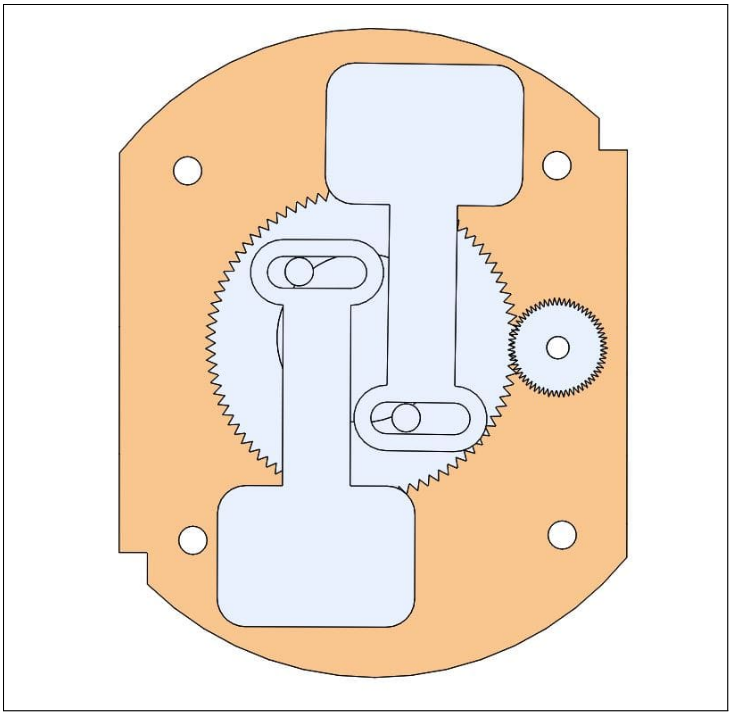

Figure 37: Sliding Pin/Scotch Yoke Design

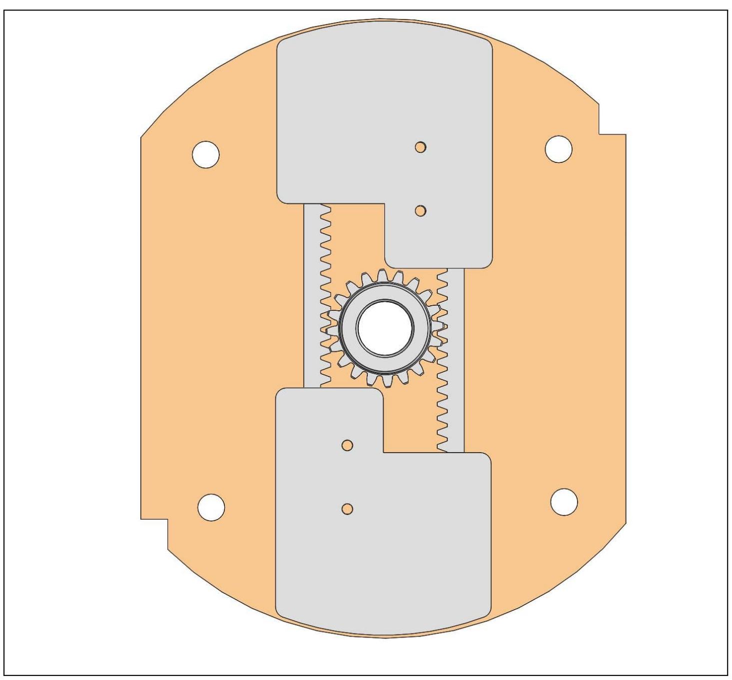

Figure 38: Rack and Pinion Design

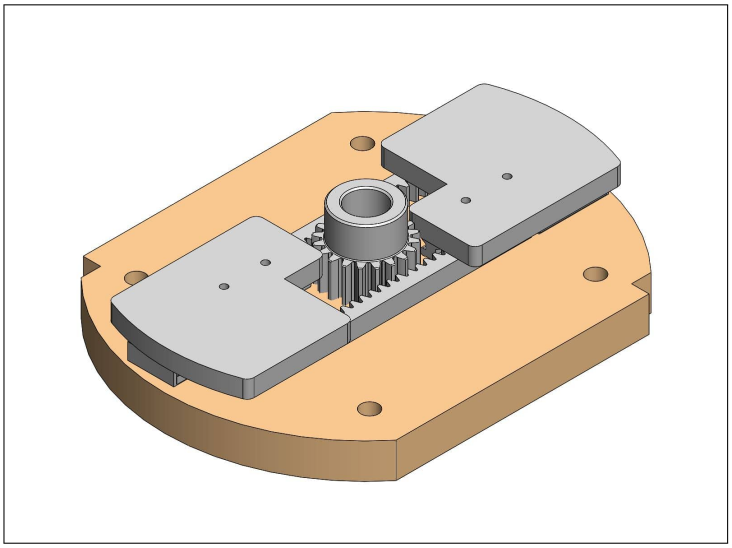

Figure 39: Rack and Pinion Retracted ISO

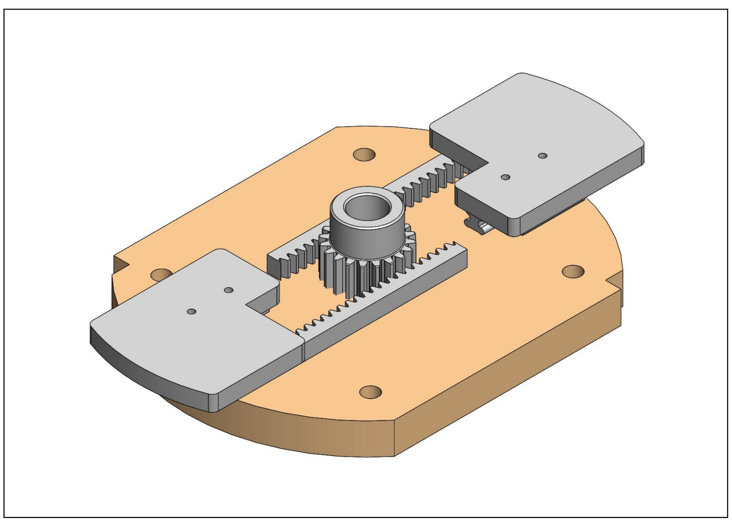

Figure 40: Rack and Pinion Extended ISO

Current coefficient of drag

With the current rack and pinion design, the increase in cross-sectional area with the blades extended results in an approximate 10% increase in the coefficient of drag of the launch vehicle. The C_d will be verified using a Star-CCM+ simulation once the OSRT obtains a license for the program, which is currently pending but has been affirmed that it is coming.

3.6.2 BEAVS 2.0 Electrical System

The electrical system for BEAVS is built to be connected to a barometric pressure sensor and accelerometer which takes in environmental data from the surrounding air during the vehicle's flight and will relay that to the microcontroller which will be running calculations that predict apogee of the vehicle. If the operations inside the microcontroller determine that the apogee is less than or greater than our desired apogee then the microcontroller will send a signal to the motor driver and into the motor to rotate the blades forwards or backwards to increase or decrease drag respectively. A list of electronics and sensors used in the system include: MMA8452Q Accelerometer, MS5802-14AB Barometric Pressure Sensor, Teensy 3.6 Microcontroller, FIT0441 Brushless DC Motor, and 12v LiPo battery.

Per the electrical system, three designs were considered:

1) Teensy 3.6
   This design is based around size constraints and simplicity. The pros of using the Teensy 3.6 is that it is reliable and a more straightforward design and can be used for redundancy for the rest of the vehicle. The clock speed and memory size are tremendous for its size and power consumption. The Teensy 3.6 includes a microSD card reader which will be used to collect the altitude data and motor position during flight which can later be analyzed to see how the system reacted during flight. The cons of this system are that is utilizes a couple different signal interfaces which can prove to be more complicated than using I2C or Serial Peripheral Interface (SPI) protocol interfaces exclusively.
2) Arduino Due
   This design was inspired by researching microcontrollers that would be effective for this system. By using the Due microcontroller, the system would have access to less and slower memory than the Teensy board, and also lacks an on-board microSD card reader like the Teensy 3.6 has. Because of this, the Printed Circuit Board (PCB) for the system would need to be designed to accommodate this component, which would also require more power being supplied to the circuit. The pros of this microcontroller are that it has more default libraries, making it easier for programming. The cons include the larger size which is one of the largest constraints of the system, there are less Inputs and Outputs (I/O) pins for connecting devices, and the absence of an on-board microSD requires more power and complexity.
3) Redboard Artemis Nano
   This microcontroller was found while researching powerful, small-scale boards that could accurately process the barometric pressure readings from the sensor during flight. This microcontroller is similar to the Teensy 3.6 with the same amount of flash memory but offers a limited selection of I/O options due to its smaller size. The pros of this board are its lower weight and size measurements and a larger Random Access Memory (RAM) size. The cons are that the clock speed is slower, therefore fewer inputs and outputs, and no integrated microSD card reader on-board. The design of the system is based

around having the quickest processing speeds so that calculations can be run in real time and can be most reactive to the change in environment, and most importantly, air pressure.

Both the "Teensy 3.6" and "Redboard Artemis Nano" designs were selected to move forward with into the design process, and KiCad schematics are pictured below in Figure 42 and Figure 43 respectively.

The DDM shown in Table 21 assisted in the process of narrowing down the designs.

Table 21: BEAVS 2.0 Electrical DDM

Design		Teensy 3.6		Ardunio Due		Artemis Nano	
Requirement	Weight	Rating	Score	Rating	Score	Rating	Score
Weight	2	3	6	2	4	4	8
Size	3	3	9	2	6	4	12
Low Cost	2	3	6	2	4	4	8
Reliable	3	5	15	4	12	4	12
Memory Size	4	4	16	4	16	5	20
Clock Speed	5	5	25	4	20	2	10
Total		77		62		70	

Figure 41 outlines the structure and interfaces of the electronics within the system.

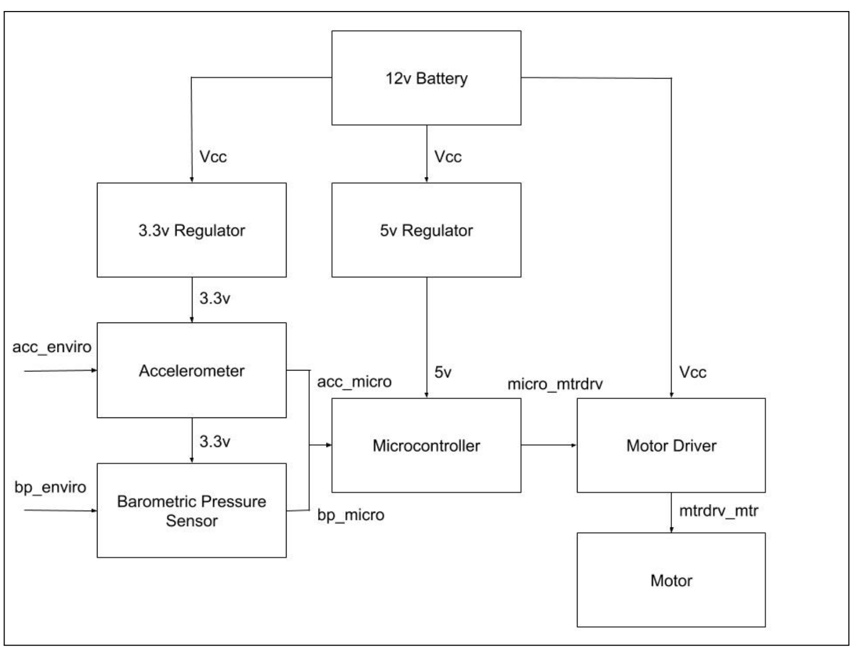

Figure 41: BEAVS Electrical Block Diagram

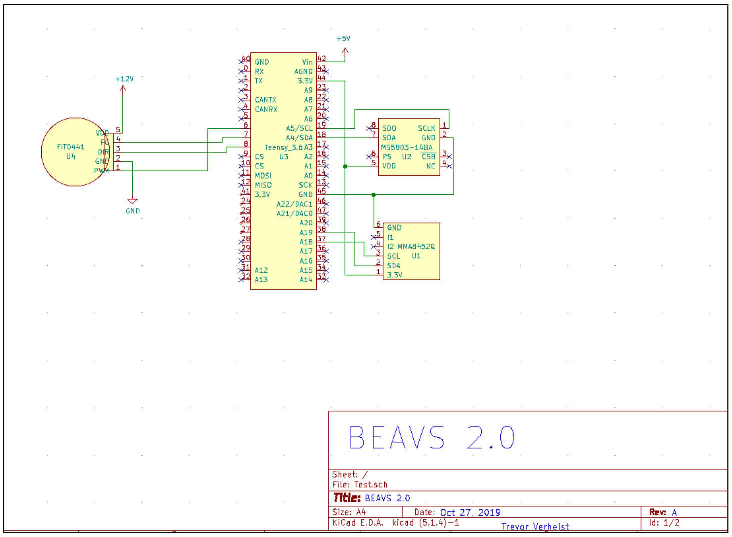

Figure 42: BEAVS 2.0 Teensy 3.6 Schematic

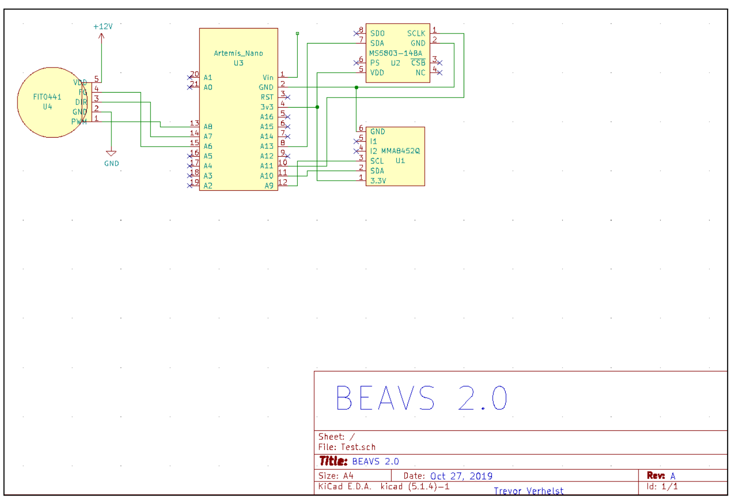

Figure 43: BEAVS 2.0 Artemis Nano Schematic

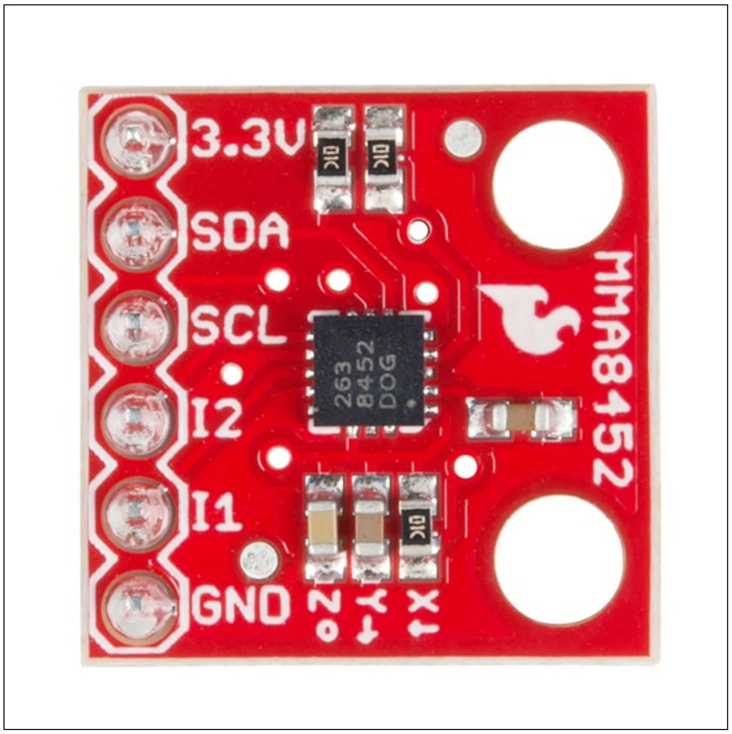

Figure 44: MMA8452Q Accelerometer

Figure 45: FIT0441 Brushless DC Motor

Figure 46: MS5802-14AB Barometric Pressure Sensor

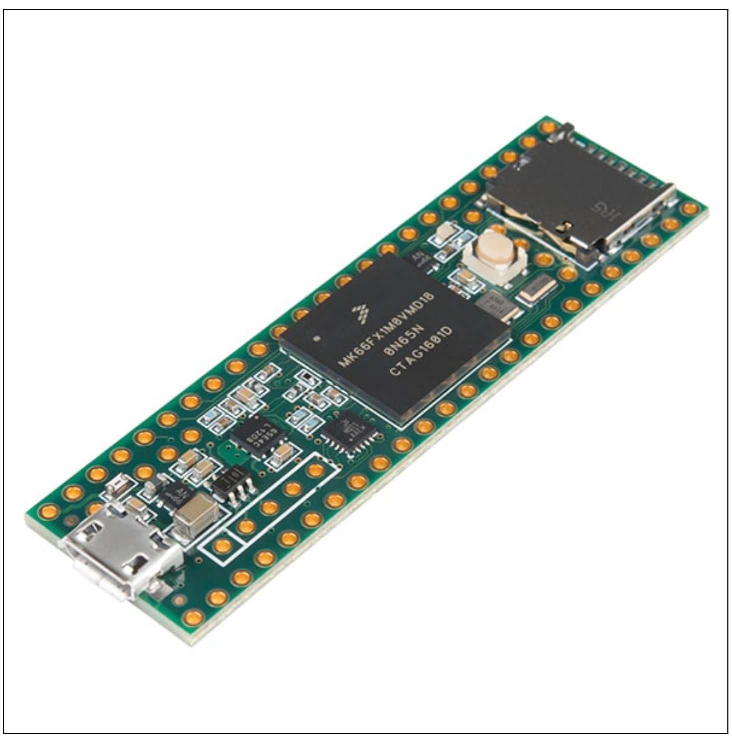

Figure 47: Teensy 3.6 Microcontroller

3.6.3 BEAVS 2.0 Controls System

To ensure the mechanical and electrical systems will operate properly during flight and will not compromise the mission, the controls system must be reliable and efficient. The BEAVS 2.0 controls system will intake data from a GPS and a barometric pressure sensor. The output will be the projected apogee altitude. The OSRT has considered multiple options for the methods of storing, processing, cleaning, and using the input data to calculate the projected apogee altitude.

Two main methods were considered for controlling the motor activation and position.

1) The first method is pre-defining the time it would take for the blades to extend fully and then send a signal to the motor encoder for that time interval. During that time interval, the program will pause until the blades are fully extended (or retracted), so as to not overwrite itself while being activated. Once the time interval is up, the blades will have either been extended or retracted and the program will resume again, gathering data from the sensors.
2) The second method is to initially estimate the altitude based off of the expected motor burnout altitude. After that altitude has been reached, the system will then rely on the sensors to run through the loop sequence.

Table 22: BEAVS 2.0 Motor Controls DDM

Design		Pause Program while Blades Extend		Switch Between Motor Burnout Altitude & Sensors	
Requirement	Weight	Rating	Score	Rating	Score
Process Time	2	3	6	4	8
Reliability	5	2	10	2	10
Complexity	3	4	12	4	8
Total		28		26	

Another two methods were considered for sensor data acquisition and transmitting that data to an Arduino to be fed into the controls system.

1) The first method is to solder the sensors onto the PCB and initialize the Arduino to intake the data and feed the values into the equations provided in 3.6.4. This control loop will then determine what the motor controls do.
2) The second method is to have the sensor controller and motor controller on separate PCB's. This is beneficial because it reduces cost and complexity so that in the event that one controller is having issues, the other controller is not compromised as a result.

These DDM's result in a motor control scheme of pausing the program while the blades are extended. This is because it is the easiest to implement and simply involves adding a timer, with the value pre-set by ground system testing, and implementing a delay to counter the added resistance of flight forces during launch.

Table 23: BEAVS 2.0 Sensor Data Acquisition Controls DDM

Design		One PCB		Two PCB's	
Requirement	Weight	Rating	Score	Rating	Score
Process Time	2	3	6	1	2
Reliability	5	3	15	1	5
Complexity	3	3	9	4	12
Total		30		19	

The sensor data acquisition control scheme will be one PCB circuit board. Having two boards results in an increased process time and complexity, but a decrease in reliability.

The airbrakes will only exist in two states: extended and retracted. This is because the cross-sectional area increase is approximately 10% and having steps in between will only complicate the system with minimal added value to controlling the apogee. With the blades deployed, the increase in cross-sectional area will increase the drag force on the launch vehicle and will therefore reduce the expected altitude.

A Proportional-Integral-Derivative (PID) control loop will be utilized along with a varying set point and a short duty cycle. The PID control loop will calculate the error value in the current set point and compare that to the desired set point. The varying set point is to ensure that in the event of a controls failure, the BEAVS 2.0 will have more opportunities to correct itself than just one with a constant set point. The short duty cycle will allow the system to recalculate the set point multiple times to fine tune the launch vehicles apogee altitude.

OSRT is approaching the controls system with preventative measures to ensure no controls or electrical failures are experienced on launch day. The controls will have fail-safes implemented that prevent the brakes from activating during motor burn, this will ensure that the blades absolutely will not be deployed under the flight forces which occur during motor burn. Teams in the past have had this happen and have suffered from airframe damage and compromised mission assurance. The airbrakes will also automatically deploy at 4,000 ft. If the change in position from the current altitude of the launch vehicle to the previous is negative (meaning it has reached apogee and has done its gravity turn), BEAVS 2.0 will not deploy. If the expected apogee altitude is projected to be less than 4,000 ft from the launch rail, BEAVS 2.0 will not deploy but will continue to collect and store flight data from the sensors.

Simulations will be run to ensure the sensors and controls are operating without error. These simulations will use calculations to solve for expected values, and compare those values to output values of the physical system. The variables that need to be defined in the system are:

- mass of launch vehicle (m)

- cross-sectional area of launch vehicle without blades extended (A_1)
- cross-sectional area of launch vehicle with blades extended (A_2)
- drag (D)
- air density (ρ)
- velocity (v)
- burn time (t)
- impulse (I)
- thrust (T)
- acceleration of gravity (g)
- coefficient of drag without blades extended (Cd_1)
- coefficient of drag with blades extended (Cd_2)
- acceleration without blades extended (a_1)
- acceleration with blades extended (a_2)
- altitude without induced drag (h_1)
- altitude with induced drag (h_2)

The following equations will allow the team to solve for each of these variables to implement into the control loop, or to run simulations to double check the values from the sensor data.

3.6.4 BEAVS 2.0 Testing and Simulation

The altitude equation (h), which solves for the projected altitude given an input of air density and drag, can be simulated for different altitude estimations using the calculation below for air densities at varying altitudes. Once the barometric pressure sensor and accelerometer are connected to the system and flown in a test launch, experimental values will be collected. These experimental values will be compared to simulated values to ensure the simulations match the flight data. This will give the OSRT confidence in the system's ability to properly track position during flight.

The mathematical relations that the system will use to simulate and validate the controls are as follows:

Density (as a function of height)

$$\rho = \frac{P}{1718(T + 459.7)} \tag{9}$$

Temperature can be approximated with respect to height using:

$$T = 59 - 0.00356h \tag{10}$$

Pressure is then approximated with respect to temperature using:

$$P = 2116 \frac{T + 459.7}{518.6}^{5.256} \tag{11}$$

Drag:

$$D = C_d \rho \frac{1}{2} v^2 A \tag{12}$$

Velocity:

$$v = \sqrt{\frac{(T - mg)}{k}} * \frac{1 - e^{\frac{kt}{m}}}{1 + e^{\frac{-2kt}{m}}} \tag{13}$$

Where q and k are respectively shorthand for:

$$q = \sqrt{\frac{(T - mg)}{12A\rho C_d}} \tag{14}$$

$$k = 12C_d A\rho \tag{15}$$

And t is time which can be solved by:

$$t = \frac{I}{T} \tag{16}$$

Where v is the velocity of the launch vehicle, m is the mass of the launch vehicle, and g is the acceleration of gravity.

Acceleration during coast phase

Without the airbrakes extended, the acceleration of the launch vehicle post motor-burnout will neglect the effects of drag:

$$a_1 = -g \tag{17}$$

With the airbrakes extended, the drag force acceleration is accounted for:

$$a_2 = \frac{-D}{m} - g \tag{18}$$

Finally, the projected altitude of the launch vehicle:

Altitude gain during motor burnout

$$h_{boost} = \frac{-m}{2k} ln(\frac{T - mg - kv^2}{T - mg}) \tag{19}$$

Altitude gain during coast phase without induced drag:

$$h_1 = h_0 + vt - \frac{1}{2} a_1 t_{coast}^2 \tag{20}$$

Altitude gain during coast phase with induced drag:

$$h_2 = h_0 + vt - \frac{1}{2} a_2 t_{coast}^2 \tag{21}$$

Where t_{coast} is:

$$t_{coast} = \frac{I}{T} \tag{22}$$

The overall altitude projection will be a super position of the altitude gained with drag and the altitude gained without drag:

Total altitude

$$h_{total} = h_1 + h_2 + h_{boost} \tag{23}$$

3.7 Avionics

The OSRT ATU is designed to fulfill the requirements of tracking the launch vehicle in the air and transmitting the data to the base station. In order to fulfill these requirements, the block diagram in Figure 48 was made. This block diagram shows the preliminary parts of fulfilling these requirements.

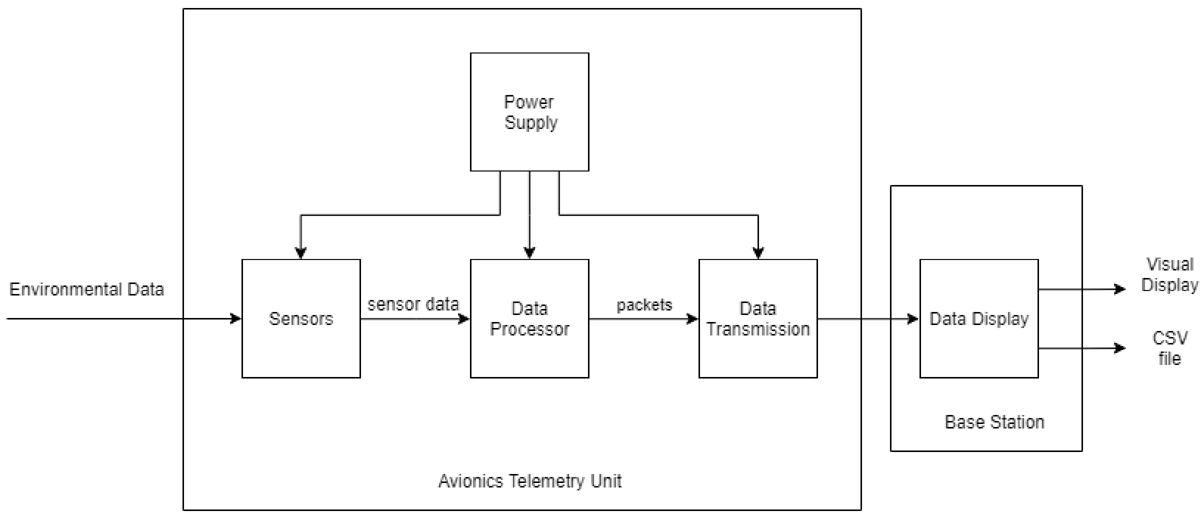

Figure 48: The Block Diagram for the Avionics System

3.7.1 Sensor Block

Data must be collected from the external environment. In order to meet the minimum requirements, location and altitude data must be gathered from the external environment. In addition to being able to track the launch vehicle's location upon landing, the ATU must also track the launch vehicle's route while airborne. As a team, it was decided to include an inertial measurement movement to gain information on the flight path and acceleration of the launch vehicle. This should help gain more information for analyzing the accuracy of predictions and give useful information for later analysis. There are many different possible options for each component, and different configurations of sensors, that have been considered for the design.

A block diagram of the system is shown in Figure 49.

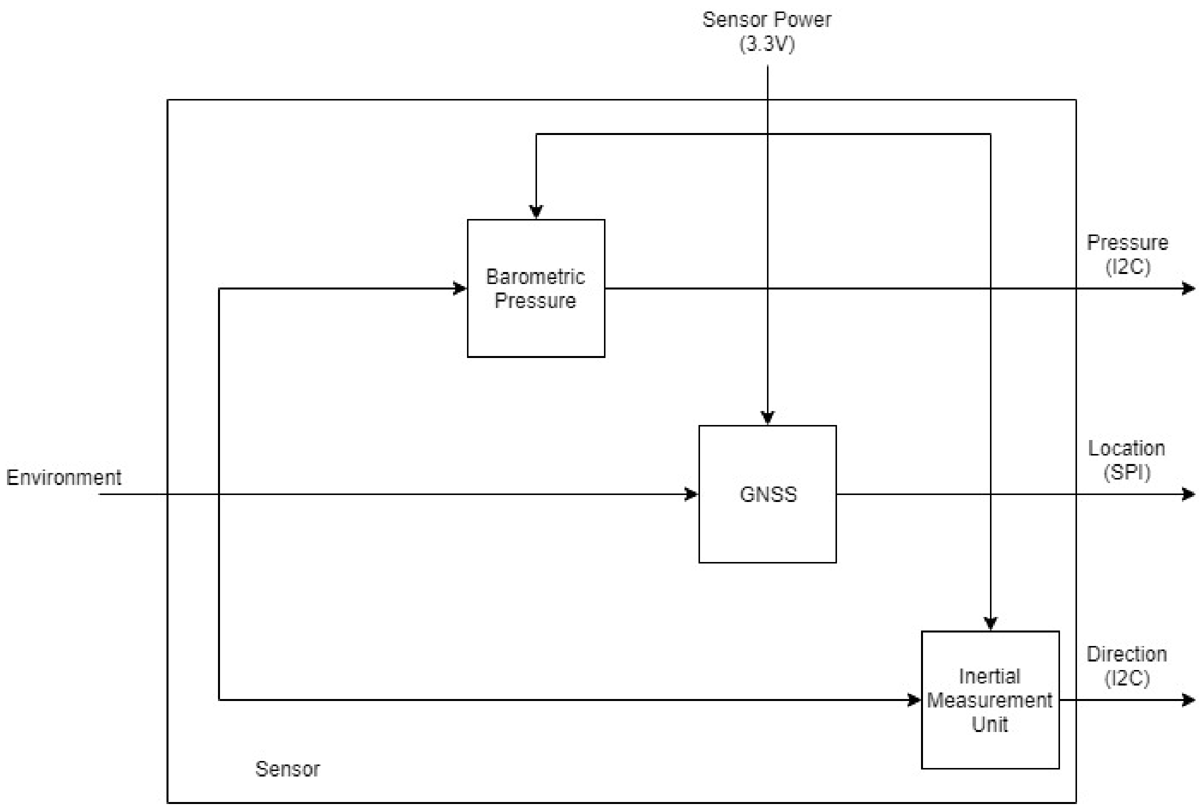

Figure 49: This shows an overview of the sensing operations that will be needed

3.7.1.1 Global Navigation Satellite System (GNSS) Modules

The avionics system will use a GNSS module to collect location data from satellites. Global GNSS constellations include GPS, GLONASS, Galileo, and BeiDou. GPS is the most robust and accurate GNSS. For GPS, the US government guarantees a global average UERE of less than or equal to 25.6 ft[16]. with 95% probability. As of May 11, 2016, actual performance for GPS shows a global average UERE or less than or equal to 2.3 ft, 95% of the time[16]. GNSS units vary in accuracy, power consumption, and ability to withstand launch forces, which are all factors to compare in determining the best option for the ATU.

GNSS receivers must track at least four satellites to determine position through trilateration. A multi-constellation receiver uses a minimum of five satellites, four from one constellation and a fifth from another. By having access to multiple GNSS systems, the ATU's accuracy and reliability increase through redundancy.

GNSS satellites communicate using signals in the L band, ranging from 1 to 2 GHz. Most GNSS receivers use a single band, but multi-frequency solutions becoming more prevalent. Using multiple bands provides additional data points from each satellite, increasing position and time accuracy. A significant source of error for GNSS comes from the delay experienced as signals travel through the ionosphere. This delay varies based on the frequency of the signal, so it may be resolved through observing multiple frequencies. Error incurred from environmental obstructions can also be mitigated as demonstrated in Figure 50.

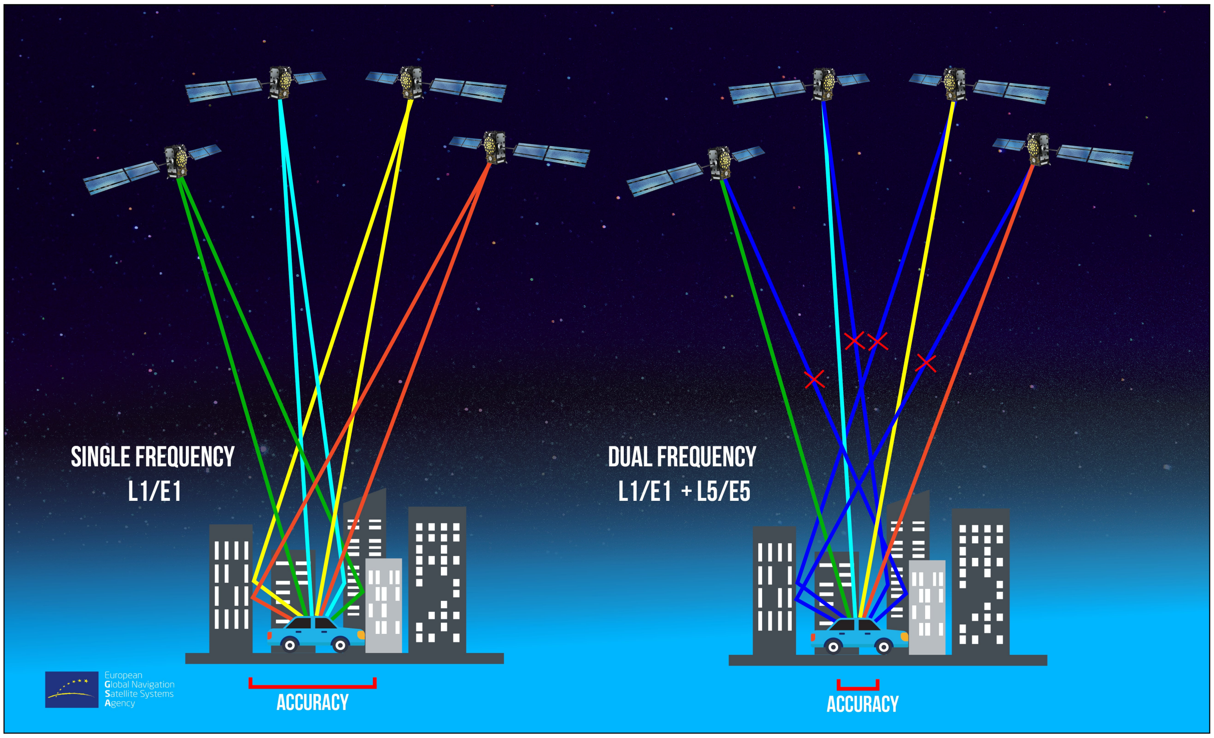

Figure 50: Dual-frequency GNSS obstruction avoidance ©European GNSS Agency

Each GNSS module has different power requirements depending on how it is operating. If a GNSS is completely without power then turned on (a cold start), it will take longer and require more power to get a fixed signal since it must search for satellites. A warm start, or starting when there is still a small amount of power that has been continuously supplied requires less time and power since it has been able to keep the location of satellites stored.

There are also time and power consumption to consider when the GNSS is operating under normal conditions. These parameters define how the device will operate normally when left on for an extended period of time in optimal conditions. However the conditions supplied are not optimal during flight, since temperature decreases as the altitude increases and there is significant force being applied during launch, parachute deployment, and landing. These conditions affect the operations of the GNSS units, and are important comparison measurements for making a decision.

A number of GNSS modules were considered. The three primary contenders, GlobalSat EM-506, U-Blox MAX-M8, and U-Blox ZED-F9P, are considered in Table 24.

Design		GlobalSat EM-506		U-Blox MAX-M8		U-Blox ZED-F9P	
Requirement	Weight	Rating	Score	Rating	Score	Rating	Score
Cost	2	3	6	4	8	1	2
Feature Inclusion	5	3	15	4	20	5	25
Position Accuracy	6	3	18	4	24	4	24
Power Consumption	3	3	9	4	12	2	6
Size	3	3	9	4	12	2	6
Total		57		76		63	

Table 24: GNSS Module DDM

After thorough consideration, the U-Blox MAX-M8 proved to be the best choice. The MAX-M8 features multi-constellation support and sports an attractively low price. While the U-Blox ZED-F9P has more features overall, one feature is unsuitable for this competition. One is RTK positioning, because using this feature requires sending location corrections to the avionics system from the base station. The ZED-F9P is also much more expensive, consumes more power, and is larger than its competitors.

3.7.1.2 Barometric Pressure

Barometric pressure is the best way to measure altitude; barometric pressure does change depending on weather conditions. Since barometric pressure is not always completely accurate, it becomes a design challenge on how to best minimize the error in data readings. This could be done by collecting more information in addition to the barometric pressure, or by taking into account the temperature at sea level. By taking temperature measurements progressively during flight, the altitude measurements can be more accurately calibrated. There are other factors that change the pressure at different altitudes, such as humidity. These are more difficult to measure accurately within the confined space of the launch vehicle.

3.7.1.3 Inertial Measurement Unit (IMU)

The IMU should measure the forces on the device in and during flight. It will allow the development of a more robust data profile of the flight and enable the GUI to be more comprehensive. The IMU will act as an acceleration, magnetism, and orientation. Doing so eliminates the need of having an accelerometer, magnetometer, and gyroscope as separate sensors.

3.7.2 Data Processing

The data processing will be divided between on-board processing and external processing. The on-board processing will need to convert the signals from analog or digital signals into number packets that can be transmitted over the corresponding transceiver. The data it transmits will additionally be logged onto an on-board microSD card.

For the microcontroller, several options were considered. The top three are a Teensy 3.6, Cortex M4 TM4C123G by TI, and ESP32. These are compared in Table 25. For these the Teensy 3.6 was considered the baseline to compare the other microcontrollers since that was used in previous years, so it received 3's in all areas.

The criteria for comparison were:

- Memory - Sensors and data processing consume a fair amount of memory especially when considering processing data from three sensors and turning it into packets.
- Quality of Documentation - If the documentation is lacking it would cause unforeseen issues, drastically increasing time.
- Number of IO - Since several sensors will be used, they all need to have an input source.
- Estimated Development Time - Since the system should be working in April, it is important that the project can be done in reasonable time.
- Size - Since the ATU must fit in a confined space, the microcontroller must leave room for the other devices.
- Power Consumption - The system must last while on the launch bay, while in flight, and while it is recovered.

Design		Teensy 3.6		Cortex M4 TM4C123		ESP32	
Requirement	Weight	Rating	Score	Rating	Score	Rating	Score
Memory	6	3	18	3	18	4	24
Quality of Documentation	5	3	15	4	20	2	10
Number of IO	6	3	18	2	12	1	6
Estimated Development Time	3	3	9	2	6	3	9
Size	3	3	9	5	15	4	12
Power Consumption	3	3	9	3	9	3	9
Total		78		80		70	

Table 25: Microcontroller DDM

It was decided to use a Teensy 3.6 and an MC4 as prototypes. They will be used for preliminary tests. Teensy 3.6 will be used for ease of development for the first subscale, and the MC4 will be tested in the following subscale.

