NASA Student Launch

2017-2018 Critical Design Review (CDR)

January 12th, 2018

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1. General Information

1.1. School Information/Project Title

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

1.1.1. Team Officials

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

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

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

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

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

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

1.1.2. Tripoli Rocketry Association Mentor

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

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

1.2. Launch Vehicle Summary

A summary of key launch vehicle parameters is shown below in Table 1.

Length (in.)	133
Diameter (in.)	6.25
Mass (lbs.)	47.22
Motor Selection	Aerotech L2200-G
Recovery System	Cruciform Drogue and Toroidal Main (4 sections)
Rail Size	1515, 144in.

Table 1: Summary of launch vehicle parameters.

The launch vehicle has been designed to allow adequate room for all payload and recovery hardware. A length of 133 in. and a diameter of 6 in. was found to provide adequate space for all flight controls, payload, and recovery subsystems. To safely launch the vehicle and provide a margin of error for mass assumptions of various components, an AeroTech L2200-G solid ammonium perchlorate motor was chosen. A Variable Drag System will allow the launch vehicle to reach 5,280 feet with accuracy within ±23 feet.

The launch vehicle will be recovered in four independent sections. The vehicle will utilize two cruciform drogue parachutes, and two toroidal main parachutes. At apogee, the launch vehicle will split into two sections, ejecting a drogue from the nose cone, and another drogue from the payload recovery coupler. The two sections will fall under drogue independently until releasing their main parachutes at the designated altitude. The payload section will use an ARRD to release the main bag and use the nose cone parachute to pull the main from the payload recovery bay. The booster main parachute bag will be connected to the bottom of the payload recovery coupler and the booster drogue will pull the booster main parachute from the booster recovery bay. The nose cone will then recover under the original drogue, and the same for the payload recovery coupler.

1.3. Recovery Summary

A separation event at apogee will decouple the vehicle into two independent sections: the payload segment, and the booster segment. Both the nosecone and the coupler will separate from their respective airframe counterparts during main deployment and be recovered under the drogue parachutes as their own independent sections. These two drogues will also act as pilot parachutes for each respective main parachute. These four independent segments are displayed below in Figure 1

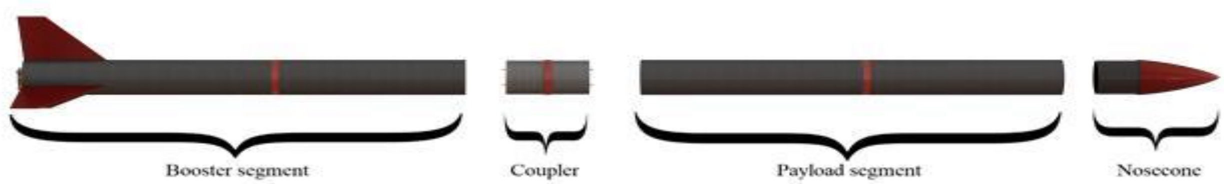

Figure 1: Independent sections of the launch vehicle upon descent

1.4. Payload Summary

This year's payload is designed to accomplish the deployable rover challenge of the NASA Student Launch Competition. The payload consists of an autonomous rover vehicle carrying a foldable solar array and an orientation correction system to ensure upright orientation of the rover prior to deployment. Deployment will be remotely activated via a unique deployment signal sent to the vehicle by a team member after receiving RSO permission to proceed. For the remainder of this document, "the payload" will refer to all subsystems and subassemblies of the payload as a whole while "the rover" will solely refer to the autonomous rover vehicle and onboard systems and assemblies.

2. Changes Since PDR

2.1. Action Items

After PDR, two main items of concern were brought up, regarding drift calculations and total mission elapsed time. These concerns have been addressed. Drift calculations are shown in Section 3.3.11.8. Official predicted mission elapsed time is shown in Table 10.

2.2. Vehicle Design Changes

Change	Justification
Nose cone material changed from carbon fiber to Nylon 12.	The team gained access to a Sinterstation 2500+ machine which allowed us to additively manufacture the nose cone. This manufacturing method is much easier and reduced manufacturing time and allows the team to focus on more demanding projects.
Nose cone overall length reduced from 18 in. to 15 in.	The team determined the 6 in. transition section associated with the nose cone could be reduced to 3 in. This reduces overall mass and surface friction of the launch vehicle.
Booster Recovery Bay shortened from 28 in. to 23 in.	The recovery hardware which will be stowed in this bay can be packed tighter than originally thought. To reduce mass and surface friction, the booster recovery bay was shortened accordingly.
Payload Recovery Bay shortened from 29 in. to 25 in.	The recovery hardware which will be stowed in this bay can be packed tighter than originally thought. To reduce mass and surface friction, the payload recovery bay was shortened accordingly.
Altimeter sled design	The design of the altimeter sled was changed to reduce mass and increase the amount of vertical space available in the coupler.

2.3. Recovery Subsystem Design Changes

Change	Justification
Laser Cut Panels	The team developed a technique to use a Universal Laser Systems laser cutter to precisely and accurately manufacture parachute panels in response to asymmetry of handmade parachutes causing twisting of shroud lines. This method produces greater symmetry and balance.
Nomex Cone	To protect sensitive payload electronics from black powder residue, two alternatives, CO_2 canisters or a protective cover, were tested. The CO_2 system was insufficient to separate the sections. The nomex cover absorbed the residue without impairing the black powder system separation.

2.4. Changes to Payload

Change	Justification
Removed SIS voltage regulator	Using a voltage regulator and powering the SIS camera module with the motor battery has been deemed unnecessary as it can be powered directly from the controller battery.
Removed SAS voltage divider	The voltage divider circuit to indicate the voltage generated by the solar panels has been deemed

	unnecessary as the panels do not provide enough power to over-current the input and can safely be connected in parallel directly to the input.
SAS Locking Base removed	The torsion spring of the spring hinge has been shown to be strong enough to hold the tower assembly of the SAS upright without the need for the lock pins removing the need for the entire locking base.
DTS requirement of 2500 ft. range removed	The 2500 foot range is no longer necessary due to the fact that team members will be allowed to be in close vicinity to the payload before sending the deployment signal.
DTS pull-apart mechanism changed to magnetic connector	It has been experimentally determined that the audio jack resists pulling apart too much for the rover to overcome and as such has been switched to a magnetic connector.
AETB and FESB snap ring retention method has been changed to a secondary inner ring-outer ring pinned method to house bearing components	Weight reduction and improved strength
AFT and FWD end crescent mounting plates material has been changed from Aluminum to d2 tool steel and are now integrated with the bearings primary inner ring	Integrating the crescent mounting plates into the primary inner ring design improved part machinability. Changing the material to a denser one lowered the center of gravity of the ROCS.
Complete RLM redesign	Increase in strength and reliability
RBS change to tread design	A new tread was designed after it was determined the previous design did not provide sufficient traction and clearance.

3. Design and Verification of Launch Vehicle

3.1. Mission Statement

River City Rocketry's mission for the 2017-2018 NASA Student Launch competition is to design, build, and launch a vehicle capable of reaching an apogee altitude of 5,280ft and then deploying a rover with foldable solar panels upon landing. After reaching apogee, the launch vehicle will safely descend under parachute and land without inflicting any damage to itself, the rover, spectators, or the surrounding environment. Additionally, River City Rocketry aims to inspire young minds in our community by exposing them to science, technology, engineering and math.

3.2. Mission Success Criteria

For our mission to be considered a success, the launch vehicle must meet the following criteria:

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

3.3. Launch Vehicle Overview

The launch vehicle has been designed to safely deliver a rover payload to an apogee altitude of 5,280ft. The launch vehicle consists of five sections: the booster, booster recovery bay, payload bay, payload recovery bay, and the nose cone. The launch vehicle will feature a rover payload as well as a Variable Drag System (VDS) that will be discussed in detail in later sections. A rendering of the fully assembled launch vehicle is shown below in Figure 2.

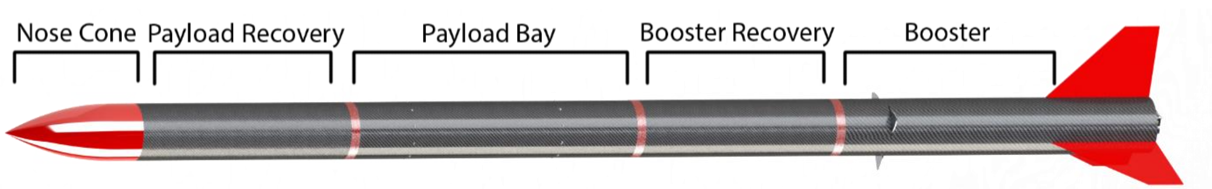

Figure 2: Launch vehicle overview.

3.3.1. Launch Vehicle Dimensions

The launch vehicle will be approximately 6.25 inches in diameter, and 133 inches in length. The dimensions of the vehicle were dictated by the size of the motor selected, the rover payload size, and the size of the recovery equipment needed. The size of the fins and nose cone were determined using software, further discussed in 3.3.4.3 and 3.3.7 respectively. The length of each section of the launch vehicle is shown below in Table 2.

Section	Length (in.)
Booster	37
Booster Recovery Bay	23

Payload Bay	33
Payload Recovery Bay	25
Nose Cone	15
Total Length	133

Table 2: Launch Vehicle Dimensions.

3.3.2. Variable Drag System

To reach the target altitude of 5,280 ± 23 ft., the team has developed and implemented a target apogee, air braking system called the "Variable Drag System" (VDS). The VDS will actively predict and alter its flight path, bringing the vehicle to the desired target altitude. While doing this, it will deliver state data to a ground station through an active telemetry system. The team is presently in the manufacturing and testing stage of the VDS to ensure peak performance.

Figure 3: Variable Drag System.

3.3.2.1. Design Overview

The VDS operates using three aluminum drag blades, actuated by a central spur gear. This spur gear is driven by a single brushless DC electric motor and can be actuated precisely to any position in its range with feedback provided by an encoder. The motor is controlled by a main electronic controller which is responsible for both actuation and the reading of sensor data. It uses this sensor data to determine the state of the vehicle and autonomously control the flight of the vehicle. Throughout the flight, a radio communication system will take this sensory data and transmit it to a ground station operated by the team.

3.3.2.2. Electronic Hardware

The electronic design of the VDS is in the optimization and manufacturing process. The electronics for this system will be split onto two stacked boards; the top board carrying the power controls components, and the bottom containing the data acquisition and motor control functionality. The schematics for these boards are shown in Figure 4 and Figure 5, respectively.

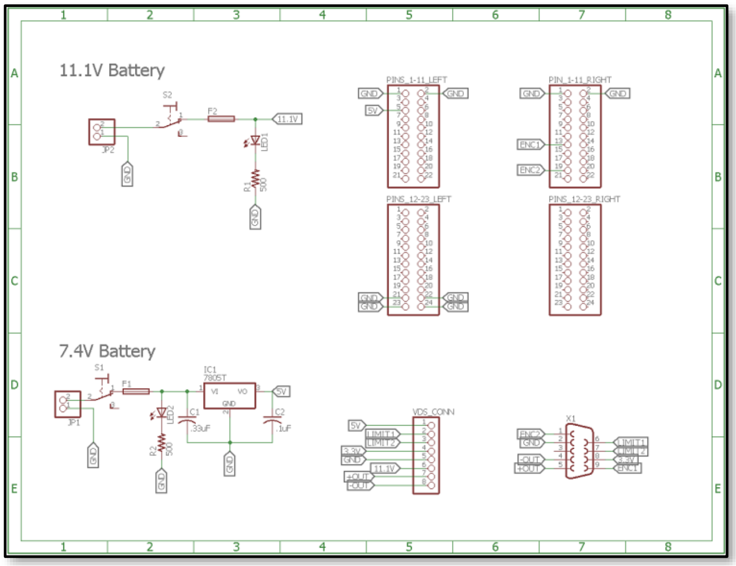

Figure 4: Power control Board Revision C schematic.

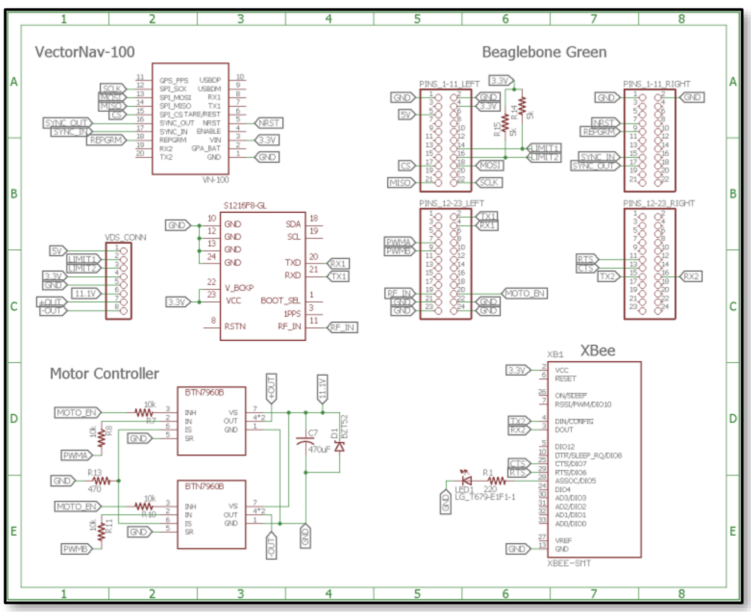

Figure 5: Data Acquisition board Revision C schematic.

3.3.2.2.1. Sensory and actuation Electronics

The VDS electronics configuration was chosen based on system requirements, ease of implementation, compatibility, and performance. The major system control components are listed below. The diagrams for the power control board and the DAQ board are displayed in Figure 6 and Figure 7.

- BeagleBone Green computer
- VN-100 Inertial Measurement Unit
- Xbee SX Pro RM radio
- BTN7960 high current half-bridge motor control Circuit
- S1216F8-GL GPS unit

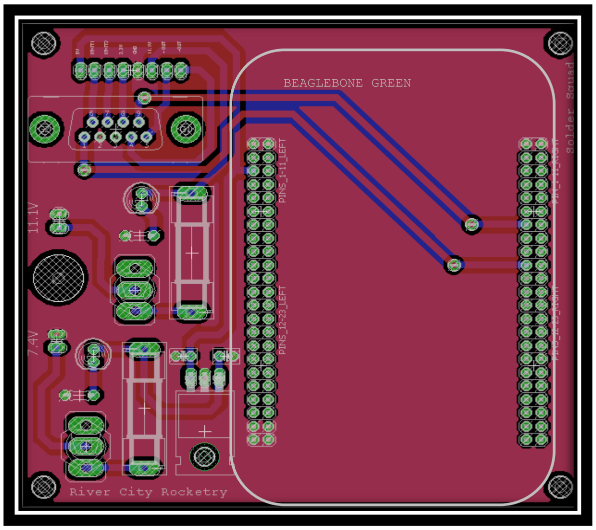

Figure 6: Power control Board Revision C schematic.

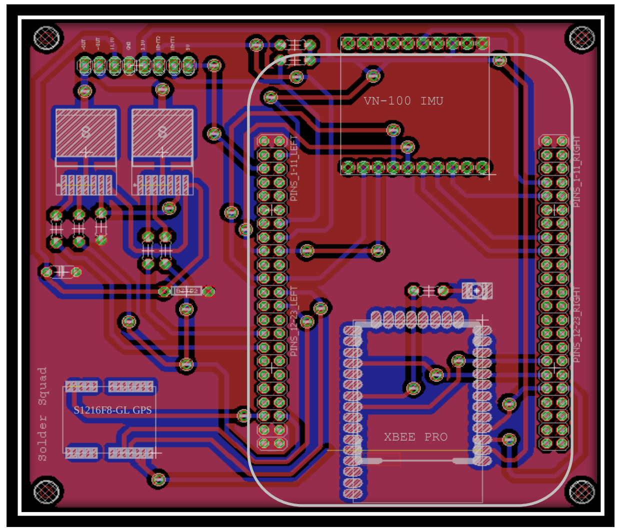

Figure 7: Data Acquisition board Revision C schematic.

The VDS controls PCB design is split into two stacked boards, one for power distribution and one for data acquisition (DAQ). The power distribution board will stack on top of the DAQ board, through which power will route to the bottom board. Output signals will route back through the top board to actuate the system. The two boards create a more modular design overall, this makes the components easier to troubleshoot and replace.

Earlier revisions of the boards contained a schematic centered on the Teensy 3.6 and a separate PCB with the Xbee electronics. Due to a limited amount of coupler space for additional battery weight, and logistical issues with integrating a third board, it was decided that the electronics would be upgraded to a computer that would be able to process the data from all electronics used for the VDS. Using a microcontroller with a greater processing speed and more memory allowed the Xbee pro to be integrated directly into the main DAQ board configuration. This configuration is shown above.

3.3.2.2.2. BeagleBone Green Wireless

Due to growth of responsibilities of the VDS flight computer, and the growing need for multitasking, the team decided to replace the existing Teensy 3.6 development board with a more-versatile, more-powerful Linux computer. The team decided upon the BeagleBone Green (BBG) as the new VDS flight computer. Equipped with floating point acceleration, 512 MB DDR3 RAM, and an ARM Cortex-A8 CPU, the BBG affords much greater processing speed than the Teensy 3.6.

Figure 8: Beaglebone Green Wireless

The BeagleBone contains 92 input and output pins that can provide Pulse Width Modulation (PWM), I2C communication, and UART serial communication. The input and output will be used to control motor actuation and data collection systems, as well as the telemetry system.

Once the PCBs are manufactured, testing will be undergone by the BeagleBone to verify that an external power supply is able to power the circuit. The board contains a 5V regulator, into which a 7.4 V supply will be run and the performance of the BeagleBone in parsing data from the VN-100 will be evaluated.

3.3.2.2.3. Telemetry Electronics

The telemetry system is integrated into the DAQ board layer of the VDS control electronics. This system consists of the Xbee SX, which will communicate via UART to the BeagleBone green, and transmit real time data from the VN-100, the S1216F8-GL GPS unit, and the state of the drag blades to the ground station.

To undergo testing, the Xbee was drafted a test printed circuit board, where the performance of the Xbee unit could be evaluated independently of the other electronics. This schematic is shown in Figure 9 and Figure 10.