3.7.3 Data Transmission

The ATU must be capable to transmitting sensor data to the base station wirelessly. LPWAN, WiFi, cellular networks are three possible ways of transmitting and receiving data.

Popular with IoT applications, LPWAN allows for wireless communication with low power consumption. LoRa transceivers use LPWAN and offer great flexibility. LoRa's communication protocol can use a low-level, PPP connection. The benefit of this is less overhead and the ability to define custom packets. Some reasons why custom packets may be desired are receipt acknowledgement and reducing interference impact. Its packets are limited to 255 bytes, but its power consumption is low. LoRa operates on lower frequencies, 915 MHz in North America, compared to 2.4 GHz or 5.8 GHz used by WiFi. By using lower frequencies, its range coverage is superior.

Pros:

- Long range
- Low power
- Receipt acknowledgement
- Reduced interference impact

Cons:

- Small packet size limit

Alternatively, the same LoRa transceivers are capable of using the LoRaWAN® network protocol, which expands LPWAN to using the internet as an intermediary to send data packets; however, this requires connecting to a network gateway. LoRaWAN® is not as reliable as LoRa because it does not guarantee that messages are received.

WiFi is well-suited for high bandwidth applications, such as video streaming, and less practical for small data packets. The disadvantage WiFi has its short range. Cellular networks can handle high speeds and long ranges, but the critical concern becomes coverage. Additionally, using 4G consumes power faster than WiFi.

3.7.4 Firmware

Many over the counter microcontrollers including the Teensy 3.6, allow for the rapid development of reliable firmware and software prototyping within the open source Arduino IDE. The Arduino IDE was chosen because it is compatible with a wide range of microcontrollers and simplifies the complexities of interfacing with hardware components within the avionics system. The Arduino IDE gives the flexibility of developing firmware using a variety of languages due to its vast library support; C, C++, and Java are only a few of many languages that can be utilized throughout the lifespan of the project, in conjunction with the Arduino IDE.

The avionics firmware will need to be able to aggregate, process, and update hardware data as quickly and reliably as possible. Several languages were considered for the development of the avionics firmware, however it was an clear choice to develop in C. The C language is fast, optimized, and portable when compared to more modern and bulky languages; it contains features from both low and high level programming languages, allowing for fast development and prototyping with both hardware and software. The language allows for quick, low-level access to memory; this is crucial for continuously updating avionics components and the software interface. Languages such as Python and Java were considered for the avionics firmware, however implementation of these languages would be more challenging and time consuming than utilizing C.

Moreover, other languages will be considered if C cannot perform within certain time and design specifications. At minimum, the firmware must update both the software and hardware peripherals ten times a second—this minimum standard is set to ensure that an accurate trajectory is followed, while providing the software with precise data artifacts for analysis. If the C language is simply incompatible with specific hardware peripherals or design, alternative languages will be implemented.

Regardless of which language is used, the design process of the avionics firmware will heavily focus on analyzing the time complexities of the code. Doing so will reduce the memory needed to store the code and decrease the amount of time needed to process input data from hardware peripherals.

3.7.5 Software

The avionics base station GUI will display transmitted data sent from the ATU on a GUI. Because the avionics base station GUI will not send data back to the ATU, control systems for this GUI are not needed; priority is placed on data visualization. The GUI will include a map plot to display the launch vehicle's location for recovery. Other graphical elements we are looking to include are flight instruments that can represent the data from the sensors in a user-friendly way. The base station will save received packets to check against the log saved on-board the ATU's microSD card for packet validation after launch.

The avionics GUI will be written in Python, resulting in a portable application that does not require installation. While C# is a popular language for GUI development, it is less compatible with operating system platforms compared to Python.

3.7.6 Schematics

From these design considerations, preliminary schematics were developed. In Figure 51, the connections selected were I2C. The power supply was selected to supply the power to the Teensy 3.6. The power regulators on the Teensy 3.6 are being used to supply power to the sensors that require 3.3 V.

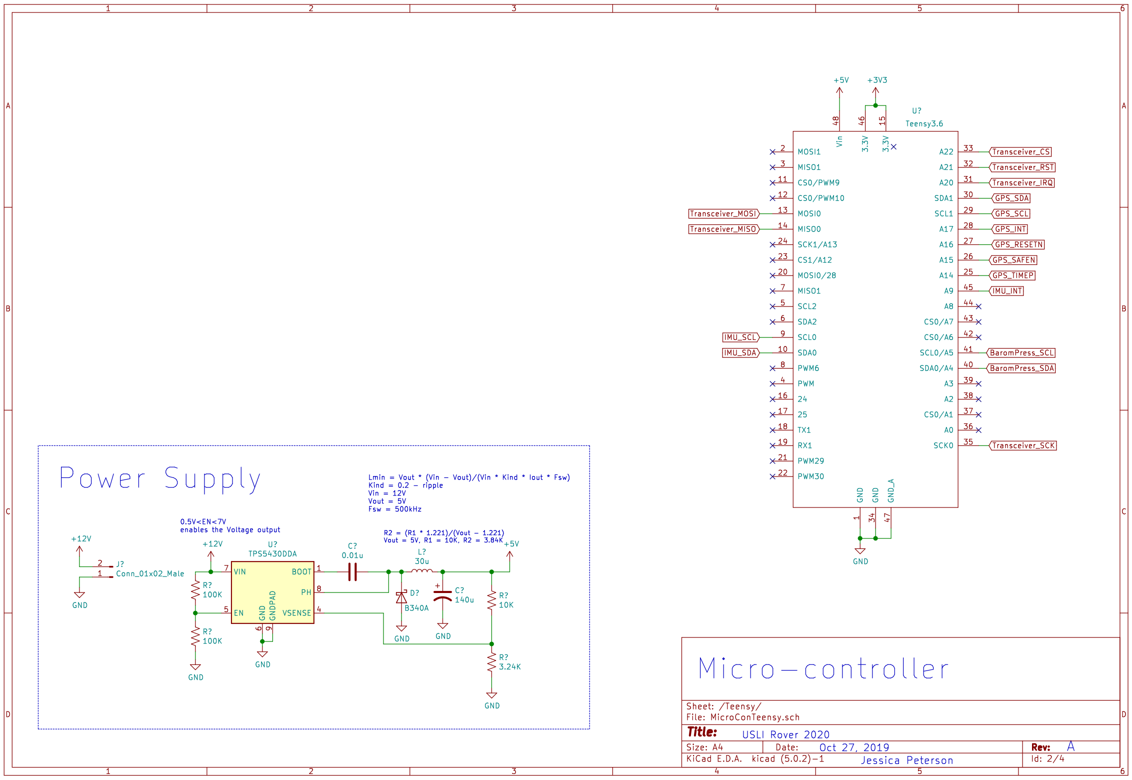

Figure 51: Avionics microcontroller and power supply

The sensors selected for the ATU, are displayed with their connections in Figure 52. These sensors are supplied with 3.3 V of power. These sensors were chosen to provide the best information to the GUI. They contain 22 ohm resistors on several of the lines to improve signal integrity, and protect from ESD.

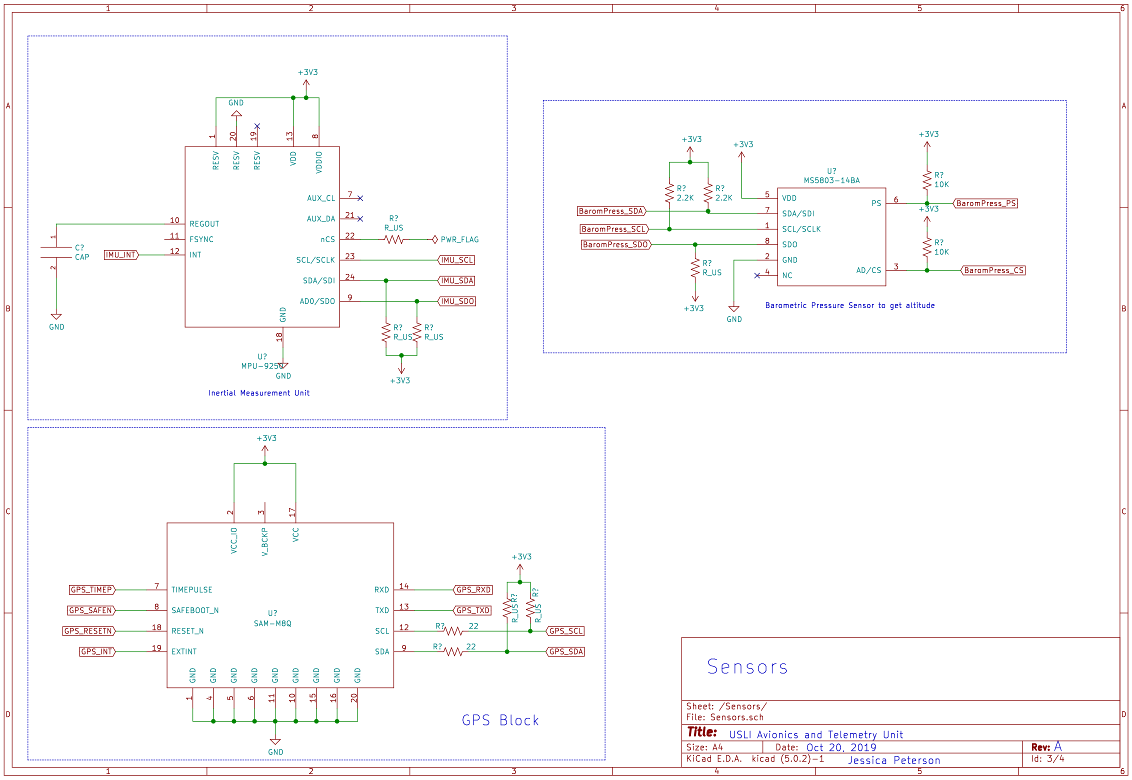

Figure 52: Avionics Sensors

The transceiver system schematic using the LoRa transceiver is shown in Figure 53. This demonstrates the system used to communicate on hardware. A capacitor is placed between ground and power to filter out noise from the power supply and keep the RF signals as isolated as possible.

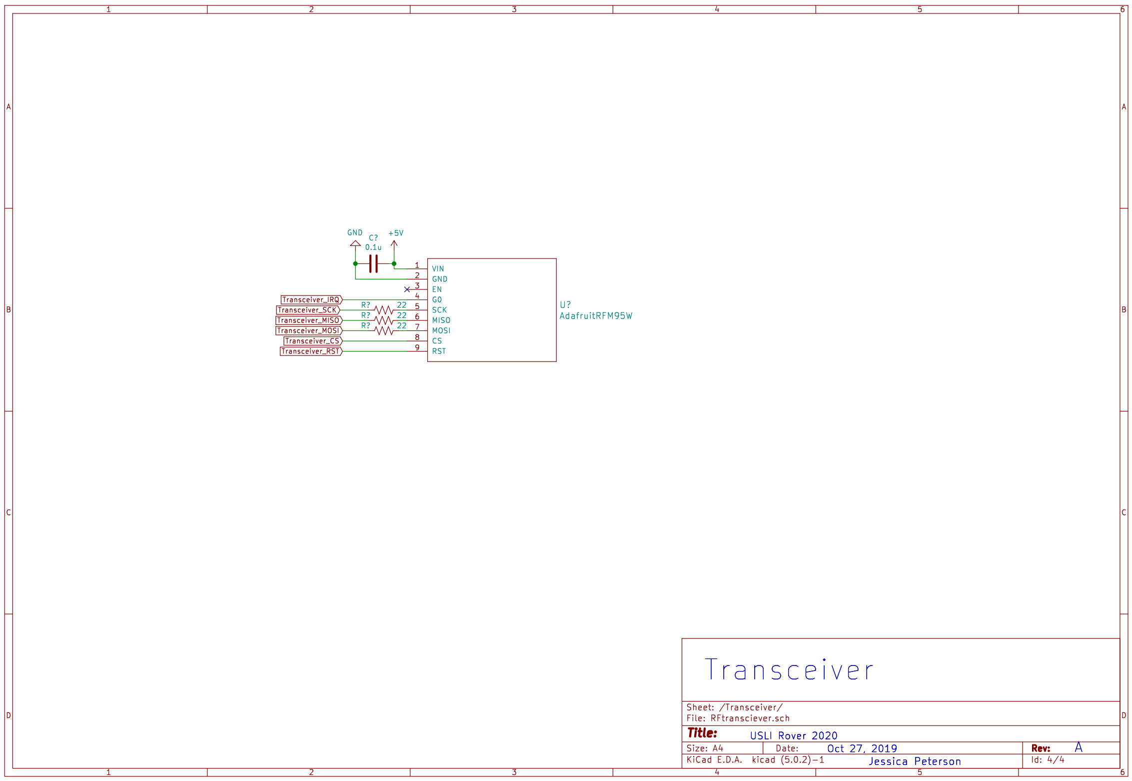

Figure 53: Avionics Transmission

3.8 Stability Analysis

The current stability of the launch vehicle is 3.01 calibers on the launch pad with Center of Pressure (CP) and Center of Gravity (CG) locations shown in Figure 54.

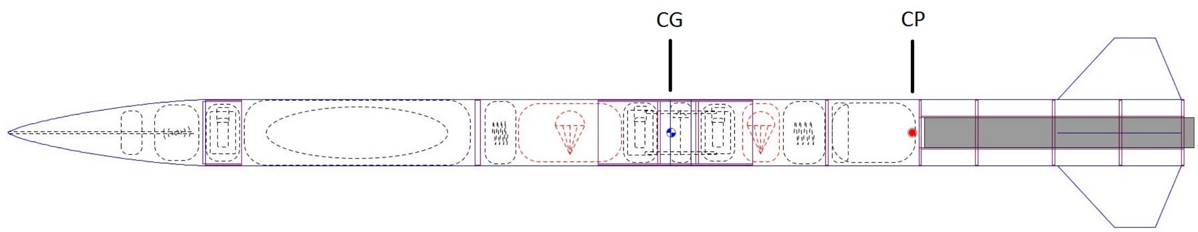

Figure 54: Diagram of CG CP Locations

The CG is located 68.51 inches from the nos cone while the CP is located 87.81 inches from the nose cone as simulated in OpenRocket.

4 PAYLOAD CRITERIA

4.1 Payload Objective

The main objectives of the payload is to survive being launched from within a high power launch vehicle, land and deploy from the launch vehicle housing, and collect a simulated lunar ice sample from one of several sites on the launch field. In order to survive launch and landing of the flight vehicle, there must be a fail-safe retention system. To deploy, there must be a mechanical ejection system. Lastly, the payload will need to travel the terrain to one of the simulated ice sample locations and collect at least 10ml of simulated ice samples. To complete the challenge, it will have to drive at least ten linear feet away with the collected samples to signify a non-existent recovery operation.

The Payload for the OSRT will be a rover with expandable wheels, a beaver tail-shaped stabilizer, and an auger-type collection system. It will be ejected from the launch vehicle by means of a lead screw motor with a built-in retention system involving two circular plates, each with a threaded section for linear motion out of the launch vehicle housing. After ejection, the wheels will expand and drive to the collection site. Once at the collection site, the auger system will move from a horizontal storage position to a vertical drilling position. Once in position, there will be a simultaneous drilling action and linear motion to bring the drill in contact with the ground. The sample will then be stored in the auger housing as the rover completes the challenge by driving ten linear feet from the collection site. Our designs will be discussed in the following sections. Along with supporting research, alternative concepts, and any relevant components.

4.2 Rover Collection

Arguably, the collection system presents one of the most challenging aspects of the rover mission. The rover has enough weight such that the collection system can dig/push its way into the ground. Designing a compact collection system has proved incredibly challenging. Once designed, the system may prove just as difficult to manufacture. A common idea is to use an auger. Manufacturing the parts required for this can prove incredibly difficult. Our team had looked into a variety ideas, some contingent on the sample properties in the field.

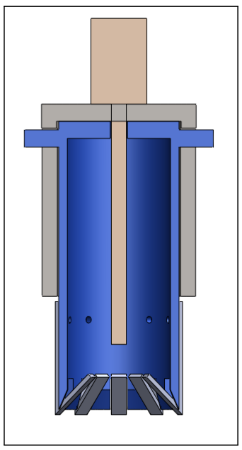

Figure 55: Rover Collection, Core Sample ISO

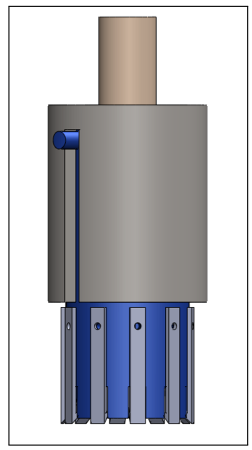

Figure 56: Rover Core

The design in Figure 55, and Figure 56 relies on friction of the ice sample. The system is inspired by core sampling. A cylinder is pushed into the soil, then, when it is pulled back out a sample will be retained inside it. Due to the granular shape of the material that will be collected, some sort of mechanism to retain the sample within the cylinder(shown in blue) is required. In this design, use of an array of thin, bent pieces of sheet metal or polymer is employed. When the cylinder is driven into the ground via the lead screw, the sheet metal is bent open, or toward the walls of the cylinder. This allows for material to enter the core sampler. When the cylinder begins to retract, the sheet metal will want to spring back toward the center of the cylinder. This action will prevent any of the ice sample from leaving the cylinder and will act as a one way valve. The design shown uses thin sheet metal, however there are other methods of achieving this. For instance, a small wire could be used, the cylinder can be lined with adhesive, or a rubber ring could be used to expand the end of the cylinder to perform the same fuction

Benefits of this design include:

- Easy to manufacture
- Easy to achieve linear motion
- Easy to test

- Simple

Challenges with this design include:

- It could require a lot of force to push into ground
- Lead screw may push material out of cylinder

The main challenge with this design is supplying the force required to push the system into the ground. This design will be relying on gravity and the weight of the rover to be enough, this may not be the case. Another challenge is the rover must remain stable as the cylinder is pushed into the ground. If the system is not located properly or if the rover is unbalanced, the lead screw may just tip the rover over, or to the side. One other big concern is whether the ice sample will simply fall out of the cylinder. The lead screw extruding though the cylinder may push material back out.

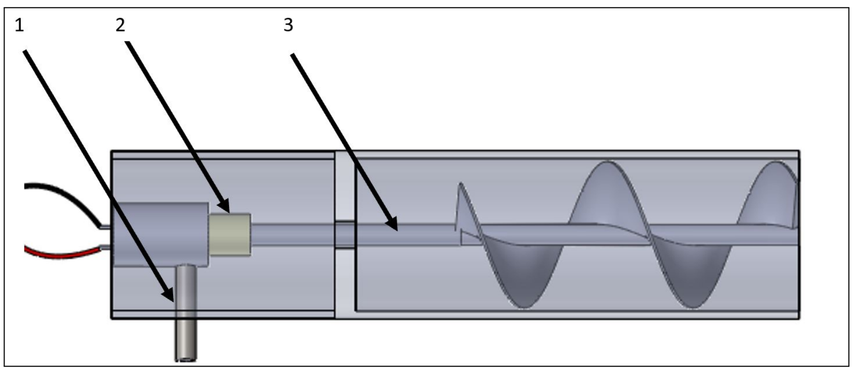

Figure 57: Rover Collection Drill

The design in Figure 57 is for an auger system. The main challenge here is designing a way to both rotate the auger and push it into the ground. This design uses a commercially available auger with an outer diameter of 1.25 in. The auger rod (item 3) will be threaded using a dye. The threaded portion of the rod will be threaded through a fixed section of a hollowed tube. The rod will then be coupled to a high torque motor (item 2). The motor is kept from turning using a pin that is kept in a slot in the tube. This allows the motor to move within the tube. When the motor is powered, the auger turns and unthreads from tube. This allows a single motor to simultaneously spin the auger, and move it linearly.

Benefits of this design include:

- Does not take as much force to penetrate ground
- Auger can penetrate a variety of soil types
- Motor is small
- Few components

Challenges with this design include:

- Auger systems have a bad track record
- Takes up a lot of space
- Ice sample may not be retained by auger

The biggest barrier to using this type of system is auger systems have had a low success ratio in the past. One challenge is that it is incredibly difficult to make the auger itself. OSRT has tried printing one with relative success, but 3D printed parts have a tendency to fracture. It is also difficult to achieve both the rotation and linear movement. Due to space constraints, it would be difficult to develop an assembly with multiple motors; then designs that can achieve the correct motion with a single motor can have geometries that are difficult to manufacture. By using a store bought auger made from steel, there is confidence that this component will not break. The other components in this system have a simple geometry that should be easy to manufacture out of a metal.

Despite potential difficulties, this system is our best option for the collection system. Augers are used across the world for digging in a variety of soil types. Assumptions as to how the soil will be found at the collection site can not be made. It is for this reason, a system that can most reliably dig into any soil type must be chosen. This is why the auger system is being pursued for use on this year's payload. The main challenge is machining the system.

The design shown in Figure 58, and Figure 59 is of a suction system. The idea being a small hollow rod will penetrate the soil. The rod will telescope into the soil, once fully extended, an opening at the end of the rod will allow the ice sample to enter the rod. Air will then be pumped down the rod, and directed back up. As the air flows back up the hollow rod, it will create suction via the Venturi effect. It is this suction that will pull the small granular ice up into a storage vessel. The design above illustrates the tip of the rod.

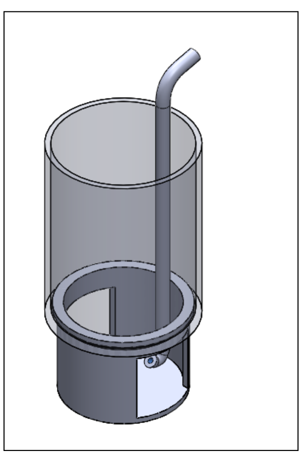

Figure 58: Rover Suction ISO

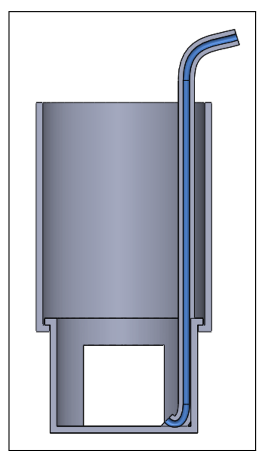

Figure 59: Rover Core

Benefits of this design include:

- Diameter of the tube can be relatively small compared to other designs
- System does not need to be as rigid
- May require less force to puncture soil
- Can collect as much sample material as possible

Challenges with this design include:

- Design is dependent on having a granular material
- Clay could easily clog system

The granular sample material that will be recovered could work perfectly with this design. However, if there is any clay mixed in with the material, then that could quickly clog the system. This system is highly dependent on soil conditions. As a result, this design has a high likelihood of not working, and as such our team will not be pursuing this idea.

The design shown in Figure 60, Figure 61, Figure 62, and Figure 63 illustrates a method for storing the auger on the rover. It also illustrates a method for potentially moving the auger linearly. In this design, the auger assembly is held within a tube, this tube is held in a hinge by a clamp, then the hinge is connected to a 90 degree servo. The servo can rotate the auger into a vertical position. The tube rack is connected to the tube, such that when in the vertical position, the rack locks into a pinion that is run by a motor. This allows the rover to rotate the collection system and drive it into the ground.

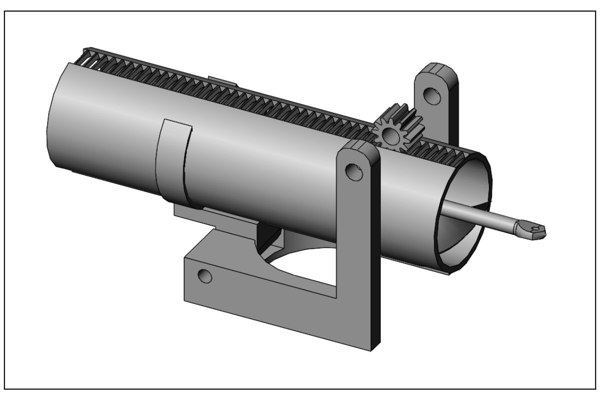

Figure 60: Auger Stowed

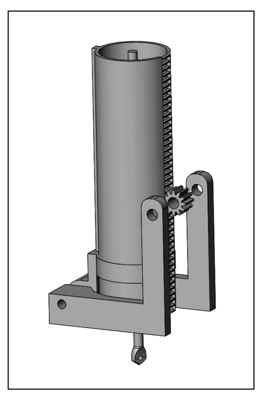

Figure 61: Auger Active

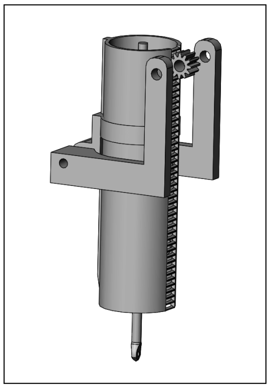

Figure 62: Auger Extended

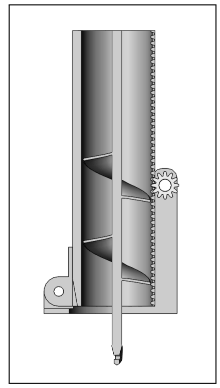

Figure 63: Auger Section View

Benefits of this design include:

- By rotating the collection system ground clearance of rover can be increased

Challenges with this design include:

- Servo may not have enough torque to secure rack and pinion
- Slightly complex geometry
- getting the clamp to release the cylinder when in the extended position could be difficult

The main challenge with this system is getting the rack and pinion to work properly. However, by re-positioning the servo hinge, the auger could rotate into better vertical position, eliminating the need for the rack and pinion.

Table 26: Collection Components

Component	Material	Weight(lbf)	Count	Subtotal Weight (lbf)
Auger	Steel	0.16	1	0.16
Housing	Aluminum	0.274	1	0.274
Auger Motor	NA	0.212	1	0.212
Servo	NA	0.212	1	0.212
Hinge	Aluminum	0.7	1	0.7
			Total Weight	1.56

4.3 Rover Drivetrain

During last year's competition, the rover was successfully transported to the designated target area, but upon ejection, it became high-centered on a small mound of dirt. As a result, the rover was unable to complete its mission. One of the key contributing factors leading to this scenario was the rover's low ground clearance that resulted from the rover's wheels.

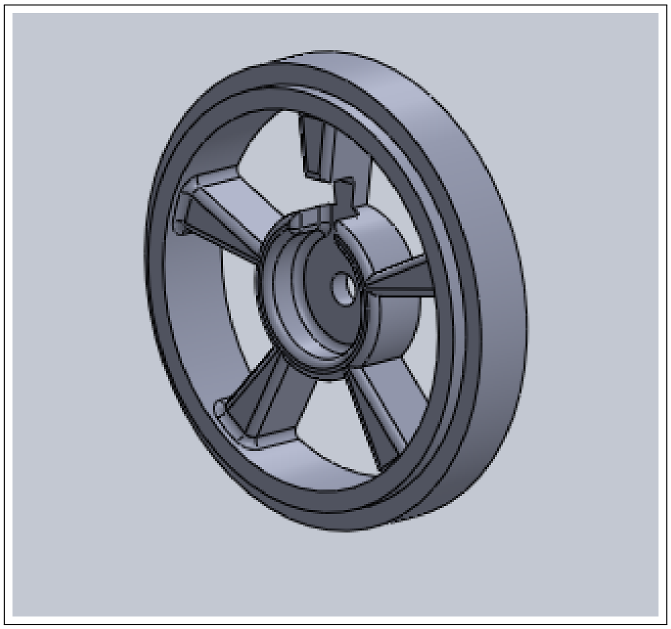

Figure 64: Previous Year's Rover Wheel

4.3.1 Designing a New Wheel

In previous years, the rover's wheel size has been capped by the launch vehicle's body diameter of approximately 6 in. This year, efforts are being made to overcome this limitation by implementing a wheel that can fold to fit inside the launch vehicle and unfold to increase ground clearance.

The primary challenge in designing expandable wheels include reaching the desired 'unfolded' diameter, maintaining their functionality, and preserving space within the launch vehicle's fore section. Each potential design has varying amounts of complexity and functionality.

The goal for each design is as follows:

- The folded wheel's diameter is 6in. or less
- The expanded wheel's diameter is 10in.
- The wheel's folded position takes up minimal space within the launch vehicle's fore section
- The wheels are lightweight

- The wheels are able to be manufactured
- The wheels roll instead of "walk" from spoke to spoke
- The wheels do not require a motor to fold/unfold

4.3.2 "Single-Hinge Spoke" Concept

The first concept for an expandable wheel involves six supports that pivot between the extended and folded positions. This is accomplished using hinges between the spoke parts and the main hub. Tension springs pull the spoke's outward-facing tabs toward the hub's center. During launch, the launch vehicle's body tube will be acting as the retaining ring, keeping the wheel in the proper folded position until it is deployed out of the launch vehicle. Upon deployment, the spokes will spring up, fanning out to the expanded position and kept in place by a combination of tension springs, mechanical stops, and normal force with the ground.

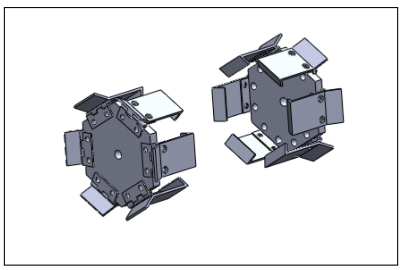

Figure 65: Rover Wheel Collapsed

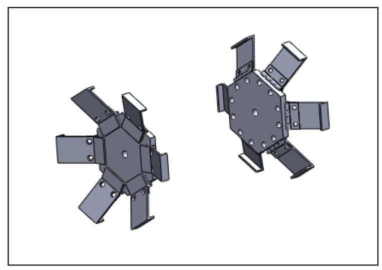

Figure 66: Rover Wheel Expanded

As seen in the design variant shown below, additional "blades" can be added to the spokes to improve the wheel's ability to roll. The blades as shown would be made out of steel, and the edges could be made thicker to provide more control surface. Using such a material requires the blade to have two radii: one about the rover's body when in the folded position, and another about the hub's center when in the expanded position.

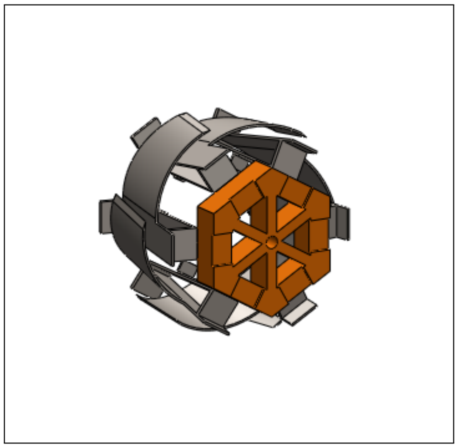

Figure 67: Rover Wheel Collapsed

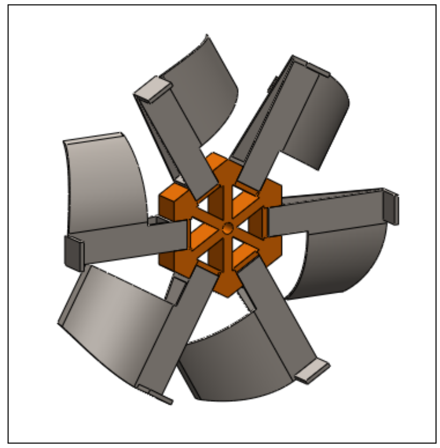

Figure 68: Rover Wheel Expanded

Alternatively, the blades could be made out of a semi-rigid rubber that can bend around the rover when in the folded position and then revert to its upright position upon release.

The spokes themselves would be constructed of steel sheet metal. The hub can be 3D-printed. All hinges and springs would be off-the-shelf parts purchased from a third party.

Benefits of this design include the following:

- Wheel diameter anywhere between 9 in. and 12 in.
- Folded diameter of 6 in. allow it to be transported in the launch vehicle
- Spring-loaded spokes will unfold upon ejection of the rover
- Spoke and hub are easily manufactured

Challenges to this design include the following:

- Depending on the blade's material, there my be difficulty making it strong enough while still fitting into the launch vehicle.
- Possible interference between the folded spokes/blades and the rest of the rover assembly
- The base assembly of the hub and spokes is relatively heavy

4.3.3 "Scissor-jack" Concept

This design emphasizes the use of multiple joints to create a compressible wheel. Inspired by the simple extending motion of a scissor jack, the wheel expands by using torsion springs located at the link joints. In its folded state, the wheel keeps inside its initial thickness. Once expanded, it provides a nearly continuous rolling surface.

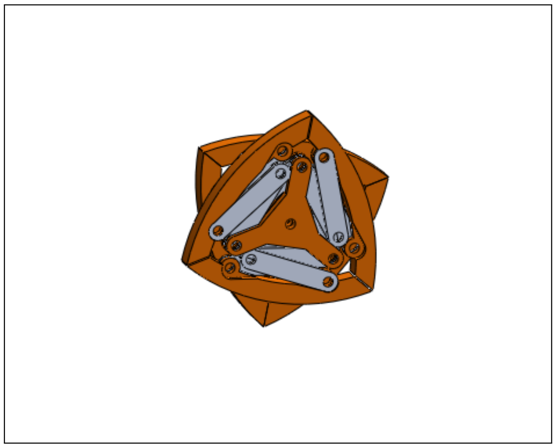

Figure 69: Rover Wheel Collapsed

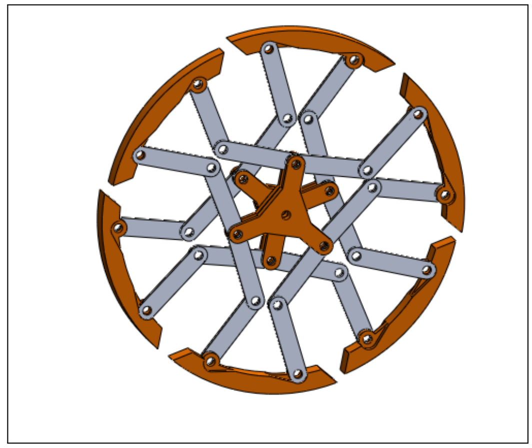

Figure 70: Rover Wheel Expanded

The hubs and rim can be 3D printed, and the links can be made from steel sheet metal. All springs and fasteners would be off-the-shelf parts purchased from third parties. Benefits of this design include the following:

- Wheel diameter expands to 10 in.
- Folded diameter of 6 in. allow it to be transported in the launch vehicle
- Folded wheel will not overhang the rover, preventing any potential interference between the two
- All parts are relatively simple to manufacture.
- The base assembly of the hub and spokes is fairly light weight
- Spring-loaded design will unfold upon ejection of the rover

Challenges to this design include the following:

- Many moving parts, meaning high amounts of potential failure points.
- Joint bearings and attachment points for springs are not yet designed
- Locking mechanism(s) are not yet designed
- 24 links per wheel — tedious to manufacture and prototype
- Relatively small surface area interfacing with ground

4.3.4 Origami Wheel Design

As team members searched for possible solutions, they came across an extremely creative approach developed by Seol Nation University (SNU) Biorobotics Lab. As illustrated in Figures 71 and 72, the rover is able to fold and unfold its wheels on command. A prototype for this design is currently being built.

Figure 71: Origami Wheel Collapsed

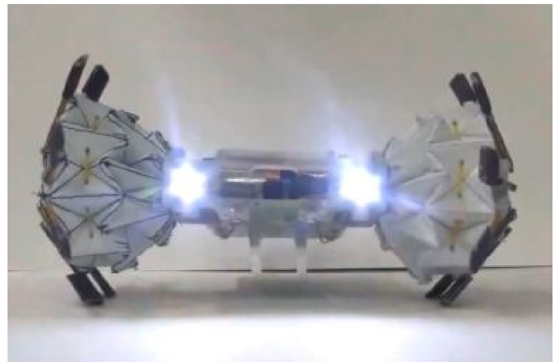

Figure 72: Origami Wheel Expanded

Benefits of this design include the following:

- Wheel diameter can be expanded
- Folded wheel diameter allows transport via launch vehicle
- Wheel can fold and unfold on command
- Flexible
- Lightweight

Challenges with this design include the following:

- Large horizontal space requirements (in folded position), leaving very little room for the electronics, motors, and collections system
- Very technically complicated
- Difficult and very time intensive construction
- Operation requires at least one more motor, using more space and power
- In order to take full advantage of the folding capabilities, the wheels 'walk' from spoke to spoke and provide minimal surface area in contact with the ground

Based on the list of benefits and challenges, the team will most likely not utilize this design. However, further investigation of this design is needed before that decision is final.

4.3.5 Proposed Design

The team will implement a "Single-Hinge Spoke" design (see Figures 67 and 68) that incorporates rubber blades to increase rolling surface while also allowing for easy folding with the launch vehicle body. As such, each wheel will be comprised out of the materials detailed in the table below.

Table 27: Wheel Components

Component	Material	Weight(lbf)	Count (per wheel)	Subtotal Weight (lbf)
Hub	ABS plastic	0.381	1	0.381
Spoke	Steel Sheet	0.213	6	1.278
Blade	Rubber	0.044	6	0.26
Hinge	Steel	0.019	6	0.114
Screws	Steel	.001	24	0.024
Extension Spring	Music wire (Steel)	0.00138	12	0.017
			Total Weight	2.074

Perhaps the greatest concern for this design is its weight. The team will be working to refine the design, remove unneeded material, and reduce the overall weight.

4.4 Rover Design

The design for last year's rover is shown in Figure 73. Our team has adapted a variety of concepts from this design to work on this year's payload.

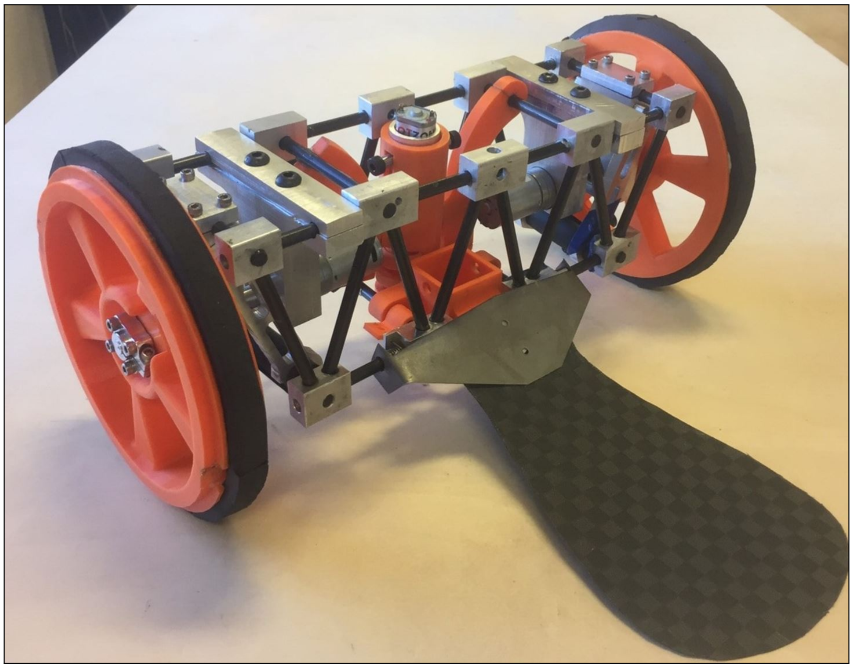

Figure 73: Previous Rover Design

Benefits of this design include:

- High durability due to the structural design
- 3D printable components
- Lightweight stabilizing tail
- Single motor collection system

Challenges with this design include:

- Complex geometry machined components
- Low ground clearance
- Low strength collection system
- High weight structural design
- Limited mounting space for electronics

The durability of this design was a requirement driven by our ejection system. This year, mechanical ejection systems, that will be less violent than the black powder system used with last years rover, will be pursued.

As a result, the rover structural design no longer needs to be as rigid. This is further explored in Section 4.6 of the report, detailing structural changes this change allows. Other lessons learned from this design include keeping machined part geometry simple, utilizing 3D printed parts when possible, and tail design. The previous year's stabilization system was very effective, as a result it can be seen in some of our designs.

An important design characteristic discovered at last year's competition is ground clearance, the lack of which heavily impacted the rover's performance on uneven terrain. As a result, significant time has been spent looking at alternative wheel designs to help with this, specifically expandable wheels. This requirement has also significantly affected structural design of the rover. After review of this design, the OSRT has decided to pursue new systems for rover drivetrain, sample collection, structure, and ejection. Overall design will be further discussed in this section, followed by our investigation to each of these systems in subsequent sections. Efforts will be focused on a two-wheeled rover payload systems that will fit lengthwise within the launch vehicle.

Following the conclusion of last years challenge, our team began to look for methods of improving our existing rover design. In this pursuit, OSRT looked not only to the shortcomings of the rover during the most recent mission, but how teams in other competitions tackled deploying a rover. One source of inspiration was our sister school, University of Oregon. Competing in A Rocket Launch For International Student Satellites (ARLISS) at the Black Rock Desert in Nevada, their objective is similar, albeit limited to autonomous maneuvering. Although there is little supplemental information beyond their blog post, the information provided allowed for basic consideration to the effectiveness of a rover design. As pictured in Figure 74 their rover has had success in maneuvering, offering an additional approach to the problem of wheel design [26].

Figure 74: The University of Oregon Rover [26]

Benefits of this design include:

- Previously proven operational functionality
- Ability to navigate hostile terrain
- Robust enough to survive ejection from launch vehicle mid-flight

Challenges with this design include:

- It is nearly twice as large as what our launch vehicle design allows for
- No onboard terrain sampling instruments
- Low strength collection system
- Terrain present in Alabama is substantially different from that present in Nevada

While the two designs are faced with different tasks, the basic platform of the two rovers are similar, and the combination of these distinct designs and purposes influenced the latest iteration of the OSRT 2019-2020 rover.

The design shown in Figures 75 and 76 illustrates drivetrain, stabilization, and structural designs. The design utilizes two expandable wheels. These wheels can expand from a 6 in. diameter to a 10 in. diameter, giving the rover an additional 2 in. of ground clearance. To stabilize the rover the system utilizes a carbon fiber tail, similar to last year's design. The structure of the rover consists of a single plate extended between the wheels, and motor mounts on either side. As shown, the collection system is most likely going to be mounted on the front of the rover. It is expected that as the collection system extends into the soil the tail will help stabilize the rover. Between the tail is the electrical compartment. Shown in dark blue are two 1300mAh 12 V Turnigy Lithium Polymer (LiPo) batteries, weighing 0.18 lb each. The motors are also shown, each with a rated torque of 7 lb-in., weighing 0.46 lb each.

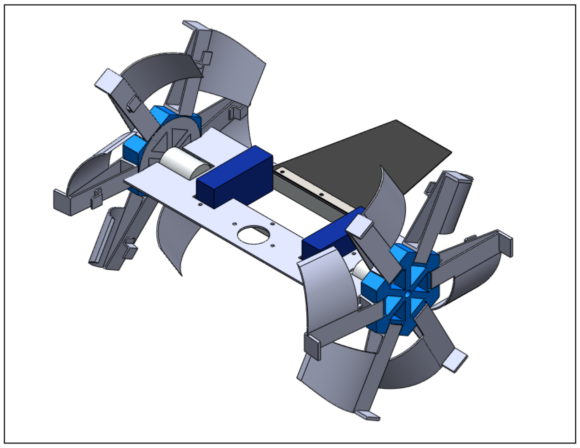

Figure 75: Rover Design 2, Expanded

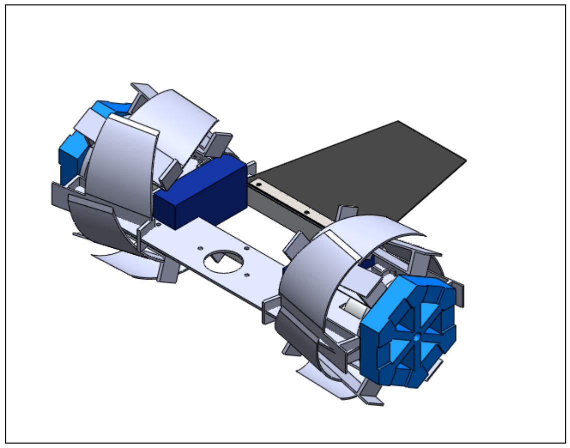

Figure 76: Rover Design 2, Collapsed

Benefits of this design include:

- High torque
- High ground clearance
- Simple mounting structure
- Protection for battery and electronics
- Space for electronics

Challenges with this design include:

- Structure stability/durability
- Wheels take up a lot of space on rover
- Collection system design

The main challenge with this design is the collection system. The system must extend farther due to the expandable wheels. Depending on how the collection device is mounted on the rover, any ground clearance gains could be lost. Another challenge is designing the wheel expansion mechanism to not interfere with the motors, or rover structure. It is unlikely that this design variant will be used, because of the collection system. As a result, future designs will employ a collection device that lies lengthwise along the rover and rotates into position for use.

The design shown in Figures 77 and 78 utilizes a structure made of commonly available polymer pipe, and expandable wheels made from collapsible sheet metal, with supports made from ductile rod instead.

Shown in the top view Figure 78, is the rover batteries and motors. Another important aspect of this design is the orientation of the wheels. Both wheels are canted inward. This design is to help make it easier for the rover to turn. Another potential benefit of canting the wheels is rover control. A common problem when the wheels of a rover are powered by separate motors, is that it is impossible to synchronize the motor movement perfectly. This can make it difficult to drive the rover in a straight line. Two aspects that may help with this problem is keeping wheel speed below 100 rpm, and then canting the wheels inward. The batteries shown in this design are Turnigy 2000 mAh, the motors are the same as listed above.

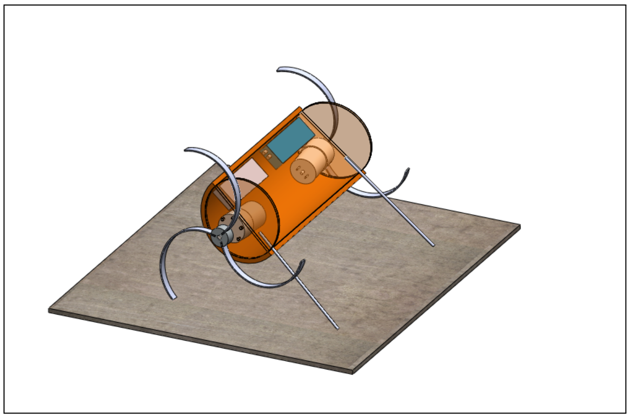

Figure 77: Rover Design 3, ISO

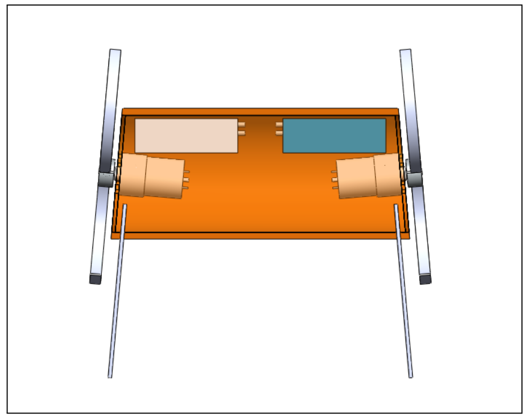

Figure 78: Rover Design 3, Top

Benefits of this design include:

- Easy to manufacture
- Battery and electronic protection
- Expandable wheels
- Durable Structure

Challenges with this design include:

- Finding wheel material that will not yield
- Commercially available tubing does not come in desired sizes
- Tubing comes in high thickness
- Collection system design
- Tubing eliminates ground clearance

One big problem with this design is commonly available Polyvinyl Chloride (PVC), High-density polyethylene (HDPE), and acrylic tubing does not come in the desired thickness and diameter. Thicker tubing eliminates mounting space with the rover, making it hard to fit batteries. Lower ground clearance does present a slight problem, however the wheels are designed for crawling. Meaning that they are not expected to hold the rover off the ground completely,instead they reach in front of the rover grabbing dirt/clay and pulling the rover forward. This design requires further testing and careful material selection to ensure metal does not yield when bent, which will determine the feasibility of the concept. the design also requires more torque than conventional wheels. One benefit of this is that the rover may be closer to the ground, reducing the distance the collection system must travel, though this runs counter to the stated aim of increased clearance.

The final rover design is depicted in Figure 79, Figure 80, and Figure 81. The design features expandable wheels, a carbon fiber tail for stability, and an auger collection system that can be rotated from a stowed position within the rover. The reason for this design is to maximize ground clearance of the rover, preventing the high centering issue previously encountered at competition. The auger collection system was chosen because of its ability to puncture a variety of soil types.This was characterized as a design requirement due to the vague description of soil/sample characteristics, and the unknown conditions at the final competition site.

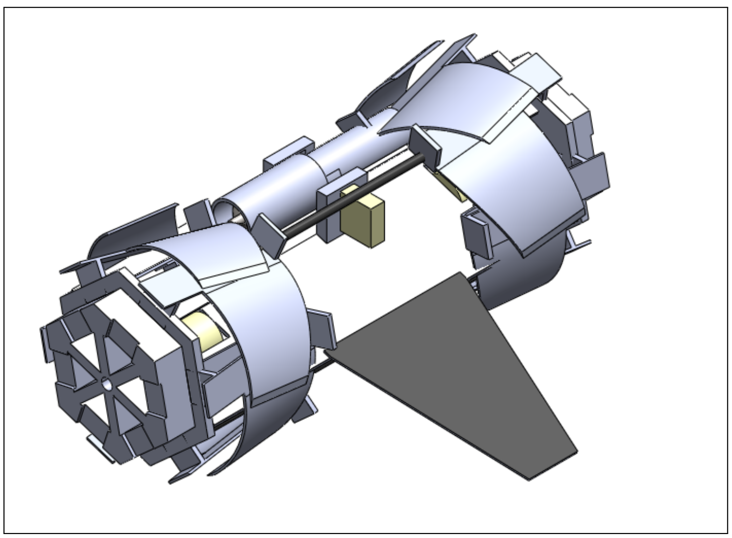

Figure 79: Final Rover Design

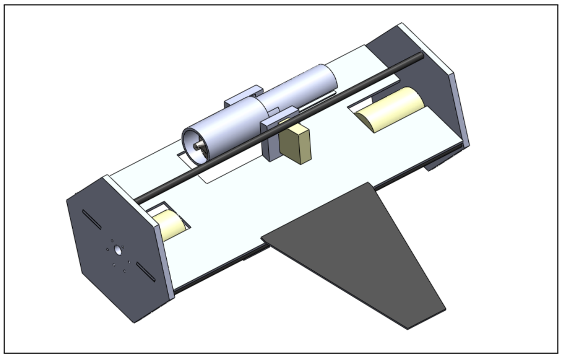

Figure 80: Final Rover Internal Structure 1

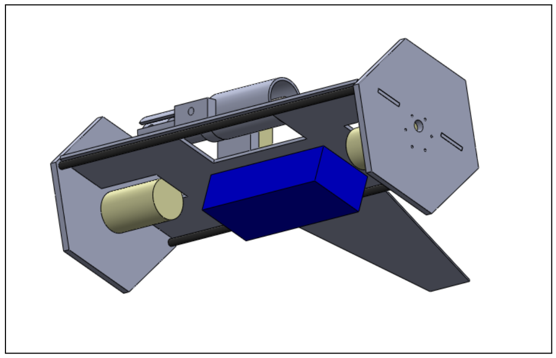

Figure 81: Final Rover Internal Structure 2

Benefits of this design include:

- Large ground clearance
- Light weight, easy to make structure
- Single battery
- Can penetrate variety of soil types

Challenges with this design include:

- Potentially difficult to implement auger system
- No bearing supports on wheels
- heavy wheel design

This design has been selected because it offers the best combination of concepts. The expandable wheels will vastly reduce the likelihood of the rover high centering. The internal structure will be lightweight, easy to manufacture, and it has sufficient room to mount all necessary electronics. Using an auger for collection will allow the rover to dig into the simulated ice sample. By drilling into the ice gradually, instead of using a core sample system seen in Figure 55, OSRT can mitigate the risk of the collection system not puncturing the ground, or requiring so much force to do so that the rover is lifted off the surface by its deployment.

Table 28: Rover Components

Component	Material	Weight(lbf)	Count	Subtotal W
Collection System	Metal	1.56	1	1
Wheels	Acrylonitrile Butadiene Styrene - A Common Thermoplastic Polymer (ABS), Metal, Rubber	2.074	2	4.
Ejection System	Wood, Metal	3.27	1	3
Structure	HDPE, Aluminum	2.52	1	2
Electronics	N/A	0.4	1	0
Hardware	Metal	2	2	
			Total Weight	1

Justification for this design is as follows:

- Drivetrain: Using a dual motor drivetrain provides more torque and makes it possible to easily turn the rover without extra components. This system can have issues with synchronizing the motor speeds. However, keeping motor Rotations per Minute (RPM) down can mitigate this. The system also eliminates the need for a center axle, clearing up space within the rover.
- Wheels: By using expandable wheels ground clearance can be increased beyond the constraints created by the launch vehicle, this comes with the added complexity and weight. However, this is crucial for our mission. The terrain being navigated is largely undefined, as it would be on a lunar ice mission. This design of wheel was selected in favor of the spiral/claw design because this design is more efficient. The claw would be lighter, but it may require more torque and power to work.
- Structure: This structure was chosen for its simplicity, and ease of manufacturing. It is much less rigid than last year's design. However, changes in the ejection system have lowered strength requirements.
- Ejection System: The lead screw ejection system was chosen for its simplicity and straightforward control. Other designs would require stored energy, or more complex systems like the pulley design. The lead screw is both capable of pushing the rover with considerable amount of force, while also having little risk of premature activation/ejection.
- Retention: The rover is retained using a coupler that is pinned to the lead screw, and flush against a bulkhead. This design was chosen because of its simplicity of manufacture and rigidity against forces at the expense of higher motor torque required. Due to frictional forces between the payload bay and the couplers, as well as the mass of the nose cone being ejected by this system, the motor must be chosen carefully and rigorously tested. This added complexity is well worth the benefit of protecting the payload from a premature ejection from the launch vehicle.
- Collection System: The lead screw motor auger system has the highest likelihood of successfully collecting the ice samples. This is because it had the longest reach and can dig itself into the ground and pull the sample back up, requiring less force or payload driving weight than other core sampling designs.

4.5 Rover Ejection & Retention

4.5.1 Ejection and Retention Leading Designs

The current leading design is for an off center lead screw. The payload ejection and retention can be handled by the same on board system, anchored to the payload bay. To control the system, a motor attached to a gearbox will drive a lead screw, which passes all the way through the payload bay, from the bay's aft to fore. The lead screw will be off center to minimize the amount of space taken away from the payload. Attached to the lead screw will be two pushers able to control the movement of two accompanying plates, each on either end of the payload. The plate on the fore end of the payload will be responsible for retention while the launch vehicle is in flight. The plate on the aft end will be responsible for ejection and will be able to physically push the payload out of the housing once landed. When the lead screw activates the pushers, it will cause both pushers to move forwards, moving the payload together. When the fore plate runs out of space on the lead screw, it will simply fall off, leaving room for the payload to be freed.

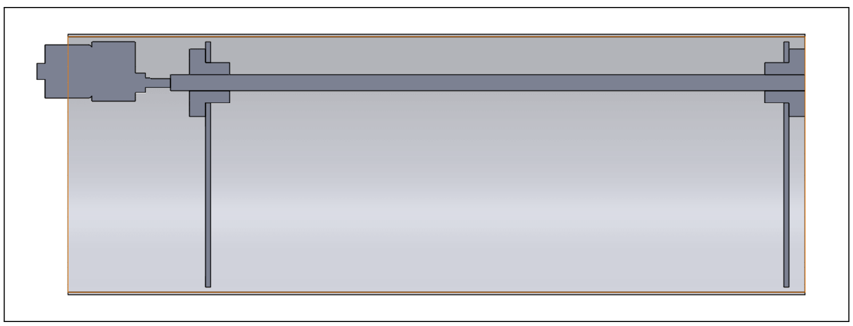

Figure 82: Lead Screw Side View

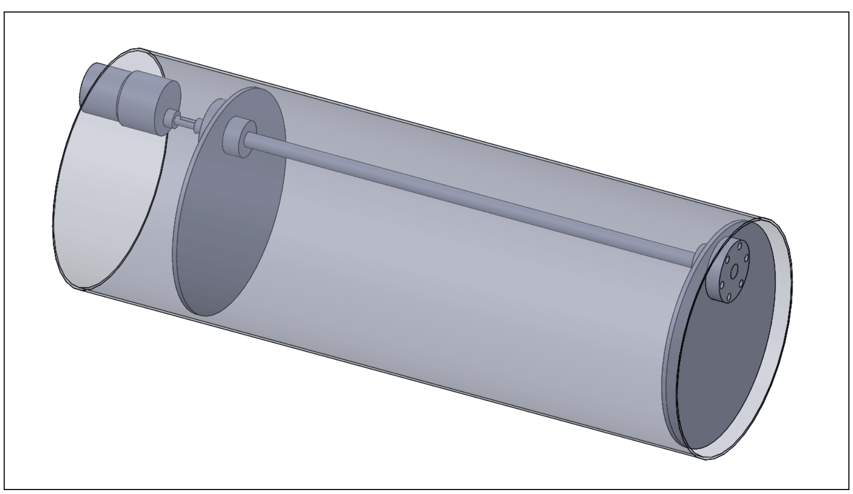

Figure 83: Lead Screw 3D View

The primary advantage of this system, and the reason it is our current leading design is due to its simplicity. By running a lead screw all the way through the payload bay, OSRT avoids the need for a complex method of ejection and retention. Additionally, requiring only one motor means that the ejection/retention system should be relatively lightweight compared to the other leading alternatives. Although the system has limited redundancy, the system itself is robust and resistant to failures. should the system fail, the payload will be stuck inside of the housing with no chance of unexpected early deployment. The solidly mounted front plate insures that there is no way for the payload to leave its housing early without a physical breakage in the metal plate or in the motor's mounting.

The next leading alternative is a "bungee jump" style shock cord attached to the payload which will be ejected during nose cone deployment. The payload will then hang outside of the housing, attached by the cord. The cord would be sufficiently long enough to keep the payload clear of being landed on by the rest of the launch vehicle when touching down. After touchdown either the payload would be able to disconnect itself from the bungee or the housing would disconnect the bungee leaving it to trail behind the payload. The overwhelming disadvantage, and reason that this is not the leading design is that this early deployment brings numerous possible risks to not only the payload but anyone on the ground as well. There is a not insignificant chance that there would be a problem during the midair deployment which causes the payload to be disconnected from the launch vehicle, creating a dangerous situation. If the team is able to find a way to ensure a safe midair ejection which does not involve the use of energetics, then this design may see more development.

The final design under serious consideration is a coiled spring locked into place with a servo. The spring would be released once the launch vehicle has touched down and be able to launch the payload out of its housing. To retain the payload, a solid plate would be held in place in front of the payload, controlled by the same servo which holds the spring. The largest disadvantage for this design is the high weight requirement needed for the spring, along with the needed structural integrity of the chassis of the payload to survive being launched from the spring.

4.5.2 Rover Retention

A critical design aspect of the payload mission is payload retention within the launch vehicle during launch and recovery. The payload not only needs to survive launch and recovery, but more importantly it must not damage the launch vehicle or experience a premature ejection. Arguably, the most important payload system is its retention within the launch vehicle. Seen in Figures 84, and 85 is a detailed view of how the lead screw will physically attach to the motor.

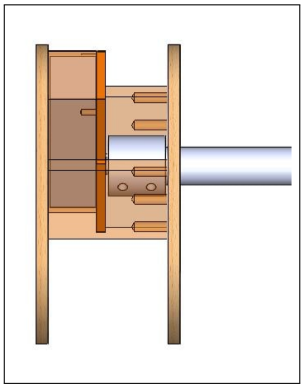

Figure 84: Lead Screw Retention

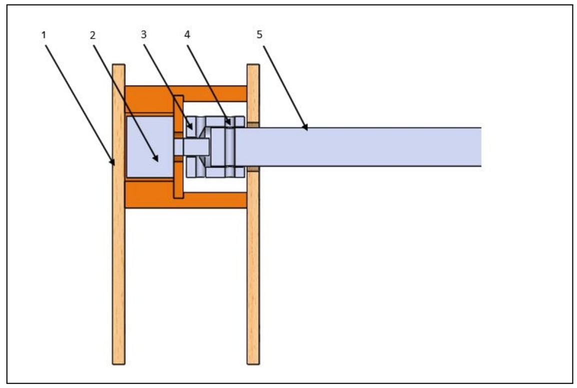

Figure 85: Lead Screw Retention Section

Our design relies on the lead screw to hold the payload within the launch vehicle. The main point of failure with this design is lead screw support. In the event of parachute deployment there will be a jolt to the launch vehicle. The worst case scenario is the payload compartment is facing downward when the jolt occurs. In this case, all the payload weight will be held via the lead screw. The design can not rely on a commercially available lead screw to motor coupler to support the payload. The above figures illustrate a design that will take the weight off of the motor coupler and transfer it to a bulkhead. Seen in Figure 85 there are two wooden bulkheads. Between the bulkheads is a worm gear motor (item 2) that drives the lead screw (item 5). The coupler that connects the lead screw to the motor (item 3&4) has some very important design characteristics. First pointed out by arrow 3 is where the coupler connects to the motor's D-shaft via a set screw. This set screw can not be expected to hold the lead screw and with it the weight of the rover; to account for this, the coupler is designed to have a diameter larger than the through hole that the lead screw fits through. If the lead screw experiences an axial load, then it will push the coupler against the bulkhead; this action will support the lead screw and with it the weight of the rover. The coupler connects to the lead screw using the pin or through bolt at arrow 4.

4.5.3 Lead Screw Calculations

The payload is currently 17 in. long. If a lead screw ejection system is pursued, then the lead screw ejecting the rover must also be 17 in. long. To appropriately size the lead screw motor, two important design constraints must be calculated. First, time required to eject payload must be estimated.

$$EjectionTime = \frac{Length}{(rpm)(Pitch)}$$

Second force provided by the lead screw must also be calculated.

$$Force = \frac{(MotorTorque)(2\pi)(Efficiency)}{(Lead)}$$

Using the following components, with a lead screw efficiency of 73%. It will take 1.68 minutes to eject, and the lead screw will push with a force of 35.4 lb.

- RB-Dfr-673, 6 VDC 160 rpm, 38.89 oz-in Worm Gear Motor
  - Operating Voltage: 3 - 9 V
  - Rated Voltage: 6 V
  - No-load Speed: 160 rpm
  - No-load Current: 40 mA
  - Rated Speed: 128 rpm
  - Rated Current: 250 mA
  - Rated Torque: 4 lb-in.
  - Rated Power: 1.3 W
  - Stall Torque: 15.6 lb-in
  - Stall Current: 1.7 A
  - Reduction Ratio: 1:37.3
  - Weight: 0.363 lb
- Lead Screw
  - Material: 304 Stainless Steel
  - Right handed.
  - Single start : 0.0787 in. (2 mm) per turn
  - Diameter: 0.3 in. (8 mm)

Table 29: Ejection Components

Component	Material	Weight(lbf)	Count	Subtotal Weight (lbf)
Lead Screw	Steel	2.23	1	2.23
Worm Gear Motor	NA	0.36	1	0.36
Motor retention	ABS	0.247	1	0.247
Coupler retention Bulkhead	Wood	0.08	1	0.08
Lead screw nut bulkhead	0.08		2	0.16
Coupler	Steel	0.25	1	0.25
			Total Weight	3.327

4.5.4 Rover Ejection Electronics

To deploy the payload utilizing the lead screw ejection method, an electrical system will need to be implemented. The preliminary design for this system is shown in Figure 86. The system will include a transceiver to receive a signal from a ground station via RF. A microcontroller will be used to compare the signals received by the transceiver with a keyword which enables rover ejection. The microcontroller will also enable a motor driver, powering the ejection motor. The final major portion of the ejection electronics is a circuit including a photoresistor to detect light entering the airframe to verify that the rover has been fully ejected.

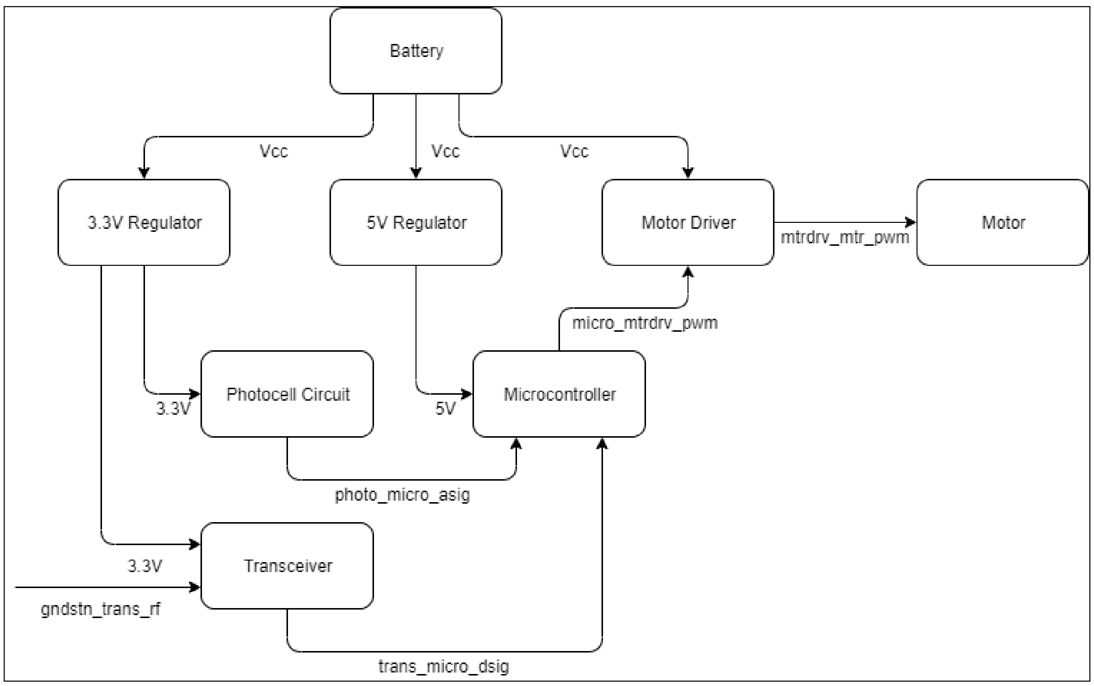

Figure 86: Rover ejection Electronics Block Diagram

4.5.4.1 Rover Ejection Electronics DDMs

For each component discussed in the Section 4.5.4 overview a DDM was made to ensure the correct components are chosen for their respective tasks. The first of these, Table 30, looks at options for microcontrollers. An emphasis is put on size of the microcontroller for this system because of limited space in the rover ejection section of the launch vehicle. General Purpose Inputs and Outputs (GPIO) and clock speed are less important to this system because the system has few blocks and none of the tasks the system performs are time sensitive.

The leading alternative for microcontroller in this system is the Teensy 3.2. This is a board with enough IO for this application and a smaller form factor than the Teensy 3.6, which was featured heavily in previous year's designs. Although it has one of the slower clock speeds of any of the options which were considered, it will be plenty fast enough because none of the operations performed by this system are time sensitive.

Table 30: Rover Ejection Microcontroller DDM

Design		**Teensy 3.6**		**Teensy 3.2**		**Arduino Pro Mini 328**		**Teensy 4.0**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score	Score	Rating (1-5)
GPIO	2	5	10	3	6	1	2	3	6
Size	4	3	12	5	20	4	16	5	20
Power Draw	5	3	15	5	25	5	25	2	10
Clock Speed	2	3	6	2	4	1	2	5	10
Total		**43**		**55**		**45**		**46**	

The next component in this system with multiple options to explore is the transceiver. Three options for this component are compared in Table 31. High value has been assigned to the software support category because success in the payload mission is contingent on the signal being received correctly every time, making ease of use a critical feature. The range category is also weighted heavily because the signal must always be received; if the team decides to pursue a team requirement that the rover mission must be performed from a distance, then having the transceiver operate from a far range will be important. Availability is being considered because one of the options has a long manufacturing lead time, which could hinder progress on the project.

Table 31: Rover Ejection Transceiver DDM

Design		**XBee-PRO 900HP**		**Adafruit RFM69HCW**		**Linx HUM-900-DT**	
Requirement	Weight	Rating (1-5)	Score	Rating (1-5)	Score	Rating (1-5)	Score
Range	4	5	20	4	16	1	4
Size	2	3	6	3	6	5	10
Software Support	5	5	25	2	10	2	10
Availability	3	5	15	4	12	1	3
Total		**66**		**44**		**27**	

The leading alternative for transceiver is the XBee-PRO 900HP. It is easy to program, use,and has been proven to receive signals at long ranges using a directional transmitting antenna.

4.5.4.2 Battery Size Selection

To determine the appropriate size for the battery a power budget was created. The power budget takes into consideration the two operating cases, idle power draw and ejection power draw. During the idle period, the motor, motor driver, and photoresistor circuit will not be considered. During this period the only thing happening is the transceiver and microcontroller waiting for the ejection command to be received. During the ejection phase, every element of the block diagram will be active. For our calculations, OSRT will be assuming a worst case scenario of six hours idle time and three minutes active ejection time. Table 32 shows the results of the power budget calculations.

Table 32: Rover Ejection Power Budget

Component	**Current (mA)**	**Voltage (V)**	**Battery Current (mA)**	**Active Time (minutes)**	**Total Power (mAh)**
Teensy 3.2	45	5	32.51	363	190.65
XBee-Pro 900HP	215	3.3	99.37	363	601.19
Motor Driver	1	3.3	0.46	3	0.02
Motor	5	25	2	10	2
Photoresistor Cricuit	5	15	4	12	1
Total			**381.35**		**804.36**

The chosen battery must meet two requirements, each of which is highlighted in Table 32. The battery must be able to continuously supply 381 mA, the maximum current draw of the system. The battery must also have a capacity of at least 804 mAh to last for the duration of the mission. For these reasons, the battery will have at least 1 Ah capacity to allow for margin of error. The battery will also be a LiPo battery because they offer excellent volumetric energy density.

4.5.4.3 Photoresistor Circuit Design

The photoresistor circuit will consist of a voltage divider, dividing 3.3 V over the photoresistor and a fixed resistor of known value. The voltage between these resistors, Vout, is then read by the microcontroller. This circuit is shown in Figure 87. Photoresistors are at their highest resistance in complete darkness and reduce their resistance in the presence of light. This means that the output of the voltage divider will be a low value when the rover has not been ejected, and once the rover has been ejected, the output voltage will rise.

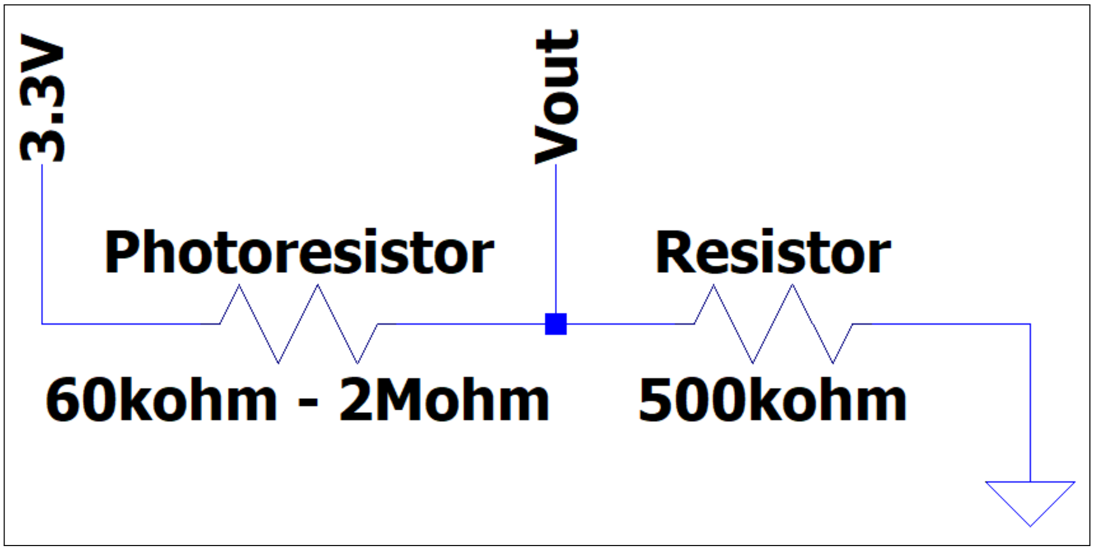

Figure 87: Photoresistor Circuit Schematic

The photoresistor has a resistance of 2 MΩ when not exposed to light and 60kΩ when exposed to bright light. Therefore, Vout varies from 0.66 V in darkness to 2.95 V in light, which will be easily detectable by the microcontroller.

4.6 Rover Structure

Last year, the OSRT used a scaffolding design for the rover structure. A benefit of this design was its high strength to weight ratio. A downside was that the design consumed a lot of space. This presented challenges with regards to mounting electronics, motors, and batteries. The design also gave the rover a very small amount of ground clearance, which resulted in the rover high-centering in last year's competition. The previous design was necessary to ensure that the rover could survive ejection from the launch vehicle via black powder, however, due to changes in the ejection system, this is no longer as much of a concern. Due to this change in the payload requirements, the OSRT will be pursuing a lighter structure design for payload. The team's goal is to simplify the design, improve maneuverability, lower weight, and increase the mounting space on the structure itself. To do this, a variety of designs have been investigated. These designs will be discussed in further detail in this section of the report.

The design shown in Figure 88, and Figure 89 illustrates one possible design for the rover structure. This design consists of three carbon fiber rods that extend the length of the rover. These rods are connected to two plates on either side of the rover. A HDPE plate also extends between the sides of the structure. This plate serves as a mounting surface for the a 7500 mAh battery, electrical components, which are not shown, and the collection system. The collection system is shown in its stored configuration. This design allows the collection system to rotate downward when it is ready to be used. The carbon fiber helps support the structure of the rover. This structure uses a carbon fiber tail to support the rover during operation.

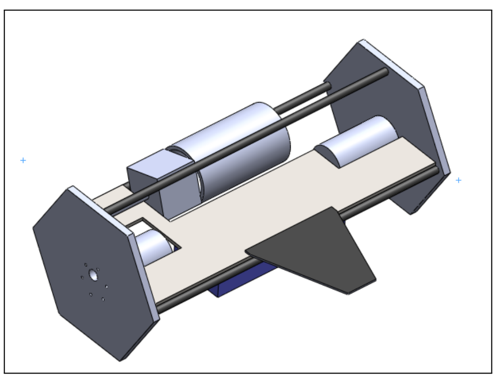

Figure 88: Rover Structure 1, ISO

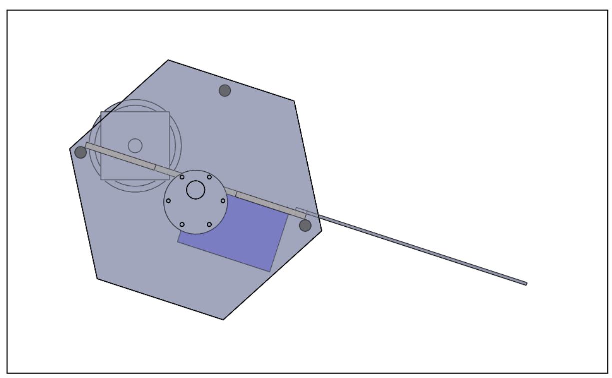

Figure 89: Rover Structure 1, Side

Benefits of this design include:

- Rigid design
- Space for electronics
- Ground clearance
- Low weight

Challenges with this design include:

- No bearing supports for motor
- Mounting holes in HDPE weaken rover structure
- No current protection for electronics
- Collection system is currently a tight fit

The main challenge with this design is fitting the collection system. If the expandable wheels are pursued, then the collection system can no longer reasonably sit fixed in the rover. To combat this, a method to rotate the collection system is being designed so it can be stored horizontally in the rover. Storing this system between the motors is difficult and takes up a lot of space. It also has to be mounted rigidly, as to prevent the system from breaking when in use. Another concern is radial loading on the motors, this design has the wheels being supported by the motor shafts. One favorable aspect of this design is it uses a single high capacity battery. This battery has exceptional storage capacity and is still relatively small, however it is still larger than the 1300mAh batteries incorporated in other designs.

The design shown in Figure 90 uses a single HDPE sheet extended between two HDPE disks for motor mounting. In the middle of the structure is a pan meant to hold the electronics. This pan will either be made out of sheet metal, or be 3D-printed. The blue blocks represent 1300mAh batteries, and the two cylinders represent motors. The main purpose behind this design is manufacturability. Each component is either 3D-printed, made out of a metal or HDPE sheet. Sheet metal or polymers are incredible easy to machine, meaning OSRT can easily build prototypes and make backup components.

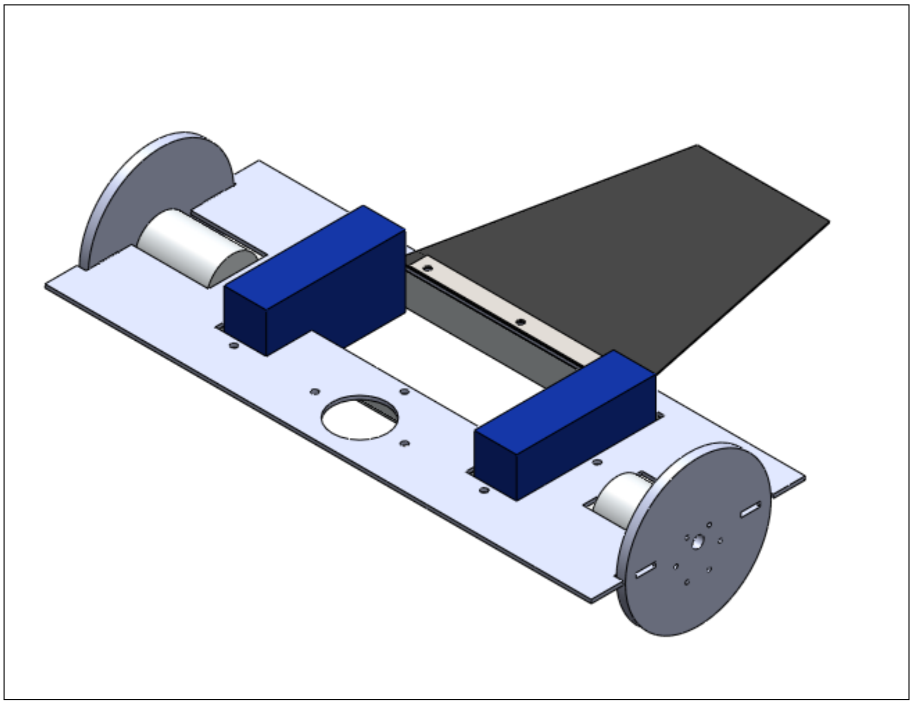

Figure 90: Rover Structure 2

Benefits of this design include:

- Easy to make
- Space for electronics
- Low weight
- Protection for electronics

Challenges with this design include:

- No bearing supports for motor
- No current protection for electronics
- Mounting collection system

The main difficulty with this design is figuring out how to mount the collection system to the structure. The current collection system designs are 5-7 in. long, meaning that they can not be rigidly mounted pointing toward the ground. As a result, this design is not feasible for the rover.