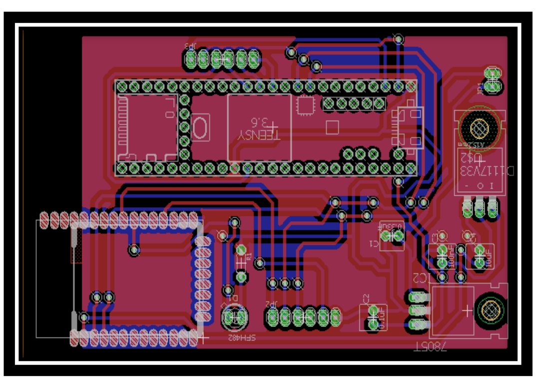

Figure 9: Xbee Pro test PCB.

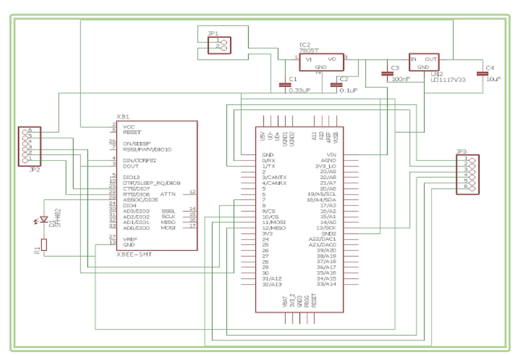

Figure 10: Xbee Pro test PCB schematic.

In testing, two of these boards would be functional. One board acts as a transceiver, and one as a transmitter. Each board runs operates through the teensy 3.6 and is powered by a 7.4 V 1000mAh li-po battery. The system will operate on a frequency between 902MHz and 928MHz, where the least noisy frequency within this range will be determined at the location of the launch, and used for transmission. A feature of the Xbee allows the user to set the power setting to meet the range needs of the application, in order to save on current consumption. The testing undergone by the Xbee is to evaluate whether the urban range of 11 miles, specified by the manufacturer, is realistic to the team's use of the unit with the team's power consumption constraints kept in mind.

The test was set up so that both the transmitting and receiving Xbee's were on identical electronics set ups with a flat patch antenna, the receiver was set at a fixed point where the transmitter was gradually moved further away and it was observed how the signal varied between the two. It was found that the antennae used through the test caused the signal to be unreliable, as the signal was varied at a distance of 0.62 mi (3274 ft.). This issue is thought to be mitigated by re-testing with an antenna more optimal for our application, such as a whip antenna. The power consumption throughout the test showed a loss of .17V over roughly an hour of solely outdoor application at sub-freezing temperatures. This battery draw was through transmission of a minimal message, and is hypothesized to be much more substantial when sending large data packets. A similar test will be completed with higher distance rated antennas, and with larger data packets to further evaluate the ability of the Telemetry system.

3.3.2.2.4. Hardware manufacturing

The hardware design will soon undergo the manufacturing phase. Due to the revisions that were made on the microcontroller circuit, the manufacturing process was delayed, as the PCBs had to be redesigned to fit the new requirements of the hardware.

The manufacturing phase will help revise and finalize the circuit design. The stages below describe the plan of manufacturing:

1. Printed Circuit Board (PCB) prototyping
2. Refining circuit design
3. Finalize and outsource PCB

The team manufactures their circuit boards with Advanced Circuits. The boards received will be made of FR-4 .062" material with a lead-free solder mask finish. Once the boards are received, the prototyping stage will consist of soldering circuit components onto the PCB's and testing system functionality. Prototyping is used to verify system design and fabrication processes. The results from prototyping will lead into the next stage of refining the design. This cycle continues to iterate until the final design is complete. The surface components required to assemble the PCBs are shown in Table 3.

VDS PCB components	
5	Long pin female header pins
2	500 Ohm resistor
2	Amber LED
2	Power Switch
2	Fuse holders
2	10 A Fuse

1	.1 uF capacitor
1	.33 uF capacitor
2	LM7805 5V regulator
4	Spring pins 8 ct.
2	BTN7960B motor control circuits
3	10k ohm resistor
2	470 ohm resistor
1	470 uF capacitor
1	Zener diode
3	5k resistor
2	220 ohm resistor
1	BeagleBone green
1	GPS 21216f8
1	VN-100 IMU
1	Xbee Pro SX

Table 3: VDS PCB component assembly list

3.3.2.3. Software

3.3.2.3.1. VDS Software

The VDS flight computer will now undertake drag-induced actuation, active flight telemetry, and on-board data acquisition. Due to the segregated nature of these responsibilities, the software will leverage Linux's ability to multitask responsibilities via separate execution contexts. The VDS drag-induced controls, for instance, will be managed from a dedicated context to prevent *locking* by other (potentially time-intensive) tasks.

While in flight, and after the termination of the thrust phase, the VDS flight computer will continuously predict the vehicle apogee based on vehicle attitude and movement through space and apply drag at all times the prediction is greater than the target apogee.

The VDS software will execute from a Linux operating system specialized for embedded use. The software, written in C++, will be written to maintain a high level of inversion of control, allowing only explicit dependencies, to ensure high testing/verification efficacy. It will implement a _mediator_, _command_ software design pattern, defining interoperability through explicit _request_ data structures, to achieve loose coupling between components to further aid in software verification/testing. The C++ code will be verified by a high level of unit test coverage across the system (using Google's _Google Test_ C++ testing framework), aggregate component software testing, and software-in-the-loop simulation testing.

3.3.2.4. Telemetry Software

The VDS software will utilize an EM (electro-magnetic) wave transceiver to transmit VDS sensor data and other auxiliary metric across the VDS (battery voltage, operating system data, etc.) to a receiver on the ground. This requires the implementation of data serialization and the construction of messages which adhere to the manufacturer-specified messaging protocol. The current design implements serialization to human-readable, delimited text. The Xbee is currently being tested via the XCTU GUI software created specifically for the Xbee, however the team intends to utilize a custom designed. In the case that the speed of the data transmission must increase, the team will implement serialization to Google Protocol Buffers binary format to decrease overall message payload size. The data sent over the transceivers will be redundantly stored onboard.

3.3.2.5. Mechanical Hardware

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

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

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

To ensure that the design will be robust enough to withstand the maximum in flight forces with a minimum acceptable factor of safety of 2.0, the mechanical components of the VDS V3 were analyzed using ANSYS Workbench 17.2. Due to uncertainties in the drag force calculation and possible changes in the maximum velocity of the launch vehicle, each drag blade of the VDS was tested to withstand a full drag force of approximately 20 lbs. multiplied by a factor of two. A minimum factor of safety of 6.51 was determined for each drag blade. The results of the analysis conducted on the drag blades are shown below in Figure 11.

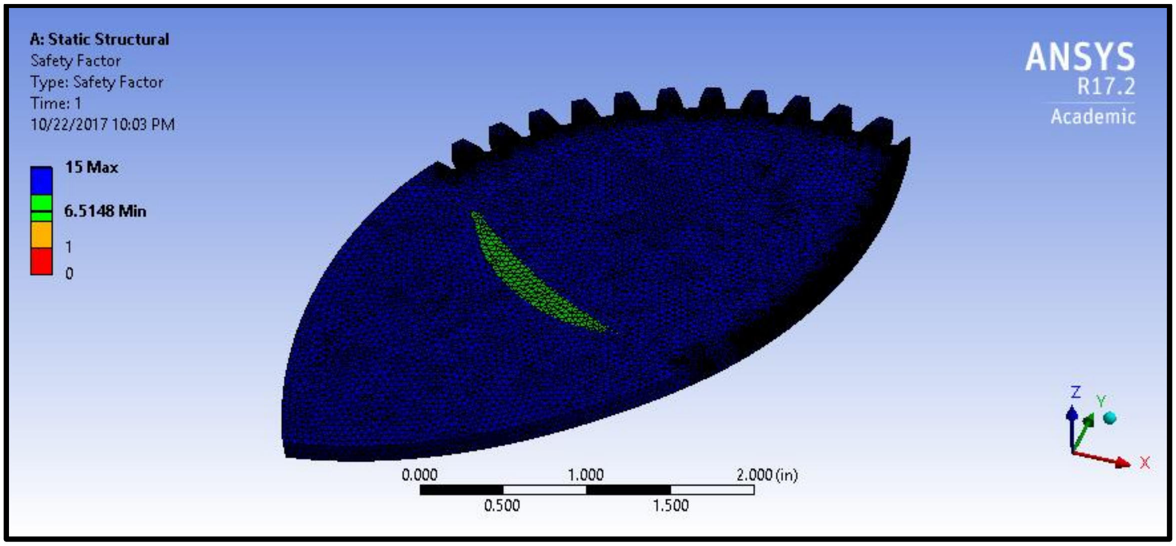

Figure 11: Drag blade factor of safety plot under maximum braking force.

The gear teeth of both the drag blades and the central spur gear were tested to ensure that they would perform safely under loading equivalent to the maximum stall torque of the NeveRest 40 DC motor applied to a single drag blade. The results of the study showed a minimum factor of safety of 1.68 for the gear design on the drag blade and 1.73 for the gear design on the central spur gear. The results of the Finite Element Analysis are shown below in

Figure 12 and Figure 13.

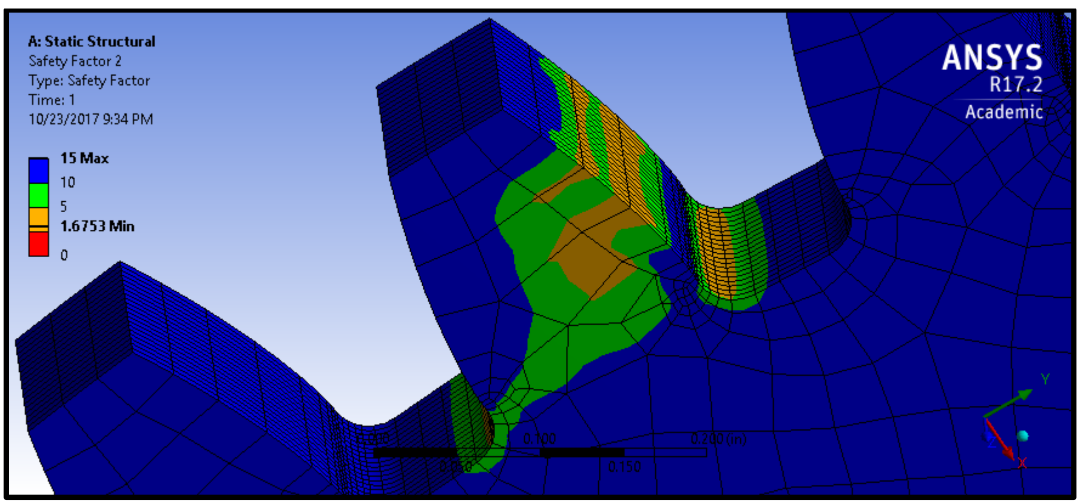

Figure 12: Drag blade gear tooth factor of safety plot.

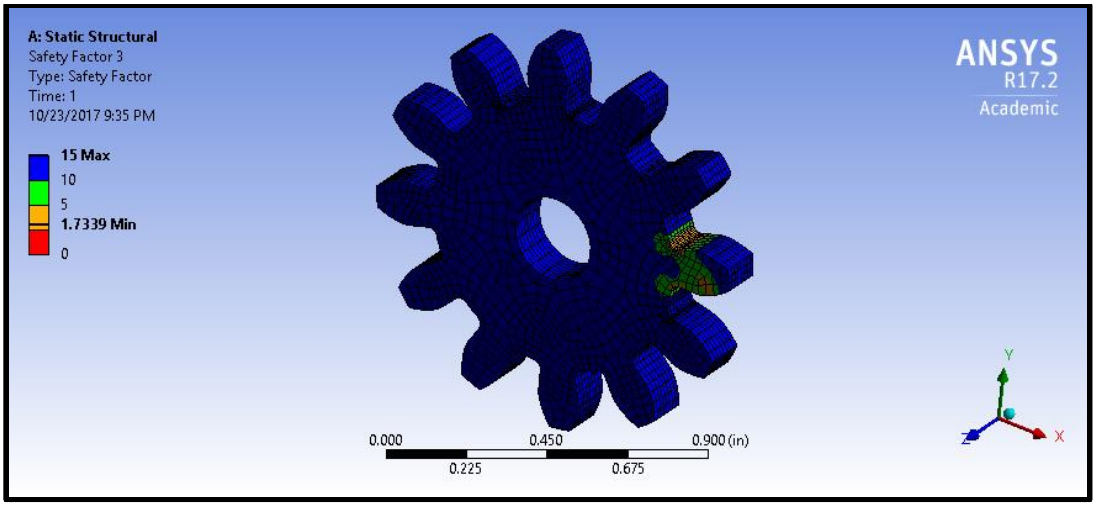

Figure 13: Central spur gear tooth factor of safety plot.

3.3.2.6. Blade Actuation

The VDS will radially actuate three drag-inducing blades from the launch vehicle's airframe using a single central spur gear press fit to the shaft of a NeveRest DC 40 Motor. Each drag blade has a set of radial gear teeth designed to mesh with the central spur gear. The blades will rotate about a single 0.125 in. Dowel pin projecting approximately half of the surface area of each blade perpendicularly into the oncoming airflow. A rendering of the VDS gear mesh configuration is shown below in Figure 14.

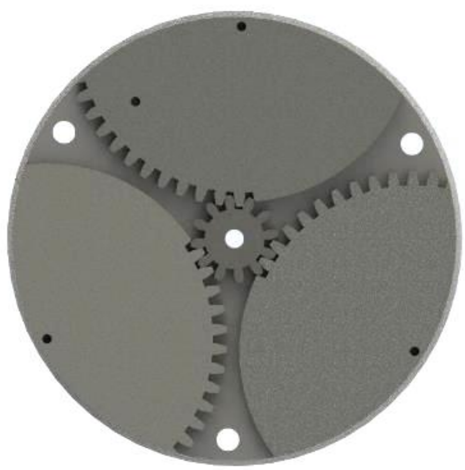

Figure 14: Rendering of VDS gear mesh configuration.

The final actuation design is optimized with respect to mass and the total projected area of the blades, and allows for continuous and simultaneous control of all drag blades using a single motor. A rendering of the VDS before and after actuation is shown in Figure 15.

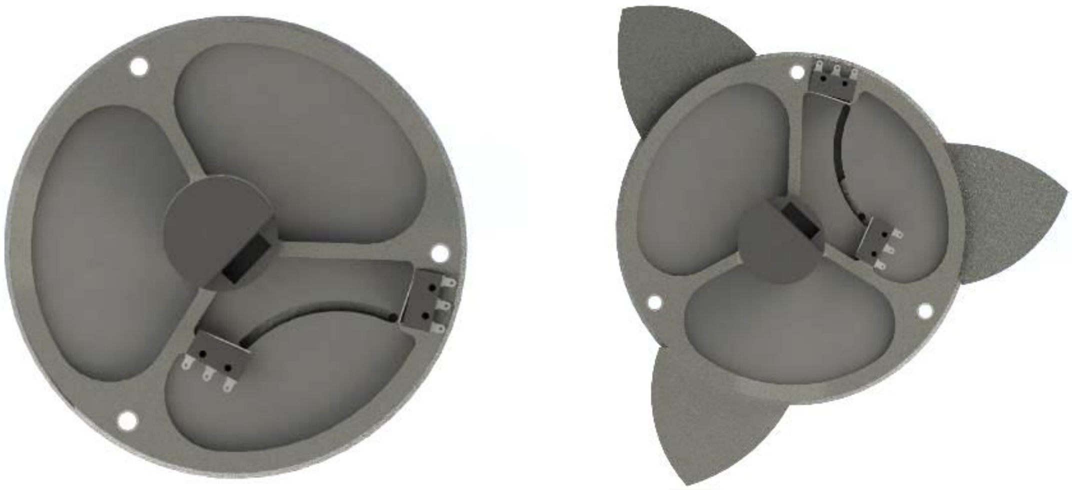

Figure 15: VDS configuation before and after drag blade actuation.

3.3.2.7. Integration

The harness that will house electronics within the VDS coupler throughout the flight will be custom designed 3D printed sleds. After considering the geometry of the coupler and the need to access the space below the electronics, the team opted to go with rectangular PCBs rather than the originally proposed circular PCBs. The square design will allow electrical wires to be routed beside the sled and into the electronics.

Additionally, a large consideration that was to be mitigated from issues stemming from the VDS v2 is the conservation of battery power while the electronics are assembled in the vehicle. The VDS coupler is in the booster section of the rocket, and therefore has to be the first assembled; the concern was that the batteries would be exhausted before launch due to the continuous current consumption. To prevent this, the vehicle will contain an external port into the electronics coupler. This will allow easy access to power the boards through the BeagleBone until the rest of the rocket has been assembled, at which point the electronics would run on battery power from then on. Several types of connectors are under consideration, with the specifications for the connector being number of data pins, rigidity, and smooth assembly into the vehicle without altering its drag force. A rendering of the avionics sled with the printed circuit boards is shown below in REF _Ref503124665 \h **Error! Reference source not found.**.

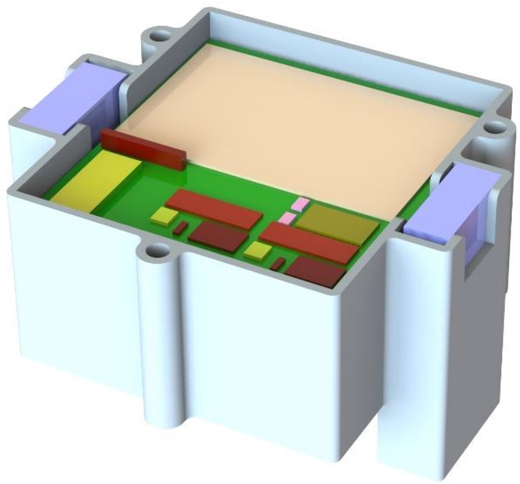

Figure 16: VDS avionics sled.