The structure design shown in Figure 91 utilizes commercially available tubing. The structure is designed to be made out of polymer like HDPE, ABS, or acrylic. The lower section of the design is meant to be made out of a commercially bought pipe, cut in half. Fitted to the top is a releasable section that is made out a polymer sheet. The bottom section is meant to protect any electronics and batteries.

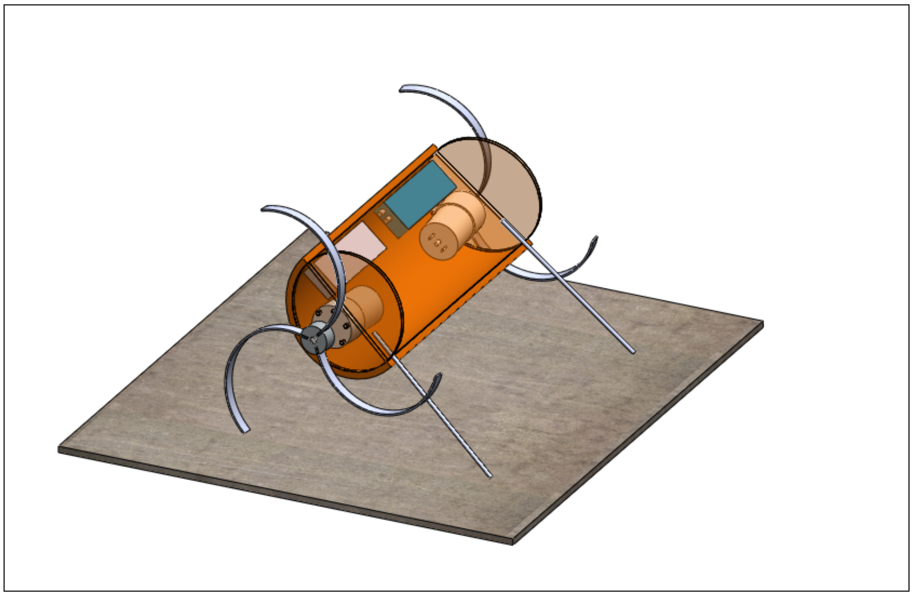

Figure 91: Rover Structure 3

- Easy to make
- Protection for electronics

Challenges with this design include:

- Piping is not readily available in desired sizes
- Pipe is excessively thick limiting space, and increasing weight
- Limited ground clearance

The main challenge with this design is piping simply is not available in desired sizes and thicknesses. It also eliminates ground clearance. It is easy to manufacture, but it is simply not applicable to the required application.

Table 33: Structure Components

Component	Material	Weight(lbf)	Count	Subtotal Weight (lbf)
Motor Mounts	HDPE	0.181	2	0.363
Struts	Carbon Fiber	0.04	3	0.1206
Mounting plate	HDPE	0.804	1	0.04
Motor	NA	0.46	2	0.92
Battery	NA	1	1	1
Tail	Carbon Fiber	0.081	1	0.081
			Total Weight	2.52

4.7 Payload Electronics

There are several possible configurations of electrical systems involving various motors to control the collection system. Depending on the collection system chosen, a different array of motors will be chosen. A simplified Block diagram is located in Figure 92. This shows the on-board electronics for the system.

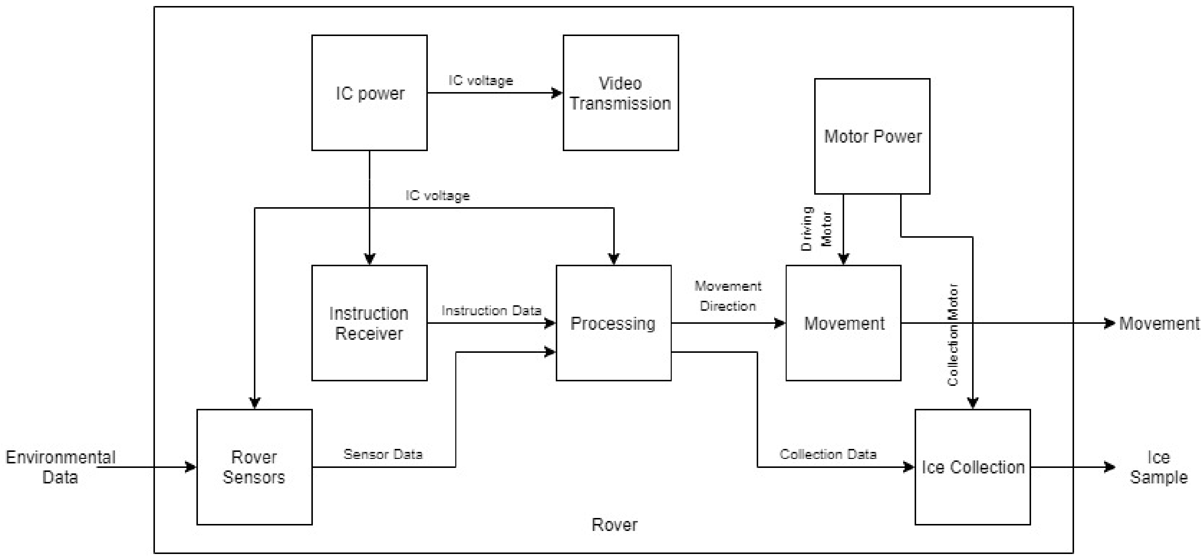

Figure 92: Block diagram of the electronics system

4.7.1 Rover Control System

In order to achieve this year's mission of maneuvering to one of the designated sample ares to collect samples. There are multiple remote control systems that our team has discussed regarding maneuvering of the rover:

- Fully Autonomous
- Manual Radio Control
- Semi-Autonomous and Radio Control

The first control system design was a fully autonomous rover maneuvering system. This design allows the rover to reach the designated area without human intervention. This system design will rely on the GPS coordinates of the designated areas and the GPS coordinates of the current rover. The system will need to implement an algorithm that allows the rover to be guided by the GPS coordinates solely to approach the designated area. The downside of this design is that the GPS location does not provide precise enough data to navigate to the exact position of the designated area, it can be anywhere within 10 meters of the designated area.

Pros:

- No human intervention
- Self-maneuvering

Cons:

- Not accurate enough GPS Location
- No control over rover if system fails

Figure 93: Image of the remote controller chosen for the project [18]

Another system design is the manual radio control system. This design allows engineers to control the direction and functionalities of the rover by sending and receiving radio frequencies. A radio frequency transceiver will be implemented on the rover and on the Remote Control (RC) control in order to send and receive data over wireless. The human controller must be able to see the rover physically in order to control the rover in the correct direction that it supposed to be heading in.

Pros:

- More certainty on heading in the correct direction
- Fully under control

Cons:

- Human controller must be in sight
- Continuously waiting for signals

The third system design that the team discussed was the semi-autonomous and radio control system, which is also our current system of choice. This system is a combined system of the autonomous and RC system that were mentioned above. The system will initial as an autonomous system which is guided by the glsGPS coordinates of the designated area. After the rover has reached the designated area, human controller will determine whether or not the rover needs extra steps and directions on getting closer to the designated area. If yes, the rover did not reach to the designated area close enough, then human controller can manually send RC signals to the rover to maneuver it to the right spot.

Pros:

- Minimize human intervention
- More certainty on location
- Human controller is in full control in case autonomous system fails

Cons:

- More power consumption

4.7.2 Rover Camera

To achieve the goal of reliable and real time video transmission, OSRT needs a camera and data transmission system that enables real-time video transmission over the range of a half mile, which is the estimated distance from the base station to the designated sample areas. After researching on the types of video transmission and robotic cameras, there were three cameras considered for this task.

The first choice is the Serial Port Digital Camera. This is a Universal Asynchronous Receiver-Transmitter (UART) Serial Port Through-the-Lens (TTL) Camera Module With Communication Protocol. Its image resolution is 640x480 and it has a night vision mode. The Vision Distance goes up to about 65 ft in daylight and up to 33 ft at night time. It communicates with a microcontroller directly. This means an additional microcontroller is needed to enable video transmission for this camera.

Pros:

- Low power Consumption
- Reasonable resolution
- Fair cost

Cons:

- Low range of transmission
- Requires additional microcontroller

The second choice is the Micro 2.4 GHz Wireless Camera. This camera uses radio frequency as transmission protocol. Its image resolution is 640*480. Its range of transmission indoors is about 33 ft and for outdoor transmission is about 330 ft. It operates on 2.4 GHz frequency and when recording video, its power consumption is 500 mA. However, the cost of this camera is extremely high, with the cost of a single

camera at$299.00.
Pros:

- Reliable data transmission
- Reasonable resolution
- Small size
- No additional microcontroller needed

Cons:

- Low range of transmission
- High cost

The third choice is the Wolfwhoop Micro First-Person View (FPV) 5.8 GHz Camera, which is also the best choice for our system. This camera uses radio frequency as transmission protocol and operates at 5.8 GHz frequency. Transmitting at 5.8 GHz, the video transmission will be less likely to interfere with other wireless communications that commonly transmit at 2.4 GHz. Its image resolution is 640*480. Its range of transmission for outdoor transmission is about 2,300 ft. When recording video, its power consumption is between 330 mA to 700 mA.This camera is small in size and light in weight. It is a popular drone camera that is commonly used due to its low cost at reliable video transmission.
Pros:

- Reliable data transmission
- Reasonable resolution
- Small size
- No additional microcontroller needed
- Long transmission range
- Reasonable cost

Cons:

- Weather dependent
- High power consumption

Figure 94: Image of the FPV camera chosen for the project [27]

Design		Micro FPV Camera 5.8 GHz		Micro 2.4 GHz Wireless Camera		Serial Port Digital Camera	
Requirement	Weight	Rating	Score	Rating	Score	Rating	Score
Transmission Range	7	5	35	2	14	2	14
Quality of Video	6	3	18	3	18	2	12
Power consumption	5	3	15	3	15	4	20
Transmission rate	7	4	28	2	14	2	14
Size	3	4	12	2	6	4	12
Cost	3	4	12	1	3	4	12
Total		120		70		84	

Table 34: Camera DDM

4.7.3 Rover Video Transmission

This year one of the rover's missions is to drive to one of the designated sample areas to collect samples. Since our team agrees on implementing a semi-autonomous control system that allows the rover to reach the designated are by utilizing GPS coordinates, our team wants to implement video transmission for the rover, so that the direction and the graphics of where the rover is heading can be seen. The video transmission feature also provides more remote freedom to our system as the human controller does not need to physically see the rover to control it when human intervention is needed. The team wants to be able to transmit video data from the rover to the base station with a distance of 0.5 miles.

There were multiple wireless video transmission protocols available:

- Radio Frequency
- WiFi
- FPV Radio Frequency

RF signals have the longest range of transmission. At a frequency of 2.4 GHz, they can send and receive signals up to multiple miles. This fully covers the range of operation that OSRT are looking for. However, RF signals were not designed for video transmission. It is very high in latency when transmitting video data which does not provide real time video data.

Pros:

- Long transmission range

Cons:

- Not suitable for video transmission

WiFi signals have the shortest range of transmission. At a frequency of 2.4 GHz, they can send and receive data for up to 300 ft. WiFi is reliable for transmitting video data, it is one of the most commonly used protocols for video transmission. It has low latency and is able to provide real time video data to the receiver. Pros:

- Reliable and stable video transmission

Cons:

- Short transmission range

FPV has a mid-range of transmission. At a frequency of 5.8 GHz, FPV can send and receive data for up to 2km. FPV is reliable for transmitting video data, it is the most commonly used protocol for drone video data transmission. It has low latency and is able to provide real time video data to the receiver. Pros:

- Reliable video transmission
- Reasonable range

Cons:

- High power consumption

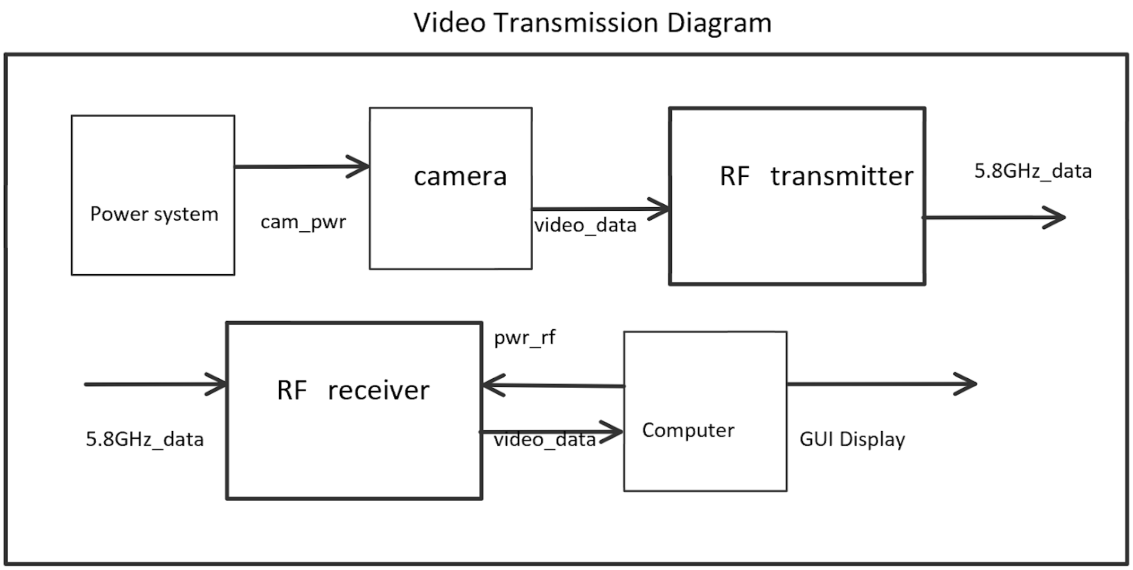

Figure 95: Block diagram of the video transmission system.

4.7.4 Rover Sensor System

The sensor system that the rover needs in order to accomplish its tasks include a magnetometer and a GPS. The magnetometer is used for spatial direction, it measures the magnetic field of the rover to determine the direction and relative change in the magnetic field. Its output data has three axes that indicate the rotation of the rover, this data allows our team to know about whether or not the rover is upright. The GPS will send the current location data of the rover to the base station so that the team knows where the rover is located remotely.

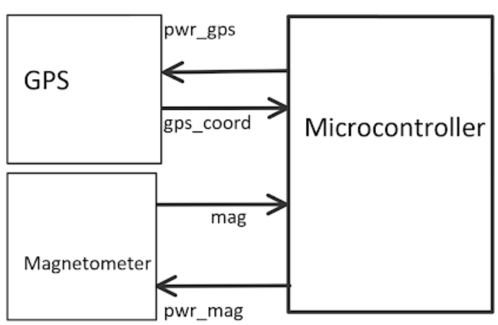

Figure 96: Block diagram of the sensor system.

4.7.5 GUI for Rover System

Since our team will be implementing video transmission for the rover, there needs to be a GUI that displays the video data and other sensor data that the rover will be sending to our base station. The GUI will have a video section, a GPS coordinates section, an auger position indicator, and data from the magnetometer. The battery life display was taken out in a later version of the GUI as it would require another sensor adding weight and more drain on the battery. To make up for this, tests will include timing the battery life under multiple conditions to have a good idea of the length of the battery life of the rover in the field. The GUI should be formatted orderly and user friendly. The team has chosen C# as the programming language to use for the GUI.

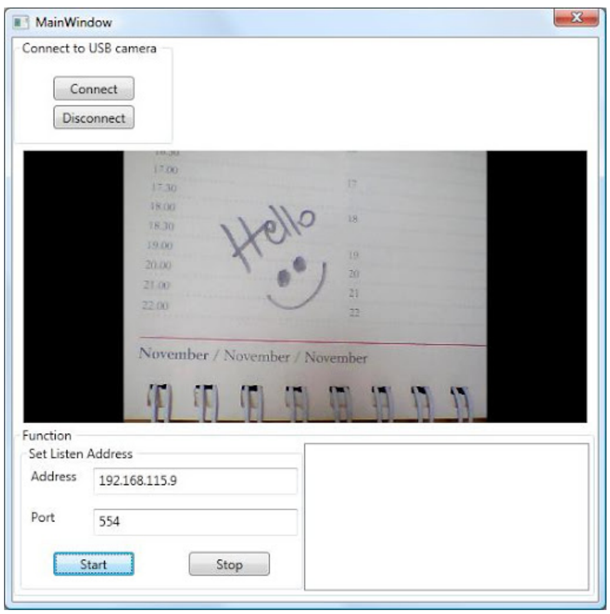

Figure 97: Illustration example of GUI for Payload [25]

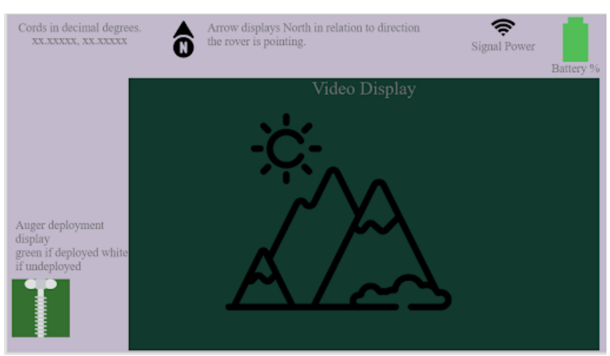

Figure 98: Illustration example of another GUI for Payload [9][17][20][10]

Figure 99: Illustration final example of a GUI for Payload [9][20][7]

4.7.6 Microcontroller Design

4.7.7 Motor Driving Electrical

There are several optional driving configurations. The motors cannot be driven directly from the microcontroller because they require a minimum of 11 Volts of input, it is necessary to use an H-bridge motor driver and a 3.3V signal from the microcontroller to drive 12 volts to the motors. This works since the H-bridge will use the smaller input voltage to turn on a transistor, enabling a higher output voltage. This meets the necessary requirements.

There are several commercial motor drivers that are available to drive motors.

There is also the concern of linear versus switching regulators. Linear regulators will consume more power as they change the voltage from 12 V to 3.3 or 5 V. They will introduce less noise than switching regulators. This means that they are ideally suited for the devices that are most sensitive and involve a fast signal transmission. Since the sensors all require 3.3 V, they do not draw much current, especially if they are supplied from the 5V supply. The sensors connected to 3.3V will also be connected to analog ground. This will further reduce the chance of any significant signal interference.

A switching regulator should be used for the 5 V supply since the devices which require 5V are not as sensitive to noise, with the exception of the radio transceiver. The Teensy 3.6 has internal voltage regulators. These regulators take the input voltage down to the necessary voltage for the internal microcontrollers. If an alternative design is chosen with a different microcontroller, then the input voltage would become more noise sensitive. The 5V supply will also draw more current with the camera requiring approximately 0.330 A. The transceivers will take approximately 0.1 A during maximum power consumption, and around 0.05 A during normal operation. The Teensy will take approximately 30 mA, and a maximum of 250 mA.This amount of current makes it impractical to use a linear voltage regulator step down from 11V. Therefore, a switching regulator to step down the voltage will reduce the power consumption.

In order to minimize the power consumption of the power regulation circuitry, the five volt switching regulator will step down the voltage from 11 V to 5 V. The 5 V source will then power the 3.3 V supply. This the minimal power consumption from power. The power budget for this layout is in Table 35.

4.7.7.1 Rover Power Budget

Device	Current (mA)	Voltage (V)	Power (W)
Motors	500	12	6
GPS	30	3.3	0.01
Transceiver	100	5	0.5
Camera	330	5	1.65
Microcontroller	10	5	0.05
IMU	3.2	3.3	0.01056
		Total	8

Table 35: Power Budget

4.7.8 Schematics

It now makes sense to describe the schematics for the chosen design. Each image describes the circuitry for a single block. The decisions and alternatives will be discussed for each schematic Figure 100.

Figure 101 shows the design for the microcontroller board. A Teensy was selected as the microcontroller for this design. As a team, it might become necessary to design a custom microcontroller breakout, depending on memory limitations. The 3.3V regulators on the Teensy are shown as being used in this schematic. The Teensy can supply approximately 250 mA, which is enough to power the selected sensors. The Teensy itself is powered directly from the 5V voltage regulator. This is a switching regulator. The capacitor and inductor values are selected to minimize the noise on the output lines. According to common convention, 0.1 micro Farad capacitors are used on the power supply lines. These capacitors should further reduce the noise on the power supply lines. The values are also set to the maximum current consumption of the system on the 5V line.

$$I_c = 740mA$$

$$I_m = 250mA$$

$$I_t \simeq 1A$$

This leads to the resistor, capacitor, and inductor values indicated. The filter is set to a corner frequency of approximately 200kHz. This should minimize the amount of ripple. The voltage divider on the enable pin is such that it will enable output with minimal current consumption. The same resistor value was chosen to keep the voltage at the EN pin consistent, since this is easier by using resistors from the same batch. A separate analog ground is used to isolate some of the signals from each other. This will ensure that analog ground will have minimal noise on it from the switching regulator. The analog and digital grounds will be connected at one point to ensure that they are at the same potential, but minimize the eddy currents changing the ground voltage.

The motor driving for both the wheel motors and the motor on collection on the auger is shown in Figure 102. The motor driver selector is a motor driver meant to supply 12V to the motors. It is connected to screw terminals which will join the motors to the board. The motor drivers are configured such that they take two inputs for each motor. An example truth table is shown in the schematic. These outputs are given by the microcontroller as 3.3V signals. The 3.3V signals will enable the 12V output on the motor drivers. The motor drivers are also configured such that they can sense over-current situations, and prevent blowouts.This will hopefully add a layer of protection and redundancy to the system. The noise that will be introduced on the 12V line from the motors should remain isolated from the analog ground. This will protect the sensors from any unseen voltage perturbations.

The sensors for the Rover are included in Figure 103. These include a GPS, an IMU, and the power connector to the remote camera. The GPS selected is the same used for avionics, and the schematic used is the same. It is connected to 3.3V, and analog ground. This is done to minimize the interference on the GPS lines which should reduce the error from such interference. Resistors are included on signal lines to clean the output, and prevent ringing. Furthermore, it will allow the flexibility of switching resistors in later designs if another value is needed to match the impedance of the signal. This method is used on other high speed signal lines on the circuitry. A low resistor value is chosen, but if a resistor later proves to be necessary then it can be shorted. Another benefit to such a resistor is to add additional ESD protection around the signal pins.

The transmission system is included in Figure 104. This uses a transceiver, which can send the GPS and IMU data back to the rover control station. It will also accept user input from the ground station and selected rover controller. The sensor data sent over will be in packets with a checksum and simple acknowledgement. This will enable the GUI to only display accurately transmitted sensor information. It will also ensure that the transceiver will not blast the air with RF interference, and that the transceiver will be able to receive control instructions. The control instructions will contain another checksum and acknowledgement. This will act as an assurance that the rover does receive all instructions, but does not respond to incorrect instructions. It will also prevent the rover from responding to any other noise on the frequency band from other signal sources. To further prevent such interference, each packet will have a prequel and ending that will enable either transceiver to quickly identify a packet's validity from other signals. It will also ensure that the rover and GUI are listening, since minimal processing will occur if a packet is immediately identified as invalid.

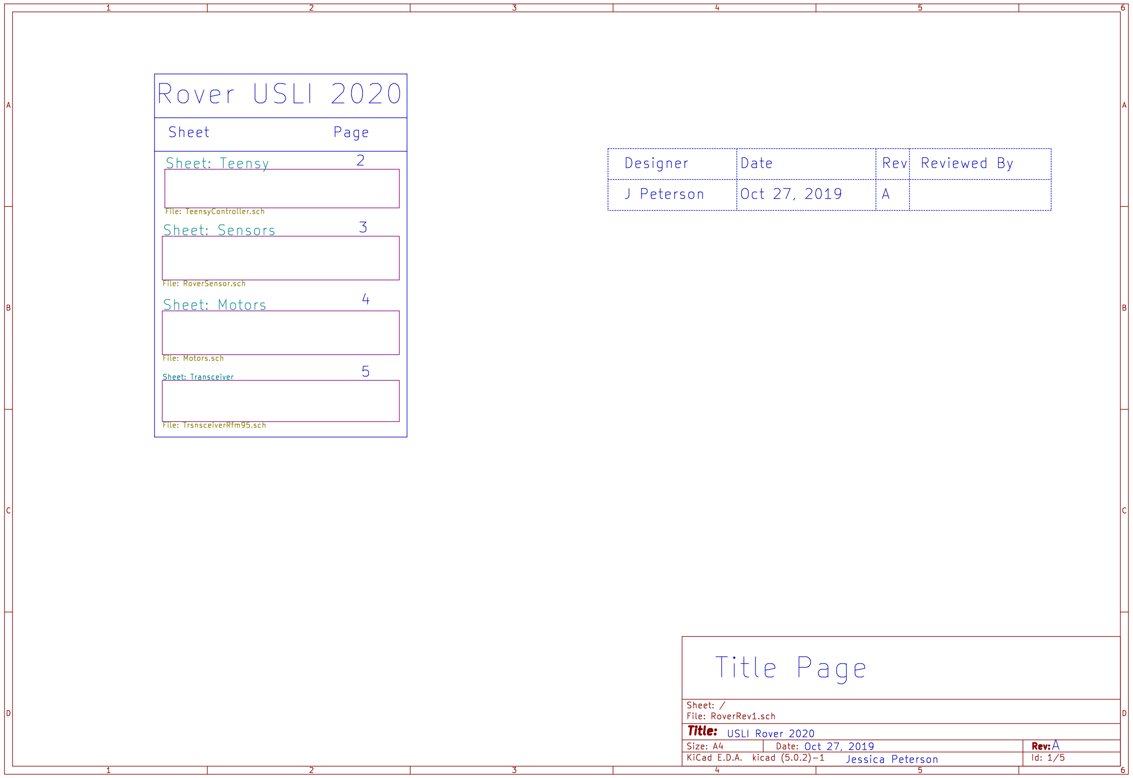

Figure 100: Rover Schematics

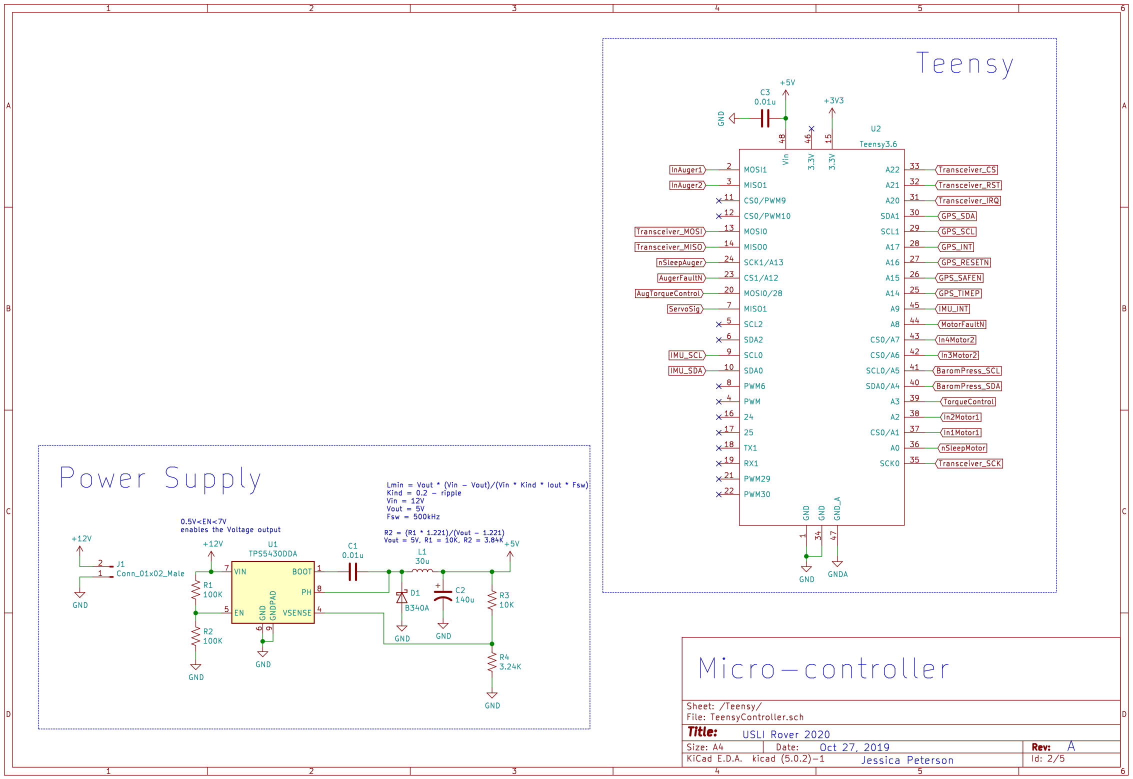

Figure 101: Rover Schematic Microcontroller block

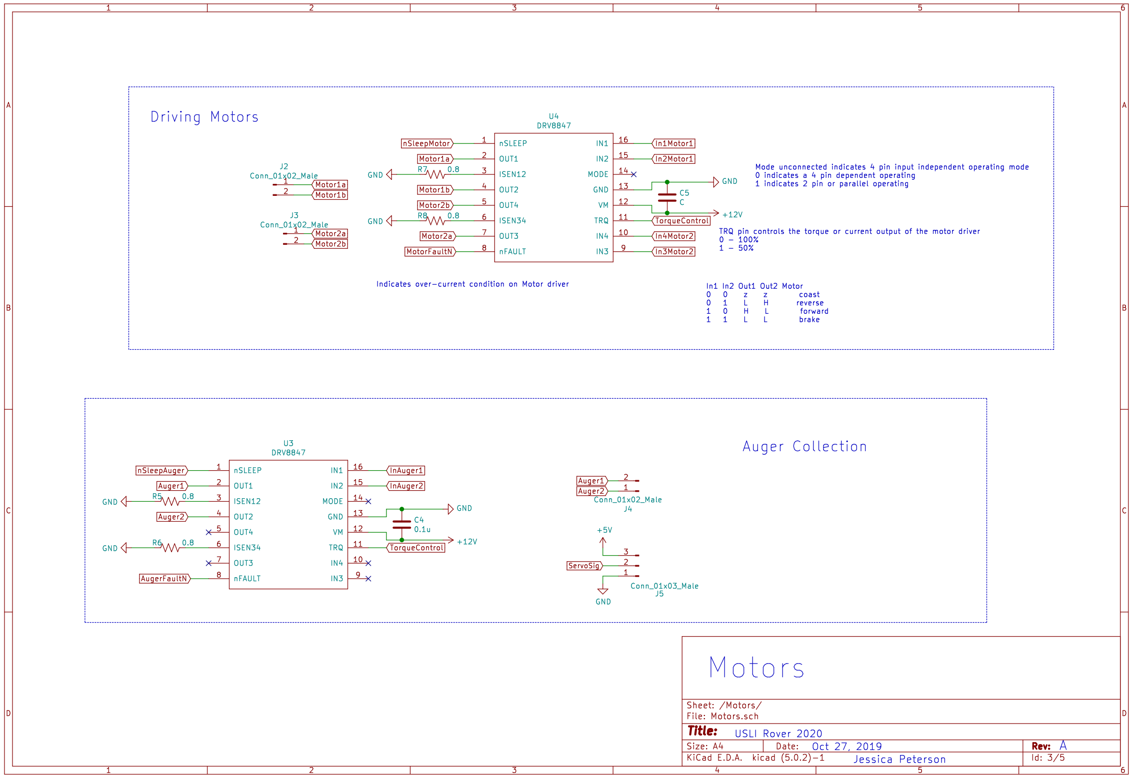

Figure 102: Rover Schematic Motor block

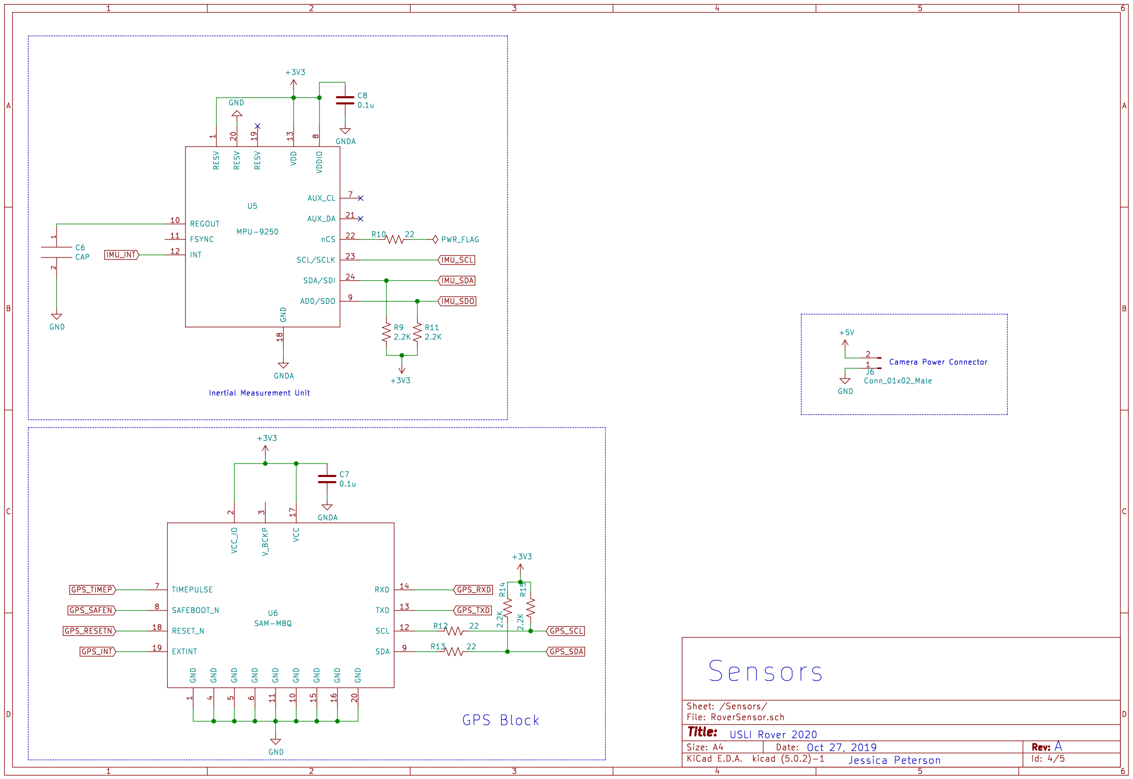

Figure 103: Rover Schematic Sensor block

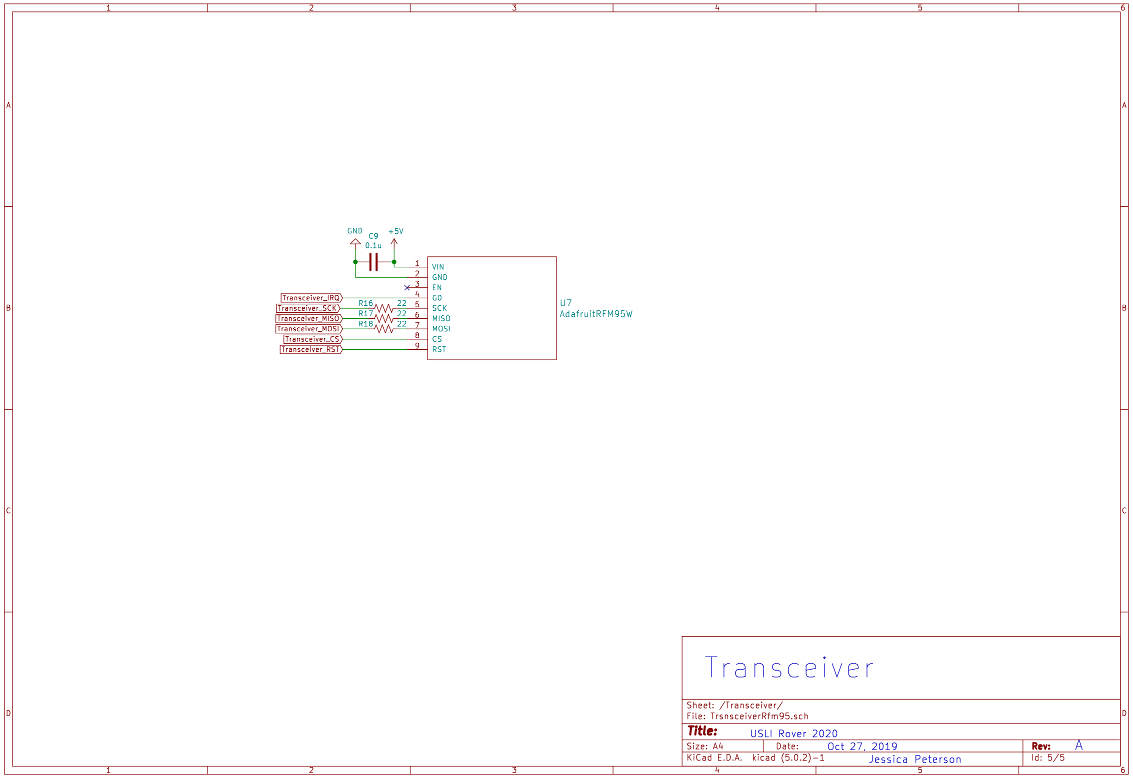

Figure 104: Rover Schematic Transceiver block

5 Safety

5.1 Personnel Hazard Analysis

Mitigation of dangers to the launch vehicle, environment, and project timeline are irrelevant if the safety, comfort, and efficiency of personnel working on the project cannot be assured. Any loss of team members to debilitating hazards, either from physical injuries, exposure to toxins, or other incident will impact all other aspects of the project by restricting available skillsets or delaying work, and threatens the reputation of OSRT; therefore, the safety of all personnel involved in every stage of the project must be of utmost importance, and all safety incidents must be deemed unacceptable regardless of severity.

For the Personnel Hazard Analysis in Table 37, each hazard has been assigned a probability and severity, as detailed in Table 36, and mitigation strategies focus on reducing the severity or frequency of the potential incident by recognizing contributing factors and high complexity of the project, with many distinct steps and different locations in which work is being performed. The overall severity and frequency of unmitigated events are ranked on a scale from 1 to 10, with 10 being an event frequent enough to happen in nearly all work sessions, or severe enough that the impacted personnel may be critically or even fatally injured. Even hazards with a frequency or severity as low as 1 must be monitored, since these risks often have a compounding effect, in which one hazard going unchecked will increase the severity or frequency rating of others.

In many instances, observation of critical rules and requirements previously identified by the safety team are sufficient to fully mitigate the frequency of hazards, for example in the proper storage of launch vehicle motors outlined by National Fire Protection Agency (NFPA) 1127 and the ignition wire installation and operation procedure National Association of Rocketry (NAR) requires. Adapting these practices for other hazards will create a consistent mitigation technique that maximizes the safety of personnel by being familiar to implement and easy to recall. An example of this adaptation process is in the lockout processes used for avionic electronics: using the same practices NAR requires to prevent active ignition wires can be used to prevent electric shock in other subcomponent operations.