3.3.2.8. Success Criteria

The success criteria of the hardware design is to provide operation of the VDS throughout an extended amount of time. The success plan is projected with three key milestones. The milestones are listed below:

1. Design of the PCB layout.
2. Finalization of the PCB manufacturing process.
3. Operation of the VDS under fully mounted and integrated PCBs.

The PCB layout design drives how the hardware system progresses. The layout is currently being reviewed and finalized with PCB prototyping. The manufacturing process gives the design a standardized way to construct multiple iterations of the design. This is important for establishing a system of revising the design or producing new designs in the future. Operating the VDS under a completely fabricated PCB system is the end goal of the hardware design. The VDS will be actuated under extended operation to ensure a robust hardware design and to establish an operational safety margin in terms of battery life, current/voltage ratings, and harness stability.

3.3.3. Vehicle Structural Components

3.3.3.1. Airframe

The launch vehicle will consist of four airframe tubes ranging in length from 23 inches to 37 inches. Each airframe tube will be constructed of quasi-isotropic carbon fiber fabric saturated with Aeropoxy laminating epoxy. The carbon fiber fabric consists of many individual filaments braided at alternating angles of 0˚, and +/- 60˚. Carbon fiber was chosen as the airframe material for its high tensile and compressive strength, resistance to harsh environmental factors, strong adhesive properties, low mass, and low cost to the team. 100% of the carbon fiber used for the airframe of the launch vehicle has been donated to the team from a sponsor, thus making it the most cost-effective material considered.

Aeropoxy PR2032 and PH3660 was chosen as the epoxy resin for the launch vehicle as it is the most practical option for the team. Aeropoxy epoxy has many advantages compared to other epoxies used in the construction of composite structures. Aeropoxy features a one hour working time with a medium viscosity, making it practical to remove excess epoxy via a heat shrink tape method. Another advantage to using Aeropoxy epoxy is that unlike other epoxies used in composite manufacturing, Aeropoxy can cure at room temperature. This is important for the team as we currently do not have access to the tooling or an oven that would be needed for epoxies used in traditional industrial applications. Material properties for Aeropoxy PR2032/PH3660 epoxy are shown below in Table 4, and were supplied by the manufacturer.

Property	Value
Cured Hardness	88 Shore D
Viscosity at 77˚F	800-875 cps
Density (in.^{3}/lb.)	0.0420
Tensile Strength (ksi)	45.35
Modulus of Elasticity (ksi)	2,800
Elongation at Break	1.91%
Flexural Modulus (ksi)	2,770

Table 4: Material propertires for Aeropoxy PR2032/PH3660.

The airframe tubes will be manufactured using a commonly used composite fabrication method known as “layup”. The carbon fiber fabric will be saturated with Aeropoxy PR2032 resin and PH3660 hardener before being wrapped onto an aluminum mandrel of the desired diameter. Upon completion of the layup process, the airframe tubes will be wrapped with heat shrink tape. The heat shrink tape will be heated with the use of a heat gun, and will shrink by up to 4%. This shrinkage will compress the carbon fiber fabric and squeeze excess epoxy from the fabric, reducing the overall mass of the finished airframe tube. This method of finishing allows for a consistently smooth outer surface finish and faster overall manufacturing times compared to a traditional vacuum bagging method.

3.3.3.2. Couplers

The launch vehicle will utilize 4 carbon fiber couplers, each 12 inches in length. Carbon fiber was chosen for the coupler material as it has a high strength to weight ratio, and thin walls resulting in more internal volume for electronics to be stored. With the team’s current manufacturing capabilities, we cannot manufacture a tube with a smooth and consistent outer diameter and thus the couplers have been ordered from a supplier. The couplers will have an outer diameter of 6 in. and an inner diameter of 5.85 in.

3.3.3.2.1. Shear Pins

During flight, a pressure differential can form within an airframe section due to the outside air pressure decreasing as altitude increases. This pressure differential can cause premature separation if improper shear pins are used. Analysis has been conducted to verify that three #4-40 Nylon 6-6 screws will prevent premature separation due to a pressure differential. The force exerted onto the shear pins by the air pressure differential was calculated using,

$$F_{SP} = (P_{MSL} - P_{apogee})(\pi r^2) \qquad \textbf{(1)}$$

where P_{MSL} is the pressure at sea level, equal to 14.7 psi, P_{apogee} is the pressure at apogee, calculated to be 11.9 psi, and r is the internal radius of the launch vehicle. Using these values, the force exerted on the bulkplate due to the pressure differential was calculated to be approximately 80 lbs. Using the average shear strength of Nylon 6-6 to be 10ksi, the pitch diameter of a #4-40 screw to be 0.09576 in., and the pitch area to be 0.00720 in.^{2}, the shear strength of one #4-40 Nylon 6-6 screw was found to be 72 lbs. Multiplying this shear strength by three results in a combined shear strength of 216 lbs., which is adequate to prevent premature separation due to a pressure differential occurring during flight.

All non-shearing connection points of airframe will utilize 3 #6-32 stainless steel button head cap screws to connect the airframe to the coupler. The screws will be threaded into a nut epoxied to the interior of the coupler. This method allows for a secure connection between the airframe and coupler and is shown below in Figure 17, where the coupler is highlighted in yellow, and the airframe in red.

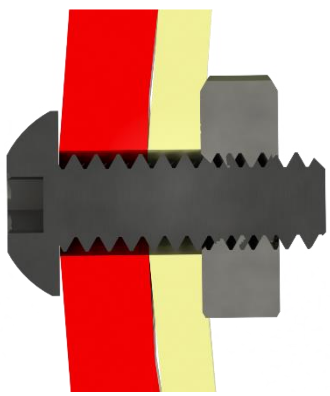

Figure 17: Non-shear pin rendering.

To alleviate pressure on the screws during burn phase, all couplers will have a 1 in. carbon fiber witness ring epoxied to the coupler at the midpoint. The witness rings will rest on the airframe below and take much of the load during burn phase. Witness rings will also be equipped with witness triangles to allow for quick and easy alignment of screw holes during assembly.

3.3.3.3. Bulkplates

To optimize the material selection and material thickness of the bulkplates that will serve as the connection points for recovery hardware, finite element analysis (FEA) was conducted on each load-bearing bulkplate design. The FEA was conducted using the estimated opening forces of the payload main parachute and booster main parachute as the applied loads. The payload bay and booster main parachute opening forces are expected to be approximately 323 lbs.-f and 412 lbs.-f respectively. The load was applied to the region of the bulkplate where the U-bolt washers are designed to rest, and the assembly was fixed where the threaded rod washers that connect the bulkplate to the coupler are designed to rest. The results of the FEA showed a minimum factor of safety of 4.8 and are shown below in **Error! Reference source not found.** and Figure 19.

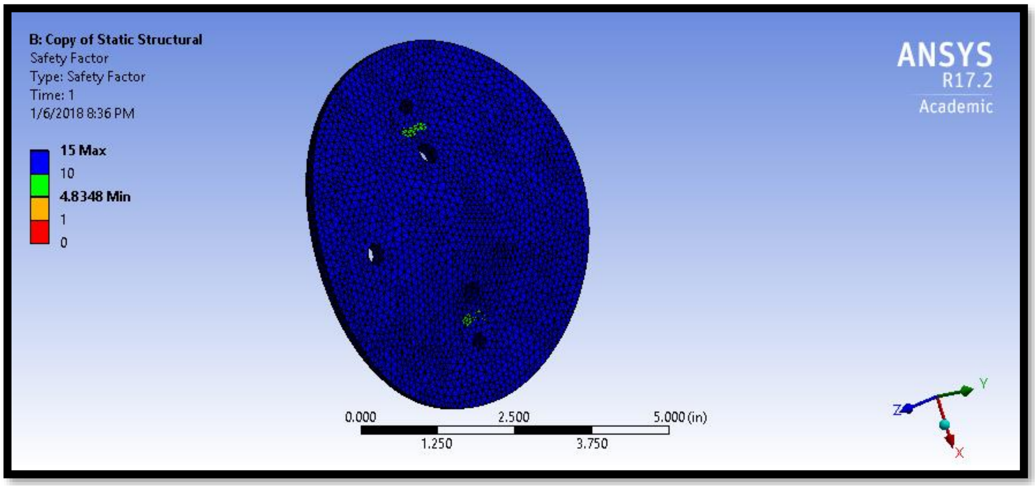

Figure 18: Payload bay bulkplate assembly factor of safety plot.

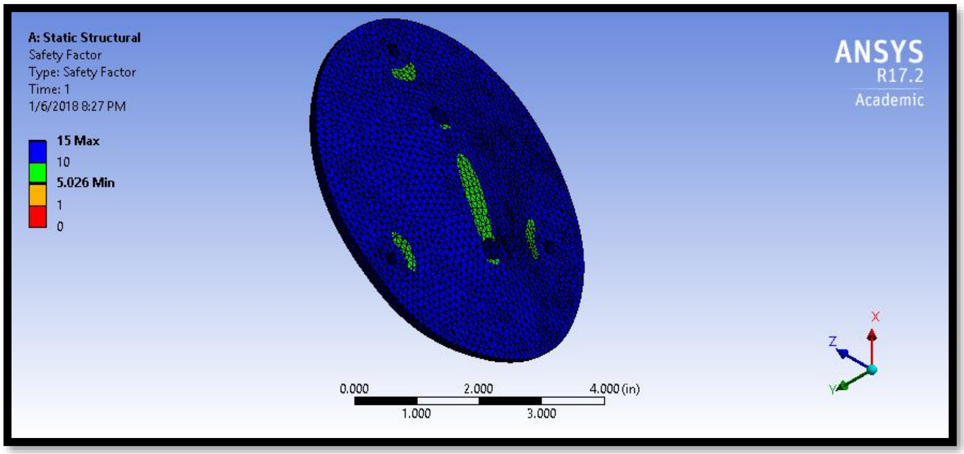

Figure 19: VDS bay bulkplate assembly factor of safety plot.

3.3.4. Booster

The booster section of the launch vehicle is responsible for securing the motor and fins. It consists of three centering rings, a removeable fin system, three fins, motor mount tube, and a motor retainer. The airframe of the booster section will be 36 inches in length to accommodate the motor hardware and VDS. A rendering of the fully assembled booster section is shown below in Figure 20.

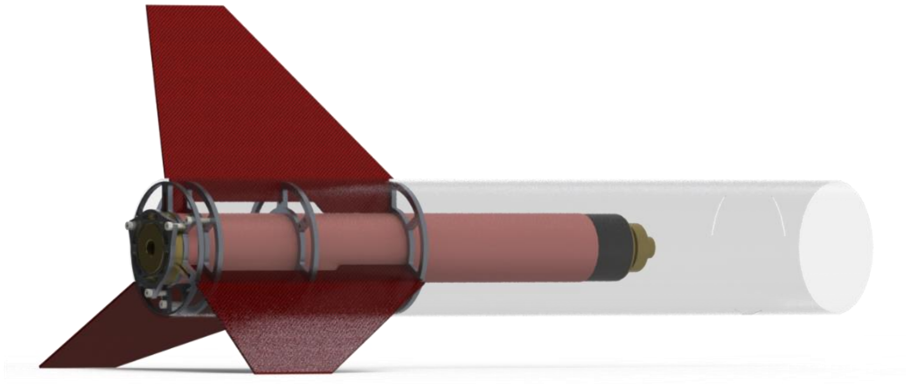

Figure 20: Fully assembled booster section with transparent airframe.

3.3.4.1. Centering Rings

The launch vehicle will utilize three, 0.25 in. thick, 6061-T6 grade aluminum centering rings to transfer the thrust from the motor to the rest of the launch vehicle. The centering rings will be cut from a sheet of aluminum using a Maxiem 450 water jet. Dimensional drawings of the fore and mid centering rings are shown below in Figure 21 and Figure 22 respectively.

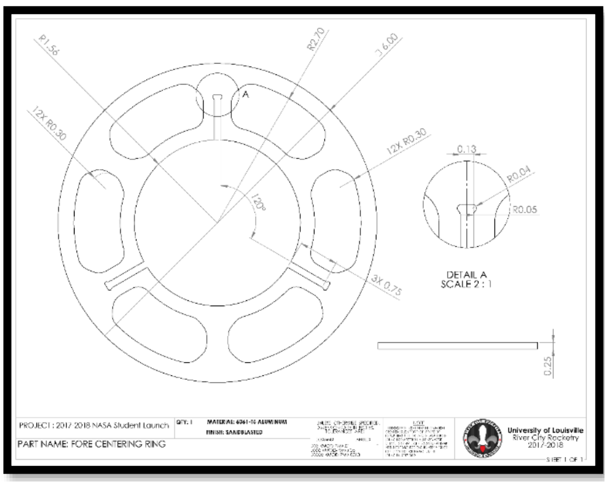

Figure 21: Dimensional drawing of the fore centering ring.

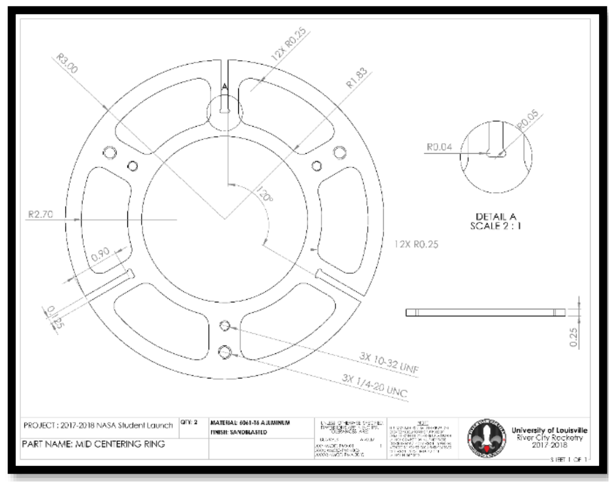

Figure 22: Dimensional drawing of the mid centering ring.

To ensure that the centering rings will withstand the forces exerted on them during motor burn, FEA was run on the fore and mid centering rings. As an additional safety measure, the centering rings were designed so that even if one fails, the remaining two centering rings could support the max load produced by the motor with a minimum factor of safety of 2.0. The centering rings were set to be fixed along their outer rim replicating how they will be once epoxied into the booster airframe. A force of 350 lbs. was applied to the inner rim of the centering rings, where the ring will be epoxied to the motor mount tube. This load is one half the expected maximum thrust we will see produced by the motor, further discussed in 3.3.11.2.2. The results of the FEA simulations showed that the minimum factor of safety of each centering ring design is greater than 2.0, as shown below in **Error! Reference source not found.** and Figure 24.

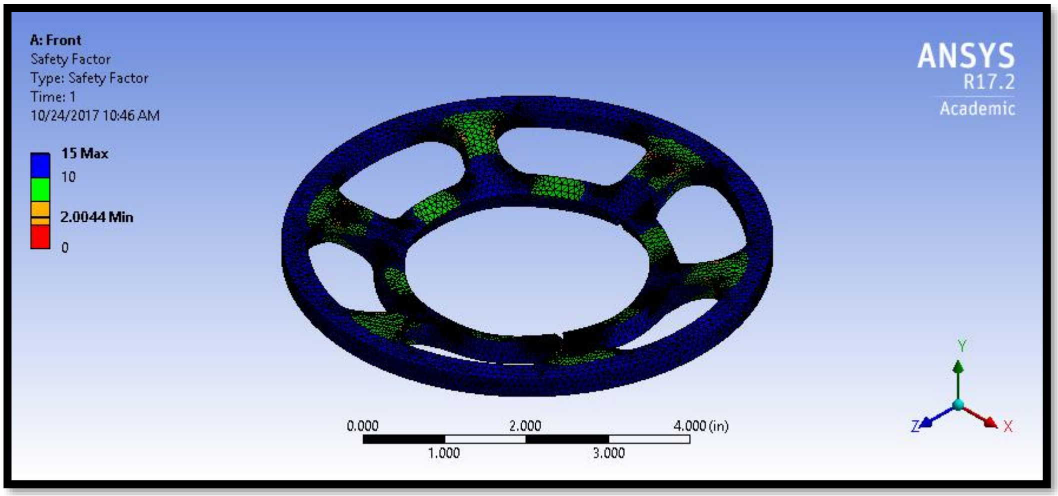

Figure 23: Fore centering ring FEA results.

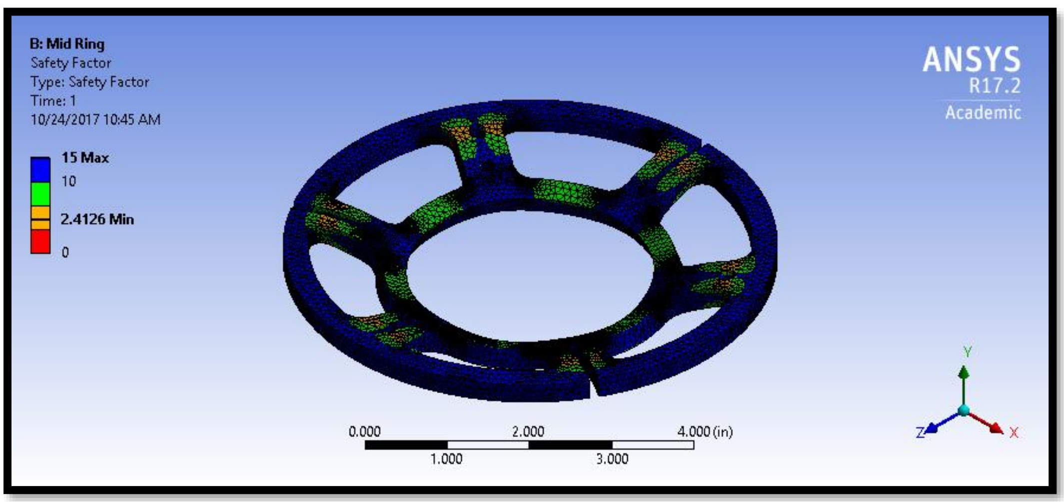

Figure 24: Mid centering ring FEA results.

3.3.4.2. Removable Fin System

The launch vehicle will utilize a custom designed Removable Fin System (RFS) that allows for easy installation and removal of the launch vehicle's fins. The RFS consists of one fore centering ring, two mid centering rings, and a fin retainer. Each centering ring will be epoxied to the motor mount tube, and booster airframe with Glenmarc G5000 high strength epoxy. The material properties of Glenmarc G5000 epoxy are shown below in Table 5.

Property	Value
Tensile Strength	7,600 Psi
Compression Strength	14,800 Psi
Elongation at Break	6.30%
Shore "D" Hardness	85

Table 5: Glenmarc G5000 epoxy material properties.

The RFS will be assembled in multiple steps. Firstly, the centering rings will be epoxied onto the motor mount. Once the centering rings are mounted to the motor mount tube, they will then be epoxied into the booster airframe. To insert the fins into the RFS, first the fore fin tab is inserted into the fore centering ring fin slot. The rest of the fin will fall into the two remaining centering ring fin slots, then the fin retainer is placed over the aft fin tab. The fin retainer is then secured to the aft centering ring with three #10-24 stainless steel shoulder bolts. Upon installation of the motor casing, the motor retainer is then secured to the fin retainer using three #10-32 stainless steel shoulder bolts. A schematic of the RFS assembly process is shown below in Figure 25.

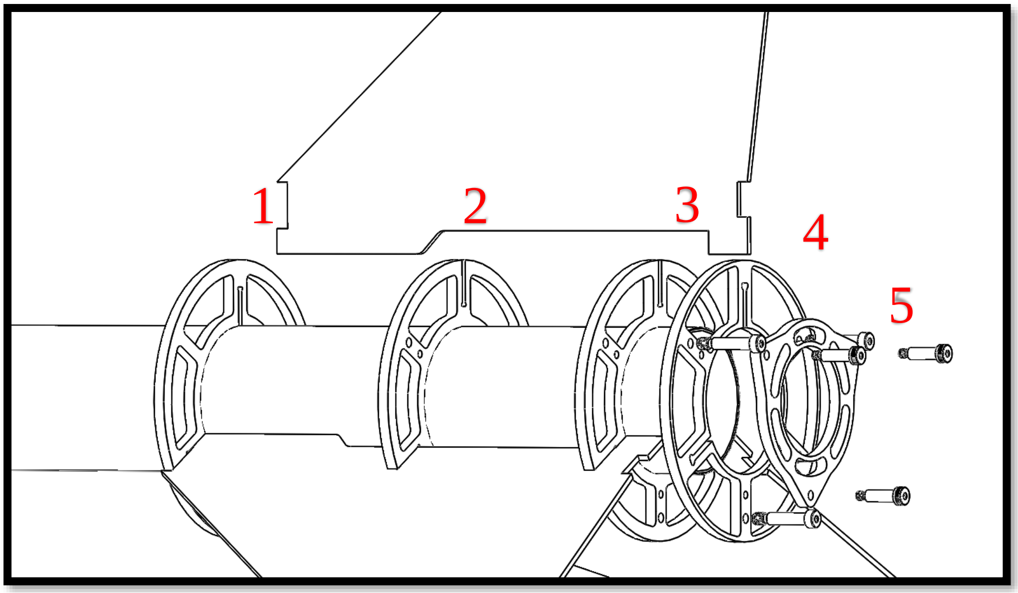

Figure 25: Schematic of the RFS assembly process.

A fin slot jig has been designed to allow for precise alignment of the fin slots. The booster section of the launch vehicle will be inserted into the jig and baffles will be used to mark where the fin slots need to be cut. The jig will be cut using a CNC laser cutter allowing for precise dimensioning. The jig will be manufactured in January and a rendering is shown below in Figure 26.

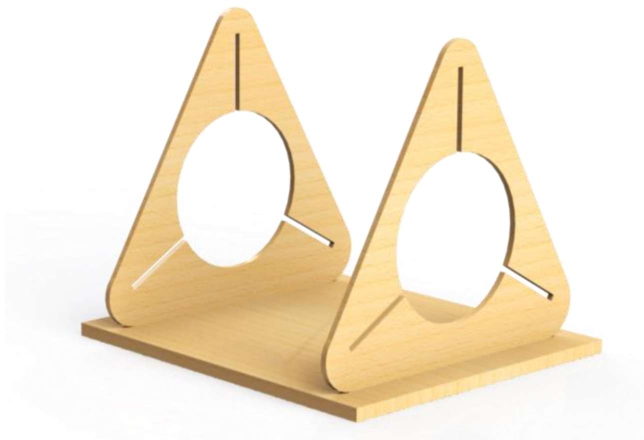

Figure 26: Fin alignment jig rendering.

The location of the fin slots will be marked on the booster airframe using a permanent marker and then cut using a milling machine and 0.125 in. diameter end mill. This method will allow for the fin slots to be cut precisely straight and with the correct width needed for the fins.

3.3.4.3. Fin Design

To reduce the overall coefficient of drag and mass of the launch vehicle, as well as compensate for the VDS, the launch vehicle will utilize 3 carbon fiber cropped delta fins. The fins will be cut from carbon fiber sheet manufactured by the team. The carbon fiber sheet was produced from a polyacrylonitrile-based resin impregnated carbon fiber unidirectional fabric. The fabric is used in the construction of the Boeing 777 tail and Boeing 787 fuselage and wings. The fabric was cut into 16 plies', of which four were +/- 0˚ to the horizontal, four at +/- 90˚ to the horizontal, and eight at +/- 45˚ to the horizontal. This layup pattern, known as a quasi-isotropic layup, was chosen as it gives the material the highest overall strength in all directions.

The fins were designed to ensure that the launch vehicle will fly in a safe and stable manner. The exterior shape of the fins was designed in OpenRocket and was dictated by stability margin simulation results. The fins were designed to meet the team derived stability margin requirement of a stability greater than 2.2 at rail exit. The mounting surfaces of the fins were designed in SolidWorks and will interface with the RFS. A dimensional drawing of the fin design is shown below in Figure 27.

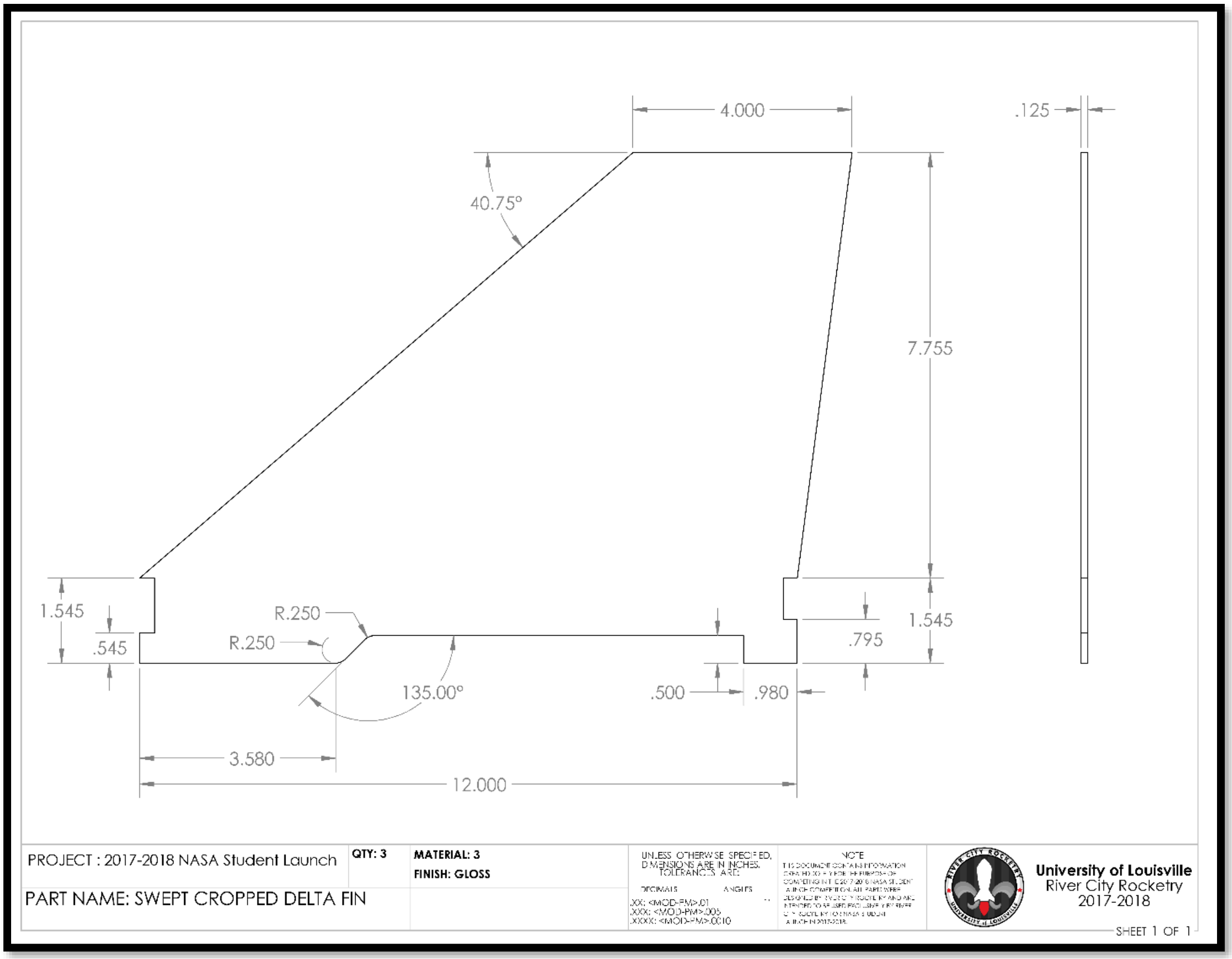

Figure 27: Dimensional drawing of the launch vehicle's fin design.

3.3.4.4. Motor Mounting

The launch vehicle's motor will be mounted in a 3 in. inner diameter carbon fiber tube. The motor mount tube will be epoxied to each of the three centering rings, which will then be epoxied to the airframe of the booster. The purpose of the motor mount tube is to allow for easy installation of the motor, and ensure that the thrust produced is centered axially with the launch vehicle.

3.3.4.5. Motor Retention

A custom motor retainer has been designed for use on the launch vehicle. The motor retainer is designed to ensure that the motor is secured within the motor mount tube during flight. The motor retainer has been designed to withstand the forces experienced during the deployment of the booster main parachute. The force that the motor retainer will have to withstand during flight was calculated using

$$F_m = m_m a_m \qquad \textbf{(2)}$$

where m_m is the mass of the motor casing after burnout, and a_m is the acceleration experienced during the opening of the main parachute. The force the motor retainer would need to withstand was calculated to be approximately 120 lbs. FEA was conducted on the motor retainer design to verify that it could withstand this force. The results of the FEA, shown below in Figure 28, show that the motor retainer design can withstand the expected opening force with a minimum factor of safety of 3.649.

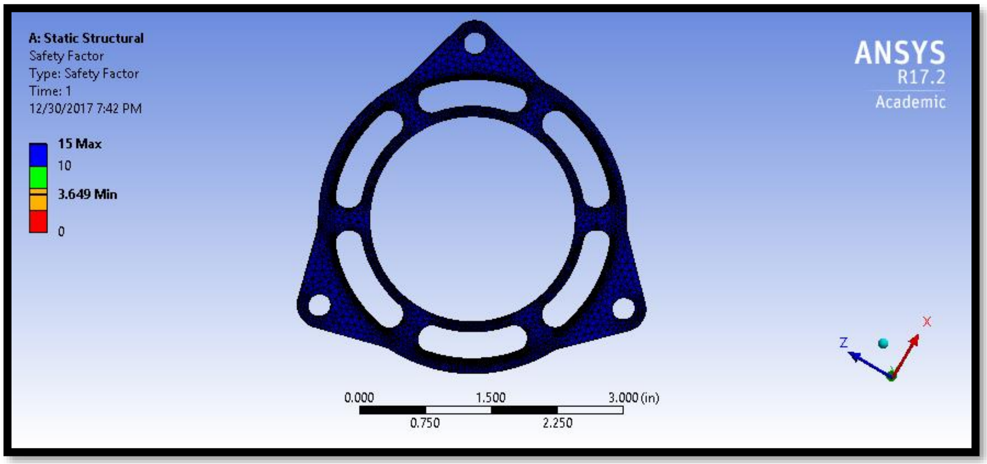

Figure 28: Motor retainer FEA simulation results.

3.3.5. Recovery Bays

The launch vehicle will utilize two recovery bays to store the main and drogue parachutes during flight. The recovery bays will be 22 and 24 inches in length providing adequate internal space for recovery equipment to be stored. The booster recovery bay will store the booster main parachute and the payload recovery bay will store the payload drogue and main parachutes.

3.3.6. Payload Bay

The payload bay is responsible for housing the rover payload and Rover Orientation Correction System (ROCS) that will be discussed in further sections. The payload bay will be 32 inches in length. The payload bay airframe will serve as the connection point for the ROCS using 20 #10-24 stainless steel button head cap screws connected from the exterior of the airframe into the ROCS. The payload bay with the rover and ROCS secured inside is shown below in Figure 29.

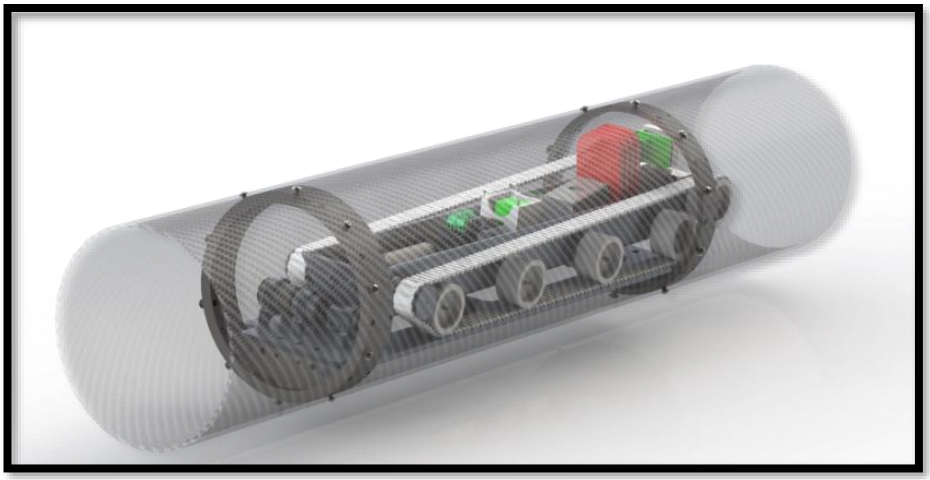

Figure 29: Payload bay with rover and ROCS secured inside.

3.3.7. Nose Cone Design

The launch vehicle will utilize a 12 in. long parabolic nose cone design with a 3 in. transition section. This design was chosen for its minimal drag and low mass. The nose cone will carry a StratoLogger altimeter and an AIM XTRA GPS device within a coupler that will be epoxied into the nose cone. A rendering of the nose cone design is shown below in Figure 30.

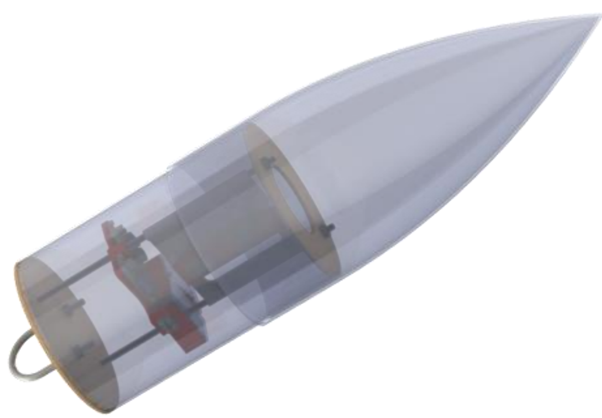

Figure 30: Parabolic nose cone with clear walls for clarity.

The nose cone has been additively manufactured from Nylon 12 using a Sinsterstaion 2500+ machine. This manufacturing method was chosen because of Nylon 12's low mass and the ease of manufacturing. The additively manufactured nose cone underwent drop testing further discussed in 6.1.2.4. Nylon 12's material properties are shown below in Table 6.

Mechanical Property	Value
Elongation at Break	15%
Tensile Modulus (ksi)	246
Tensile Strength (Psi)	6,815

Density (lb/in^3)	0.034

Table 6: Mechanical properties of Nylon 12.

3.3.8. Avionics

3.3.8.1. Separation Events

To reliably and redundantly separate the launch vehicle to deploy parachutes, each separation event will be triggered using two PerfectFlite StratologgerCF altimeters. The StratoLogger, shown below in Figure 31, will be programmed prior to launch day to detonate a pyrotechnic charge that will cause the launch vehicle to separate and deploy a parachute. A second Stratologger will be used to act as a failsafe if the first Stratologger malfunctions. To prevent over pressurization of the airframe, the redundant StratoLogger will be programmed to trigger either 1 second or 100 feet later than the first. The StratologgerCF is highly reliable and records altitude at a rate of 20Hz with a 99.9% accuracy. These altimeters have been found to be accurate to ±1 feet

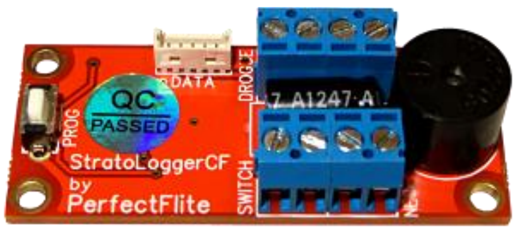

Figure 31: StratoLoggerCF altimeter.

The StratoLogger altimeters will be mounted onto a custom designed 3D printed sled. The sleds will be printed from PLA plastic using a IIIP 3D printer. The altimeters will be mounted onto the sled using four #4-40 nylon 6-6 screws. The altimeters will be powered by a Duracell 9-volt battery, secured in the battery cavity of the sled with three #4-40 nylon 6-6 screws. The launch vehicle will carry five StratoLogger altimeters, one in the nose cone for altitude scoring, two in the payload coupler, and two in the payload recovery coupler. The altimeters will be activated on the launch rail via a set of Featherweight PCB screw switches that will be accessible through a hole drilled in the airframe. A rendering of the altimeters secured to a sled is shown below in Figure 32.