Table 36: Risk Assessment Code (RAC)

	Severity				
Probability	**1 - Catastrophic**	**2 - High**	**3 - Moderate**	**4 - Low**	**5 - Negligable**
A - Frequent	1A	2A	3A	4A	5A
B - Probable	1B	2B	3B	4B	5B
C - Occasional	1C	2C	3C	4C	5C
D - Remote	1D	2D	3D	4D	5D
E - Improbable	1E	2E	3E	4E	5E

Table 37: Personnel Hazard Analysis

Risk	Cause	Effect	Pre-RAC	Mitigation Strategy	Mitigation Verification	Post-RAC
Laceration	Sharp and/or rough Edges left on fabricated parts. Sharp metal chips in machine shop from improper clean up	Injury	3C	Team members will wear protective gear and will clean workplace thoroughly	Avoiding lacerations	3D
Heatstroke	Long-term exposure to heat in field work or poorly ventilated workshops	Permanent brain damage	2C	Hydration, paired/supervised work, education on warning signs	Non-dry skin, functional communications	1D
Frostbite	Low temperature exposure during field work or in composite freezers	Dysfunctional physical coordination	1D	Proper clothing, weather forecast data	No shivers or slow speech or mumbling	1D
Hypothermia	Low temperature exposure during field work or in composite freezers	Dysfunctional physical coordination	1D	Proper clothing, 'no-go' thresholds for extreme weather	No shivers slow speech or mumbling	5D

Continued on next page

Table 37 – continued from previous page

Risk	Cause	Effect	Pre-RAC	Mitigation Strategy	Mitigation Verification	Post-RAC
Epoxy Fumes	Concentrated fumes from binding agents are highly toxic from improper handling of epoxy	Damage to lungs	2C	Ventilated spaces, restrict epoxy use to designated spaces as per Material Safety Data Sheet (MSDS)[13]	No coughing or hard breathing	5D
Particulate Detonation	Fine particles like sawdust may detonate when exposed to flame	Eye damage and burns	2D	Clean workspaces regularly, keep areas ventilated, control ignition sources, wear safety glasses as per OSU's Machine Shop Safety (MSS) rule 1	No burns or eye injuries	3D
Sleep Deprivation	Insufficient rest disorients workers and increases frequency of other hazards	Difficulty concentrating	2B	Reasonable work expectations, paired/supervised work, self care expectations	Alert and responsive team members	4C
Tripping	from falling over discarded objects or exposed cables	Bruises or other injuries	3C	Keep workspaces clean, walk site before use to note danger regions	No fall related injuries and clean floors	3D

Continued on next page

Table 37 – continued from previous page

Risk	Cause	Effect	Pre-RAC	Mitigation Strategy	Mitigation Verification	Post-RAC
Falling Objects	Dislodged tools or material could strike workers	Severe injury or damaged launch vehicle parts	3C	Secure all materials	Undamaged team members and/or launch vehicle parts	3D
Car Accident	Traffic incidents can range from minor delays to sever injury	Property damage, injury and or fatalities	1E	Well-rested drivers, OSRT transportation in good repair and known weather forecasts	Safe transportation with no car damage	3D
Saw Blades Human Incision and/or damage to launch vehicle items	Improper education for saw blade use	Fatalities and/or permanent manufacturing damage, accidental severing or cuts into worker extremities	1D	Check safety guards, only trained operators of tools as per OSU's MSS rule 13	Mint condition materials, tools and team members	4D
Premature Ignition	Faulty charge wiring; shear pins do not break	Accidental firing of launch vehicle motors could strike personnel with high-speed debris or exhaust	3C	NFPA storage rules, firing circuit lockouts, no open flames near launch vehicle	Flight	3D

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Table 37 – continued from previous page

Risk	Cause	Effect	Pre-RAC	Mitigation Strategy	Mitigation Verification	Post-RAC
Pinch of fingers or skin when securing fasteners, seating couplers, or closing bulkheads	Extreme, unpredictable spasticity	Bruising or scratches visible on skin	4B	Proper tool usage, clear handholds or grip points, use multiple people where needed	able-bodied fingers	5D
Spray Paint	Inhalation of toxic fumes	Incapacitation of user which could lead to hospitalization	2C	Ventilated spaces and use of masks as per product instructions and MSDS	No coughing or hard breathing	4E
Ladder Falls	Falling as a result of slips or ladder instability	Uncoordinated and well thought-out physical movements	2C	Limit solo ladder use, follow printed instructions	Contact with falling parts does not result in physical injury	3D
Slips	Falling as a result of fluid spills	Injury which could be serious if around dangerous equipment	3C	Announce and clean spills immediately, avoid icy surfaces, and wear proper footwear	Ground has proper friction for comfortable walking	4D

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Table 37 – continued from previous page

Risk	Cause	Effect	Pre-RAC	Mitigation Strategy	Mitigation Verification	Post-RAC
Struck by Launch Vehicle	Improper structural integration and flight path analysis. Falling debris or uncontrolled launch vehicles carry significant kinetic energy	Fatality	1E	Stay alert, listen to Range Safety Officer (RSO), maintain clear range boundaries	Unharmed team members and spectators	5E
Electrical Shock	Discharge of firing circuits or other onboard electronics	Mild to moderate electrocution of nearby persons	2D	Lockout-tagout procedures, insulated wiring, and inspect batteries and connections	Devices may be shaken without connections coming loose and devices run without creating arcs	4E
Mental Health	Consistent high-stress environments can cause attrition of personnel	Depression, anxiety, nervousness	3B	Emphasis on team cohesion, work-life balance and self care	Stimulated contributions and positive engagement in team morale	4D
Chemical Burns	Improper education on handling chemicals. Skin contact with solvents and other caustic chemicals	Burning of skin	2D	Consult MSDS sheets, know chemical shower and washing procedures where necessary	Unharmed upon exposure to chemicals	4C

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Table 37 – continued from previous page

Risk	Cause	Effect	Pre-RAC	Mitigation Strategy	Mitigation Verification	Post-RAC
Sunburn	Ultraviolet (UV) exposure during extended field work	First or Second degree burns	3B	Sunscreen, shaded awnings, and hats	Skin is not red	5D
Splinters	Wood fragments lodged in skin can cause infections	May cause infection	3C	Use gloves and sand down edges of wood	Fingers may run across surface without getting splinters	5E
Ergonomic Strain	Back, wrist, and finger strain from awkward angles, lifting, and exerting excessive force on shrouded parts	Injury, sore muscles and joints	3C	Subcomponent accessibility, correct tool for given task, proper lifting and carrying techniques	Suitable body mechanic motion to remain uninjured	5E
Uneven Terrain	Holes, ruts, and unstable rubble	Could cause sprained ankles, falls, or twisted joints	3C	Move slowly, wear correct footwear, and use established paths	No injuries due to uneven terrain	4E
Wildlife Attack	Field work may involve regions with wild animals, including possible rabies vectors	Animal inflicted wounds and possible infection	2D	Avoid deep foliage, wear long pants, do not run	Unharmed team members	4D

Table 38: Environmental Hazard Analysis

Risk	Cause	Effect	Pre-RAC	Mitigation Strategy	Verification Mitigation	Post-RAC
Motor catches fire	Direct exposure extreme to sunlight.	Vehicle catches on fire, becomes inoperable, and fails competition.	1E	Store motors and explosives away from any and all heat sources.	There will be a trailer stationed at the launch site. This will contain secure containers for motors and explosives away from heat sources.	1E
Launch vehicle goes past radius of 2,500 ft	High winds past 20 mph at apogee.	The vehicle flies well beyond its calculated drift, It is unrecoverable and unsalvageable.	1D	Launch rail should be adjusted to go against current calculated wind speeds. Will follow all launch procedures and safety checklists prior to ignition as per NAR High Power Rocket Safety Code (HPRSC) 7 and Tripoli Rocketry Association, Inc. (TRA) 6-5.	Launch rail will be angled against the wind direction to counteract wind speeds.	1E

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Table 38 – continued from previous page

Risk	Cause	Effect	Pre-RAC	Mitigation Strategy	Verification Mitigation	Post-RAC
Launch vehicle damage	High winds during descent.	Vehicle drifts into obstacles and/or the ground.	3C	Ensure large area of clearance around launch site as per as per NAR HPRSC 7 and TRA HPRSC 6-5. Range Safety Officer (SO) cancels flight under high wind conditions as per TRA High Power Safety Code (HPSC) 1-1.2 .	Develop launch safety requirements and checklists.	3D
Electrical component failure	High humidity, rain, or lightning as per TRA HPRSC 6-4.	Electronics fail, posing safety hazard and subsequent systems failure	2D	Water-resistant enclosure of each relevant subsystem cancel flight if conditions permit	Systems tests and checklists, launch if weather permits	2E

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Table 38 – continued from previous page

Risk	Cause	Effect	Pre-RAC	Mitigation Strategy	Verification Mitigation	Post-RAC
Loss of visibility	Inclement weather reducing visibility.	Difficulty visually tracking launch vehicle during flight and after touch down. Safety hazard for mid-air systems failure.	2C	Verify weather conditions for launch day as per TRA HPRSC 6-5.	Cross reference Federal Aviation Administration (FAA), NAR, TRA , and NASA launch safety requirements.	2D
Electrical ignition failure	Rainy or humid conditions.	Vehicle does not launch.	2C	Use water-resistant enclosed wiring.	Ground test in wet conditions.	2D
Structure and external component malfunction	Exposure to rain or snow.	Material properties or functions are altered and loss of structural integrity.	1E	Build structure with water-resistant materials and test wet conditions of deployment systems.	Pre-flight checklists and data from inclement weather simulation tests.	1E
Unexpected or excessive weather-cocking or launch rail failure	High winds, as per TRA HPRSC 6-5.	Trajectory alters toward horizontal, becomes a safety hazard, and recovery/deployment failure	3C	Analysis through calculation and simulation to ensure stable flight	Redundant calculations and simulations. Launch only in accordance with safe flight conditions of FAA, NAR, TRA, and NASA	3D

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Table 38 – continued from previous page

Risk	Cause	Effect	Pre-RAC	Mitigation Strategy	Verification Mitigation	Post-RAC
Launch vehicle descends outside of launch range and collection radius	High winds, as per TRA HPRSC 6.5, or early recovery deployment.	Violation of competition rules, loss of line of sight with vehicle, loss of tracking, and a difficult recovery.	2D	Adjust launch rail to preemptively counter the effects of wind during flight.	Current wind measuring prior to launch, and vehicle allowed to descend as far as possible before deployment of main recovery parachute.	2E
Improper motor burn	Humidity, rain, direct sunlight, or other inclement weather, such as what is detailed in TRA HPRSC 6-4.	Motor does not reach projected altitude, and loss of altitude points.	3D	Project altitudes within margin of safety and ensure safe storage of motor.	Design motor system to reach higher than projected apogee and allow BEAVS to control final apogee.	3E
Improper apogee variance control	Varying or extreme air density, humidity, disparity in drag calculations used to control BEAVS.	BEAVS does not adjust altitude properly and loss of altitude points.	3D	Use various weather and air conditions to program BEAVS system.	Ensure that BEAVS works in various conditions or allows for modifications based on launch day weather measurements.	3E
Hazardous waste leak	Battery malfunction or broken component leaks chemicals.	Exposing surrounding flora and fauna to hazardous materials.	3D	Inspect and test all batteries prior to use.	Completion of required safety checklists.	3E

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Table 38 – continued from previous page

Risk	Cause	Effect	Pre-RAC	Mitigation Strategy	Verification Mitigation	Post-RAC
Fire started upon motor ignition	Motor ignition flame spreads to surrounding brush.	Brush wildfire presents safety hazard and immediate environmental damage.	2C	Clear flammable brush surrounding launch pad and launch in isolated area clear of all large brush.	Verification of safe launch requirements as set forth by the FAA, NAR, TRA, and NASA.	3D
Recovery deployment failure	Failure in deployment system, insufficient deployment forces	Uncontrolled vehicle landing and jettison of parts/debris.	1C	Ground testing of all recovery systems paired with extensive calculations prior to launching.	Consistent successful ground tests and successful prototype launch.	1D
Jettison of wadding	Insufficient securing or enclosure of wadding.	Spreading of wadding material into surrounding environment.	4B	Use wadding that is biodegradable and not harmful to environment and collect jettisoned debris.	Line of sight maintained with jettisoned material such that it can be collected.	5B

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Table 38 – continued from previous page

Risk	Cause	Effect	Pre-RAC	Mitigation Strategy	Verification Mitigation	Post-RAC
Vehicle collision with structure on descent	High winds, as detailed in TRA HPRSC 6-5 or early recovery deployment.	Damage to structure and/or surroundings as well as launch vehicle.	2D	Ground test of recovery systems and avionics.	Pre-launch checklists to ensure system functionality and verification of safe launch requirements as set forth by the FAA, NAR, TRA, and NASA.	2E
Injury or death to animal/wildlife	Animals within launch zone are struck by falling debris.	Injury or death of the animal.	2E	Launch conducted in area most likely away from wildlife, and active monitoring of surroundings at launch site.	Verify clear range prior to launch.	2E
Hazardous material in launch debris	Improper motor burn or excessive expelling of shrapnel or fuel leak from the motor.	Fire hazard for launch and surrounding brush and chemical hazard to wildlife and people.	2C	Inspection of motor systems prior to launch, inspection of launch zone after launch, and safe disposal of debris if necessary.	Pre-launch checklists to ensure system integrity.	2D

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Risk	Cause	Effect	Pre-RAC	Mitigation Strategy	Verification Mitigation	Post-RAC
Parachute ejection energetic or motor explosion	Malfunction of energetic system and motor retention failure.	Explosion and expulsion of debris as shrapnel, as well as fire and safety risk.	1C	Extensive ground testing of ejection energetics, all energetics systems checked prior to installation.	Pre-launch checklists, safety verification checklists prior to installation.	1D
Expulsion of debris mid-flight	Improper fastening of launch vehicle and constituent parts.	Debris littered in area surrounding launch site.	3C	Fastener and hardware securing checklist	Redundant checks and verifications pre-flight.	3D
Launch vehicle breaks apart	Zippering or insufficient structural integrity.	Vehicle breaks apart, dispersing debris at launch site surroundings.	2C	Structural integrity calculations, simulations at event extremes, designed and implemented structural integrity measures.	Redundant calculations, positive taper of airframe by incrementally increased carbon fiber layer depth.	2E
Improper disposal of waste	Team members not utilising proper garbage disposal procedure at launch site.	Exposes wildlife and landscape to litter and garbage.	4C	Have defined garbage disposal locations/procedures, known by all team members.	Visual inspection of launch site prior to setup and before departure by team leaders.	4E

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Table 38 – continued from previous page

Risk	Cause	Effect	Pre-RAC	Mitigation Strategy	Verification Mitigation	Post-RAC
Destruction of environment due to airframe retrieval	Failure of recovery systems or erroneous flight.	Foliage and ground cleared in effort to retrieve launch vehicle from ground after high speed impact	3A	Ensure nominal flight and recovery by following all checklists, procedures, and applicable rules and laws.	Successful launches due to adherence from checks and safety guidelines, with paperwork available for flight readiness verification.	3D
Fire to surroundings and launch vehicle	Motor fails to fully ignite, and therefore, launch vehicle does not leave launch rail.	Brush fire safety hazard to wildlife, environmental damage, launch vehicle damage.	1D	Have fire extinguishers on-hand in event of fire, checks and verifications during motor build.	Safety checklists and pre-launch verifications.	2D

5.1.1 Failure Modes & Effects Analysis

Risk Priority Number (RPN) is a the product of severity, occurrence, and detection. Higher scores in this category indicate a greater effect of the given failure mode.

Table 39: Structures Failure Mode Effects Analysis (FMEA)

System	Failure Mode	Failure Effect	Failure Cause(s)	Severity	Occurrence	Detection	RPN	Mitigation
Nose Cone	Non-Uniform, non-straight nose cone	Unpredictable Flight Path, Increases drag	Damaged nose cone	8	1	8	64	Inspection before and after the use of nose cone as described in checklists required by Requirement 1.2.
	Nose cone detaches during flight.	Flight failure, loss of nose cone, and loss of avionics.	Tip is not properly secured and/or not correctly attached to recovery system.	8	1	5	40	Follow installation checklists methodically. Inspect launch vehicle before sealing using checklists created in compliance with Requirement 1.2.
Airframe	Delamination of airframe materials from temperature.	Launch vehicle is not recoverable or reusable.	Long term storage in unsuitable temperature conditions	9	1	6	54	Choose proper size by calculating thermal stress of materials for long term use in compliance with Requirement 6.1.
	Airframe buckles from high stress.	Launch vehicle is destroyed.	Recovery system fails to deploy.	9	1	8	72	Modify safety factor for adjustments in launch vehicle assembly as described in Requirement 6.1.
	Zippering along the edges from shock cord	Destruction of the launch vehicle	Shock cords pulling across edge of tube	8	2	8	128	Ensure CO_2 and/or black powder charges are properly sized according to Equations 7 and 8.

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Table 39 – continued from previous page

System	Failure Mode	Failure Effect	Failure Cause(s)	Severity	Occurrence	Detection	RPN	Mitigation
Fins	Fins are misaligned.	Unpredictable flight pattern or loss of launch vehicle causing damage to surroundings.	Damage of fins from shipping and handling and/or hard landing. Fins not cured and/or aligned perfectly.	7	3	5	105	Inspect fins before and after launches. Handle fins carefully. Use an OSRT designed fin-alignment guide as per Requirement 6.12.
	Fins fall off	Unpredictable flight pattern or loss of launch vehicle causing damage to surroundings.	Insufficient amount of epoxy and/or cured improperly. Forces cause epoxy failure. Insufficient amount of epoxy and/or epoxy cured improperly.	8	3	6	144	Inspect fins before and after launches. Handle fins carefully. Use fin-alignment guide as per Requirement 6.12.
Coupler	Overall Vehicle is bent	Loss of launch vehicle. Unable to use launch vehicle again.	Bending forces from harsh landing or forces experienced from improper integration.	8	4	3	96	Composite layers will be thick in regions that will accommodate higher stresses and achieve a safety factor as per Requirement 6.1.
Bulkhead	Premature ejection	Destruction of airframe.	Shear pins released before ejection	9	4	9	324	Adequate number of shear pins to satisfy Requirement 6.1.

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Table 39 – continued from previous page

System	Failure Mode	Failure Effect	Failure Cause(s)	Severity	Occurrence	Detection	RPN	Mitigation
	Fracture	Internal components damaged and unrecoverable launch vehicle	Internal components damaged and unrecoverable launch vehicle	8	2	2	32	Select plywood from reputable sources, making sure the plywood is thick and has large washers. Bulkheads will be shown to exceed safety factor as per Requirement 6.1.
Threaded Rod	Fracture	Loss of recovery system and launch vehicle	Force is greater than what the strength of the rod is designed for	9	4	5	180	Design the size of threaded rod to a suitable factor of safety as per Requirement 6.1.
Shear Pins	Shear pins fail to break	Loss of vehicle and recovery system does not deploy	Insufficient pressure is created to break shear pins	10	3	5	150	Have black powder ejection charges more powerful than primary ejection charges and test all charges with ejection test.
E-Matches	Poor E-match connection	Loss of coupler detachment	E-match does not light	10	2	6	120	Inspect E-match and wiring thoroughly before integration and launch using checklists as per Requirement 1.2.

Table 40: Recovery FMEA

System	Failure Mode	Failure Effect	Failure Cause(s)	Severity	Occurrence	Detection	RPN	Mitigation
			CO_2 Ejection System					
Solenoid	Solenoid Fails to actuate	Pin does not puncture canister	Wire Disconnection	8	2	4	64	Inspect the wiring to ensure that all leads are tightly secured using checklists per Requirement 1.2.
	Solenoid fails to retract	Canister cannot vent due to blockage	Pin jams in canister	8	1	7	56	Test solenoid extensively before flight using checklists per Requirement 1.2.
	Solenoid is pushed back by the force of escaping CO_2	Altimeter breaks	CO_2 canister vents too quickly	7	2	8	112	Adequate space will be given to allow the canister to vent without impacting the solenoid and electronics as described in Section 3.4.
	Solenoid fails to reach CO_2 canister	Pin does not puncture canister	Poorly dimensioned casing and pin	8	1	8	64	Ensure that the solenoid can get the pin to reach the canister in the housing as described in Section 3.4.
	Solenoid plunges the pin into the canister multiple times	CO_2 venting slowed	Electronics error giving multiple pulses of electricity	3	1	5	15	Ensure through testing that the solenoid will only actuate and retract once.

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System	Failure Mode	Failure Effect	Failure Cause(s)	Severity	Occurrence	Detection	RPN	Mitigation
	Solenoid becomes dislodged	Pin does not puncture canister	Launch Vibrations	8	3	8	192	Casing will be manufactured to adequately hold the solenoid. If needed, set screws will be implemented and thread locking liquid will be used to ensure that the solenoid does not fall out as per Requirement 6.1.
	Solenoid shaft bends	Solenoid is nonoperational	Pin and solenoid misalignment	8	3	2	48	Test solenoid actuation at least once immediately before flight using checklists per Requirement 1.2.
	Solenoid shaft breaks	Pin does not puncture canister	Pin and solenoid misalignment impact from other object	8	2	9	144	Ensure solenoid is appropriately aligned and is protected from impacts using checklists per Requirement 1.2.
			CO_2 venting makes the shaft brittle	8	3	9	216	Ensure solenoid is insulated and can survive CO_2 venting temperatures through testing.

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System	Failure Mode	Failure Effect	Failure Cause(s)	Severity	Occurrence	Detection	RPN	Mitigation
	Solenoid does not deliver enough force to puncture the canister	Canister cannot vent	Solenoid selection	8	1	2	16	Ensure solenoid can provide the force needed to puncture the canister through testing.
Pin	Pin gets stuck in the canister	Canister cannot vent due to blockage	Set screw failure	8	4	2	64	Inspect the pin to ensure that it is tightly secured onto the solenoid shaft using checklists per Requirement 1.2.
	Pin fails to puncture canister	Canister cannot vent	Pin tip failure	8	5	5	200	Ensure pin is sharp enough before each canister puncture using checklists per Requirement 1.2.
	Pin falls off of solenoid shaft	Pin does not puncture canister	Launch Vibrations	8	5	8	320	Inspect the pin to ensure that it is tightly secured onto the solenoid shaft using checklists per Requirement 1.2.
			Condensation	8	1	2	16	Ensure solenoid shaft and pin are dry and at ambient temperature before integrating into the launch vehicle using checklists per Requirement 1.2.

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System	Failure Mode	Failure Effect	Failure Cause(s)	Severity	Occurrence	Detection	RPN	Mitigation
	Pin becomes misaligned	Pin does not puncture canister	Launch Vibrations	8	3	4	96	Inspect the pin to ensure that it is tightly secured onto the solenoid shaft and ensure that the solenoid is secured in the casing using checklists per Requirement 1.2.
	Pin breaks	Pin does not puncture canister	Force from the solenoid	8	3	8	192	Manufacture pin so that it is strong enough to sustain the forces induced by the solenoid as per Requirement 6.1.
	Pin is too short	Pin does not puncture canister	Resharpening the pin results in it shortening too much	8	5	8	320	Ensure the pin is in a length range that can satisfactorily puncture the canister as described in Section 3.4.
Canister	Canister explodes	Parachutes deploy prematurely	Extreme heat	8	2	9	144	Insulate the canister against temperatures exceeding the maximum probable temperature as per Requirement 6.1.
	Canister shears	Canister is destroyed	Force applied while screwing it into the casing	8	1	9	72	Ensure that the canister is not screwed in too tightly using checklists per Requirement 1.2.

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System	Failure Mode	Failure Effect	Failure Cause(s)	Severity	Occurrence	Detection	RPN	Mitigation
	Canister leaks	Canister is rendered useless	Puncture due to impact of another object	8	1	1	8	Ensure that there are no loose parts that can damage the canister using checklists per Requirement 1.2.
	Canister breaks free of the casing	Canister flies around the parachute bay and potentially ejects itself.	Failure of the casing	10	1	9	90	Ensure the casing is strong enough to support the venting canister as per Requirement 6.1.
Casing	Casing becomes dislodged	Casing becomes free in the parachute bay, only tethered by the solenoid's power cord	Launch Vibrations	5	3	8	120	Ensure that bolts are tightened and thread locking liquid is applied using checklists per Requirement 1.2.
		Casing travels throughout parachute bay, potentially ejecting itself	CO_2 Venting	10	2	9	180	Ensure that bolts are tightened and thread locking liquid is applied using checklists per Requirement 1.2.
	Casing becomes brittle	Casing cracks due to forces applied by bolts and launch vibrations	CO_2 Venting	6	2	2	24	Coat interior of casing in a thermally insulating paint and ensure casing is not brittle using checklists via Requirement 1.2.
Recovery System								

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System	Failure Mode	Failure Effect	Failure Cause(s)	Severity	Occurrence	Detection	RPN	Mitigation
Parachute	Parachute tears	Parachute rips reducing drag and increasing speeds landing too fast	Parachute tear	8	3	4	96	Check parachutes after every launch using checklists as per Requirement 1.2.
	Parachute breaks off shock cord	Parachute rips off, causing launch vehicle to plummet at terminal velocity	Break in shock or shroud lines	8	3	3	72	Ensure connections between shock cords and parachutes maintain a safety factor in compliance with Requirement 6.1.
	Parachute tangles	Parachute does not fully deploy, the parachute is tangled drag reduced	Packing bag issue	6	3	3	54	Pack parachutes as defined in checklists per Requirement 1.2.
Quick link	Quick link breaks	Parachute breaks off, drag reduced vehicle hits at high kinetic energy	Quick link ultimate strength issue	7	3	3	63	Calculate impulse prior and select quick links in compliance with Requirement 6.1.
Tender descender	Tender descender fails to release	Main parachute does not deploy. Bag stays in bay vehicle falls at high speeds	Tender descender e-match or failure	9	4	4	144	Check tender descenders and e-matches using checklists as per Requirement 1.2.
E-match avionics	E-match fails to ignite	Main parachute does not deploy. Bag stays in bay vehicle falls at high speeds	Tender descender e-match or failure	9	4	4	144	Check e-matches and avionics using checklists as per Requirement 1.2.
Drogue Parachute	Drogue parachute fails to deploy	Drift becomes exponential speed increases vehicle lands too hard	Ejection charge fails to ignite and/or parachute becomes tangled.	7	4	5	140	Check ejection charges and parachutes using checklists per Requirement 1.2.

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System	Failure Mode	Failure Effect	Failure Cause(s)	Severity	Occurrence	Detection	RPN	Mitigation
	Drogue parachute breaks off	Drift becomes exponential, speed increases, and the vehicle goes ballistic	Shock cord or shroud line failure	7	4	5	140	Ensure shroud lines and shock cords meet safety factor per Requirement 6.1.
Nylon Tether	Nylon tether breaks	Vehicle breaks apart one or both pieces hits the ground with no parachute	Shock cord and tether failure	9	5	2	90	Ensure nylon tether can withstand all impulses with sufficient safety factor as per Requirement 6.1.
Bulkhead	Bulkhead Eyebolt breaks	One or more of the parachutes separate from the launch vehicle causing catastrophic damage.	Epoxy failure force analysis failure	10	1	2	20	Ensure bulkheads withstand the impulse of recovery in compliance with Requirement 6.1.
Ejection Charges	Ejection charges fail to separate coupler	Vehicle fails to separate and deploy one or both parachutes	Friction between coupler and airframe is greater or ejection charge is less powerful than expected	9	2	2	36	Having backup charges which are larger than the primary charges set to deploy at a delay of no more than 2 seconds after apogee as per Requirement 3.1.2.

Table 41: BEAVS 2.0 FMEA

System	Failure Mode	Failure Effect	Failure Cause(s)	Severity	Occurrence	Detection	RPN	Mitigation
BEAVS 2.0 Mechanical System								
Rack: Airbrake linear actuator	Experiences bending moment	Rack bends or breaks	Exposure to flight forces out of structural tolerance	4	2	5	40	Maximize rod thickness and internal structural support in compliance with Requirement 6.1.
Linear Guide Sleeve: Guides rack in/out of launch vehicle body	Disconnects from rack	Rack is unable to extend or retract from position, potentially flies out of airframe	Pushed off rack due to flight forces	7	5	1	35	Maximize material strength and fasteners in compliance with Requirement 6.1.
Gear: Translate power from motor to motion in rack	Gets jammed between racks	Rack is unable to extend or retract from position	Gear and rack teeth misaligned	3	2	1	6	Use Computer Numerical Control (CNC) machine to manufacture parts for tight tolerances. Use checklists to ensure the system is operational before launch as per Requirement 1.2.
Motor shaft: Connects motor and gear	Experiences losses due to friction or play	Rack extends and retracts with reduced speeds	Friction and/or play between motor shaft and gear	2	2	5	20	Use CNC machine to manufacture parts for tight tolerances. Use checklists to ensure the system is operational before launch as per Requirement 1.2.

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System	Failure Mode	Failure Effect	Failure Cause(s)	Severity	Occurrence	Detection	RPN	Mitigation
Bulkhead: Platform for all components to sit on	Splinters	All mechanicals on bulkhead are compromised	Moment from flight forces on the blades when extended	8	4	2	64	Add hard points and reinforcements sufficiently to satisfy Requirement 6.1.
BEAVS 2.0 Electrical System								
Battery: Provides power to mechanical system	Runs out of charge	Entire system fails to receive power and does not operate	Battery is drained from sitting on launch rail too long before flight	8	6	4	168	Choose larger than required battery/have back up battery connected. All batteries will be fully charged prior to each launch in accordance with Requirement 6.14.
Accelerometer: Measures acceleration of launch vehicle	Incorrect measurements relayed to controls system	Data input to control loop produces false values and system acts according to false data	End of product life cycle	5	5	5	125	Accelerometer will be proven to be working via testing prior to launches.
Barometric Pressure Sensor: Measures atmospheric pressure	Incorrect pressure values relayed to controls system	Throws off calculations in control system	Intaking noisy data misaligned with venting holes	6	5	5	150	Implement a Kalman filter and PID control loop to ensure clean data is being utilized. Testing will be performed to ensure data is clean and reliable prior to launch.

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System	Failure Mode	Failure Effect	Failure Cause(s)	Severity	Occurrence	Detection	RPN	Mitigation
Motor: Drives the rack and pinion assembly	Disconnects from power source	Entire system fails to work	Battery dies	8	2	2	32	Choose larger than required battery/have back up battery connected. All batteries will be fully charged prior to each launch in accordance with Requirement 6.14.
PCB: Mechanically supports and electrically connects electrical components	Components disconnect	Entire or portion of system fails to work	Excessive flight forces and not enough electrical potting material	7	4	4	106	Electrical potting material will be used to dampen vibrations as per Requirement 6.15.
BEAVS 2.0 Controls System								
Control System: Controls mechanical system with inputs from electrical system	Blades extend during motor burnout	Structural damages to airframe and BEAVS 2.0 mechanical systems	Input data provides incorrect numbers	10	3	5	150	Implement fail-safe to prevent system activation before motor burnout has been completed.
	Apogee altitude hits over 4,000 ft	Apogee altitude is over shot	Insufficient drag is produced in time to reduce apogee	6	9	9	486	Implement control feature where blades automatically extend once altitude achieves 4,000 ft.
	Apogee altitude does not reach 4,000 ft	Apogee altitude is never reached	Launch day wind conditions, improper ballast, flight angle	7	7	8	392	Do not activate system, but continue to collect data.

Table 42: Payload FMEA

System	Failure Mode	Failure Effect	Failure Cause(s)	Severity	Occurrence	Detection	RPN	Mitigation
Coupler Retention Bulkhead	Fracture	Bulkhead failure	Jolt from parachute ejection	8	2	6	96	Appropriately size component to have a safety factor of at least 2 in accordance with requirement 6.1.
	Bulkhead release	Loss of payload retention	Fastener failure	8	1	1	8	Add additional mounting fasteners to bulkhead to have a safety factor of at least 2 in accordance with requirement 6.1.
	Lead screw coupler shear	Loss of payload retention	Too soft of material	8	3	1	24	Support coupler with washer to ensure a safety factor of at least 2 in accordance with Requirement 6.1.
Lead Screw Nut Bulkhead	Breaks	Lead screw jam	Bolt failure	5	1	3	15	Make bulkhead out of durable material and test ejection system to ensure proper function with a safety factor of at least 2 in accordance with Requirement 6.1.
	Twists	Lead screw jam	Jolt from parachute ejection	2	3	2	12	Use a large lead screw that can withstand the forces of flight and recovery without bending in accordance with Requirement 6.1.

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System	Failure Mode	Failure Effect	Failure Cause(s)	Severity	Occurrence	Detection	RPN	Mitigation
	Spins with lead screw	Ejection failure	Bulkhead pushed out of airframe	6	3	1	18	Lead screw must be shorter than the length of the airframe.
	Twists in airframe	Ejection failure	Thin bulkhead	6	3	1	18	Manufacture to a tight clearance between bulkhead and airframe, and thicken bulkhead to be in compliance with Requirement 6.1.
Lead Screw Coupler	Slips off motor shaft	Ejection failure	High loading on lead screw	4	4	2	32	Support lead screw with bulkhead to ensure a safety factor of at least 2 in accordance with Requirement 6.1.
	Shaft misalignment	Poor lead screw efficiency	Improper Mounting	3	7	3	63	Select a motor with sufficient torque to establish a safety margin of at least 2 as per Requirement 6.1.
	Slips through bulkhead	Loss of payload retention	Improper outer diameter	9	2	1	18	Properly size coupler diameter to have a safety margin of at least 2 as per Requirement 6.1.
	Friction between coupler and bulkhead	Motor stall	Coupler shifted during recovery	6	2	4	48	Modify contact surface to reduce friction and allow for a safety margin of at least 2 as per Requirement 6.1.

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System	Failure Mode	Failure Effect	Failure Cause(s)	Severity	Occurrence	Detection	RPN	Mitigation
Coupler to Lead Screw Pin	Shear	Loss of payload retention	High loading on lead screw	8	1	2	16	Maximize pin shear strength to allow for a safety margin of at least 2 as per Requirement 6.1.
	Pin ejection	Loss of payload retention	Improperly secured	8	1	1	8	Use bolt and nut for pin to allow for a safety margin of at least 2 as per Requirement 6.1.
Lead Screw Nut	Jam/seize	Ejection failure	Poor lubrication	4	1	1	4	Internally lubricated lead screw nuts to ensure a safety margin of at least 2 as per Requirement 6.1.
	Spins with Lead screw	Ejection failure	Improper retention	6	4	3	72	Properly fit bulkheads to launch vehicle to prevent them from spinning during ejection.
	Spins with Lead screw	Ejection failure	Fasteners shear	6	4	3	72	Use rigid fasteners and bulkhead material.
Lead Screw	Lead screw bending	Ejection failure	Side loading on lead screw	7	4	1	28	Appropriately size lead screw, if necessary add supporting structure to ensure a safety margin of at least 2 in accordance with Requirement 6.1.

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System	Failure Mode	Failure Effect	Failure Cause(s)	Severity	Occurrence	Detection	RPN	Mitigation
Ejection Motor	Stall	Ejection failure	Motor generates insufficient force to push the payload	2	5	2	20	Use motor with enough torque to push the rover from the airframe as calculated in Section 4.5.3
	Overheat	Ejection failure and airframe damage	Motor causes damage to itself and surrounding components	8	1	4	32	Only use motor within voltage specifications found in the datasheet. The motors will be proven to not overheat via testing.
Ejection Electronics	Violent disassembly during flight	Ejection failure	Flight and recovery forces jar electrical components loose	2	6	2	24	Electrical potting material will be used to reinforce the connections between the components and the board as per Requirement 6.15.
	Current damages PCB traces	Ejection failure and PCB damage	PCB traces are improperly sized	4	1	3	12	PCB traces will be appropriately sized for the expected currents going through them to a safety margin of 2 in accordance with Requirement 6.1.

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System	Failure Mode	Failure Effect	Failure Cause(s)	Severity	Occurrence	Detection	RPN	Mitigation
Chassis	Breaks	Parts become misaligned	Stress during flight	6	1	4	24	Testing will be performed to ensure the chassis can withstand forces of launch and recovery to a safety factor of 2 per Requirement 6.1.
	Component fastener failure	Rover failure	Flight and recovery forces	6	3	3	54	Utilize rigid connections between components to ensure a safety factor of 2 per Requirement 6.1.
	High centering	Drivetrain failure	Low ground clearance	6	5	1	30	Expandable wheels will be used to maximize ground clearance in accordance with Requirement 6.5.
Stabilizing tail	Breaks	Speed of payload lowered	Stress during flight	3	2	6	36	Test strength of tail as per Requirement 6.13
	Excessive Bending	Rover structure spins	To ductile of material	6	3	1	18	Test variable tail thicknesses as per Requirement 6.13
Battery	Fully Discharged	Payload loses function	Payload active for too long	6	4	1	24	Develop power budget to appropriately size battery and fully charge battery before each launch as per a checklist

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Table 42 – continued from previous page

System	Failure Mode	Failure Effect	Failure Cause(s)	Severity	Occurrence	Detection	RPN	Mitigation
	Battery is pierced by Foreign Object Debris (FOD)	Battery explodes destroying itself, payload, and launch vehicle if rover is not yet ejected	Improperly protected battery	10	2	4	80	The battery will be protected from impact and brightly colored as per Requirement 2.21
Camera	Signal corruption	Loss of feed	Magnetic Interference	6	3	1	18	Incorporate magnetic protection as per Requirement 6.2
Wheels	Slips off motor	Wheels unpowered	Shifting during flight	7	3	7	147	Create a firm padded section for the payload in which it cannot shift as per requirement 4.3.7.1
	Does not unfold and expand	Wheels difficult to turn	Broken/Jammed Hinge	4	4	4	64	Inspect hinges, ensure adequate lubrication in accordance with a checklist as per Requirement 5.1
	Sinks into soft dirt	Rover cannot move	Not enough surface area	5	3	4	60	Increase surface area in contact with the ground as per Requirement 6.5
Drive motors	Breaks	Wheels unpowered	Damaged in flight	7	1	7	49	Have motors attached firmly, make sure rover is securely fixed in launch vehicle and well padded as per Requirement 6.1
Auger	Fails to collect	No sample collected	Not enough power	7	4	4	112	Make sure the motor has ample strength as per Requirement 6.6

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System	Failure Mode	Failure Effect	Failure Cause(s)	Severity	Occurrence	Detection	RPN	Mitigation
	Sample falls out of auger	No sample collected	Not enough friction between auger and sample	7	5	6	210	Design sample retention system as per Requirements 4.3.4 and 6.6
	Failure to dig into soil	No sample collected	7	Not enough weight behind rover	6	3	126	Angle auger to gain mechanical advantage as per Requirement 6.4
Auger servo	Jam	No sample collected	Misaligned Parts	7	4	6	168	Ensure proper assembly as per a checklist and repeated tests on auger rotation as per Requirement 6.13
	Auger Rotates when digging	No sample collected	Servo torque fails to hold auger in position	7	4	1	28	Scale servo appropriately during testing as per Requirement 6.13

5.2 Analysis of Project Risk

5.2.1 Safety Responsibilities

Safety for 2019-2020 OSRT is lead by Lead Safety Officer Wyatt Hougham, supported by safety officers Nicholas Drachnik and Jessica Peterson for personnel hazard protection, and James Felsher for environmental hazard protection. The safety team shall be responsible for OSRT member compliance with all OSU, NASA, NAR, TRA, NFPA, and FAA safety regulations. Safety will also be in charge of organizing and managing a database of checklists pertaining to personnel, environmental, and launch vehicle safety. Although safety is the responsibility of all team members, it is the safety officers who will be responsible for ensuring that all safety rules set forth by the team and all regulatory bodies are followed at all times.