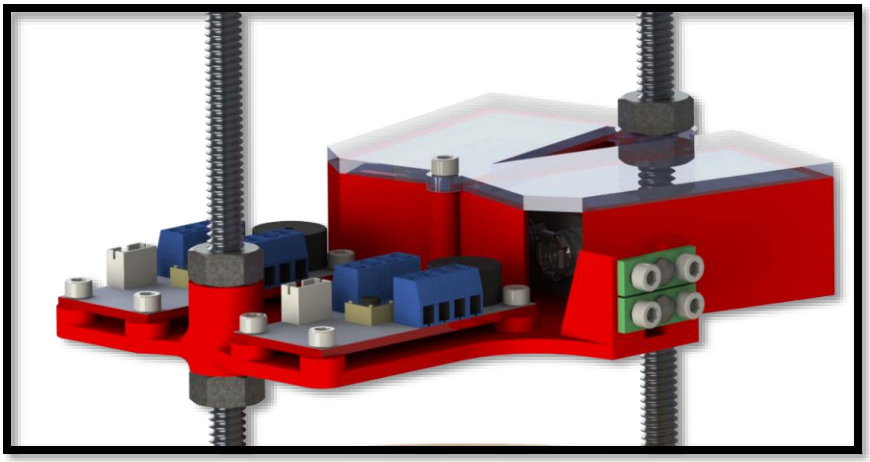

Figure 32: StratoLogger sled secured inside a coupler on two threaded rods.

3.3.8.2. Vent Hole Sizing

To prevent the altimeters from registering any pressure anomalies during flight, vent holes must be drilled in the avionics couplers. The diameter of the vent hole needed for each coupler was calculated using

$$D_{vent} = D_i \sqrt{\frac{L * A_{ref}}{V_{ref}}} \quad \textbf{(3)}$$

where D_i is inner diameter of the coupler, L is the length of the coupler, A_{ref} is the reference area of the vent hole, and V_{ref} is the reference volume of the coupler. According to Vern's Rocketry, it is recommended that a 0.25 in. diameter vent hole be created for every 100 in.^{3} of avionics coupler. Therefore, 0.049 in.^{2} was used for A_{ref} and 100 in.^{3} was used for V_{ref}. The dimensions of each avionics coupler, and their respective calculated vent hole diameter, are shown below in Table 7.

Avionics Coupler	Length (in.)	Interior Diameter (in.)	Calculated Vent Hole Diameter (in.)
Booster Recovery	5.75	5.85	0.310
Payload Recovery	5.875		0.314

Table 7: Vent hole sizes required for each avionics coupler.

3.3.8.3. GPS Tracking

Each independent section of the launch vehicle will carry a GPS tracking device. The launch vehicle will utilize four different types of GPS tracking devices, outlined below in Table 8.

Independent Section	GPS Tracker
Booster	Skytraq S1216F8-GL
Payload Coupler	Trackimo TRKM010
Payload Bay	Eggfinder GPS Tracking System
Nose Cone	AIM Xtra

Table 8: GPS tracking devices used in each independent section.

3.3.8.3.1. Skytraq S1216F8-GL

The Skytraq S1216F8-GL will be integrated into the VDS electronics coupler via surface mount on the printed circuit boards, and will serve as the GPS unit for the booster section of the vehicle. This unit can track up to 12 satellite signals at any given position, and it's -148dBm cold start sensitivity feature allows the unit to acquire, track, and fix onto any signal autonomously in weak signal environments. The positioning data from the Skytraq will be transmitted through the Xbee RF active telemetry system to the VDS ground station, allowing the team to monitor the position of this unit throughout the flight and upon recovery. The telemetry system will operate on a frequency between 902MHz and 928MHz, where the least noisy frequency within this range will be determined at the location of the launch, and used for transmission.

Figure 33: Skytraq S1216F8-GL

3.3.8.3.2. Trackimo TRKM010

The Trackimo TRKM010, shown below in **Error! Reference source not found.**, will be used to track the payload coupler as it can be easily mounted on the exterior of the upper bulkplate facing the sky during descent. The bulkplate will likely be exposed during descent and upon landing, giving a high likelihood of a successful position read-out. The Trackimo has been ordered and testing will be conducted to verify that the position can be read when mounted onto the top of a bulkplate.

Figure 34: Trackimo TRKM010 GPS tracking device.

The Trackimo uses GPS satellites and a worldwide cellular network to track the launch vehicle with an accuracy of up to 30 feet. The Trackimo operates on 850, 900, 1800, and 1900 MHz frequencies. The device's location can be viewed via a mobile app. To verify that the Trackimo would have cellular service on launch day, the AT&T 3G coverage map shown below in Figure 35, was checked and showed that there will be cellular service at Braggs Farms.

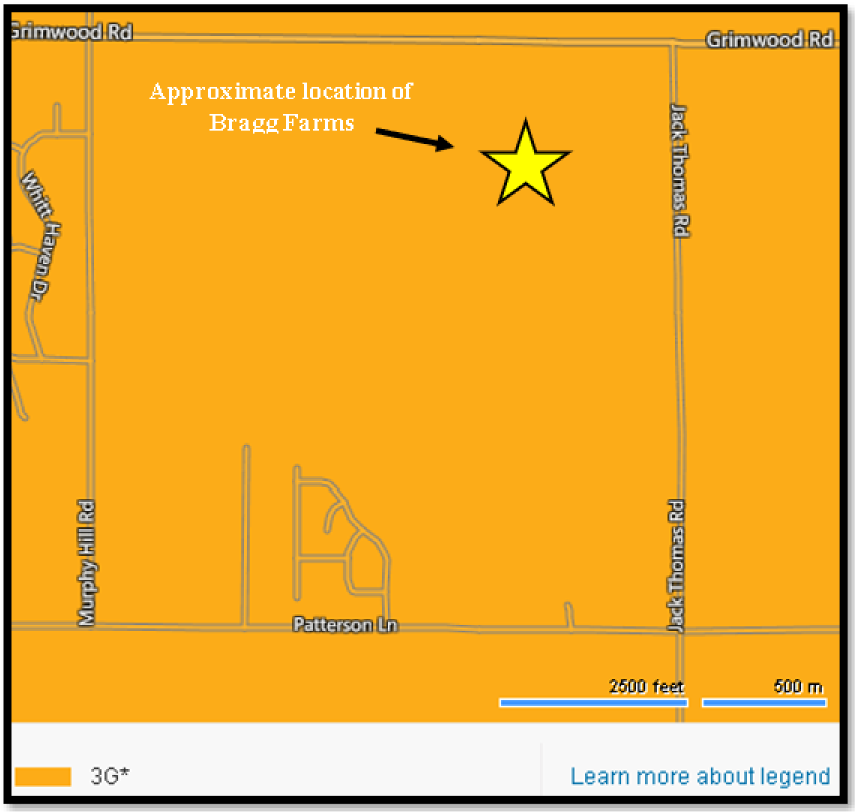

Figure 35: Map showing that Bragg Farms is within the AT&T 3G coverage zone.

3.3.8.3.3. Eggfinder GPS Tracking System

The Eggfinder GPS tracking system will be located in the payload recovery coupler of the launch vehicle. The Eggfinder is only 20 grams and has a range of 8000 feet from the tracking receiver and operates at a frequency of 900 MHz at 100mW. Due to the fact that the airframe is carbon fiber, an antenna connected to the Eggfinder will be secured to the exterior of the launch vehicle through a vent hole in the airframe. Placing the antenna on the exterior of the launch vehicle reduces the risk of signal attenuation from the carbon fiber airframe. The Eggfinder GPS tracking system is shown below in Figure 36.

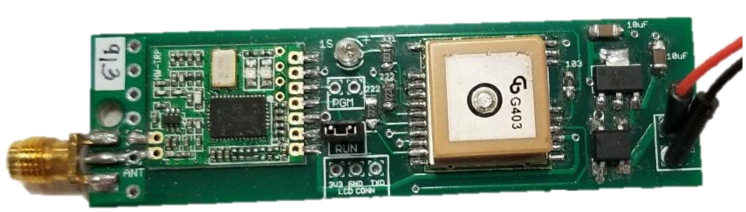

Figure 36: Eggfinder GPS tracking system.

3.3.8.3.4. AIM XTRA

The AIM XTRA will be stowed in the nose cone of the launch vehicle during flight. The AIM XTRA is an electronic device capable of GPS tracking, telemetry, and separation event triggering. The team will only make use of the GPS tracking capability to track the nose cone section of the vehicle during recovery. The AIM XTRA can be operated in the 432-434MHz band without the need for a HAM radio license and makes use of the AIM BASE connected to a laptop computer. The AIM XTRA is shown below in Figure 37.

Figure 37: AIM XTRA GPS tracking device.

3.3.9. Recovery Subsystem

3.3.9.1. Mission Overview

The launch vehicle separations will be staged in such a way that allows for each parachute to be deployed at safe distances from each other while also obeying the recovery distance requirements (RDR) outlined in SOW 3.9 of section 6.1.1. The recovery events are detailed in Table 9 below.

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

Table 9: Parachute deployment sequence of events

These steps are visualized below in Figure 38.

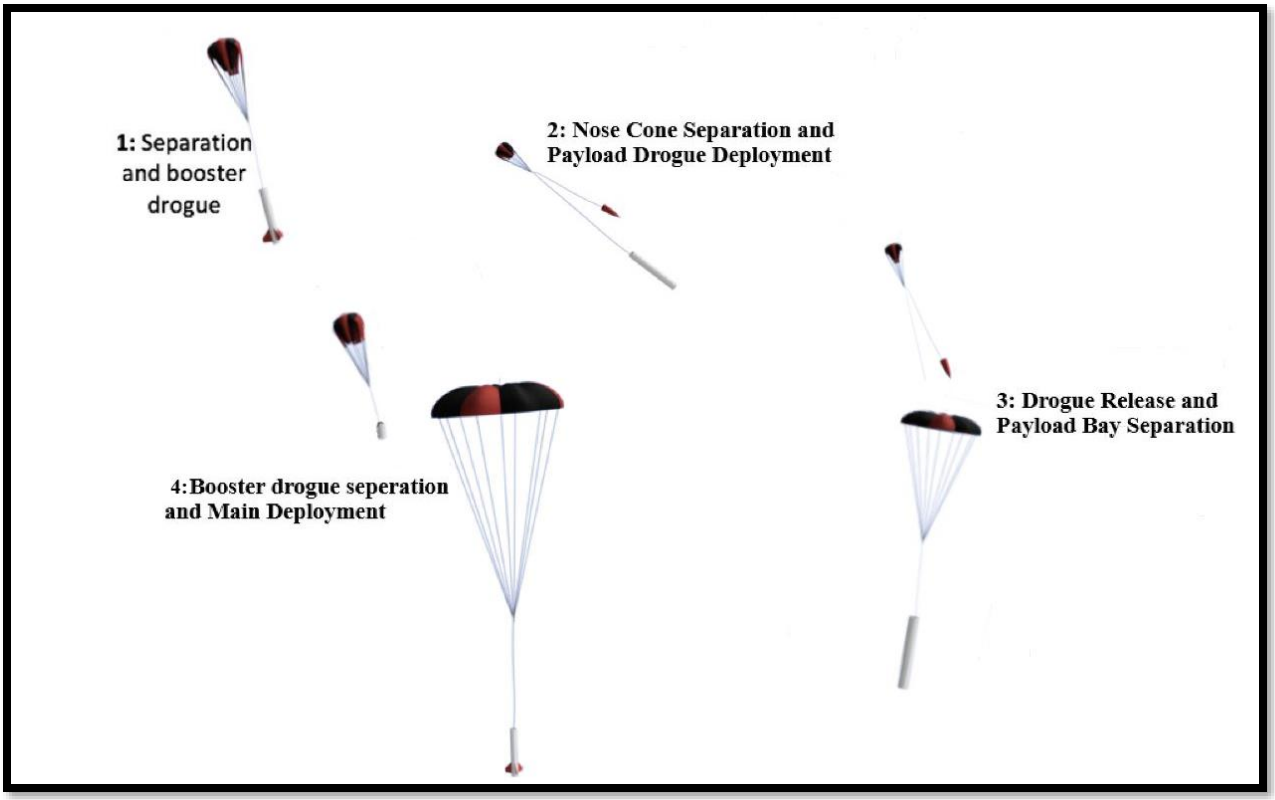

Figure 38: Parachute deployment procedure

The total mission elapsed time (MET) as well as the durations of the counterparts of the mission are shown below in Table 10 where the main decent phase of the nosecone and coupler will be under the drogue parachute acting as a main.

MET	94.8s			
Section	**Nosecone**	**Payload**	**Coupler**	**Booster**
Ascent	17.8s			
Drogue Decent	53s		53s	
Main Decent	20s	24s	20s	24s

Table 10: Mission elapsed time

Each drogue will act as a pilot parachute for its respective main parachute. The payload segment's drogue and main will be connected in series to an Advanced Retention and Release Device (ARRD) as shown in Figure 39. It is activated by two StratologgerCFs at the proper altitude using two redundant e-matches in the ignition chamber. The ignition of the black powder charge forces a piston inside the ARRD to release an attachment point out of the top of the body which the drogue is attached to. This will ensure that the drogue does not act as the pilot parachute until the main deployment event. The ARRD has been load tested to 2,000 lbs. and all components are recovered and cleaned after each flight and test. The ARRD has been tested both on the ground and during all the subscale flights with an exceptional success rate showing only one faulty deployment due to human error which will be mitigated in all future tests and launches with TDR 3.1.

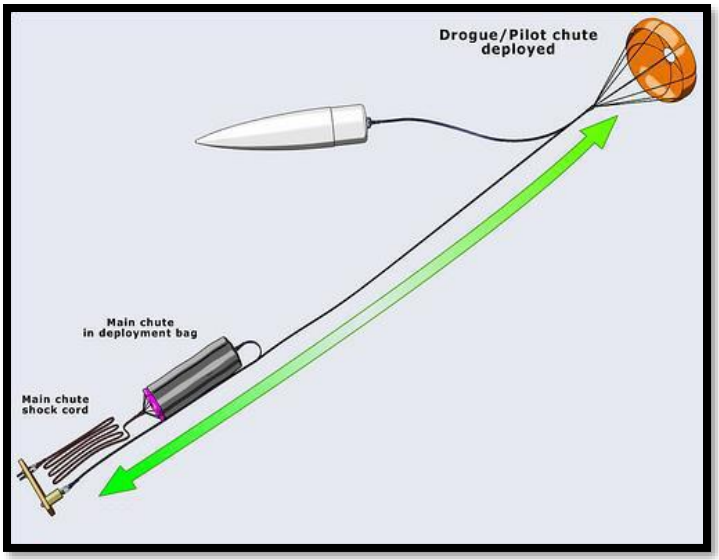

Figure 39: Steady state during drogue decent

The release device lets the drogue free during the main event. This moves tension to the tether attached to the main deployment bag and allows the drogue to now act as a pilot parachute for the bag. The drogue pulls the deployment bag from the bay, and at line stretch, pulls the bag from the main parachute, and frees the nosecone to descend under the retired drogue. This is detailed below in Figure 40.

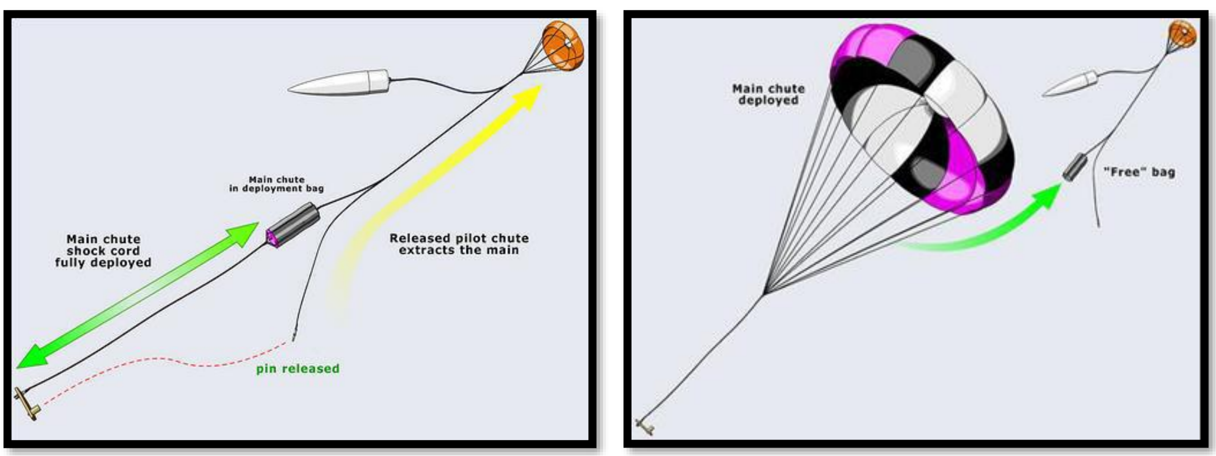

Figure 40: ARRD activation and main deployment

Unlike the payload section, the booster section of the launch vehicle will be recovered using a drogue and main separated into two bays. The drogue will be stowed in the coupler aft of the payload bay. During separation at apogee, the drogue will be loosely tethered to the payload section of the launch vehicle using weak adhesive tape. As the payload bay separates from the coupler, it will pull the drogue away from its stowed position. Once the maximum length in the shock cord is reached, the tether will sever, and the drogue will be fully deployed. At main event, the coupler will be separated from the booster recovery bay via black powder separation charges. The main parachute's deployment bag will be drawn out of the bay by the drogue as it is connected to the coupler. Once the maximum shock cord length is reached, the main will be pulled from the deployment bag and the coupler and drogue will become their own independent section. These Black powder charges are also activated by two redundant StratologgerCFs using two redundant black powder charges which are separated by a two second delay as to not over pressurize the airframe.

3.3.9.2. Design

The team has decided to design and manufacture custom parachutes as well as deployment bags and rigging for each possible application within the recovery subsystem. The design process and manufacturing process, as well as the materials used, are explained below.