5.2.2 Safety Methods

For high-level project safety as well as risk prevention and mitigation, FMEA will be used to identify and evaluate any technical risks to the launch vehicle and all of the constituent parts. This will be conducted through a standard scoring system factoring severity, occurrence, and detection. Preliminary FMEA will be conducted by each launch vehicle sub-team such that insight into the intricacies of each subsystem can be considered. Early FMEA will afford the safety team the best opportunity to prevent hazards in every aspect of the project.

Administrative controls will be a significant focus of the Safety Officers. Engineering controls will be utilized in every mode possible, however, the administrative controls will work to regulate behavior such that a minimum level of safety is maintained even in the event of an engineering control failure. Administrative controls will be implemented through but not limited to: Safety Checklists, Job Hazard Analysis (JHA) forms, as well as enforcing adherence to Material Safety Data Sheet recommendations MSDS, OSU Environmental Health & Safety (EHS) standards, and Personal Protective Equipment Personal Protective Equipment (PPE) requirements.

5.2.3 Project Risks and Risk Management

Safety Officers will also work closely with team management to prevent and mitigate risks to the project. Some major risks to the project include timeline delays, project creep, and resource constraints. To avoid timeline delays, safety will work with sub-team leads to ensure creating and maintaining maximum lead time in any area where applicable, although there may be inevitable delays over which the team has no control. To mitigate these delays, alternatives to any applicable process will be considered as early as possible. To create lead time and prevent delays, ordering will be done as early as possible, checklists and other administrative controls will be completed in advance to all relevant team activities, and scheduling will be done early enough to minimize time conflicts. To avoid project creep, project and team goals have been set forth to better focus and synthesize team member efforts and resources toward relevant objectives. Tables 43 and 44 give more of a look into OSRT's analysis into potential risks, their liklihood and severity, and how they could be mitigated.

Table 43: Risk Assessment Matrix

	Impact		
Liklihood	**1 - High**	**2 - Medium**	**3 - Low**
A - High	1A	2A	3A
B - Medium	1B	2B	3B
C - Low	1C	2C	3C

Table 44: Risk Analysis

Risk	**Likelihood and Impact Rating**	**Mitigation Technique**
Timeline pushed backward	3A	Break up large deadlines into smaller, more manageable, internal checkpoints that can be accomplished more readily.
OSRT cannot raise enough money to finance the project	1C	OSRT will reach out to as many potential donors as possible and look for frugal ways to construct the launch vehicle and payload, including, but not limited to, finding companies to donate old materials and sorting through scrap bins around campus.
Mistake in manufacturing causes a part's completion to be delayed	2B	Procedure checklists and blueprints will be created prior to manufacturing to ensure that careless or misinformed errors are not made.
Poor weather at the launch site causes the launch to be moved backward to another day.	2B	Be prepared to launch both earlier and later in the selected day to be able to take any possible opportunity of the flight waiver opening.

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Risk	Likelihood and Impact Rating	Mitigation Technique
Team member's illness causes manufacturing to be delayed	2B	Ensure that more than one individual on the team is certified to work on any necessary machine and understands the relevant part drawings and procedure checklists.
Delay in part delivery and/or damaged orders	2B	Place orders as soon as possible and from reputable carriers
Launch vehicle does not comply to NASA USLI operational requirements	1C	Compliance will be checked and ensured at every milestone and design decision, with simulations and calculations to verify launch vehicle adherence to operational requirements
Team members create more goals and team derived operational requirements then reasonably feasible	1C	Project scope will be defined by every subteam and verified by team leads in all milestones
Lack of funding requires team members to fund their trip out of pocket, causing some members to not be able to attend launches	2c	Possible sponsors will be reached out too, and spending on materials greatly monitored

6 Project Plan

6.1 Project Requirements Verification

Shown in Tables 45-50 is a breakdown of all USLI competition requirements outlined in the handbook, a brief description of how OSRT plans to verify these requirements will be completed, and the current status of the verification implementation.

6.1.1 General Requirements

Table 45: General Requirement Verification Matrix

Requirement	Verification Plan	Status
1.1.1 Students on the team will do all of the work on the project, except when it comes to assembling motors and handling black powder or any other kind of ejection charge.	Individuals who are not students on the team will be prohibited from doing work on the project at any point in time, unless it is for motor assembly or ejection charge purposes, and only team members will be granted access to the team's shared drive and LaTeX documents.	In progress - This will be a daily practice starting from the time the handbook is released to when the Post-Launch Assessment Review is submitted and included in the timeline, as everything in the timeline is student work.
1.1.2 The team will submit new work.	The team will not copy and paste large sections of material from previous documents into new documents without significantly modifying that which is copied.	In progress - This will be completed at each deliverable submission and is included in the timeline with ample time to write, compile, and edit all deliverables before submission.

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Table 45 – continued from previous page

Requirement	Verification Plan	Status
1.2 The team will provide and maintain a project plan, including project milestones, budget and community support, checklists, personnel assignments, STEM Engagement events, and risks and mitigations.	The Team Lead will maintain the project plan, monitor the work of the team done by project milestones, and keep track of personnel assignments, while Budget and Finance keeps the budget up-to-date and works to maintain community support. STEM Engagement will keep track of all STEM Engagement activities and record and report the outcome of each activity to NASA via the STEM Engagement Activity Report, and Safety will be responsible for keeping risks and mitigations up-to-date. All aforementioned subteams will update these respective pieces of information in each deliverable submitted to RSO	In progress - A project plan with the milestones, budget and community support, checklists, personnel assignments, STEM Engagement events, and risks and mitigations will be repeatedly updated and submitted to NASA personnel up to and including to when the Post-Launch Assessment Review is submitted. This is included in the timeline in the compiling and editing phase of each deliverable.
1.3 Foreign National team members must be identified by the Preliminary Design Review.	Team members will be required to report to the Team Lead that they are a Foreign National before the Preliminary Design Review submission deadline.	Complete - OSRT does not have any Foreign National (FN)s to report to NASA, and that has been reported to NASA.
1.4 The team must identify all team members attending launch week activities by the Critical Design Review.	The Team Lead will collect a list of members and one mentor who will be attending launch week activities and submit it by the Critical Design Review submission deadline.	Not complete - This will be completed closer to the Critical Design Review submission deadline, and is accounted for in the timeline in the editing and compiling schedule of the Critical Design Review.
1.5 The team will engage a minimum of 200 participants in educational, hands-on science, technology, engineering, and math activities, as defined by the STEM Engagement activity report between project acceptance and Flight Readiness Review (FRR).	The team will keep a tally of how many students participate in each educational activity, and will submit this number, along with the STEM Engagement Activity Report, within two weeks of the STEM Engagement activity to RSO	In progress - The OSRT has started working on STEM Engagement projects and reaching out to schools and activities, and will continue to do so throughout the project. This is accounted for in the timeline under all of the STEM Engagement events and lesson plans.

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Requirement	Verification Plan	Status
1.6 The team will establish a social media presence to inform the public about team activities.	The team will create a Snapchat and Instagram account, and will post on these platforms regularly to keep followers up-to-date on team activities.	Complete - Snapchat and Instagram accounts have been created.
1.7 Teams will e-mail all deliverables to the NASA project management team by the deadline specified in the handbook for each milestone either by e-mailing the file directly, or, if the file is too big, by including a link to download the file.	The team will send all deliverables to the NASA project management team, and then will screenshot the sent e-mail with a timestamp and keep the screenshots in a file on the team's shared drive.	In progress - This will be completed at each deliverable submission and is accounted for in the timeline on the day of deliverable deadlines.
1.8 All deliverables must be in PDF format.	All deliverables submitted to NASA will be saved in PDF format, and one copy of each deliverable will be saved to the team's shared drive.	In progress - This will be completed at each deliverable submission and is accounted for in the timeline on the day of the deliverable deadlines.
1.9 In every report, the team will provide a table of contents with major sections and their respective subsections.	A table of contents will be submitted within each deliverable that details the deliverables sections and subsections.	In progress - This will be completed at each deliverable submission and is accounted for in the timeline in the compiling and editing schedule of each deliverable.
1.10 In every report, the team will include a page number at the bottom of each page.	A page number will be included at the bottom of each page in every deliverable.	In progress - This will be completed at each deliverable submission and is accounted for in the timeline in the compiling and editing schedule of each deliverable.
1.11 The team will provide any computer equipment necessary to perform a video teleconference with the review panel.	The team will reserve a conference room on the Oregon State University campus for the duration of the video teleconference that has a speaker and projector system. The team members will provide a camera, a microphone, and a telephone.	Not complete - This will be completed before each video teleconference, and is accounted for in the timeline in the video teleconference schedule of each deliverable.

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Requirement	Verification Plan	Status
1.12 The team will use a launch pad provided by Student Launch's launch services provider.	The team will only use Student Launch's launch services provider's launch pad, and will design the launch vehicle so that it is compatible with either 8-foot 1010 or 12-foot 1515 rails.	In progress - This will be incorporated into the launch vehicle designs, and is accounted for in the timeline with the launch vehicle design schedule, but will officially be completed at the competition launch.
1.13 The team will identify a "mentor".	The team will identify their mentor and report their mentor to the NASA project management team by the time the Proposal is submitted.	Completed - The team's mentor is Joe Bevier and the team's advisor is Dr. Nancy Squires.

6.1.2 Vehicle Requirements

Table 46: Vehicle Requirement Verification Matrix

Requirement	Verification Plan	Status
2.1 The launch vehicle will deliver the payload to an apogee between 3,500 feet and 5,000 feet above ground level.	The Aerodynamics/Recovery and Structures/Propulsion Teams will maintain altitude simulations as the launch vehicle is constructed in order to accurately select a motor that will deliver the launch vehicle into the given altitude window. Aerodynamics/Recovery will also develop an altitude control system to hone in on our declared altitude during flight.	In progress - A projected altitude is defined in the projected altitude section, but will be officially declared by the Preliminary Design Review submission deadline, and this is accounted for in the timeline in the compiling and editing schedule of the Preliminary Design Review.

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Requirement	Verification Plan	Status
2.2 Teams shall identify their target altitude goal at the Preliminary Design Review milestone.	The team will report our target altitude on our submission for the Preliminary Design Review.	In progress - A projected altitude is defined in the projected altitude section and was defined in the Aerodynamics/Recovery and Introduction. This is accounted for in the timeline in the compiling and editing schedule of the Preliminary Design Review.
2.3 The vehicle will carry one commercially available, barometric altimeter for recording the official altitude used in determining the Altitude Award winner.	The team will select a commercially available barometric altimeter for implementation into the recovery system by the Preliminary Design Review submission deadline.	In progress - Altimeter research has been conducted, such as in the parachute ejection section, and was officially defined in the Aerodynamics/Recovery Section. This is accounted for in the timeline in the design schedule for recovery.
2.4 The launch vehicle will be designed to be recoverable and reusable.	The launch vehicle will be designed so that it has a recovery system that allows the launch vehicle to land softly, and an interchangeable motor and ejection charges that allow for the launch vehicle to relaunch within a reasonable time frame.	In progress - The initial design is complete, as detailed in the launch vehicle section, but is yet to be finalized, and this requirement will be completed when the launch vehicle design is complete. This is accounted for in the timeline in the launch vehicle design schedule.
2.5 The launch vehicle will have a maximum of four (4) independent sections.	The launch vehicle will be designed to have three (3) independent sections.	Complete - The launch vehicle will have three (3) independent sections, the nose cone, the fore body section, and the aft body section.

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Requirement	Verification Plan	Status
2.5.1 Coupler/airframe shoulders which are located at in-flight separation points will be at least one body diameter in length.	The team will design the airframe shoulders to be at least 6.25 in. in length.	Complete - The airframe shoulders are designed to be 6.5 in. in length on both the fore and the aft breaks of the launch vehicle.
2.5.2 Nose cone shoulders which are located at in-flight separation points will be at least 1⁄2 body diameter in length.	The team will design the airframe nose cone shoulders to be at least 3.125 in. long.	Complete - The nose cone shoulders are designed to be 4 in. long.
2.6 The launch vehicle will be capable of being prepared for flight at the launch site within 2 hours of the time the Federal Aviation Administration flight waiver opens.	The team will arrive to the launch site at least two hours before the flight waiver opens, and will practice integration of all of the systems into the launch vehicle no later than 1 day in advance of the launch day in order to ensure that the assembly of the launch vehicle takes no longer than 2 hours.	Not completed - This will be completed when the subscale systems are completed and is accounted for in the timeline in each launch's integration schedule.
2.7 The launch vehicle and payload 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.	The team will design all on-board electronics to last a minimum of 10 hours, and the payload to last a minimum of 18 hours on the Launch Pad.	In progress - The initial design has been outlined in the launch vehicle and payload sections, but both are yet to be finalized. This requirement will be completed when the design of launch vehicle and payload electronics are complete, and is accounted for in the timeline in launch vehicle and payload design schedule.
2.8 The launch vehicle will be capable of being launched by a standard 12-volt direct current firing system.	The team will use a standard, commercially available motor and will ensure that it will be able to be ignited with a standard 12-volt direct current firing system by launching it at least once with a 12-volt direct current firing system.	Not complete - This will be completed when the motor has been integrated and successfully launched with a standard 12-volt direct current firing system, and is accounted for on the timeline in the integration and launch schedules.

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Requirement	Verification Plan	Status
2.9 The launch vehicle will require no extraneous external circuitry or special ground support equipment to initiate launch.	The team will use a standard, commercially-available motor, will not make any modifications to it, and will ensure that the motor can be launched without extraneous external circuitry by launching an identical motor at least once.	Not complete - This will be completed when the motor has been integrated and launched once without extraneous external circuitry which is accounted for on the timeline in the integration and launch schedules.
2.10 The launch vehicle will use a commercially available solid motor propulsion system using Ammonium Perchlorate Composite Propellant (APCP) which is approved and certified by the NAR, TRA, and/or the Canadian Association of Rocketry (CAR).	The team will only select a motor that uses ammonium perchlorate composite propellant that is approved and certified by NAR, and will have the team mentor approve of the purchase as a representative of NAR before purchase of the motor.	In progress - A motor has been selected, as shown in the propulsion section, however, this will be completed when the motor is purchased after approval of a NAR mentor, and is accounted for in the section for ordering launch vehicle components schedule.
2.10.1 Final motor choices will be declared by the Critical Design Review (CDR) milestone.	The team will include their declaration of their final motor by the team's Critical Design Review deliverable submission.	In progress - A motor has been selected, as shown in the propulsion section, however, it will be officially declared after a scaled-down motor is purchased after approval of a NAR mentor, and flown successfully before the Critical Design Review deliverable submission, which is accounted for in ordering launch vehicle components, integration, and launch schedules.
2.10.2 Any motor change after CDR must be approved by the NASA RSO.	The team will not change their motor after their Critical Design Review deliverable submission unless it is absolutely necessary, in which case, the team will seek approval from the NASA RSO before finalizing any motor changes.	Not completed - This will be completed when the launch vehicle is launched at competition, but if needed, is incorporated on the timeline in the modifying/repairing the launch vehicle section.

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Requirement	Verification Plan	Status
2.11 The launch vehicle will be limited to a single stage.	The launch vehicle will be designed to only hold one motor in the aft section of the launch vehicle.	In progress - A single-staged launch vehicle has been designed in the launch vehicle, and this requirement will be complete when the design for the launch vehicle as been finalized, which is accounted for on the timeline in the design of the launch vehicle.
2.12 The total impulse provided by a College or University launch vehicle will not exceed 5,120 Newton-seconds (L-class).	The team will not select a motor larger than an L-class for implementation into the launch vehicle, and will keep the launch vehicle and rover weight low enough that an L-class motor or smaller can carry the launch vehicle to the predetermined altitude.	Complete - The team has selected a AeroTech L2200 with a total impulse of 5104 N-sec.
2.13 Pressure vessels on the vehicle will be approved by the RSO	The team will have all checklists that involve pressure vessels require a signature from the RSO after the RSO approves the vessel in order for the checklist to be complete.	In progress - The checklists will be completed while doing design and preparing for construction of the system and will be, and has been accounted for in the timeline during the design and early construction schedules of the recovery system.
2.13.1 The minimum factor of safety (Burst or Ultimate pressure versus Max Expected Operating Pressure) will be 4:1 with supporting design documentation included in all milestone reviews.	The team will ensure that all pressure vessels have a minimum factor of safety of 4:1, and will supply updated calculations in each deliverable to demonstrate that the factor of safety remains at least 4:1.	In progress - This will be completed at each deliverable submission starting with Preliminary Design Review (PDR)and is accounted for in the timeline in the compiling and editing schedule of each deliverable.

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Requirement	Verification Plan	Status
2.13.2 Each pressure vessel will include a pressure relief valve that sees the full pressure of the tank and is capable of withstanding the maximum pressure and flow rate of the tank.	The team will incorporate a pressure relief valve into all pressure vessels, and test it to ensure that the valve can withstand the maximum pressure and flow rate of the vessel.	In progress - This will be complete when pressure relief valves are incorporated into the parachute ejection charge design, which is accounted for in the timeline in the recovery design section.
2.13.3 The full pedigree of the tank will be described, including the application for which the tank was designed and the history of the tank. This will include the number of pressure cycles put on the tank, the dates of pressurization/depressurization, and the name of the person or entity administering each pressure event.	The team will keep a log explicitly for each tank used, and will record the number of pressure cycles and the dates of pressurization/depressurization, along with requiring the signature of the individual who administered each pressure event. This documentation, along with the description of the application for which the tank was designed will be included in all deliverables.	In progress - This will be complete when the tanks are incorporated into the parachute ejection charge system and go through testing, which is accounted for in the timeline in the recovery design, manufacturing, and testing schedules.
2.14 The launch vehicle will have a minimum static stability margin of 2.0 at the point of rail exit.	The team will ensure that the static stability margin of the launch vehicle will at least be 2.0, and the updated calculations will be included in each deliverable submitted.	In progress - Completed in current design with a rail exit stability of 3.12. This will be completed at each deliverable submission and is accounted for in the timeline in the compiling and editing schedule of each deliverable.
2.15 Any structural protuberance on the launch vehicle will be located aft of the burnout center of gravity.	The burnout center of gravity will be calculated twice: first, prior to finishing the design of the launch vehicle to ensure that all protuberances are designed to be aft of the burnout center of gravity, and second, after finishing the design of the launch vehicle to ensure that the protuberances are still aft of the burnout center of gravity after their addition.	In progress - The launch vehicle has been designed without any protuberances thus far, but this will be completed when the launch vehicle design is finalized, and is accounted for in the timeline in the launch vehicle design schedule.

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Requirement	Verification Plan	Status
2.16 The launch vehicle will accelerate to a minimum velocity of 52 fps at rail exit.	Simulations will be conducted in OpenRocket throughout the development of the launch vehicle, recovery system, and payload to ensure that the launch vehicle will accelerate to a minimum velocity of 52 fps at rail exit. If it cannot, the motor will either be increased, or the weight of the overall launch vehicle will be decreased.	Not completed - This will be completed when the manufacturing and testing of the launch vehicle, recovery system, and payload are completed, and is accounted for in the timeline in the launch vehicle, recovery, and payload manufacturing and testing schedule.
2.17 All teams will successfully launch and recover a subscale model of their launch vehicle prior to CDR.	The team will have three subscale launches to test the launch vehicle and recovery system designs, which will all be photographed and signed off by two members in every checklist leading up to the launch, one on Saturday, November 9th, another on Saturday, November 30th, and a third on Sunday, December 15th.	Not completed - This will be completed by the final subscale launch day, Sunday, December 15th.
2.17.1 The subscale model should resemble and perform as similarly as possible to the full-scale model.	The subscale model will be designed to be $^{2}/_{3}$rds scale replica of the full-scale launch vehicle.	In progress - The design of the subscale model is detailed in the launch vehicle section but this will be complete when the subscale launch vehicle and recovery system and full-scale launch vehicle and recovery system are designed, and this is accounted for in the design and launching of the subscale model and the design of the full-scale model schedules.

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Requirement	Verification Plan	Status
2.17.2 The subscale model will carry an altimeter capable of recording the model's apogee altitude.	The subscale model will use the same altimeters as the full-scale model.	In progress - Altimeters have been explored, such as in the parachute ejection section, but this will be complete when the altimeters are selected for the full-scale launch vehicle, and implemented in the subscale launch vehicle, all of which is accounted for in the timeline for purchasing both launch vehicle and recovery components, ground testing the recovery system, and building the subscale schedules.
2.17.3 The subscale launch vehicle must be a newly constructed launch vehicle, designed and built specifically for this year's project.	The launch vehicle and recovery system will be constructed from all-new materials, ensuring that these systems are built specifically for this year's project.	In Progress - Materials for the subscale launch vehicle have started arriving, and part manufacture has begun.
2.17.4 Proof of a successful flight shall be supplied in the CDR report. Altimeter data output may be used to meet this requirement.	Altimeter data and photos of the launch day, including of the launch and of the recovery, will be provided as proof of a successful flight.	Not completed - This will be completed when a successful flight is completed and the proof is submitted in the Critical Design Review deliverable, which is accounted for in the timeline under launch and compiling and editing the Critical Design Review deliverable.
2.18.1.1 All teams will successfully launch and recover their full-scale launch vehicle prior to FRR in its final flight configuration.	The team has scheduled two launch days to do a successful launch and recovery of the launch vehicle and recovery system, which will all be photographed and signed off by two members in every checklist leading up to the launch, one on Saturday, February 1st and another on Saturday, February 22nd.	Not completed - This will be completed upon successfully launching and recovering the launch vehicle and recovery system in February, which is accounted for in the timeline with scheduled full-scale launches.

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Requirement	Verification Plan	Status
2.18.1.2 The launch vehicle flown must be the same launch vehicle to be flown on launch day.	The team will not change the launch vehicle or recovery system between its final flight before FRR and its flight on launch day, and will use the same checklists as will be used on launch day to ensure this.	Not completed - This will be completed between the final launch before the Flight Readiness Review and launch day, where no technical modifications will be made to the launch vehicle or recovery system between the two flights, and has been accounted for in the timeline with a scheduled launch vehicle repair time, but not a launch vehicle modification time.
2.18.1.3 The vehicle and recovery system will have functioned as designed.	The vehicle will meet all speed and energy requirements, will separate at the correct times, and the recovery system will deploy and inflate its parachutes at the correct times to ensure that the vehicle lands under the energy requirements as well.	Not completed - This will be completed at full-scale launches in February, and has been accounted for in the timeline in the scheduled launch and, if needed, the modify/repair schedules as well.
2.18.1.4 The full-scale launch vehicle must be a newly constructed launch vehicle, designed and built specifically for this year's project.	The launch vehicle and recovery system will be constructed from all-new materials, ensuring that these systems are built specifically for this year's project.	Not completed - This will be completed when the full-scale launch vehicle and recovery system are built and is accounted for in the timeline both in material and components purchasing for the launch vehicle and recovery system, and in both of their manufacturing schedules as well.
2.18.1.5 If the payload is not flown, mass simulators will be used to simulate the payload mass, and will be located in approximately the same location as where the payload would be.	If the Payload Team is not ready to fly to payload, they will manufacture a mass that is the same size and basic shape of the payload, and can be retained by the same retention and ejection system in the launch vehicle.	Not complete - This will be completed when the Payload Team manufactures a mass representative of the payload that is flown in a full-scale flight, which is accounted for in the timeline in the manufacturing schedule of a full-scale payload.

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Requirement	Verification Plan	Status
2.18.1.6 If the payload changes the external surfaces of the launch vehicle (such as with camera housings or external probes) or manages the total energy of the vehicle, those systems will be active during the full-scale Vehicle Demonstration Flight.	The team will activate all payload features that change the external surface of the launch vehicle and/or manage the total energy of the vehicle, and will have two team members sign off on the checklist for this system, along with photos taken of the system, to verify its actuation later.	Not completed - This will be completed when Payload actuates these features before a full-scale launch, which is accounted for in the timeline in both the integration schedule and the February launch days' schedules.
2.18.1.7 Teams shall fly the launch day motor for the Vehicle Demonstration Flight.	The team will fly the launch day motor at both full-scale launches and will have two people sign off on its checklist for this feature, along with photos taken of this system, to be able to verify this later if need be.	Not completed - This will not be completed until both full-scale launches are completed and is accounted for in the timeline in the ordering schedule for launch vehicle components, the integration schedule the day before launch, and the February launch days' schedules.
2.18.1.8 The vehicle must be flown in its fully ballasted configuration during the full-scale test flight.	The team will fly the vehicle in its fully ballasted configuration, and will have two people sign off on its checklist for this feature, along with photos taken of this system, to be able to verify this later if need be, and if it is necessary to change the ballasted configuration after the second full-scale flight, a third full-scale flight will be conducted to ensure that the final ballasted configuration is flown before the Flight Readiness Review.	Not completed - This will be completed when the vehicle has a fully ballasted full-scale flight in February, which has been accounted for in the timeline in the integration schedule and the February launch days' schedules.

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Requirement	Verification Plan	Status
2.18.1.9 After successfully completing the full-scale demonstration flight, the launch vehicle or any of its components will not be modified without the concurrence of the NASA RSO.	The team will not modify any portion of the launch vehicle and its components without a signature or written consent of the NASA RSO	Not completed - If necessary, this will be completed when the NASA RSO either sends a letter of approval in the form of an e-mail or letter, or signs a form stating their approval, and has been accounted for in the timeline in the modify/repair schedules between the two February launches should the first launch be successful, or in the repair schedules should the second launch be successful.
2.18.1.10 Proof of a successful flight shall be supplied in the FRR report. Altimeter data output is required to meet this requirement.	Altimeter data and photos of the launch day, including of the launch and of the recovery, will be provided as proof of a successful flight.	Not completed - This will be completed when a successful flight is completed and the proof is submitted in the Flight Readiness Review report, and has been accounted for in the timeline in the February launch days' schedules and in the compiling and editing schedule for the Flight Readiness Review deliverable.
2.18.1.11 Vehicle Demonstration flights must be completed by the FRR submission deadline.	The team will have two full-scale vehicle demonstration flights which will all be photographed and signed off by two members in every checklist leading up to the launch, the first on Saturday, February 1st and the second on Saturday, February 22nd.	Not completed - This will be completed when both launches are completed before the FRR submission deadline, which is accounted for in the timeline in the two scheduled February launch days and their respective integration days.

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Requirement	Verification Plan	Status
2.18.2.1 All teams will successfully launch and recover their full-scale launch vehicle containing the completed payload prior to the Payload Demonstration Flight deadline.	The team will launch and recover the full-scale launch vehicle with the payload twice while the launch vehicle and recovery system meet all requirements, and will photograph the payload's flight during launch, after landing before payload deployment, after payload deployment, and after payload actuation.	Not completed - This will be completed when the payload is flown and documented in February, and is accounted for in the timeline in the two scheduled February launch days and their respective integration days.
2.18.2.2 The launch vehicle flown must be the same launch vehicle to be flown on launch day.	The team will not change the launch vehicle or recovery system between its final flight before FRR and its flight on launch day, and will use the same checklists as will be used on launch day to ensure this.	Not completed - This will be completed between the final launch before the Flight Readiness Review and launch day, where no technical modifications will be made to the launch vehicle or recovery system between the two flights, and has been accounted for in the timeline with a scheduled launch vehicle repair time, but not a launch vehicle modification time.
2.18.2.3 The payload must be fully retained until the intended point of deployment, all retention mechanisms must function as designed, and the retention mechanism must not sustain damage requiring repair.	The team will design and test the retention system before flying it in the launch vehicle it ensure the robustness of this system.	In progress - The Payload team has worked on designing a payload retention and ejection system, which is demonstrated in the ejection system subsection of the Payload section. This requirement will be completed once the Payload team finishes the design, and has manufactured and tested their retention system, all of which has been accounted for in the timeline in the designing, ordering, manufacturing, and testing schedules for the payload.

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Requirement	Verification Plan	Status
2.18.2.4 The payload flown must be the final, active version.	The team will fly the payload in its final, active state, with photographs being taken, along with two people signing off on all of the payload checklist leading up to the flight to be able to ensure that the payload was flown in its final and active state.	Not completed - This will be completed between the final launch before the Flight Readiness Review and launch day, where no technical modifications will be made to the payload between the two flights, and has been accounted for in the timeline with a scheduled payload repair and testing time, but not a payload modification time.
2.18.2.5 Payload Demonstration Flights must be completed by the FRR Addendum deadline.	The team will complete two Payload Demonstration Flights which will all be photographed and signed off by two members in every checklist leading up to the launch, the first on Saturday, February 1st and the second on Saturday, February 22nd.	Not completed - This will be completed when both launches are completed before the FRR Submission deadline and therefore, will also be completed before the FRR Addendum deadline, which is accounted for in the timeline in the two scheduled February launch days and their respective integration days.
2.19 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 Report.	If an FRR Addendum is necessary to submit to NASA project management, the team will complete the addendum and submit it by the due date of March 23rd, 2020.	Not completed - This will be completed when it the FRR Addendum is submitted to NASA project management, and time will be allotted, if necessary, to ensure that the FRR Addendum is completed by the time it is due.

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Requirement	Verification Plan	Status
2.19.1 Teams who complete a Payload Demonstration Flight which is not fully successful may petition the NASA RSO for permission to fly the payload at launch week.	If it is necessary to petition the NASA RSO for permission to fly the payload, the team will petition the NASA RSO no later than March 25th, 2020 with a letter from the team and a detailed report of what caused the Payload Demonstration Flight failure, and what the team has done to fix those causes.	Not completed - This will be completed when the team sends the letter and report to the NASA RSO, and time will be allotted, if necessary, to ensure that the letter and report are sent to the NASA RSO no later than March 25th, 2020.
2.20 The team's name and launch day contact information shall be in or on the launch vehicle airframe as well as in or on any section of the vehicle that separates during flight and is not tethered to the main airframe. This information shall be included in a manner that allows the information to be retrieved without the need to open or separate the vehicle.	The team will place a label detailing the team's name and the launch day contact information of the team lead to the outside of the fore and aft sections of the launch vehicle, and on the underside of the payload in an area that is clearly visible.	Not completed - This will be completed when the launch vehicle and payload are built, and has been accounted for in the timeline in the launch vehicle manufacturing schedule.
2.21 All Lithium Polymer batteries will be sufficiently protected from impact with the ground and will be brightly colored, clearly marked as a fire hazard, and easily distinguishable from other payload hardware.	The team will develop a uniform method of brightly coloring and clearly labeling all Lithium Polymer batteries so that they are easily distinguished from other electronics components and payload hardware. This method will be turned into a checklist, which, once completed, will need to be signed by the person in charge of the system, the team SO, and the Team Lead.	Not completed - This will be completed when a uniform method of clearly marking the Lithium Polymer batteries is developed and properly executed, and is accounted for in the design, order, and manufacturing of the launch vehicle.

Table 47: Vehicle Prohibition Verification Matrix

Requirement	Verification Plan	Status
2.22.1. The launch vehicle will not utilize forward canards. Camera housings will be exempted, provided the team can show that the housing(s) causes minimal aerodynamic effect on the launch vehicle's stability.	The launch vehicle will be designed without the need for forward canards excluding camera housings.	In progress - Completed in current designs, all future design iterations will not include forward canards. This requirement will be completed when manufacturing of the full scale launch vehicle is complete.
2.22.2. The launch vehicle will not utilize forward firing motors.	The launch vehicle will be designed without the need for forward firing motors, using drag characteristics to slow down instead.	In progress - Completed in current design, all future design iterations will not include forward firing motors. This requirement will be completed when manufacturing of the full scale launch vehicle is complete.
2.22.3. The launch vehicle will not utilize motors that expel titanium sponges (Sparky, Skidmark, Metal- Storm, etc.)	The launch vehicle will be designed without utilizing motors that expel sponges.	In progress - Completed in current design, all future design iterations will not include motors that expel titanium sponges. This requirement will be completed at CDR, when the final competition motor is selected.
2.22.4. The launch vehicle will not utilize hybrid motors.	The launch vehicle will be designed without the need for hybrid motors.	In progress - Completed in current design, all future design iterations will not include hybrid motors. This requirement will be completed upon manufacturing of the full scale launch vehicle.

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Requirement	Verification Plan	Status
2.22.5. The launch vehicle will not utilize a cluster of motors.	The launch vehicle will be designed without the need for cluster of motors.	In progress - Completed in current design, all future design iterations will not include clusters of motors. This requirement will be completed upon manufacturing of the full scale launch vehicle.
2.22.6. The launch vehicle will not utilize friction fitting for motors.	The launch vehicle will be designed without using friction fitting for motor retention.	In progress - Completed in current design, all future design iterations will not utilize friction fitting for motors. This requirement will be completed upon manufacturing of the full scale launch vehicle.
2.22.7. The launch vehicle will not exceed Mach 1 at any point during flight.	The launch vehicle will use a motor with a long enough burn time and low enough thrust to stay below Mach 1 but reach apogee.	In progress - OpenRocket simulations verify the vehicle will not exceed Mach 1 at any point during flight. Flight data from the Vehicle Demonstration Flight will be used to ensure the vehicle does not exceed Mach 1.
2.22.8. Vehicle ballast will not exceed 10 percent of the total unballasted weight of the launch vehicle as it would sit on the pad (i.e. a launch vehicle with an unballasted weight of 40 lbf on the pad may contain a maximum of 4 lbf of ballast).	The launch vehicle will have its fully fueled weight measured prior to flight to determine the max ballast available to use.	In progress - Completed in current design, less than 10 percent ballast is required for all planned launch conditions. All future design iterations will utilize less than 10 percent ballast for all flight conditions.
2.22.9. Transmissions from onboard transmitters will not exceed 250 mW of power (per transmitter).	The launch vehicle will be designed without transmitters that use more than 250 mW of power	In progress - Completed in current design, the transmitters selected do not transmit more than 250 mW of power. Any future changes to the transmitters will require no more than 250 mW of power.

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Requirement	Verification Plan	Status
2.22.10 Transmitters will not create excessive interference. Teams will utilize unique frequencies, hand-shake/passcode systems, or other means to mitigate interference caused to or received from other teams.	The launch vehicles transmitters will not create excessive interference and will use frequency unique to OSRT.	In progress - Completed in current design, OSRT tracking systems have implemented this previously, similar methods will be utilized this year. Any changes to the systems will have means to mitigate interference.
2.22.11. Excessive and/or dense metal will not be utilized in the construction of the vehicle. Use of light- weight metal will be permitted but limited to the amount necessary to ensure structural integrity of the airframe under the expected operating stresses.	The launch vehicle will be designed mainly using composites, wood, plastics. Use of metals will be minimized to critical components which cannot be manufactured from other materials.	In progress - Completed in current design, all future designs will be made with the intent to minimize metal usage.

6.1.3 Recovery System Requirements

Table 48: Recovery System Requirement Verification Matrix

Requirement	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.	The team has designed the recovery system to deploy its drogue parachute at apogee and its main parachutes at 600 feet in altitude, which is what the system will be set to for all of the launches, which will be completed by checklists which will be signed by the team member in charge of this system, the Team SO, and the Recovery Team Lead. Video from the ground will be taken for verification that the parachutes ejected at the appropriate time and altitude.	In progress - The recovery system has been designed and is detailed in the recovery system subsection. This requirement will be completed when recovery system's design is finalized, and the system is manufactured and implemented into the subscale and full-scale launch vehicles, which will both launch and complete their recovery sequences at the correct altitudes. All of this has been accounted for in the timeline in the design, ordering, manufacturing, and testing schedule of the recovery system, as well as in the integration and launch schedules for the subscale and full-scale launches.

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Requirement	Verification Plan	Status
3.1.1. The main parachute shall be deployed no lower than 500 feet.	The team has decided to deploy the parachute at 600 feet, and that is what the system will be set to while it is being assembled, which will be completed by checklists which will be signed by the team member in charge of this system, the Team SO, and the Recovery Team Lead, and the process will be photographed as well in the event that the team needs to reexamine the parachute set-up.	In progress - The initial recovery system is detailed in the recovery system section, however for this requirement to be completed, the recovery system will be finalized, manufactured, and implemented into the subscale and full-scale launch vehicles, which will complete their recovery sequences at the correct altitudes. This is accounted for in the timeline in the recovery system design, ordering, manufacturing, and testing schedules, along with the integration and launch schedules of both the subscale and full-scale launch vehicles.
3.1.2. The apogee event may contain a delay of no more than 2 s.	The team will set the apogee event to contain as little to no delay as possible during the subscale launches in order to allow for both a CO2 and BP charge to fire. The CO2 will fire first, as that is what the team is trying to perfect, but until it is perfected, a back-up black powder charge will be ignited 2 s later to ensure that the drogue parachute is deployed. This is what the system will be set to while it is being assembled, which will be completed by checklists, which will be signed by the team member in charge of this system, the Team SO, and the Recovery Team Lead, and the process will be photographed as well as in the event that the team needs to reexamine the parachute ejection charge set-up.	Not completed - This will be completed when the subscale and full-scale launches complete their apogee events within the first two seconds of reaching apogee. This is accounted for in the timeline in the integration and launch day schedules of the subscale and full-scale launch vehicles.

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Requirement	Verification Plan	Status
3.1.3. Motor ejection is not a permissible form of primary or secondary deployment.	The team will design the motor retention system to ensure that the motor will not fall out either during or directly after launch, and will test their system at the first subscale launch for verification that it securely holds the motor in place.	In progress - A motor retention system has been initially designed, as seen in the vehicle specifications section, however, this requirement will be completed when the motor retention system's design is further developed and finalized, the system is built, and has successfully completed one subscale launch. All of this is accounted for in the timeline in the launch vehicle design, component ordering, and manufacturing schedules, as well as in the integration and launch day schedules.
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.	The team will conduct at least 5 ground ejection tests for both the drogue and the main parachutes in order to ensure that both the parachute and ejection system work, and that the operators receive enough practice to minimize the chance of making a error that could cause the recovery system to fail when it comes time to deploy the parachutes during a launch. These tests will be photographed, and a summary of what was done and what happened will be written no later than 3 days after the testing took place.	Not completed - This will be completed when the parachutes and their ejection system's designs are finalized, and are constructed to the point that they can be tested.
3.3. Each independent section of the launch vehicle will have a maximum kinetic energy of 75 ft-lbf at landing.	The team will run simulations in OpenRocket in order to determine the correct size and shape of the parachutes, as well as when they need to deploy, to determine how to land with a maximum kinetic energy equal to or less than 75 ft-lbf.	Complete - For the full-scale launch, the drogue will open at apogee, and the main will open at 600 ft, greatly reducing the kinetic energy of of landing to below 75 ft-lbf

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Requirement	Verification Plan	Status
3.4. The recovery system will contain redundant, commercially available altimeters. The term “altimeters” includes both simple altimeters and more sophisticated flight computers.	The team will purchase and install at least four different altimeters, one for each ejection charge and one for each back up ejection charge.	Not completed - This requirement will be complete when altimeters are purchased, tested, and installed. This is accounted for in the timeline in the recovery system design, ordering, manufacturing, testing, and launch vehicle integration schedules.
3.5. Each altimeter will have a dedicated power supply, and all recovery electronics will be powered by commercially available batteries.	The team will ensure that each altimeter has its own battery, and will select a battery that can be purchased either online or from local hobby stores.	Not completed - This requirement will be completed when a battery has been selected and purchased for each altimeter and installed. If these batteries are Lithium Polymer batteries, this will be completed when the battery is also properly marked and labeled. This has been accounted for in the timeline in the recovery system design, ordering, manufacturing, and testing schedules
3.6. Each altimeter will be armed by a dedicated mechanical arming switch that is accessible from the exterior of the launch vehicle airframe when the launch vehicle is in the launch configuration on the launch pad.	The team will design each avionics bay so that the mechanical arming switch is easily accessible from the exterior of the launch vehicle and the altimeters can be armed within 10 seconds.	Not completed - This requirement will be complete when the avionics bays are laid out, built, and tested to ensure that the arming switches are easy to find and turn on. This has been accounted for in the timeline in the recovery and launch vehicle designs, ordering, manufacturing, and testing schedules.

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Requirement	Verification Plan	Status
3.7. Each arming switch will be capable of being locked in the ON position for launch (i.e. cannot be disarmed due to flight forces).	The team will ground test the arming switches to ensure that the switches cannot be disarmed by flight forces.	Not completed - This requirement will be completed when arming switches are selected, purchased, and tested. This has been accounted for in the timeline in the recovery design, ordering, manufacturing, and testing.
3.8. The recovery system electrical circuits will be completely independent of any payload electrical circuits.	The team will design the payload and recovery system electrical circuits will be independently designed and built by separate member on the team as to not depend on each other electrically in any way.	Not completed - This requirement will be complete when the payload and recovery system electrical circuits are designed and manufactured so that the recovery electrical system does not rely on the payload electrical system. This has been accounted for in recovery system design, ordering, and manufacturing schedules.
3.9. Removable shear pins will be used for both the main parachute compartment and the drogue parachute compartment.	The team will use 2-56 1/4-in. nylon shear pins to fix the nose cone to the fore section of the launch vehicle, and the fore section of the vehicle to the aft section of the vehicle.	In progress - While shear pins have been selected, this requirement Will be complete when shear pins have been purchased, tested, and installed in the launch vehicle. This has been accounted for in the timeline in the recovery system ordering and testing schedules, and integration schedules of the launch vehicle.

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Requirement	Verification Plan	Status
3.10. The recovery area will be limited to a 2,500 ft. radius from the launch pads.	The team will deploy the main parachutes as low as possible in order to minimize the mount of drift while still keeping the landing kinetic energy less than or equal to 52 fps.	In progress - The altitude at which the main parachutes deploy has been selected, as shown in the recovery system section, Therefore, this requirement Will be complete when the recovery system components are purchased and assembled to deploy at 600 feet. This has been accounted for in the timeline in the recovery system design, ordering, and manufacturing schedules.
3.11. Descent time will be limited to 90 seconds (apogee to touch down).	The team will deploy the main parachutes as low as possible in order to minimize the amount of time it takes the launch vehicle to go from apogee to landing while still keeping the landing kinetic energy less than or equal to 52 fps.	In progress - The altitude at which the main parachutes deploy has been selected, Therefore, this requirement Will be complete when the recovery system components are purchased and assembled to deploy at 600 feet. This has been accounted for in the timeline in the recovery system design, ordering, and manufacturing schedules.

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Requirement	Verification Plan	Status
3.12. 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.	The team will install an active electronic tracking device in the aft section of the airframe, and either in the fore section of the airframe or in the nose cone, and will have a life of at least 18 hours to ensure that the active tracking is still functional by the time it is launched, even if the launch vehicle waits for several hours on the launch pad.	In progress - the size of the recovery system is already determined through initial recovery system designs, however, this requirement will be completed when the active electronic tracking devices are purchased and installed. This has been accounted for in the timeline in the recovery system's design, ordering, manufacturing, and testing schedules, and the team's integration system.
3.12.1. Any section or payload component, which lands untethered to the launch vehicle, will contain an active electronic tracking device.	The team will install an active electronic tracking device in the aft section of the airframe, and either in the fore section of the airframe or in the nose cone, and will have a life of at least 18 hours to ensure that the active tracking is still functional by the time it is launched, even if the launch vehicle waits for several hours on the launch pad.	In progress - the size of the recovery system is already determined through initial recovery system designs, however, this requirement will be completed when the active electronic tracking devices are purchased and installed. This has been accounted for in the timeline in the recovery system's design, ordering, manufacturing, and testing schedules, and the team's integration system.

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Requirement	Verification Plan	Status
3.12.2. The electronic tracking device(s) will be fully functional during the official flight on launch day.	The team will test extensively electronic tracking devices before launch day and will record what tests were conducted and the results of those tests within 3 days after conducting the tests.	In progress - the type of electronic tracking device has already been determined through initial recovery system designs, however, this requirement will be completed when the active electronic tracking devices are purchased and installed. This has been accounted for in the timeline for recovery system's design, ordering, manufacturing, and testing schedules. It has also been accounted for in the team's integration system.
3.13. The recovery system electronics will not be adversely affected by any other on-board electronic devices during flight (from launch until landing).	The team will design a shield that can be easily placed and secured around the recovery system electronics to prevent them from being adversely affected by other on-board electronic devices during flight.	Not completed - This requirement will be complete when the shielding system is designed, finalized, manufactured, and tested. This is accounted for in the timeline in the recovery system design, ordering, manufacturing, and testing schedules.
3.13.1. 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.	The team will design the sections of the launch vehicle to allow space to for the recovery system altimeters to be separated from other radio frequency transmitting device.	In progress - The initial designs of the recovery system and launch vehicle have been completed, but, the requirement will be complete when the avionics and parachute bay's designs are finalized. This is accounted for in the timeline in both recovery and launch vehicle design.

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Requirement	Verification Plan	Status
3.13.2. The recovery system electronics will be shielded from all onboard transmitting devices to avoid inadvertent excitation of the recovery system electronics.	The team will design a shield that can be easily placed and secured around the recovery system electronics to prevent the early excitation of the recovery system due to transmitting devices.	Not completed - This requirement will be complete when the shielding system is designed, finalized, manufactured, and tested. This is accounted for in the timeline in the recovery system design, ordering, manufacturing, and testing schedules.
3.13.3. The recovery system electronics will be shielded from all onboard devices which may generate magnetic waves (such as generators, solenoid valves, and Tesla coils) to avoid inadvertent excitation of the recovery system.	The team will design a shield that can be easily placed and secured around the recovery system electronics to prevent the early excitation of the recovery system due to magnetic waves.	Not completed - This requirement will be complete when the shielding system is designed, finalized, manufactured, and tested. This is accounted for in the timeline in the recovery system design, ordering, manufacturing, and testing schedules.
3.13.4. The recovery system electronics will be shielded from any other onboard devices which may adversely affect the proper operation of the recovery system electronics.	The team will design a shield that can be easily placed and secured around the recovery system electronics to prevent the early excitation of the recovery system due to all other sources other than transmitting devices and magnetic waves.	Not completed - This requirement will be complete when the shielding system is designed, finalized, manufactured, and tested. This is accounted for in the timeline in the recovery system design, ordering, manufacturing, and testing schedules.

6.1.4 Payload Requirements

Table 49: Payload Experiment Requirement Verification Matrix

Requirement	Verification Plan	Status
4.2 Teams will design a system capable of being launched in a high power launch vehicle, landing safely, and recover simulated lunar ice from one of several locations on the surface of the launch field.	The team will design, build and test a payload that can fit within the airframe of the launch vehicle sustaining launch forces. Once the launch vehicle has landed, it will be able to be deployed and navigate to one of the predetermined locations to retrieve a lunar ice sample; carrying the sample away from the location from which it was taken.	In progress - The initial payload design is small enough to be launched inside a launch vehicle however, this requirement will be completed when the payload design is finalized and it is built and tested. All of this has been accounted for in the timeline, the payload design, ordering, manufacturing and testing schedules.
4.3.1. The launch vehicle will be launched from the NASA-designated launch area using the provided Launch pad. All hardware utilized at the recovery site must launch on or within the launch vehicle.	The launch vehicle will entirely contain the payload within the airframe and be able to deploy from the launch vehicle once it has landed. It will be able to navigate to one of the predetermined collection areas to collect a lunar ice sample without any exterior hardware.	In progress - The initial payload design, as depicted in the payload section, is small enough to fit all required hardware for operation into the airframe of the launch vehicle however, this requirement will be completed when the payload design is finalized and built and has been integrated with the launch vehicle. This is accounted for in the payload design, ordering, manufacturing, and testing schedules.

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Requirement	Verification Plan	Status
4.3.2. Five recovery areas will be located on the surface of the launch field. Teams may recover a sample from any of the recovery areas. Each recovery site will be at least 3 ft in diameter and contain sample material extending from ground level to at least 2 in. below the surface.	The team will design, build, and test a payload that can navigate to the recovery site, and dig down three in. below the surface in order to access an ice sample.	In progress - The initial payload design has been completed, and is depicted in the payload section. This requirement will be completed when the payload design is finalized and built, the payload test bed is designed and built and the payload navigation and digging capabilities are tested. This has been accounted for in the proposed timeline in the payload design, ordering, manufacturing, and testing schedules.
4.3.3. The recovered ice sample will be a minimum of 10 milliliters (mL).	The team will design a payload that has an ice sample storage capacity of at least 15 milliliters to ensure that it can collect and store the required amount of ice.	In progress - The ice collection system has an initial design, as depicted in the ice collection system section, but this requirement will be completed when the ice collection system design is finalized, built, and its storage capacity is tested. All of this is accounted for in the proposed timeline in the payload design, ordering, manufacturing, and testing schedules.
4.3.4. Once the sample is recovered, it must be stored and transported at least 10 linear feet from the recovery area.	The team will design the payload to have a storage system with the capability to securely hold the ice sample and will have the capability to drive at least 20 linear feet from the recovery area to ensure that it makes it far enough from the recovery area.	In progress - The design has been worked on, but needs to be finalized and the payload needs to be built and tested. All of this has been accounted for in the proposed timeline.

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Requirement	Verification Plan	Status
4.3.5. Teams must abide by all FAA and NAR rules and regulations.	The team will familiarize itself with FAA and NAR rules and regulations and design a system that does not violate any rules or regulations from the FAA and NAR.	In progress - The payload is being designed to FAA and NAR standards. This design time has been accounted for in the proposed timeline.
4.3.6. Black Powder and/or similar energetics are only permitted for deployment of in-flight recovery systems. Any ground deployments must utilize mechanical systems.	The team will design, build, and test a mechanical payload deployment system that uses a motor to turn a threaded rod which will contact a metal cylinder that pushes the payload out of the airframe.	In progress - The design needs to be finalized, built, and tested. All of this has been accounted for in the proposed timeline.
4.3.7. Any part of the payload or vehicle that is designed to be deployed, whether on the ground or in the air, must be fully retained until it is deployed as designed.	The team will design, build, and test a payload retention system that will retain the payload until it is intended to be deployed.	In progress - The payload retention system design needs to be finalized, built and tested. All of this has been accounted for in the proposed timeline.
4.3.7.1. A mechanical retention system will be designed to prohibit premature deployment.	The team will design and build a mechanical retention system that will prohibit the payload from prematurely deploying both in flight, recovery, and on the ground until it is supposed to do so.	In progress - The payload retention system design needs to be finalized, built, and tested. All of this has been accounted for in the proposed timeline.
4.3.7.2. The retention system will be robust enough to successfully endure flight forces experienced during both typical and atypical flights.	Once the retention system is designed, the team will put it through rigorous testing to ensure that it can withstand anything from launch forces to ballistic impact forces.	Not completed - the payload retention system needs to finished being designed and built before it can be tested. Testing is accounted for in the proposed timeline.

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Requirement	Verification Plan	Status
4.3.7.3. The designed system will be fail-safe.	The team will design the retention system so that if the system loses power before the launch vehicle lands, the rover will still be secured inside the airframe and not pose a safety threat to spectators.	In progress - The payload retention system design needs to be finalized, built, and tested. All of this has been accounted for in the proposed timeline.

6.1.5 Safety Requirements

Table 50: Safety Requirement Verification Matrix

Requirement	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.	The team will develop a launch and safety checklist for each system that is designed and implemented into the launch vehicle, the recovery system, and/or the payload.	In progress - Checklists will be developed and finalized with the designs. This portion of the design process has been accounted for in the proposed timeline.
5.2. Each team must identify a student SO who will be responsible for all items in section 5.3.	The team will identify a student SO by September 14th, 2019.	Completed - The team student SO is Wyatt Hougham

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Requirement	Verification Plan	Status
5.3. The role and responsibilities of the SO will include, but are not limited to: Monitor team activities with an emphasis on safety during: Design of vehicle and payload, Construction of vehicle and payload components, Assembly of vehicle and payload, Ground testing of vehicle and payload, Subscale launch test(s), Full-scale launch test(s), Launch day, Recovery activities, and STEM Engagement Activities.	The SO will approve final designs before orders are placed, The SO and their safety team will help subteams draft safety documents for the construction process of their respective system, and either The SO, or a member of the team appointed and trained by The SO, will be present at all construction and assemblies of systems, as well as at all ground tests, launches, recovery activities, and The STEM Engagement activities.	In progress - the designs will be approved as they are finalized, the safety team is working on drafting up documents to help the subteams complete them more efficiently and effectively, and The SO is working on training members of the safety team, and members of the team in general, so that they can act as SO in the event that The SO is unable to.
5.3.2. Implement procedures developed by the team for construction, assembly, launch, and recovery activities.	The Safety team will work alongside the subteams to develop procedures that are both safe and effective for construction, assembly, launch, and recovery activities.	Not completed - This will start after the designs are finalized, and is accounted for in the construction time for each portion of this project.
5.3.3. Manage and maintain current revisions of the team's hazard analyses, failure modes analyses, procedures, and MSDS/chemical inventory data.	The Safety team will maintain a binder that will be stored in OSRT's main work space in an easily accessible area. This binder will hold all of the team's current hazard analyses, failure modes and analyses, procedures, and MSDS/chemical inventory data, and will be updated weekly.	In progress - Hazard analyses have been started, and the rest of the binder will be created as subteam's designs become finalized and as they create construction plans, and has been included in the timeline.

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Requirement	Verification Plan	Status
5.3.4. Assist in the writing and development of the team's hazard analyses, failure modes analyses, and procedures.	The Safety team will assist the subteams with writing and developing their hazard analyses, failure modes and analyses, and procedures throughout the duration of the project.	In progress - Hazard analyses have been started, and failure modes and analyses and procedures will be created as the designs are finalized and construction procedures are created. The construction of these safety documents is included in the design and construction phases of the timeline.
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 does not give explicit or implicit authority for teams to fly those 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.	The SO and Team Lead will work together to contact the local club's President of Prefect and RSO at least a week before attending any NAR or TRA launch.	Not completed - This will be completed as needed, and therefore, will not be included in the timeline.
5.5. Teams will abide by all rules set forth by the RSO	The SO and Safety Team will familiarize themselves with the FAA rules regarding rocketry, and help the rest of the team familiarize themselves with the FAA rules. The Safety Team will be responsible for holding the entire team to the FAA rules.	In progress - The Safety Team is in the process of familiarizing themselves with the FAA rules and is included on the timeline as looking over last years documentation, and working on research and design of the launch vehicle, recovery system, and payload.

6.1.6 Team Derived Requirements

Table 51: Team Derived Requirement Verification Matrix

Requirement	**Verification Plan**	**Status**
6.1. Maintain a minimum safety factor of 2 on all systems	Calculations and simulations of designs and will be conducted ensuring a minimum factor of safety of all components and systems on the launch vehicle.	In progress - All current designs determined to have minimum factor of safety of 2. will be continually checked and updated for every deliverable, change in design or new design
6.2. Team can deploy and operate rover from up to 1/2 mile away	Test rover and ejection systems at appropriate distance	In progress - Electrical system and rover controls are currently being designed to be controlled remotely from afar
6.3. Both full scale and sub scale launch vehicles will be checked post flight for defects or wear	A checklist will be created for post flights analysis of launch vehicles, with areas of interest to check for wear, or excessive stresses on the airframe and subsystems. Any defects or issues with subsystems will be reported and fixed before the next flight.	Incomplete - Will be completed before first subscale launch and improved on for each consecutive launch. with documentation available for review at any point to verify launch readiness
6.4. Rover collection system must be capable of digging into various soil types	Auger will be tested at on a variety of soil types	In progress - Auger system is being designed to be capable of collecting a variety of soil types
6.5. Rover wheels must be able to expand to increase ground clearance	Rover will be observed being ejected from the airframe to confirm that the wheels expand	In progress - Designs have been developed which allow for the wheels to expand upon exiting the airframe
6.6. Rover collection system must be designed to securely retain sample	System will be tested to ensure sample is securely stored transport	In progress - A cap or one way valve is currently being designed for collection system
6.7. Two step Dynamic Ultra Once-over (DUO) verification of checklists prior to flight will be completed for both full scale and subscale launches	Checklists will be completed and signed by two team members to ensure all steps are completed properly, and any mistakes are identified. Two signatures will be required on each checklist prior to each launch	Incomplete - Will be completed during each launch with documentation available for review at any point to verify launch readiness

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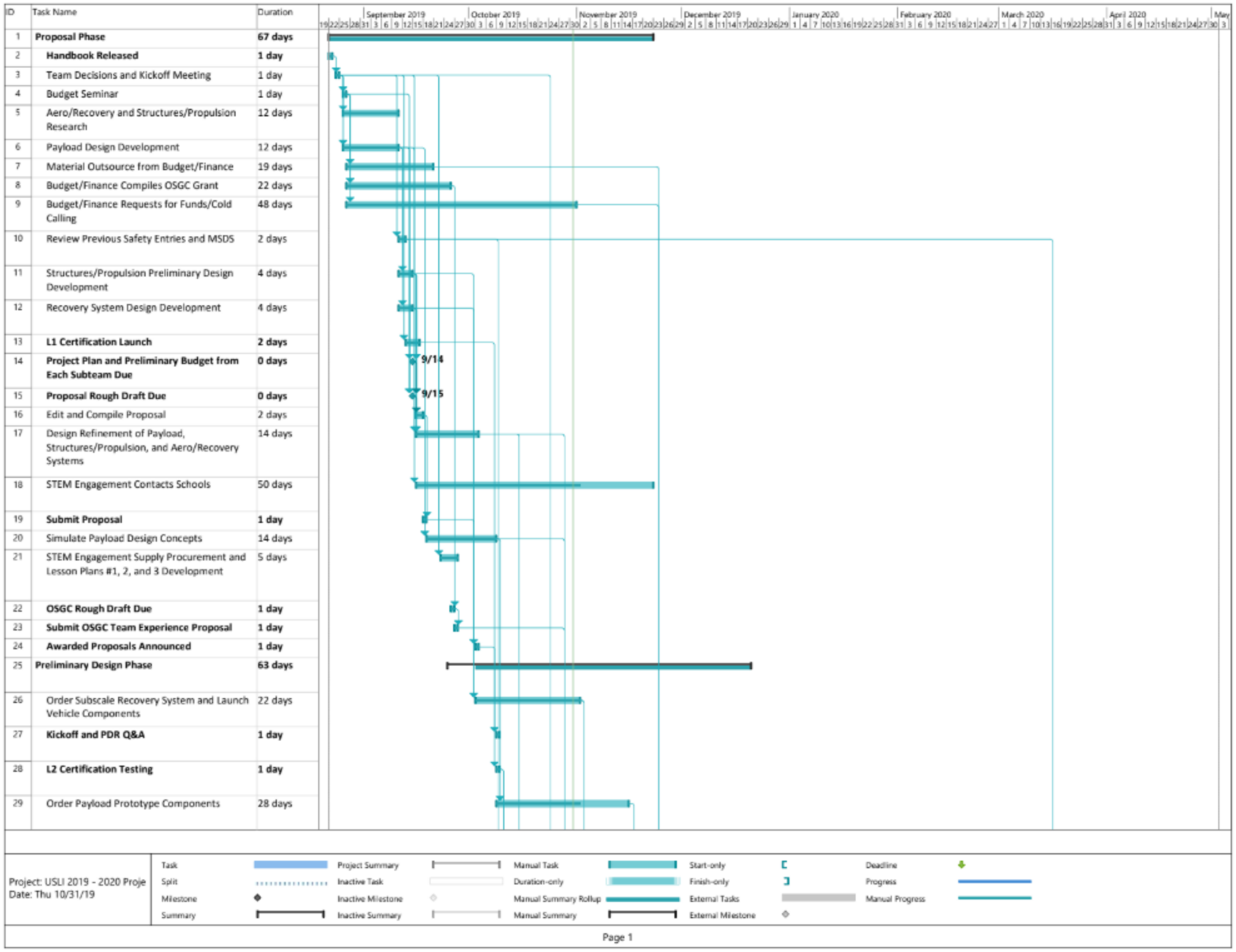
Table 51 – continued from previous page

Requirement	Verification Plan	Status
6.8. OpenRocket simulations will be performed prior to each launch to hit a predetermined altitude	OpenRocket will be used to verify apogee and the weight adjusted accordingly before every flight using measured weight of the launch vehicle	Incomplete - Will be completed before each launch, before arriving to and on the launch site
6.9. All screws and threaded rods will be analysed for stress to ensure proper structural integration	Analysis will be conducted on all threaded rods and screws prior to design integration to ensure they will hold the loads and stresses required of them	In progress - All screws and threaded rods currently in design are calculated to withstand their required stresses to a factor of safety. Will be updated as parts are added or taken away.
6.10. All materials to be used for the manufacturing process will have credible analysed properties	All materials used will required data sheet either from the manufacturing company or verified through testing	In progress - All current materials used adhere to these requirements, and all future materials will be verified before integration
6.11. Safety officer will verify and check all preflight checklists prior to launch	Checklists will require the signature of a safety officer before launch of launch vehicle	Incomplete - Will be completed prior to each launch with documentation available to verify launch readiness
6.12. When no existing tool will allow for manufacturing within desired specifications, OSRT will manufacture such a tool.	Analysis will be performed on existing manufacturing options and, if none are sufficient, proprietary means will be pursued.	In Progress - Tools will be manufactured as need arises.
6.13. Testing will be conducted with the goal and expectation of developing meaningful data and information in order to improve a part, system, or method	All test data will be accessible to team members for use and verification	In Progress - Testing data is made available via team storage drives as testes are conducted.
6.14. All batteries used for a flight will be charged, if rechargeable, or never before used, if single use, prior to each flight.	Rechargeable batteries will be marked with the date of charging once charged prior to launch. Single use batteries will be removed from manufacturer packaging during integration on the day of launch. This will be verified via checklists per Requirement 1.2	Incomplete - Will be enacted prior to the first subscale launch.
6.15. All PCBs used in the launch vehicle will use electrical potting material to cover components.	Electrical potting material will be confirmed to be present on all PCBs during launch vehicle integration via checklists as per Requirement 1.2	Incomplete - Electrical potting will be done once PCBs have arrived and are assembled.

6.2 Project Plan



ID	Task Name	Duration
1	**Proposal Phase**	**67 days**
2	**Handbook Released**	**1 day**
3	Team Decisions and Kickoff Meeting	1 day
4	Budget Seminar	1 day
5	Aero/Recovery and Structures/Propulsion Research	12 days
6	Payload Design Development	12 days
7	Material Outsource from Budget/Finance	19 days
8	Budget/Finance Compiles OSGC Grant	22 days
9	Budget/Finance Requests for Funds/Cold Calling	48 days
10	Review Previous Safety Entries and MSDS	2 days
11	Structures/Propulsion Preliminary Design Development	4 days
12	Recovery System Design Development	4 days
13	**L1 Certification Launch**	**2 days**
14	**Project Plan and Preliminary Budget from Each Subteam Due**	**0 days**
15	**Proposal Rough Draft Due**	**0 days**
16	Edit and Compile Proposal	2 days
17	Design Refinement of Payload, Structures/Propulsion, and Aero/Recovery Systems	14 days
18	STEM Engagement Contacts Schools	50 days
19	**Submit Proposal**	**1 day**
20	Simulate Payload Design Concepts	14 days
21	STEM Engagement Supply Procurement and Lesson Plans #1, 2, and 3 Development	5 days
22	**OSGC Rough Draft Due**	**1 day**
23	**Submit OSGC Team Experience Proposal**	**1 day**
24	**Awarded Proposals Announced**	**1 day**
25	**Preliminary Design Phase**	**63 days**
26	Order Subscale Recovery System and Launch Vehicle Components	22 days
27	**Kickoff and PDR Q&A**	**1 day**
28	**L2 Certification Testing**	**1 day**
29	Order Payload Prototype Components	28 days

Figure 105: OSRT Project Plan 1/5

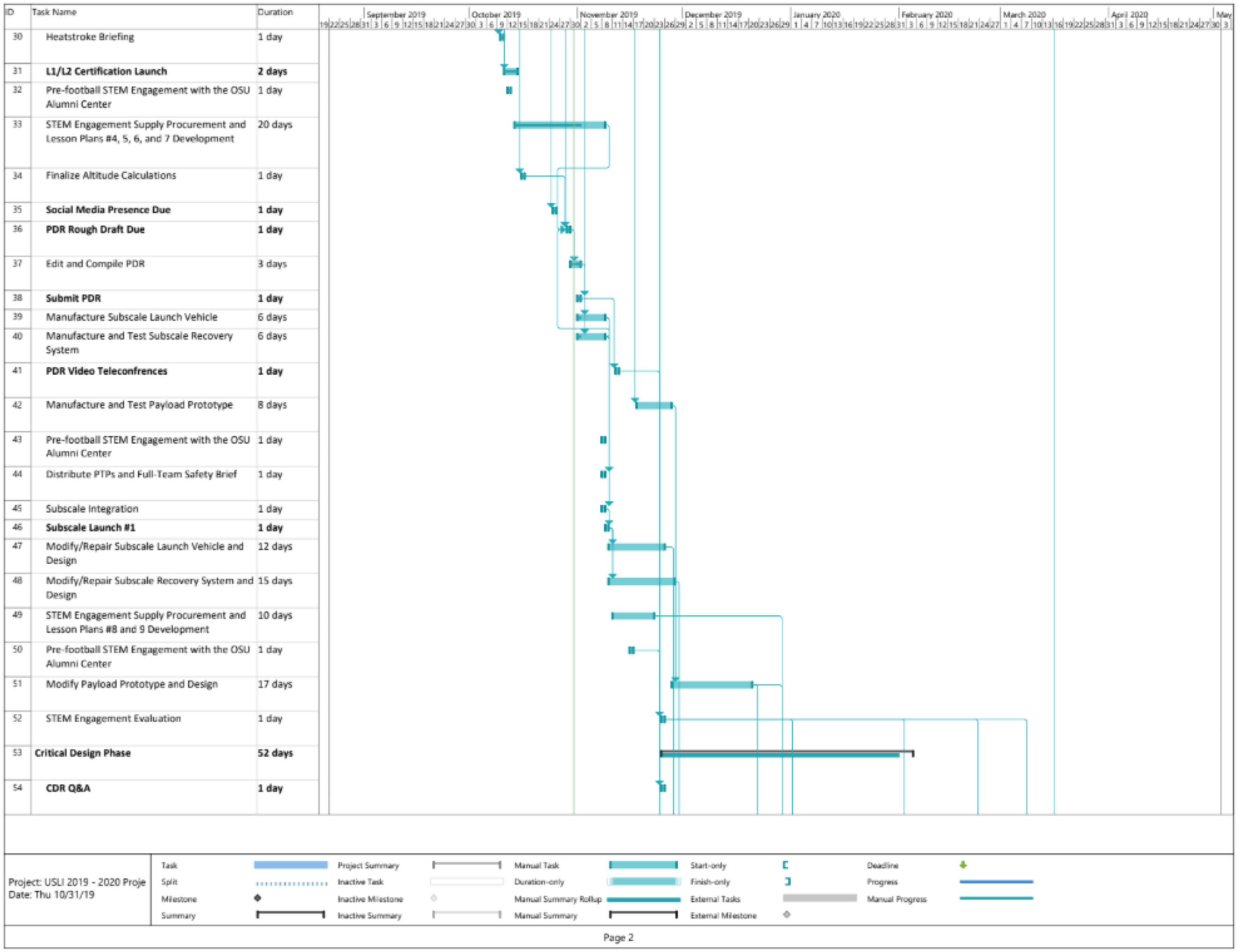
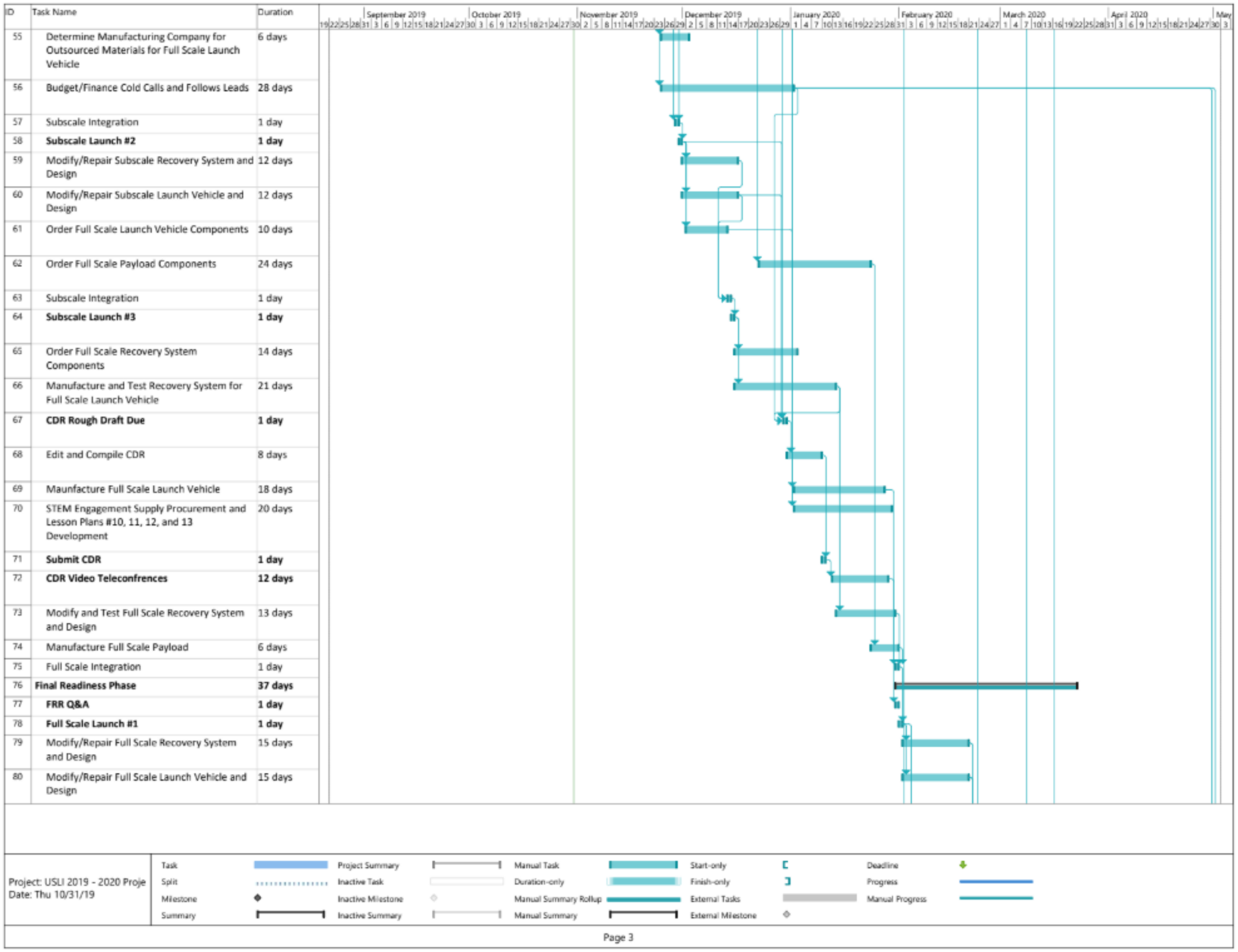


Figure 106: OSRT Project Plan 2/5



ID	Task Name	Duration
55	Determine Manufacturing Company for Outsourced Materials for Full Scale Launch Vehicle	6 days
56	Budget/Finance Cold Calls and Follows Leads	28 days
57	Subscale Integration	1 day
58	**Subscale Launch #2**	**1 day**
59	Modify/Repair Subscale Recovery System and Design	12 days
60	Modify/Repair Subscale Launch Vehicle and Design	12 days
61	Order Full Scale Launch Vehicle Components	10 days
62	Order Full Scale Payload Components	24 days
63	Subscale Integration	1 day
64	**Subscale Launch #3**	**1 day**
65	Order Full Scale Recovery System Components	14 days
66	Manufacture and Test Recovery System for Full Scale Launch Vehicle	21 days
67	**CDR Rough Draft Due**	**1 day**
68	Edit and Compile CDR	8 days
69	Maunfacture Full Scale Launch Vehicle	18 days
70	STEM Engagement Supply Procurement and Lesson Plans #10, 11, 12, and 13 Development	20 days
71	**Submit CDR**	**1 day**
72	**CDR Video Teleconfrences**	**12 days**
73	Modify and Test Full Scale Recovery System and Design	13 days
74	Manufacture Full Scale Payload	6 days
75	Full Scale Integration	1 day
76	**Final Readiness Phase**	**37 days**
77	**FRR Q&A**	**1 day**
78	**Full Scale Launch #1**	**1 day**
79	Modify/Repair Full Scale Recovery System and Design	15 days
80	Modify/Repair Full Scale Launch Vehicle and Design	15 days

Figure 107: OSRT Project Plan 3/5

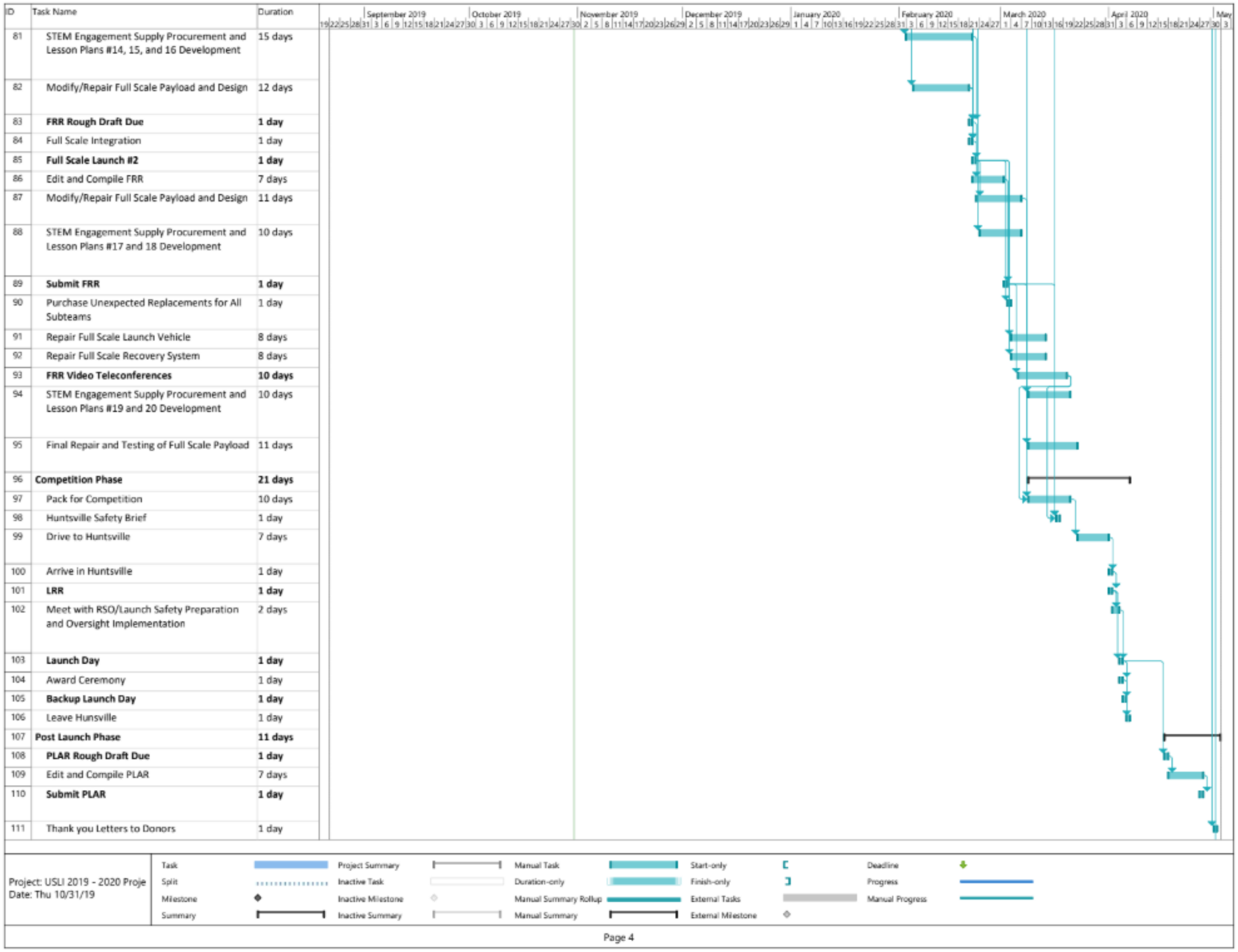


Figure 108: OSRT Project Plan 4/5

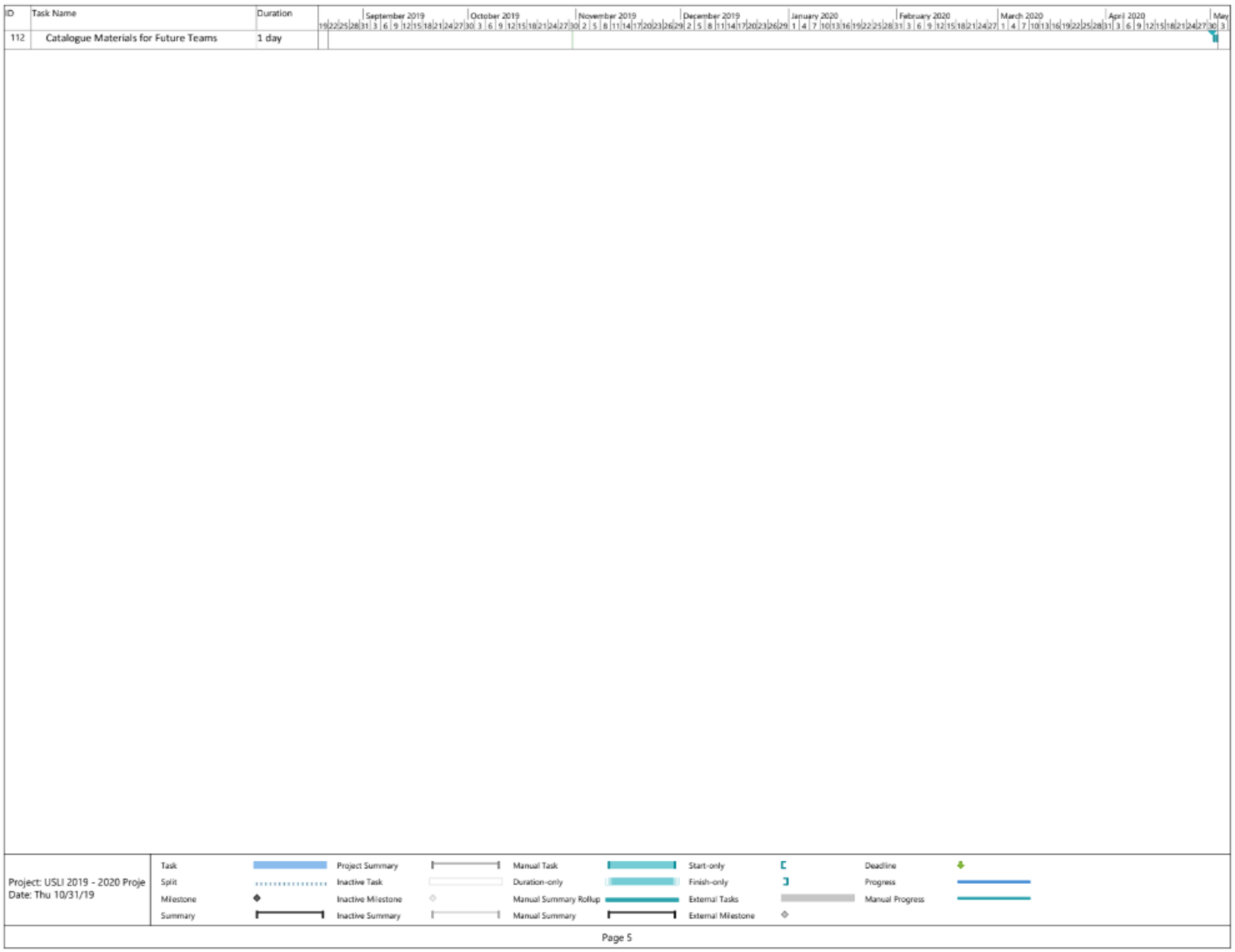


Figure 109: OSRT Project Plan 5/5

6.3 Project Budget

The updated budget is presented in table 52 and it is depicted in Figure 110. They represent the distribution of anticipated launch vehicle materials that will be used for fabrication, which include Structural, Payload and Aero and Recovery fabrication. Budget distribution will also anticipate budget use for traveling, STEM engagement and testing. The most reliable and economical suppliers were considered for material procurement.