3.3.9.2.1. Main

The two main parachutes will be toroidal in design. Toroidal parachutes are modified 0.707 elliptical parachutes where the venthole is pulled lower into the canopy by a centerline to increase the drag coefficient without affecting weight or volume. These parachutes consist of 10 panels of ripstop nylon. Each panel is designed based on the diameter of the parachute and the number of panels. The diameter of each parachute is derived to meet the Kinetic Energy Requirements (KER) in SOW 2.3 outlined in section 6.1.1. Nominal diameter for a parachute can be found using

$$\boldsymbol{D_o = \sqrt{\frac{4m_v m_s g}{\pi E C_D \rho}}} \quad \textbf{(4)}$$

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

$$\boldsymbol{D_o = \sqrt{\frac{4m_s^2 g}{\pi E C_D \rho}}} \quad (5)$$

The venthole size is found using

$$\boldsymbol{D_v = \frac{D_o}{5}} \quad (6)$$

where D_v is the diameter of the vent hole. The payload section's main parachute will be an 88 in. diameter with an 18 in. diameter vent hole and the booster section's main parachute will be a 99 in. diameter with a 20 in. diameter vent hole. These parachutes feature an outer set and an inner set of shroud lines to both the outer canopy and the venthole respectively. The lengths for both sets of these shroud lines are calculated using

$$\boldsymbol{L = 2D_o} \quad (7)$$

where L is the length of the shroud line from the canopy to the attachment point and D_o is the nominal diameter. The Centerline is approximately 0.81% of the shroud line length. This has been found by adjusting the length and measuring for maximum drag force. These dimensions are shown below in Table 11.

Component	Material	Break Strength (lbs.)	Payload Length	Booster Length
Outer shroud lines	Spectra Line	300	176	198
Inner shroud lines	Spectra Line	300	35	40
Centerline	Paracord	320	136	153

Table 11: Rigging break strength and lengths

Ten panels were chosen as a balance between number of shroud lines and amount of manufacturing time. A larger number of shroud lines means that the parachute can absorb more force without the possibility of a shroud line snapping. however, this also means that more panels must be sewn together, creating more room for manufacturing error. The rigging for the main parachutes are visualized below in Figure 41 and Figure 42 respectively.

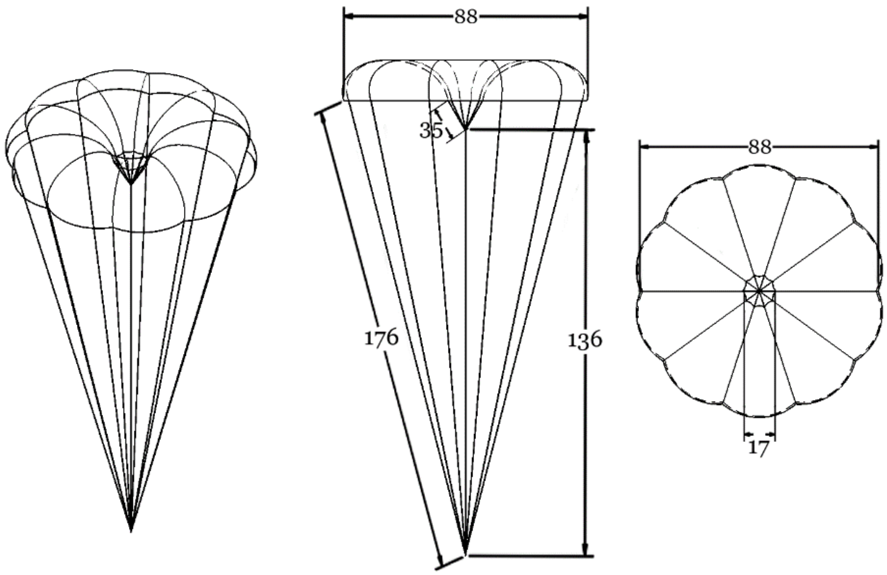

Figure 41: Payload main parachute

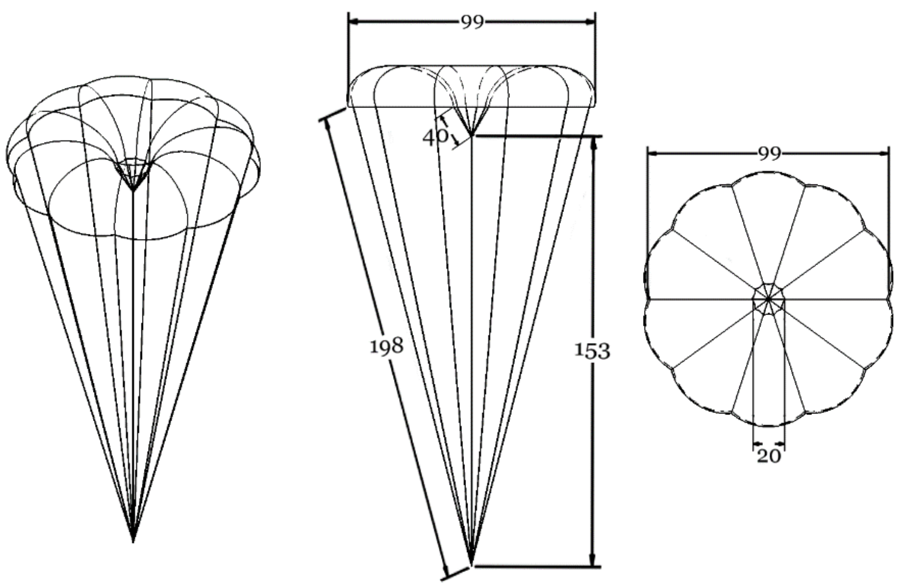

Figure 42: Booster main parachute

Opening Forces were calculated for the toroidal main parachutes to ensure the rigging materials would not break during deployment. Using the weights of both the booster segment and payload segment, opening force was found using

$$\boldsymbol{F_x = \frac{(C_D S)_p \rho V^2 C_x X_1}{2}} \tag{8}$$

Where F_x is the opening force of the main parachute, $(C_D S)_p$ is the drag coefficient proportional to the surface area of the parachute, ρ is the density of air at sea level, V is terminal velocity at opening, C_x is the opening force coefficient, and X_1 is the opening force reduction factor. C_x and X_1 are scaleable constants reliant on the design of the parachute. Using data acquired from subscale flight results, we can obtain an X_1 and a C_x for a toroidal parachute. First the C_x is found using

$$\boldsymbol{C_x = \frac{F_x}{F_c}} \tag{9}$$

where F_x is the measured opening force of a subscale parachute and F_c is the measured steady state force of a subscale parachute. Next the equation is solved backwards to find X_1 which is now the only unknown. The force reduction factor and the opening force coefficient of a toroidal parachute design can be scaled. Applying data from a full scale parachute design, we can solve for predicted opening force values. These values are shown below in Table 12. The shock cord then has a factor of safety of 2.8 and the shroud lines have a factor of safety of 6 when the strengths are combined.

Section	Newtons	lbs-f	G-force	m/s^2	ft/s^2
Payload main	1945.9	437.4	25.6	251.1	823.8
Booster main	2480.5	557.6	28.9	283.5	930.1

Table 12: Opening forces and accelerations

3.3.9.2.2. Drogue

The cruciform design was chosen for the Drogue parachutes, which is dependent upon two criteria. The first of which is the Kinetic Energy Requirement (KER) outlined in SOW 3.3 section 6.1.4.1 where each drogue parachute will become the main parachute for the coupler and nosecone after main deployment. The minimum diameter was calculated using

$$\boldsymbol{D_o = \sqrt{\frac{4m_v m_s g}{\pi E C_D \rho}}} \tag{10}$$

where D_o is the nominal diameter, m_v is the mass of the vehicle, m_s is the mass of the subsection, g gravitational acceleration, E is the kinetic energy, C_D is the coefficient of drag and ρ is the air pressure at sea level. This gives us a floor diameter.

Second the parachutes were sized to descend under a constant velocity where the amount of drift in 20 MPH winds would remain within 2,500 ft. The drifts calculated are shown in Table 34 of section 3.3.11.8 as well as the explanation of how they were calculated. With the vertical velocity requirement found from these calculations, we can solve for the surface area of the parachute using this equation.

$$S_o = \frac{2mg}{C_D V_e^2 \rho} \tag{11}$$

Where V_e is terminal velocity, and S_o is the surface area of the drogue. The cruciform design is rectangular where the width of a panel is $\frac{1}{3}$ the size of the length. given the surface area, the length is solved for using

$$L = \frac{3}{5}\sqrt{5S_o} \tag{12}$$

where L in the length across the long side of a panel. This gives us a ceiling diameter. Applying both constraints a diameter range is derived and shown below in Table 13 below.

Parachute	Minimum Diameter (KE)	Maximum Diameter (Drift)
Payload Drogue	25	36
Booster Drogue	16	35

Table 13: Drogue Minimum and maximum diameters

For simplicity and safety, a single size of 30 in. was chosen for both sections that would satisfy all requirements. Shroud lines for each drogue parachute will lead from a central attachment point to each corner of the parachute as shown in Figure 43.

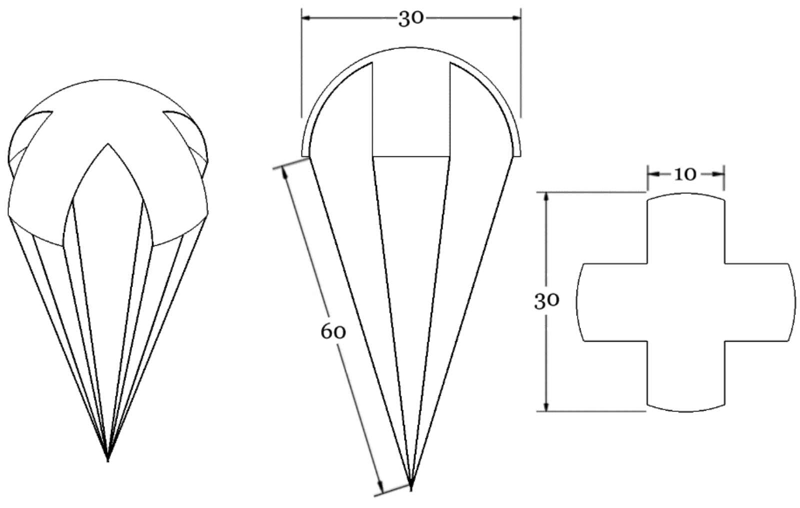

Figure 43: Drogue dimentions

3.3.9.3. Methods and Manufacturing

3.3.9.3.1. Panel gores

Each parachute will be processed from ripstop nylon fabric and sewn together in house. Ripstop nylon was chosen due to its exceptionally low porosity, high tensile strength and low weight to surface area ratio. ripstop nylon is named accordingly for the crosshatched reinforcing threads that are woven into the fabric. Due to this, cutting out parachute panels using traditional methods such as scissors are impractical and causes fraying. To avoid any potential failures, all panels are modeled in Adobe Illustrator and cut out using the Universal Laser System shown below in Figure 44.

Figure 44: Universal Laser Systems

Cutting out the panels in this manner has two primary benefits. First, it adds a level of precision, speed and repeatability that cannot be matched by cutting the panels out manually. Secondly, as nylon is a thermoplastic made from synthetic polymers, it can be melted to fuse edges and retain strength; using a laser to cut out the panels therefore also serves to cauterize the panel edges. To use the Universal Laser System (ULS), a board of eighth inch plywood is cut to the exact dimensions of the ULS. The ripstop is then stretched out across the board and pinned into place using thumbtacks as can be seen in Figure 45.

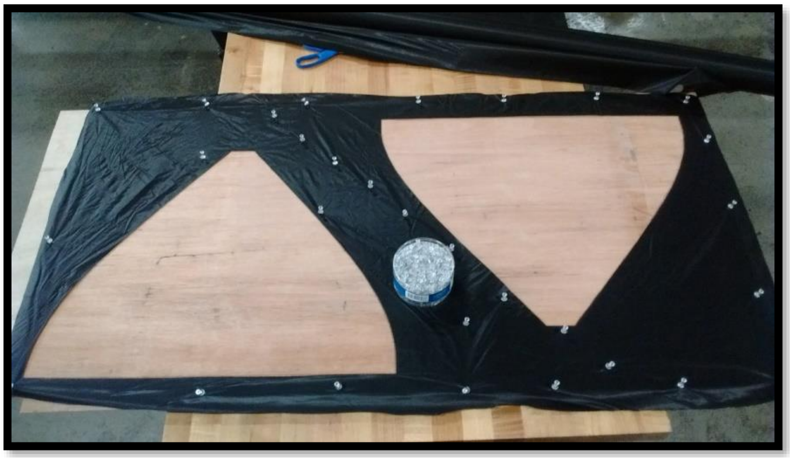

Figure 45: Pinned Ripstop after being cut on the ULS

Using the geometric symmetry of both the cruciform drogue and toroidal gore patterns, panels much larger than the 48in x 24in cutting area of the ULS can be manufactured with relative ease by folding along these symmetry lines. Once the panels have been cut out, they are hemmed to prevent fraying. They are either hemmed by folding the fabric onto itself twice, as shown in the top of Figure 46, or with a thicker material which provides strength at high-stress points and increases stability by decreasing the average angle of oscillation during deployment and descent. Protective ribbon hems are done using double straight stiches which firmly secures the material to the Ripstop shown in the middle of Figure 46. The fabric chosen for these protective hems is a lightweight and semi-flexible cotton fabric which is folded from two inches down to a quarter inch wide four-layer bias tape. For all ripstop hems, a single straight stich is used due to the lesser number of holes it punctures into the fabric. Panels are sewn together in a 0.5 in. wide clasping style which maximizes the amount of grip the two panels have when inflated and stretched and reduces the chances of the seams being undone. This is shown below in Figure 46 on the bottom. All stiches were done using Anefil Bonded Nylon Fire Resistant thread.

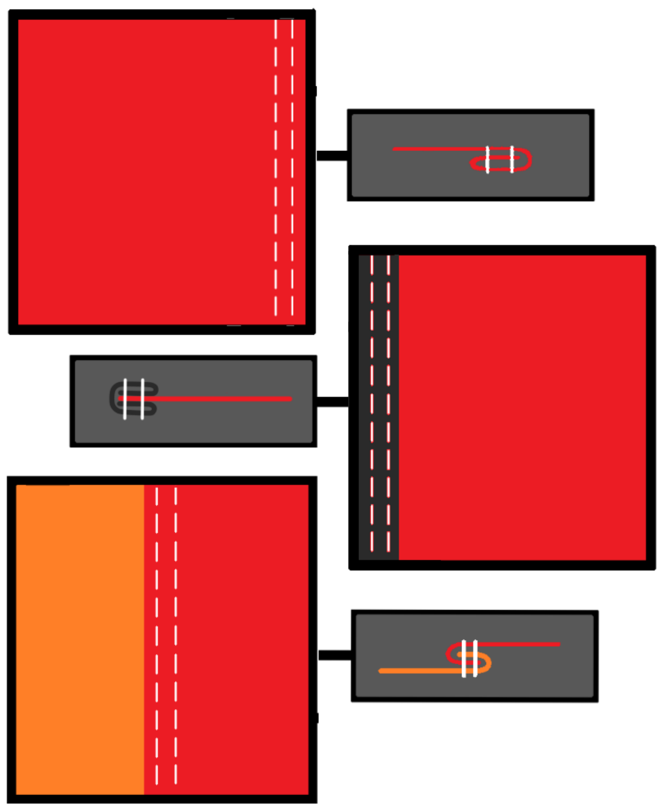

Figure 46: Stitching style for hems and attachment

3.3.9.3.2. Rigging

The shroud lines used for all parachutes are 0.047 in. fire resistant bonded white nylon, commonly referred to as spectra lines. The centerlines of each main parachute are 0.118 in. woven nylon paracord. These materials' break strengths and their uses are shown earlier in Table 11. The shock cord chosen for the launch vehicle is a 9/16 in. tubular nylon cord which will be connected to the vehicle using 5/16 in. zinc plated steel quick-links, rated for 1200 lbs. with a bowline hitch knot on each end of the cord. The bowline hitch was chosen for its self-tightening characteristics. The drogue parachute shock cord and deployment bag tether for the payload section will be directly tied to the attachment point of the ARRD and will not utilize a quick-link. The shock cord configuration is shown below in Figure 47.

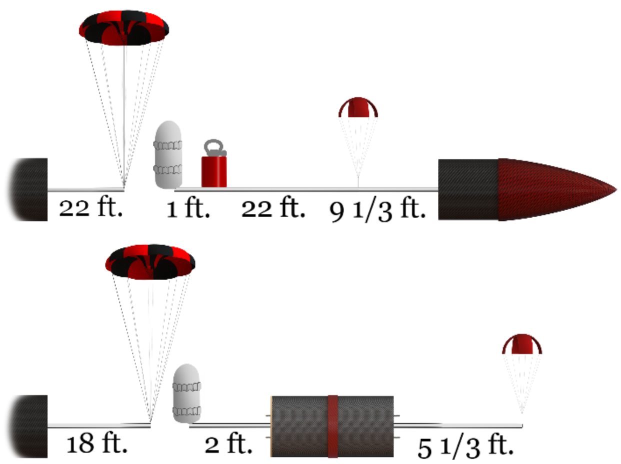

Figure 47: Shock cord configuration

3.3.9.3.3. Deployment bags

Each main parachute will be contained in a custom deployment bag made from fire retardant canvas as shown in Figure 48 that will hold the main parachutes but will not be tethered to them. The deployment bags will be pulled off by the drogue parachutes and will land separately with the nosecone or coupler that the drogues will carry.

Figure 48: Deployment bag line stows

The deployment bags feature loops to stow the shroud lines of each parachute along the side of the bag to ensure an orderly deployment. These looped shroud lines also hold a flap of canvas over the open end of the bag such that it only opens when the shroud lines are fully deployed as shown below in Figure 49.

Figure 49: Line stow latch

This also ensures the parachute will not deploy until all shock cord and shroud lines are at full length.

3.3.9.3.4. Nomex blast cone

The payload bay will also house a 3.0 cc black powder charge. This is a hazard to the payload's mission and protective methods were researched. CO2 deployment systems were tested but could not separate the vehicle. The difference in pressure did not provide enough force to fracture the shear pins and the gas seeped through coupler section of the airframe. Cylindrical charge wells were considered but subsequently also declared a hazard due to the possibility of fragmentation.

Figure 50: Nomex blast cone

Finally, a flame retardant fabric charge well was considered as shown in Figure 50 above. A nomex cloth was sewn along two edges to create a cone shape, and two corners were also sewn to create a small opening at the base of the cone. The opening is oriented away from the payload. This shape allows the chemical reaction of a black powder charge to be directed away from the payload to avoid residue. The design was tested with 3.0 cc of black powder and nearly all the residue was absorbed by the nomex, with only a small amount of residue exiting the mouth of the cone. The pressure change within the airframe was still sufficient and separated the vehicle. This method of separation will be further tested for reliability and consistency.

3.3.10. Subscale Vehicle Flight Results

3.3.10.1. Design Overview

To verify the overall design of the launch vehicle, and estimate the coefficient of drag, a one-half scale model was designed, manufactured, and flown. To replicate the recovery procedure that will be used on the full-scale vehicle, the subscale vehicle featured a single recovery bay in place of the full-scale vehicle's payload and payload recovery bays. The subscale vehicle also did not feature any VDS, and thus the VDS bay was replaced with an avionics bay. The avionics bay held two PerfectFlite StratoLogger altimeters, and an AIM XTRA flight computer. An Aerotech I300 motor was used on the subscale vehicle. An OpenRocket model of the subscale vehicle is shown below in Figure 51.

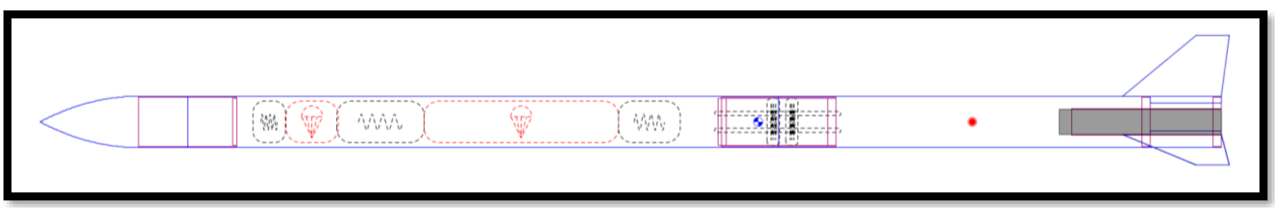

Figure 51: OpenRocket model of the subscale launch vehicle.

The subscale vehicle's airframe was constructed from 3 in. diameter BlueTube. The bulk plates and fins were constructed from 0.25 in. birch plywood cut with a CNC laser cutter. The nose cone, altimeter sled, and AIM XTRA sled were all 3D printed from PLA plastic with an IIIP 3D printer. A photo of the fully assembled subscale vehicle is shown below in Figure 52.

Figure 52: Fully assembled subscale launch vehicle.

A scaling factor of one-half was chosen for the subscale vehicle as an airframe diameter of 3in. is the minimum size that still allows for the integration of recovery hardware and flight electronics that will be flown in the full-scale vehicle. The one-half scaling factor also allows us to achieve very similar predicted flight characteristics to that of the full-scale vehicle, as shown in Table 14.

Characteristic	Subscale	Full Scale
Stability Margin at Rail Exit (cal.)	2.23	2.21
Exit Rail Velocity (ft./s)	94.9	94.5
Maximum Velocity (ft./s)	515	702
Maximum Acceleration (ft./s^2)	595	456

Table 14: Predicted flight characteristics of the subscale vehicle compared to that of the full scale vehicle.

A primary goal of the subscale vehicle is to accurately estimate the drag coefficient of the full-scale launch vehicle. The subscale vehicle was designed so that all exterior dimensions, including the nose cone design, were kept proportional to that of the full-scale vehicle. This allows the team to calculate an estimate of the full scale's coefficient of drag from the subscale's flight performance. Another parameter that was kept proportional to that of the full-scale vehicle, was the overall mass distribution of the vehicle. This allowed for a nearly identical stability margin at rail exit for the two launch vehicles and resulted in very similar overall flight trajectories.

Vehicle parameters that were not kept proportional to that of the full-scale vehicle were overall mass, material selections, and surface finish. The overall mass of the subscale vehicle was not kept proportional to that of the full scale as the mass does not affect the coefficient of drag. While the team could've designed the subscale vehicle to be ½ the overall mass of the full-scale vehicle, the benefits of doing so were minimal and would've only increased overall cost and difficulty of the project. The subscale vehicle experiences much less severe loads during flight than the full-scale vehicle and for this reason, lower strength materials were used. Using lower strength materials lowered the overall cost and decreased the difficulty in manufacturing the vehicle. Due to the differing airframe, nose cone, and fin materials, the surface finish of the subscale vehicle is likely to be different from that of the full-scale vehicle. While it is desirable to have identical surface finishes between the two vehicles, the team does not have the ability to accurately measure, and replicate surface finishes. The differing surface finishes will result in a propagation of errors in our coefficient of drag estimations for the subscale and full-scale vehicles, thus resulting in less accurate predictions and simulations. These errors should be easily corrected for on the full-scale vehicle with the use of the VDS.

3.3.10.2. Recovery Subsystem Design Overview

The recovery subsystem for the subscale vehicle was designed with two main goals. The first was to replicate the staged parachute deployment characteristic to both the payload and booster sections of the full-scale recovery system. Seeing as these are almost identical, it was deemed that only one main and one drogue parachute were required to properly test the design. The second goal was to obtain opening force data for the main parachute so that design specific variables, such as the opening force reduction factor, could be quantitatively measured. This is discussed further in 3.3.9.2.1. This data was found using the AIM XTRA accelerometer on the December 2nd launch. With these two goals in mind, the recovery system was designed with the deployment procedure shown below in Table 15.

Event	Altitude	Phase	Description
1	Apogee	Drogue Event	The Nose-Cone was ejected using a black-powder charge and pulled the drogue parachute out.
2	600 ft.	Nosecone Untether	The activation of the retention/release device untethers the nosecone from the recovery bay and becomes an independent section under the drogue parachute.
3	600 ft.	Main Deployment	The main parachute is deployed from the recovery bay by a retention/release device, where the drogue acts as a pilot parachute for the main parachute’s deployment bag.
4	0 ft.	Separate Landing	The nose-cone under the drogue’s influence lands alongside the main launch vehicle under the main parachute.

Table 15: Subscale vehicle recovery deployment procedure.

As with the full-scale recovery system, a single ARRD (Advanced Retention and Release Device) was used to untether the drogue and nose cone at the appropriate altitude. The subscale vehicle configuration for the ARRD is shown below in Figure 53.

Figure 53: Subscale ARRD Configuration

The drogue parachute was originally cut from ripstop nylon using a soldering iron to help seal the edges of the material and prevent ripping. This method is a less efficient method and lead to instability during decent. The instability caused the drogue to lose drag force and subsequently failed to deploy the main parachute. The improved method is discussed in 3.3.9.3.1. Table 16 shows the various specifications for both main and drogue parachute.

Subscale Vehicle Recovery Specifications			
	Main		**Drogue**
Diameter	30in		20in
C_D	1.4		.6
Shroud Line Length	45in (Outer)	8.73in (Inner)	40in

Table 16: Subscale vehicle recovery specifications.

All Parachute sizes and rigging specifications were found using the same methods discussed in section 3.3.9.2.

3.3.10.3. Launch Results

3.3.10.3.1. Launch Conditions

The subscale vehicle was launched in Elizabethtown, KY on November 11th and in Cedarville, OH on December 2nd. Various properties concerning the launches are shown below in Table 17.

Property	**November 11th in Elizabethtown, KY**	**December 2nd in Cedarville, OH**

Vehicle Mass on Pad	5.703 lbs.	5.978 lbs.
Temperature (°F)	46	50
Wind Speed	3 mph	4 mph
Wind Direction	NE	NW
Air Density at Ground Level (kg/m^3)	1.2850	1.2853

Table 17: Properties at time of launch on November 11th.

On the second flight of the subscale vehicle, the AIM XTRA flight computer was added to the avionics bay and is responsible for the small difference in overall mass of the vehicle on the two launches. The AIM XTRA provided the team with much more useful data that will allow us to better estimate the vehicle's coefficient of drag, further discussed in 3.3.10.4.

3.3.10.3.2. November 11th Launch

3.3.10.3.2.1. Flight Profile

On the November 11th flight of the subscale vehicle, the subscale vehicle exited the rail without incident and ascended stably to an apogee of 2,353 ft. The StratoLogger altimeters recorded the vehicle's altitude over the course of the flight. An altitude vs. time graph is shown below in Figure 54.

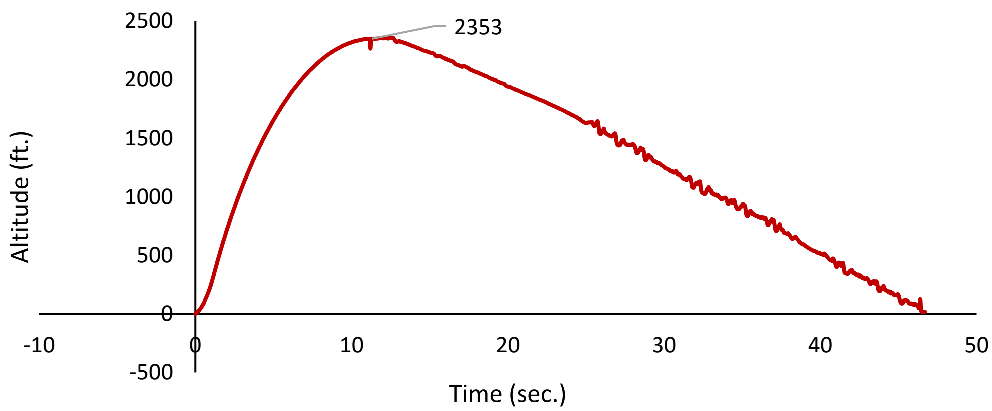

Figure 54: Altiude vs. Time plot of the November 11th launch.

At apogee, the nose cone was successfully ejected from the recovery bay and deployed the drogue parachute as designed. Following drogue deployment however, the nose cone and drogue began spinning rapidly. This spinning resulted in the drogue being unable to fully inflate. When the vehicle reached 600 ft. AGL, the StratoLogger altimeter successfully triggered the release of the main parachute bag via an ARRD. However, due to the drogue not being fully inflated, the drogue did not exert enough force on the tightly packed main parachute bag to pull it out of the recovery bay. This resulted in the main parachute failing to deploy and thus the subscale vehicle hit the ground at a greater than nominal speed. The subscale vehicle experienced severe damage including a broken fin, nose cone, and recovery bay as shown in Figure 55.

Figure 55: Damage sustained on November 11th flight after failed recovery.

3.3.10.3.2.2. Recovery Failure Analysis

Upon recovery of the subscale vehicle, it was observed that severe tangling had occurred between the nose cone shock cord and the drogue parachute's shroud lines. This tangling was caused by an imperfect manufacturing technique of the drogue parachute that resulted in an asymmetrical shape. The lack of symmetry in the drogue parachute caused the drogue to spin and wrap the nose cone around it's shroud lines. The tangling caused by the spinning resulted in the drogue not fully inflating. When the vehicle reached 600 ft. AGL, the StratoLogger altimeter fired a black powder charge and activated the ARRD which released the main parachute bag. The ARRD performed as designed and released the bag from the bulkplate as shown in Figure 56.

Figure 56: Photo showing that the ARRD released the main parachute bag as designed.

A combination of the drogue not being fully inflated due to spinning, and the main parachute bag being too large, caused the main parachute to fail to deploy.

3.3.10.3.2.3. Simulation Evaluation

To evaluate the accuracy of the OpenRocket simulation software, the altitude data from the November 11th launch was compared to a simulated launch using the weather conditions described in Table 17. An altitude versus time plot of the recorded altitude data and the simulated data is shown below in Figure 57.

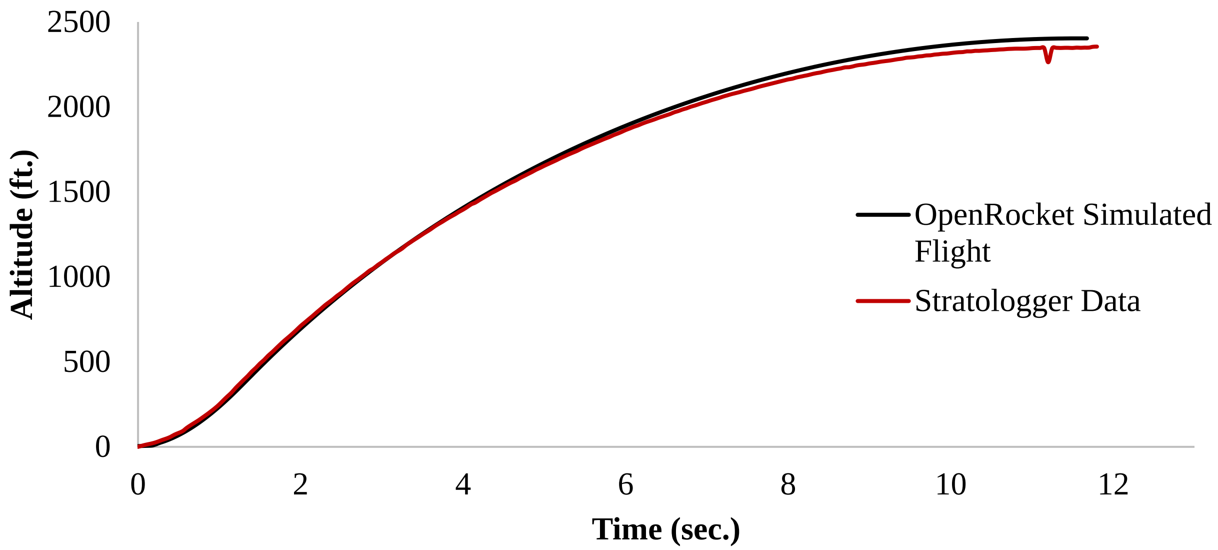

Figure 57: OpenRocket simualted altitude data compared to recorded StratoLogger altitude data.

As shown above, the OpenRocket simulation was quite accurate in predicting the overall ascent of the subscale vehicle. Statistical calculations were conducted to calculate the percent differences between the simulation and the actual flight data. The results are shown below in Table 18.

Property	OpenRocket Simulation	StratoLogger Data
Apogee Altitude (ft.)	2,404	2,353
Apogee Altitude Percent Difference	2.14%	
Average Percent Difference Between Altitudes	5.98%	
Post-Burn Phase Average Percent Difference Altitudes	3.17%	

Table 18: Percent differences between OpenRocket simulation and StratoLogger data.

3.3.10.3.3. December 2nd Launch

The subscale launch vehicle was launched a second time on December 2nd in Cedarville, OH. The vehicle flew with two Stratologger altimeters and an AIM XTRA flight computer. The AIM XTRA was flown on this flight to allow for a more precise coefficient of drag estimation.

3.3.10.3.3.1. Recovery Subsystem Design Changes

The drogue parachute used for the second subscale launch vehicle was the first laser-cut prototype and as a proof of concept met expectations with minimal tangling and suitable stability. Number of gores was increased from 8 to 10 to better conform to toroidal shape.

3.3.10.3.3.2. Flight Profile

The subscale vehicle exited the rail once again without incident and ascended stably to an apogee altitude of 2,258 ft. An altitude vs time graph is shown below in Figure 58.

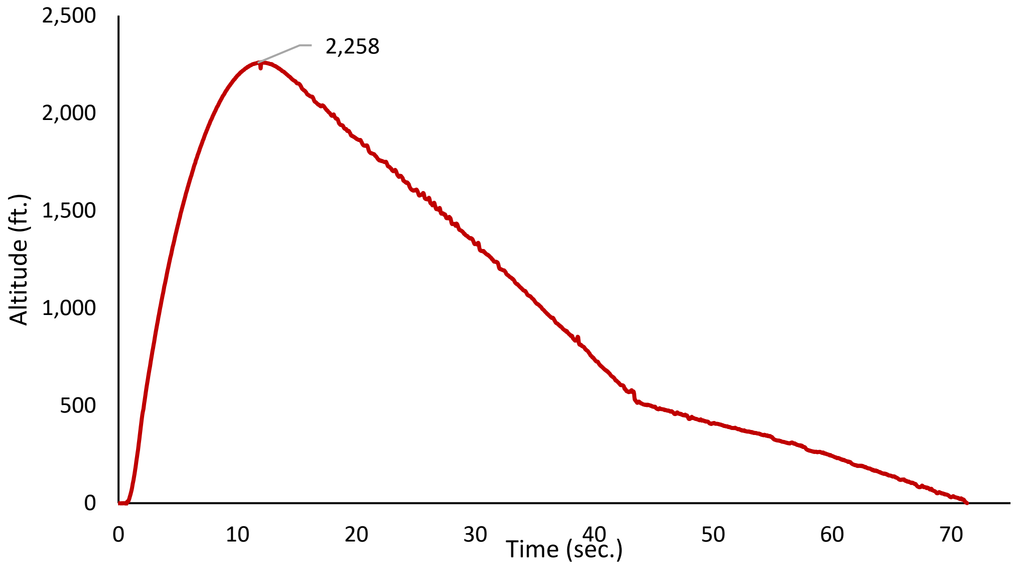

Figure 58: Altitude vs Time graph of the December 2nd launch.

At apogee, the nose cone was ejected from the recovery bay, and successfully deployed the drogue parachute. Unlike the November 11th launch, the nose cone and drogue spun minimally and allowed the drogue to fully inflate. At 600 ft. AGL, the ARRD released the main parachute bag and the main parachute was successfully deployed, as shown below in Figure 59. The nose cone and booster section landed gently and sustained no damage.

Figure 59: Photo of the subscale vehicle under parachute.

3.3.10.3.3.3. Simulation Evaluation

To evaluate how accurate the OpenRocket simulation software is, the altitude data from the December 2nd launch was compared to a OpenRocket simulated launch with the weather conditions specified in Table 17. An altitude versus time plot of the recorded altitude data and the simulated data is shown below in Figure 57.