Table 52: Budget

Quantity	Object	Cost per Unit	Vendor	Total Cost
Aerodynamics and Recovery				
2	8 ft Main Parachute	$348.15	Fruity Chutes	$696.30
2	7 ft Main Parachute	$296.96	Fruity Chutes	$593.92
8	Drogue Parachute	$55.97	Fruity Chutes	$447.76
10	Shock Cord (per yard)	$23.13	Fruity Chutes	$231.30
6	Shear Pins	$5.50	Home depot	$33.00
4	Nomex Fire Proof Blankets	$27.00	Fruity Chutes	$108.00
15	Quick Release Hooks	$4.10	Apogee components	$61.50
6	Descenders	$81.43	Apogee components	$488.58
2	6 ft Chute	$225.75	Fruity Chutes	$451.50
2	5 ft Chute	$193.50	Fruity Chutes	$387.00
1	8 ft Deployment Bag	$74.18	Fruity Chutes	$74.18
1	7 ft Deployment Bag	$46.23	Fruity Chutes	$46.23
1	6 ft Deployment Bag	$46.23	Fruity Chutes	$46.23
1	5 ft Deployment Bag	$46.23	Fruity Chutes	$46.23
BEAVS				
1	MS5802-14AB Barometric Pressure Sensor	$4.87	Mouser	$4.87
1	MMA8452Q Accelerometer	$34.95	Sparkfun	$34.95
1	Turnigy 12v LiPo	$10.99	HobbyKing	$10.99
1	OSRT Designed PCB	$0.00	OSRT	$0.00
1	FIT0441 Brushless DC Motor	$19.90	DFRobot	$19.90
1	Teensy 3.6 Microcontroller	$29.25	Sparkfun	$29.25

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Quantity	Object	Cost per unit	Vendor	Total Cost
2	1/8 in. aluminum plate (2 in. x 24 in. bar)	$11.08	McMaster	$11.08
4	1/4-20 fasteners	$0.00	OSRT	$0.00
4	8-32 Threaded Rod	$1.67	McMaster	$6.68
3	1/2 in. Aerospace Grade Plywood Bulkhead	$1.53	Wicks	$4.59
1	PLA 3D Printer Filament (1 kg)	$19.99	Amazon	$19.99
100	M2 Fasteners	$0.00	OSRT Machine Shop	$0.00
2	7mm Linear Guide Block	$41.33	McMaster	$82.66
2	7mm Linear Rail (24 mm)	1.06	McMaster	$42.12
10	GA Steel Plate	$2.37	JCI	$23.70
1	Gear	$34.69	McMaster	$34.69
		Parachute Ejection		
1	Yakamoz 14pcs 0.5-3mm Small Electric Drill Bit	$8.99	Amazon	$8.99
3	Turnigy D1104-4000 kv 5.5g Brushless Motor	$7.09	HobbyKing	$21.27
1	Hobbywing Quicrun 60 A 2S-3S Waterproof Brushed ESC for 1/10	$20.99	HobbyKing	$20.99
2	Turnigy 1700 mAh 2S 20C Lipo Pack (Suits 1/16th Monster Beatle, SCT & Buggy)	$8.82	HobbyKing	$17.64
1	Turnigy 12 v 2-3S Basic Balance Charger	$5.10	HobbyKing	$5.10
1	Turnigy TGY-i6 AFHDS Transmitter and 6CH Receiver (Mode 1)	$57.80	HobbyKing	$57.80
1	2 in. x 2 in. x 3 ft Aluminum Stock	$78.72	McMaster	$78.72
1	INTOO Mini Drill Bit Set 60 Pcs+12 Pcs	$11.99	Amazon	$11.99
5	12gm CO2 Cartridge (each)	$4.50	Tinder Rocketry	$22.50
1	NYLON SHEAR PINS - 20 PACK	$3.22	Apogee Components	$3.22

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Quantity	Object	Cost per unit	Vendor	Total Cost
1	Fantasycart Fiberglass Cloth Plain Weave 4.12 Oz 39 in. wide in 16.6 yards long	$36.99	Amazon	$36.99
1	3M 20124 All Purpose Fiberglass Resin, 1 Gallon	$57.40	Amazon	$57.40
6	1 ft of 1/2 in. ID Surgical Tubing	$1.78	McMaster	$10.68
2	Pack of 24 E-Matches	$15.60	MJG Technologies	$31.20
4	1 ft of 1/2 in. OD Silicone Rubber Rod	$7.45	McMaster	$29.80
1	1 lb of GOEX FFFFg Black Powder	$17.95	Powder Valley Inc.	$17.95
1	8 in. Black Cable Ties 100 Pk.	$1.99	Harbor Freight	$1.99
1	Hornady G2-1500 Digital Powder Scale 1500 Grain Capacity	$39.49	Midway USA	$39.49
1	5 ft of 1/2 in. ID Surgical Tubing	$8.90	McMaster	$8.90
1	80 3 ft Long Firewire Ignitors	$60.00	Aircraft Spruce Co	$60.00
1	1 ft of 1/2 in. OD Silicone Rubber Rod	$7.45	McMaster	$7.45
1	8 in. Black Cable Ties 100 Pk.	$1.99	Harbor Freight	$1.99
1	1 in. 10 yd Nylon Shock cord	$30.13	Fruity Chutes	$30.13
3	1 in. 5 yd Nylon Shock cord	$23.13	Fruity Chutes	$69.93
			Recovery Total:	$4,711.18
Payload				
Rover Parts				
5	Driver Motors	$80.00	RobotShop	$80.00
1	Battery Charger	$35.00	Hobby King	$35.00
2	Battery	$100.00	Hobby King	$160.00
1	RC Remote/Reciever	$150	Hobby King	$150

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Quantity	Object	Cost per unit	Vendor	Total Cost
#	Structural Material	$100.00	Self Machined	$100.00
1	Camera/Reciver	$50.00	RobotShop	$50.00
#	Wire/Assorted Bits	$30.00	Home Depot	$30.00
1	3/4 in. Drill Bit	$17.00	Home Depot	$17.00
1	PBC Pipe	$2.00	Home Depot	$2.00
1	Collection Assembly	$20.00	Self Machined	$20.00
1	Lead Screw Motor	$15.00	Amazon	$15.00
Testing Accessories				
1	Terrain Bed Material	$30.00	Self Built	$30.00
Rocket Ejection				
1	Ejection Motor	$40.00	RobotShop	$40.00
1	Ejector	$20.00	Self Machined	$20.00
#	Composite Ejection Material	$60.00	Self Machined	$60.00
1	Payload Housing	$50.00	Self Machined	$50.00
			Payload Total:	$859.00
Structures/Propulsion				
Structure				
3	Tubes (fore, aft, motor)	$1666.67	Innovative Composite Engineering	$5000
1	Epoxy	$81	ApogeeRockets	$81
1	Epoxy Resin	$59	Fiberglass Supply Depot	$59.00
1	4 ft X 8 ft Carbon Sheet	$395.00	Tim McAmis Performance Parts	$395.00

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Quantity	Object	Cost per unit	Vendor	Total Cost
4	Fiberglass sheets	$6.50	Fiberglass Supply Depot	$26.00
1	Plywood	$43.50	Aircraft Spruce Co	$43.50
1	Nose Cone	$169.95	Madcowrocketry	$169.95
2	Aluminum "Pipe" Stock 4 in. OD, 1 ft thick, 3 in. long	$16.05	McMaster	$32.10
2	Threaded Rod	$5.97	McMaster	$11.94
8	Eye Bolt 1/4-28	$4.83	McMaster	$38.64
1	Coupler	$32.00	Madcowrocketry	$32.00
1	Bulkhead	$0	OSRT	$0
1	Switch Band	$5.00	Madcowrocketry	$5.00
1	Airfoiled Rail Buttons	$7.83	ApogeeRockets	$7.83
1	Aluminum Bar	$28.84	ApogeeRockets	$28.84
1	Aluminum "Pipe" Stock	$7.22	McMaster	$7.22
1	Motor Tube (22 in. long)	$27.00	McMaster	$27.00
1	G5000 Rocket Epoxy 2 Quart	$81.25	ApogeeRockets	$81.25
Propulsion				
3	AeroTech L2200G-P	$279.99	AeroTech	$839.97
1	RMS-75/5120 Casing W/Forward Seal Disk	$459.03	Apogee	$459.03
Devices				
4	Altimeter	$69.95	MissleWorks	$279.80
[redacted]			Structure/Propulsion Total:	$8,618.24
Testing Accessories				
1	Terrain Bed Material	$30.00	Self Built	$30.00

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Quantity	Object	Cost per unit	Vendor	Total Cost
		Educational Outreach		
#	Office Supplies, Stickers, PPE, High Vis. Clothing/Material	#	As Needed	$500.00
			Outreach Total:	$500.00
		Budget/Finance		
#	Office Supplies, Shipping and Handling	#	As Needed	$320.00
			Finance Total:	$320.00
		Administrative		
#	Office Supplies	#	As Needed	$100.00
			Administrative Total:	$100.00
		Safety		
#	Office Supplies, Stickers, PPE, High Vis. Clothing/Material	#	As Needed	$200.00
			Safety Total:	$200.00
		Traveling Expenses		
25	Plane Tickets	$400.00	United Airlines	$10,000.00
25	Lodging	$1,200.00	Hilton Hotels	$6,300.00
			Traveling Total:	$16,300.00

Project funding will come from a few different sources. OSRT has applied for the maximum of $12,000 from the Oregon Space Grant Consortium (OSGC). Innovative Composite Engineering (ICE) has agreed to donate approximately $5,000 by means of airframe materials. The students will bring in $10,000 ($400 each) by purchasing their own airfare. OSRT expects to need approximately $34,500 for the entirety of the project, including the cost to bring the team to Huntsville, Alabama for the competition.

OSU has multiple rocket teams on campus which work together to create a unified point of contact for American Institute of Aeronautics and Astronautics (AIAA) and large companies which want to support the team. This is important to making sure that companies are not contacted by multiple teams from OSU. In addition to the contacts brought in from the university the team will also personally contact as many local companies as possible for sponsorships or donations. The OSRT plans to bring in the remaining $7,500 through these sponsorships and donations. All sponsors will be featured on the OSRT website under a sponsors page and will also have a high resolution logo placed onto the body of the launch vehicle. Additionally, at all STEM educational outreach events, a billboard with a list of all the sponsors will be present.

Last year, the OSRT had a budget of $30,000 with $11,500 of that coming from OSGC. The remaining $18,500 came mostly through sponsorship. however, students covered the cost of their plane tickets from out-of-pocket expenses ranging from $300-$800. 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.

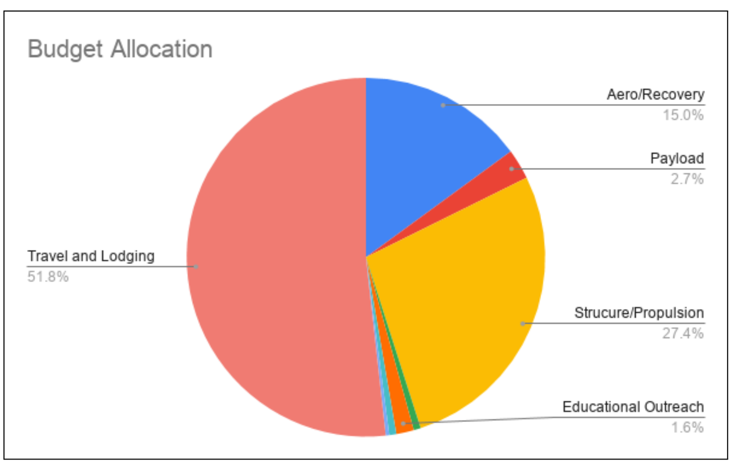

Figure 110: Budget Allocation

6.4 STEM Engagement

This year, STEM Engagement is participating in recurrent and new event opportunities. The team lead noticed a new opportunity provided by the Alumni Center of our university to host a table before the home football games. This year, OSRT is also participating in Science Saturday at the Clackamas Public Library. OSRT is also utilizing previous opportunities, such as continuing to participate in Discovery Days here at OSU. Each participating organization creates fun STEM activities for elementary school students while reaching out to schools to present STEM lesson plans.

The most recent event involved a display board giving general information about the team and the launch vehicle project, and featured a paper airplane making activity. The activity of creating paper airplanes seems relatively simple, but it was able to create a large impact. On that day our team drew the largest crowd that OSRT had at either of the two previous tables, and encouraged a large number of both children and adults to learn more about the club. Actual interest in what OSRT is doing as a club and as a team is existent, as shown when one audience member asked for the club's business card to offer future opportunities at their youth group; educational outreach was able to spur even more educational outreach!

There were multiple events in October. They were the football table described above and an FIRST Robotics Competition (FRC) robotics team in Albany, Oregon. Many events are scheduled for November, such as Science Saturday, Discovery Days, and more football tables. In addition to November presentations planned at five schools, January already has three confirmed school visits, with multiple leads being pursued for the following months. The minimum goal of two hundred students reached through STEM engagement is well within reach, with continual planning and efforts to exceed that goal and inspire as many people as possible to explore the possibilities of STEM education and involvement.

The OSRT is prepared for an important upcoming event, Science Saturday at the Clackamas Library. A short presentation and three interactive activity stations are planned: balloon rocket races, stretchy universe slime, and build your own rover. Rover building in particular will allow students to have fun while working on engineering skills. Engineering is only one part of STEM, but it is still so important to get children interested in STEM because people in STEM do so many different and incredible things, such as launch people into space; build our infrastructure; create technology and formulas that help solve the world's problems; and teach the next generation of innovators.

The STEM Engagement subteam strives to give students every opportunity that should be available to them. If they are never given the opportunity to engage in STEM then they will never know whether they like it or not. What would the world be if people did not realize that they enjoyed subjects that are so important to our future? OSRT is so proud to be a part of STEM Engagement so OSRT can build a better future. This will keep OSRT striving toward as many engagement opportunities that OSRT can find.

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8 Appendix

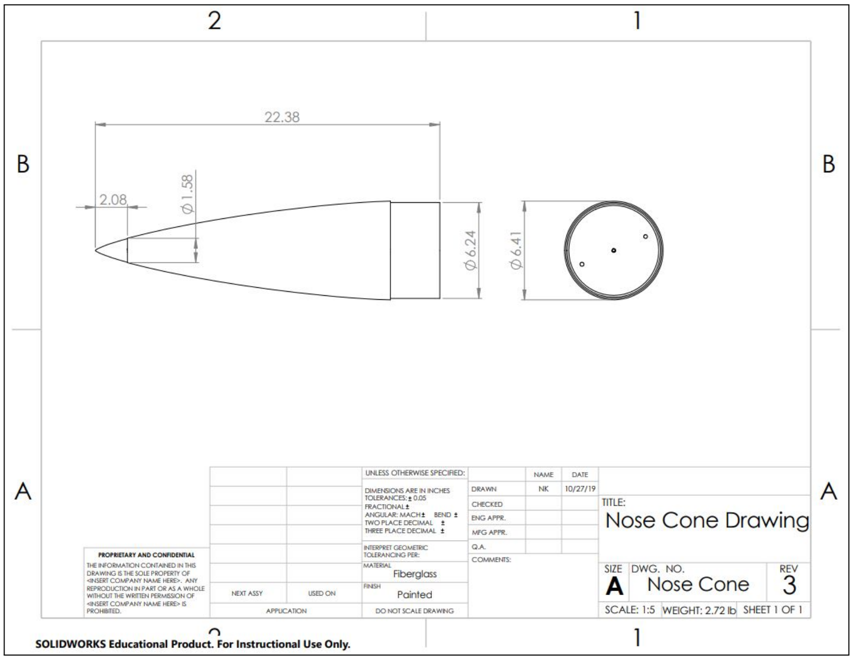

Figure 111: Nose Cone Drawing

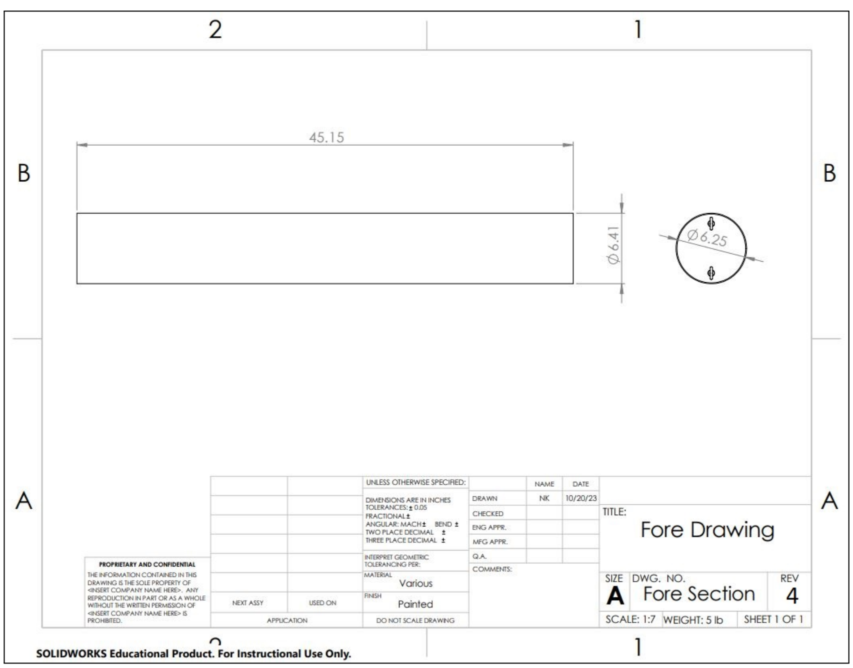

Figure 112: Fore Drawing

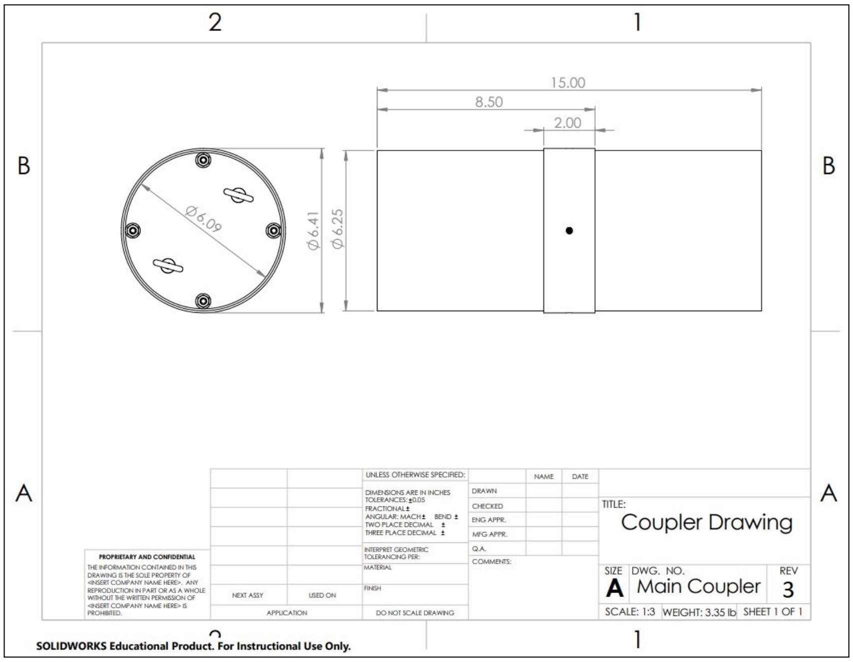

Figure 113: Coupler Drawing

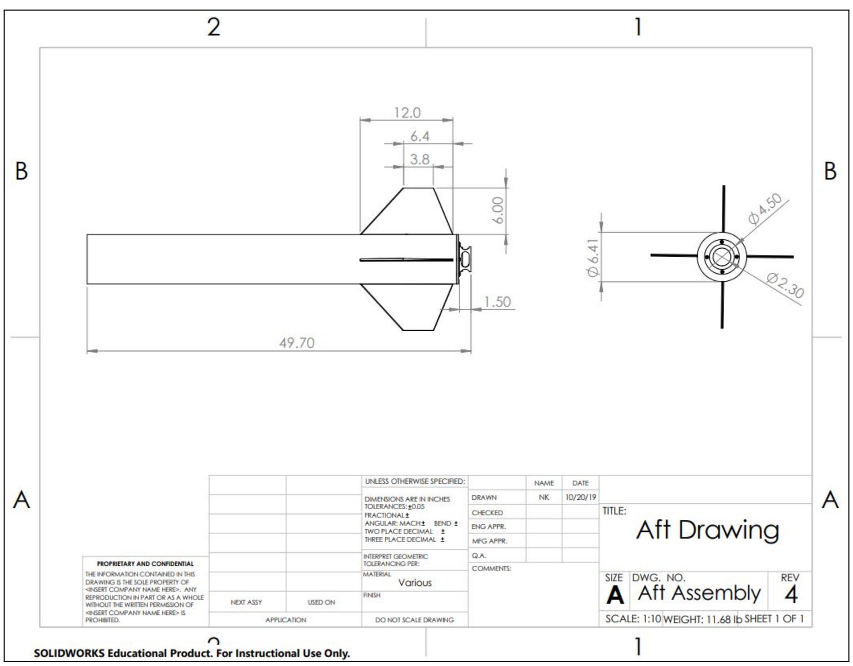

Figure 114: Aft Drawing

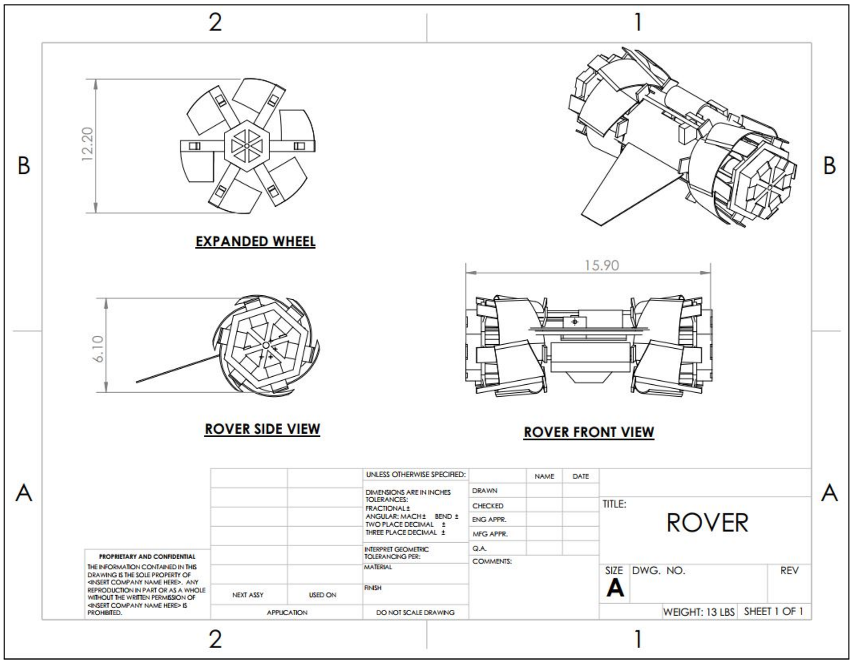

Figure 115: Rover Drawing

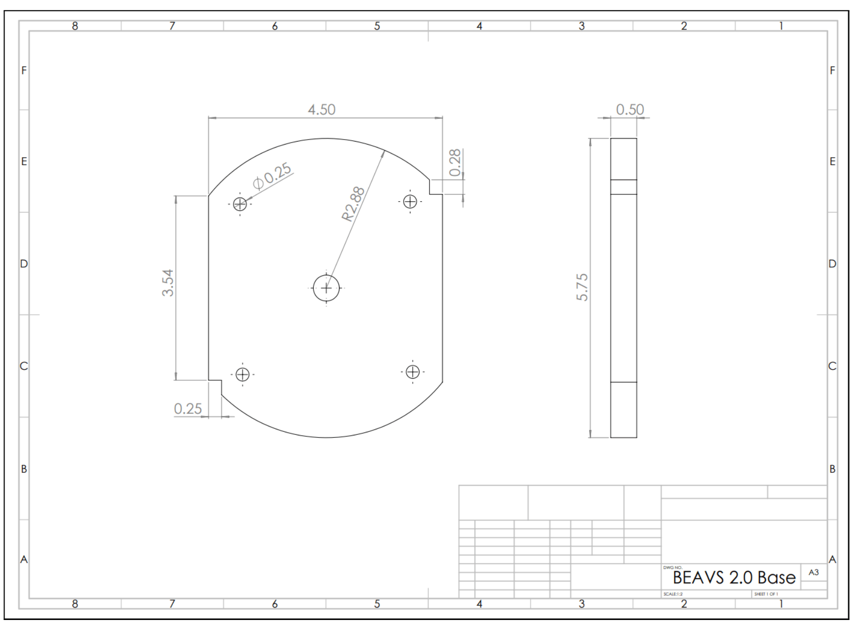

Figure 116: BEAVS 2.0 Base

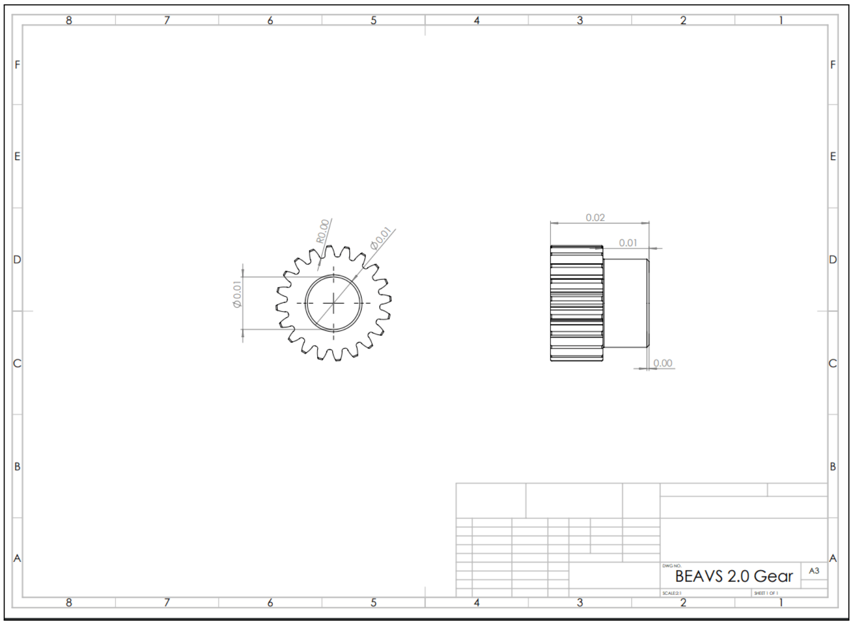

Figure 117: BEAVS 2.0 Gear

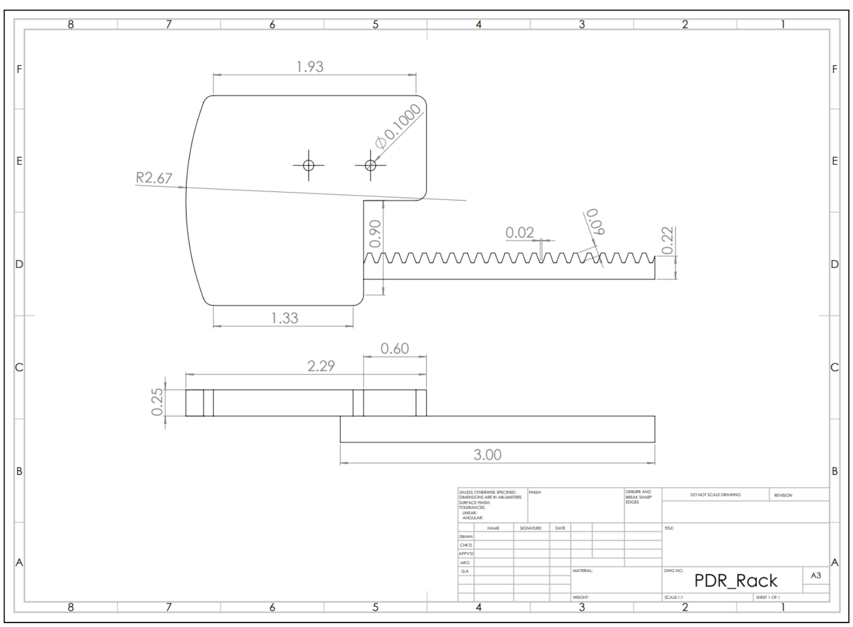

Figure 118: BEAVS 2.0 Rack

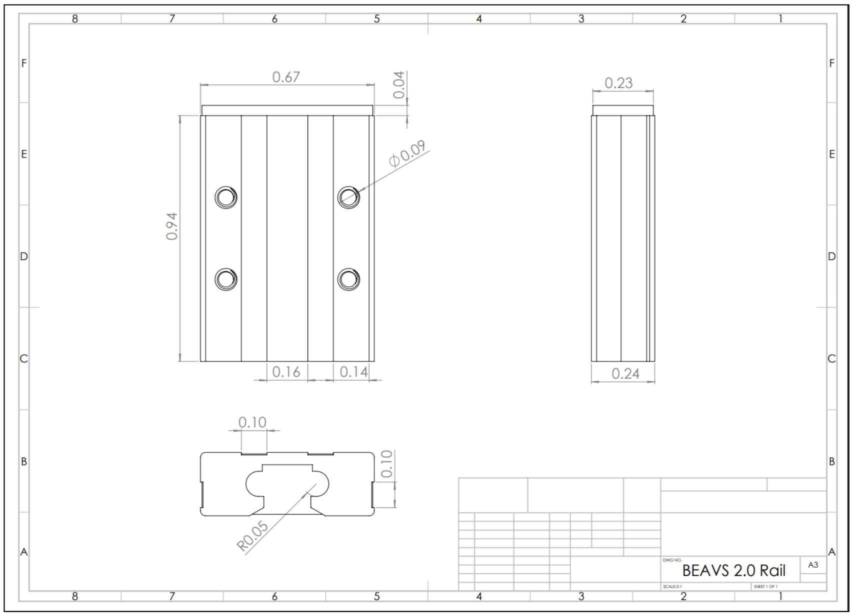

Figure 119: BEAVS 2.0 Rail

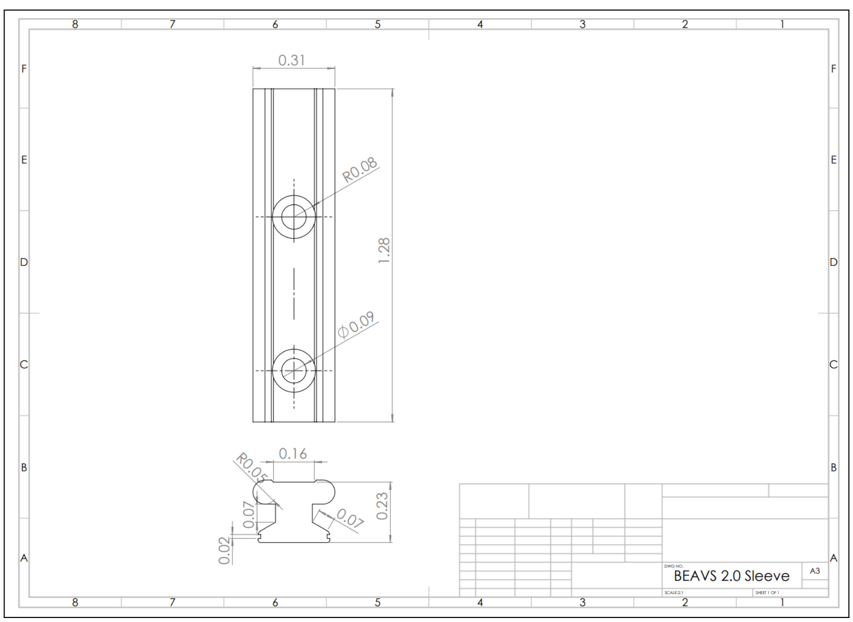

Figure 120: BEAVS 2.0 Sleeve

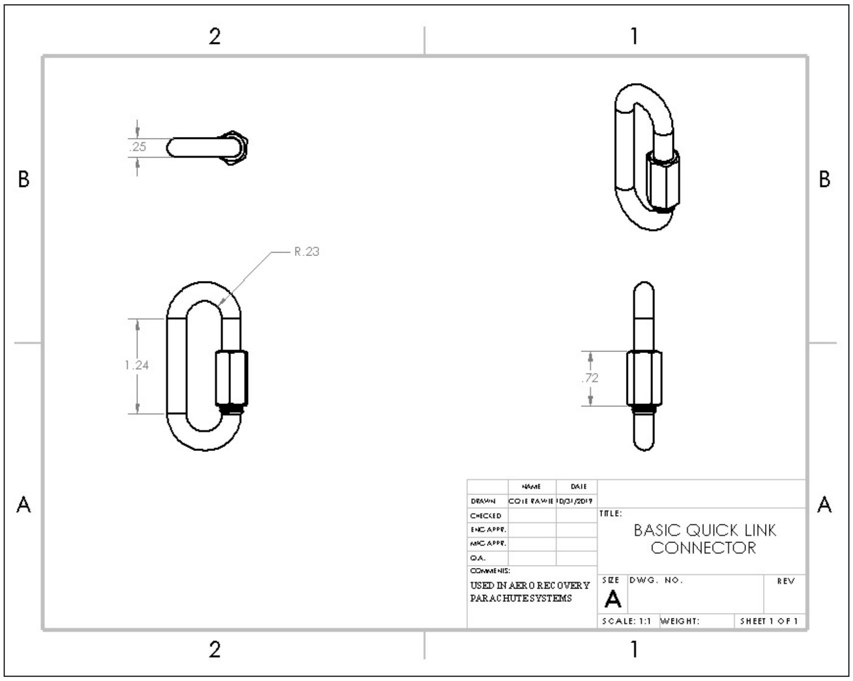

Figure 121: Quick Link

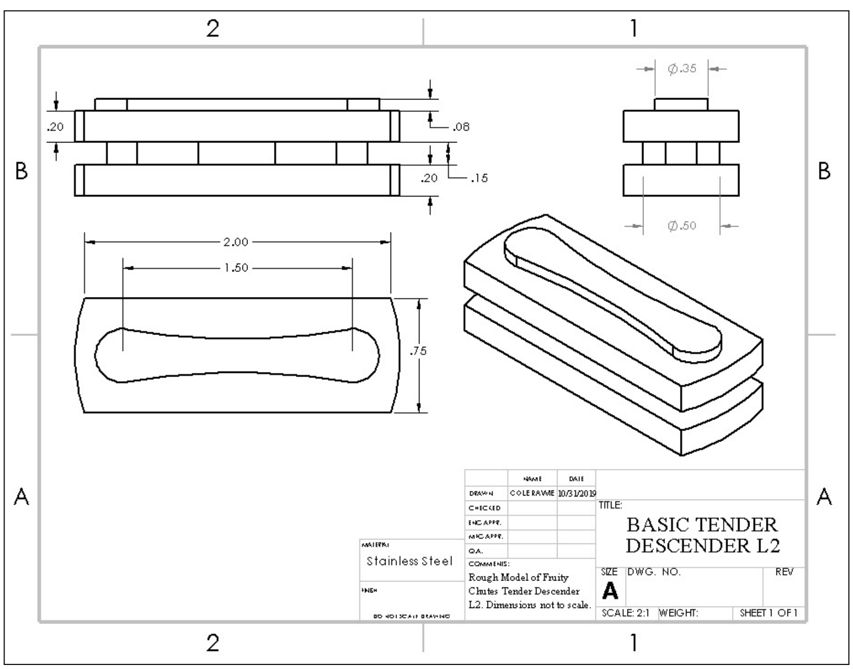

Figure 122: Tender Descender

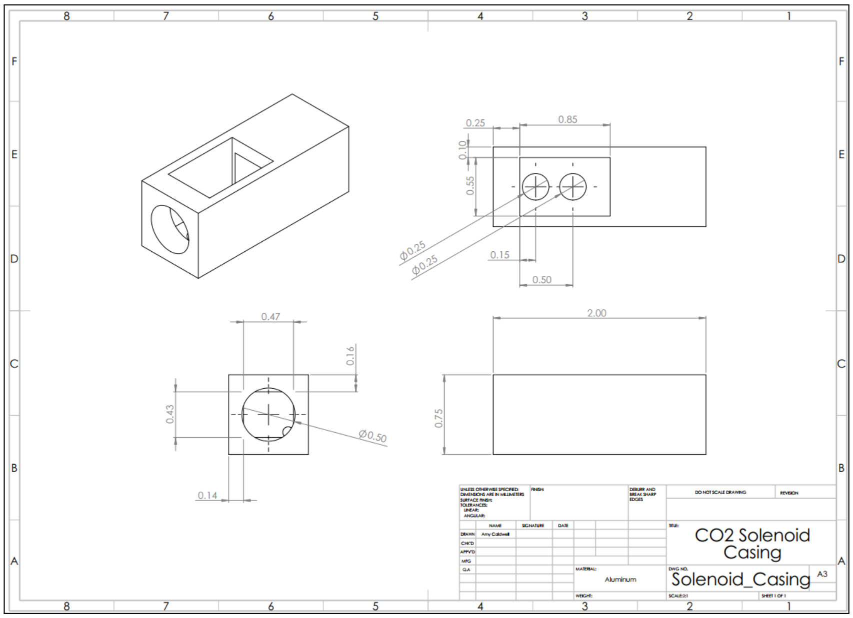

Figure 123: CO2 Solenoid Casing