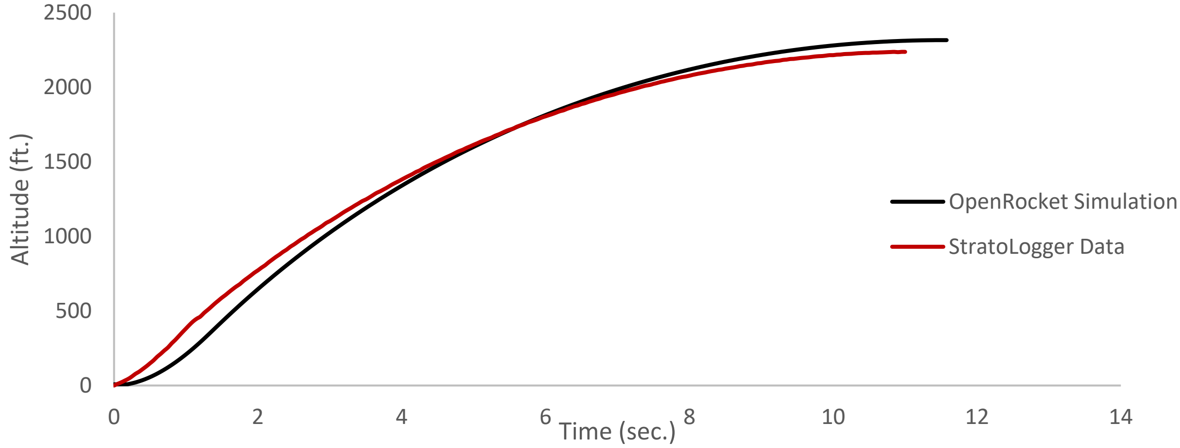

Figure 60: OpenRocket simulated altitude data compared to recorded StratoLogger altitude data.

In contrast to the November 11th flight, the simulation data differs from the recorded data significantly early in the flight during motor burn. After burn phase however, the simulation realigns with the recorded data. Statistical calculations were conducted to calculate the percent differences between the simulation and the actual flight data. The results are shown below in Table 19.

Property	OpenRocket Simulation	StratoLogger Data
Apogee Altitude (ft.)	2,315	2,258
Apogee Altitude Percent Difference	2.49%	
Average Percent Difference Between Altitudes	10.49%	
Post-Burn Phase Average Percent Difference Altitudes	3.40%	

Table 19: Percent differences between simulated altitude data and recorded altitude data.

3.3.10.4. Coefficient of Drag Estimation

To estimate the coefficient of drag for the subscale vehicle, the data recorded during the December 2nd launch with the AIM XTRA was analyzed. The AIM XTRA recorded the acceleration and velocity of the vehicle during coast phase, which allowed the team to estimate the coefficient of drag using

$$C_d = \frac{2m(a-g)}{\rho V^2 A} \quad \textbf{(13)}$$

where m is the mass of the vehicle, a is the acceleration of the vehicle, g is the acceleration due to gravity, ρ is the density of air, V is the velocity of the vehicle, and A is the cross-sectional area of the vehicle. To properly use this equation, only the coast phase of the subscale vehicle's flight was evaluated, so that the only forces acting on the vehicle are gravity and the drag force. To do this in a way which evaluated the entire coast phase and not just a single data point, the acceleration was plotted against the velocity squared and then a trendline was generated, as shown below in Figure 61.

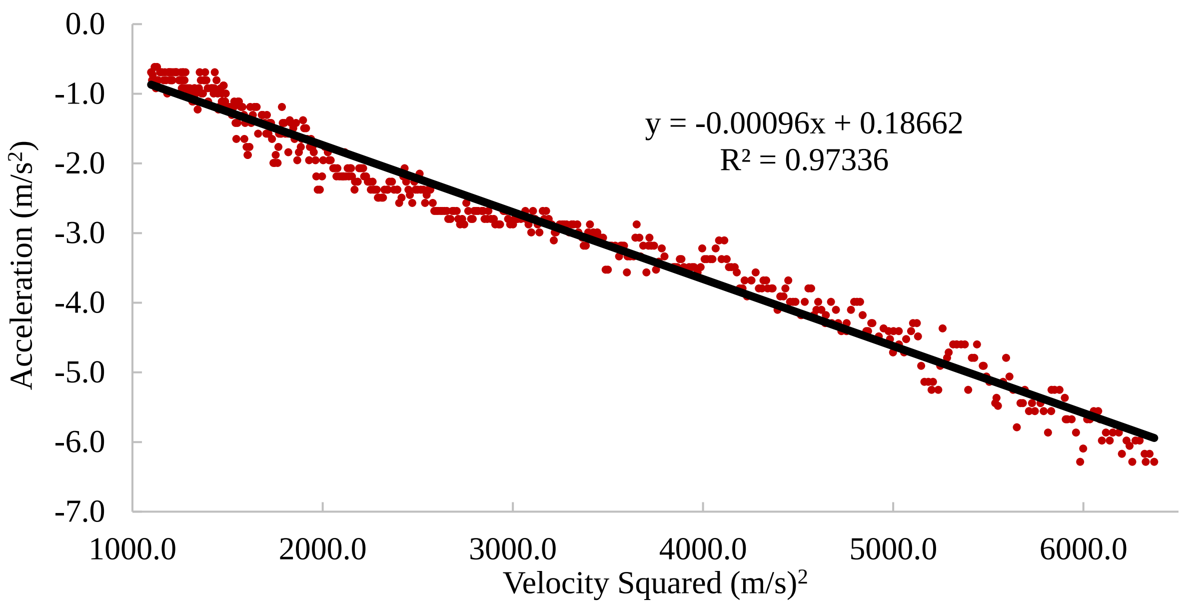

Figure 61: Acceleration plotted against velocity squared during the coast phase of the December 2nd flight.

By using the equation for the generated trendline, an acceleration term could now be calculated for any given velocity seen during the coast phase of the subscale vehicle. The density of air, ρ, was calculated using an online calculator to be 1.13 kg/m^3 at the average altitude during coast phase. The cross-sectional area of the subscale launch vehicle was calculated to be 0.006661277 m^2. Using these values, the trendline's equation for acceleration, and (13), the vehicle's estimated coefficient of drag was calculated to be 0.59.

In the OpenRocket simulations, the software used a coefficient of drag of 0.73, or roughly 23% higher than the estimated value the team calculated. This result did not reinforce what the data showed, as the subscale achieved a higher altitude in the OpenRocket simulations than what it achieved during a real flight, with a lower estimated coefficient of drag.

An explanation of this discrepancy in the coefficient of drag and altitude relationship lies in the overall thrust produced by the Aerotech I300 motor. The OpenRocket simulations used the motor data provided by the manufacturer. On the December 2nd launch, the AIM XTRA was flown onboard the subscale vehicle and recorded impulse and thrust data. The motor thrust data used in the OpenRocket simulations and the motor thrust data recorded by the AIM XTRA on December 2nd, is shown below in Table 20.

Parameter	Simulated Value	Recorded Value	Percent Difference
Average Thrust (N)	300.4	272.4	9.78%
Peak Thrust (N)	474.4	439.6	7.61%
Total Impulse (Ns)	426.5	387.5	9.58%
Burn Time (s)	1.40	1.41	0.71%

Table 20: Simulated motor characteristics compared to recorded motor characteristics.

According to the data recorded by the AIM XTRA, the Aerotech I300 rocket motor was observed to produce a maximum of 439.5 N of thrust and an average thrust of 272.4 N. Both data points are close to 10% lower than the value's used by OpenRocket in the simulations. This could potentially be the cause of the lower apogees experienced in the two flights, when compared to the simulations, despite a lower coefficient of drag.

3.3.11. Mission Performance Predictions

3.3.11.1. Applicable Equations

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

$$\boldsymbol{m_a = m_r + m_e - \frac{m_p}{2}} \tag{14}$$

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

$$\boldsymbol{k = \frac{1}{2}\rho C_D A} \tag{15}$$

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

$$\boldsymbol{q_1 = \sqrt{\frac{T - m_a g}{k}}} \tag{16}$$

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

$$\boldsymbol{x_1 = \frac{2kq_1}{m_a}} \tag{17}$$

The burnout velocity (m/s) is calculated using

$$v_1 = q_1 \frac{1 - e^{-x_1 t}}{1 + e^{-x_1 t}} \quad (18)$$

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

$$y_1 = \frac{-m_a}{2k} \ln\left(\frac{T - m_a g - k v_1^2}{T - m_a g}\right) \quad (19)$$

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

$$m_c = m_r + m_e - m_p \quad (20)$$

Using coasting mass, the coasting velocity coefficient is calculated using

$$q_c = \sqrt{\frac{T - m_c g}{k}} \quad (21)$$

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

$$x_c = \frac{2k q_c}{m_c} \quad (22)$$

The rocket's coasting velocity is then found using

$$v_c = q_c \frac{1 - e^{-x_c t}}{1 + e^{-x_c t}} \quad (23)$$

The coasting distance is found using

$$y_c = \frac{m_c}{2k} ln\left(\frac{m_c g + k {v_c}^2}{T - m_c g}\right) \quad (24)$$

The peak altitude of the rocket can then be found using

$$PA = y_1 + y_c \quad (25)$$

The rocket's center of gravity location is calculated using

$$cg = \frac{d_n w_n + d_r w_r + d_b w_b + d_e w_e + d_f w_f}{W} \quad (26)$$

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

$$X = \frac{(C_N)_N X_N + (C_N)_F X_F}{(C_N)_N + (C_N)_F} \tag{27}$$

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

$$X_N = \frac{2}{3} L_N \tag{28}$$

where L_N is the nose cone's length. Variable C_{NF} of equation 14 is defined by the fin center of pressure coefficient calculated using

$$(C_N)_F = \left[1 + \frac{R}{S + R}\right] \left[\frac{4N\left(\frac{S}{d}\right)^2}{1 + \sqrt{1 + \left(\frac{2L_f}{C_R + C_T}\right)^2}} \right] \tag{29}$$

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

$$X_F = X_B + \frac{X_R(C_R + 2C_T)}{3(C_R + C_T)} + \frac{1}{6}\left[(C_R + C_T) - \frac{(C_R C_T)}{(C_R + C_T)}\right] \tag{30}$$

where X_B is the distance from the nose tip to the leading edge of the fin root chord, X_R is the distance between the fin root leading edge and the fin tip leading edge measured parallel to the vehicle body. Equations 14 through 17 are also known as the Barrowman Equations (The Theoretical Prediction of the Center of Pressure, 1966).

Note that Equation 14 makes use of a simplified form because the vehicle makes no transition in the body diameter, thus the transitional terms have been omitted. These equations are used to verify the OpenRocket simulation conducted of the full-scale launch vehicle.

3.3.11.2. OpenRocket Flight Simulations

To simulate the flight of the launch vehicle, the OpenRocket software was used to create a model of the launch vehicle. The OpenRocket model of the full-scale launch vehicle is shown below in Figure 62.

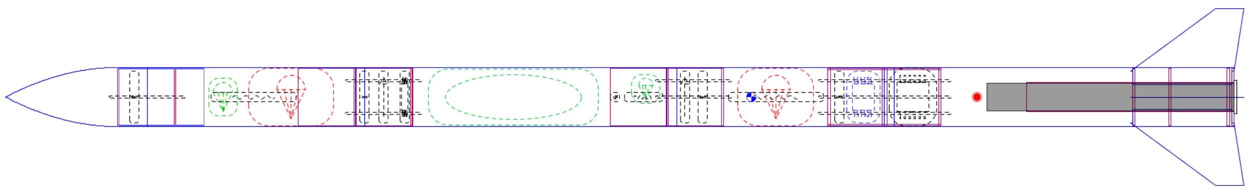

Figure 62: OpenRocket model of the full-scale launch vehicle.

3.3.11.2.1. Component Mass Estimates

To most accurately simulate the flight of the launch vehicle the masses of each component of the vehicle have been estimated or measured. These mass estimates were generated by weighing identical or very similar components, and using SolidWorks to generate a mass estimate based on material densities. The mass estimates were input into the OpenRocket model and were used to predict the flight of the launch vehicle and help dictate the overall design. The component mass estimates have been broken down by each independent section and are shown below in Table 21, Table 22,Table 23, and Table 24.

Nose Cone Section			
Component	**Mass (lbs.)**	**Quantity**	**Total Mass (lbs.)**
9-Volt battery	0.11	1	0.11
Altimeter Sled	0.11	1	0.11
Nose Cone	1.5	1	1.5
Nose cone all thread	0.03	2	0.06
Nose cone coupler	1.15	1	1.15
Nose cone recovery u bolt	0.065	1	0.065
Outer bulkplates	0.055	1	0.055
Stratologger	0.03	1	0.03
Inner bulkplates	0.05	1	0.05
AIM XTRA + sled	0.185	1	0.185
AIM XTRA battery	0.15	1	0.15
Total Section Mass (lbs.)			**3.465**

Table 21: Nose cone section component masses.

Payload Section			
Component	**Mass (lbs.)**	**Quantity**	**Total Mass**
Payload bay airframe	2.4	1	2.4
Payload recovery bay airframe	1.8	1	1.8
Payload recovery coupler all thread	0.05	2	0.1
Payload recovery coupler U-bolt	0.08	1	0.08
Outer bulkplates	0.055	2	0.11
Inner bulkplates	0.05	2	0.1
Payload coupler Nuts	0.008	2	0.016
Payload Drogue	0.2	1	0.2

Stratologger	0.03	2	0.06
Payload recovery coupler	1.15	1	1.15
ARRD	0.37	1	0.37
Altimeter Sled	0.11	1	0.11
GPS Receiver, Sled, Battery	0.183	1	0.1825
Payload Main	1.6	1	1.6
ROCS	3.967	1	3.967
RLM	1	1	1
DTS	0.139	1	0.139
RBS	0.988	1	0.988
RDS	1.636	1	1.636
OAS	0.063	1	0.063
SAS	0.498	1	0.498
SIS	0.044	1	0.044
CES	0.204	1	0.204
Total Mass (lbs.)			**16.8175**

Table 22: Payload section component masses.

Lone Coupler			
Component	**Mass (lbs.)**	**Quantity**	**Total Mass (lbs.)**
Payload coupler	1.15	1	1.15
Payload coupler all thread	0.02	2	0.04
Payload coupler CO2 plate	0.04	1	0.04
Payload coupler Nuts	0.01	16	0.128
Payload coupler U-bolt	0.08	1	0.08
GPS Receiver, Sled, Battery	0.18	1	0.1825
Altimeter Sled	0.11	1	0.11
Stratologger	0.03	2	0.06
Total Section Mass (lbs.)			**1.7905**

Table 23: Lone coupler section component masses.

Booster Section			
Component	**Mass (lbs.)**	**Quantity**	**Total Mass (lbs.)**
Booster airframe	2.7	1	2.7
Booster Recovery Equipment	1.7	1	0.8
Booster recovery bay airframe	1.65	1	1.65
Centering Ring	0.27	3	0.81
Epoxy	0.5	1	0.5
Fin	0.7	3	2.1
Fin Retainer	0.13	1	0.13
Inner bulkplates	0.05	2	0.1
Motor Retainer	0.1	1	0.1
Outer bulkplates	0.06	2	0.11

Aerotech L2200	10.6	1	10.6
Batteries	0.23	2	0.45
VDS blade	0.09	3	0.27
VDS bottom AL plate	0.11	1	0.1073
VDS coupler	1.15	1	1.15
VDS coupler all thread	0.05	3	0.15
VDS custom spacers	0	3	0.0105
VDS Delrin plate	0.17	2	0.34
VDS electronics	0.2	2	0.4
VDS limit switch	0	2	0.008
VDS middle wooden plate	0.05	1	0.05
VDS motor	0.76	1	0.7605
VDS Motor Shim	0.02	1	0.0185
VDS placeholder	1		0
VDS sled	0.3	1	0.3
VDS sled lid	0.05	1	0.05
VDS top AL plate	0.1	1	0.1
Shoulder Bolt large	0.02	3	0.06
Shoulder Bolt small	0.02	3	0.045
Telemetry electronics	0.2	1	0.2
Nuts	0.01	18	0.18
Total Section Mass (lbs.)			**25.1498**

Table 24: Booster section component masses.

The launch vehicle's overall on the pad mass is shown below in Table 25.

Total Vehicle	
Section	**Mass (lbs.)**
Nose Cone	3.465
Lone Coupler	1.7905
Payload Section	16.8175
Booster Section	25.1498
Total Vehicle Mass (lbs.)	**47.2228**

Table 25: Overall launch vehicle mass estimate.

3.3.11.2.2. Motor Selection

As discussed in PDR, several OpenRocket simulations were performed to determine which motor would be most optimal to deliver the launch vehicle to an apogee of approximately 5,500ft with an inactive VDS. Following design changes made since PDR, the launch vehicle's predicted overall mass has slightly increased and therefore new OpenRocket simulations were conducted. The new simulation results with each motor considered are shown below in Table 26.

Motor	Total Impulse (Ns)	Average Thrust (N)	Predicted Apogee Altitude (ft.)

Aerotech L2200	5,104	2,200	5,435
Cesaroni L2375	4,905	2,375	5,265
Cesaroni L3150	4,806	3,150	5,026

Table 26: OpenRocket simulation results comparing different motors considered and their simuated apogee altitudes.

The simulation results show that the Aerotech L2200 motor will deliver the launch vehicle to an apogee altitude just below 5,500 ft. with an inactive VDS. While the Cesaroni L2375 motor would deliver the launch vehicle to a reasonable apogee altitude, it does not reach our 5,500 ft. goal and is out of stock on many rocketry suppliers websites. The Cesaroni L3150 does not deliver the vehicle to our target apogee and therefore has been eliminated from consideration. For these reasons, the Aerotech L2200 has been selected for the launch vehicle's motor and several motor reloads have been ordered for use during the test launches and competition. The Aerotech L2200's specifications are shown below in Table 27.

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

Table 27: Aerotech L2200 rocket motor specifications.

3.3.11.2.2.1. Simulated Motor Thrust Curve

The OpenRocket software includes hundreds of rocket motors' thrust curves that it uses for conducting simulations. In the OpenRocket software's data sheets, it lists that the Aerotech L2200's thrust curve was acquired from the manufacturer, Aerotech Consumer Aerospace. The simulated motor thrust curve used in all simulations is shown below in **Error! Reference source not found.**.

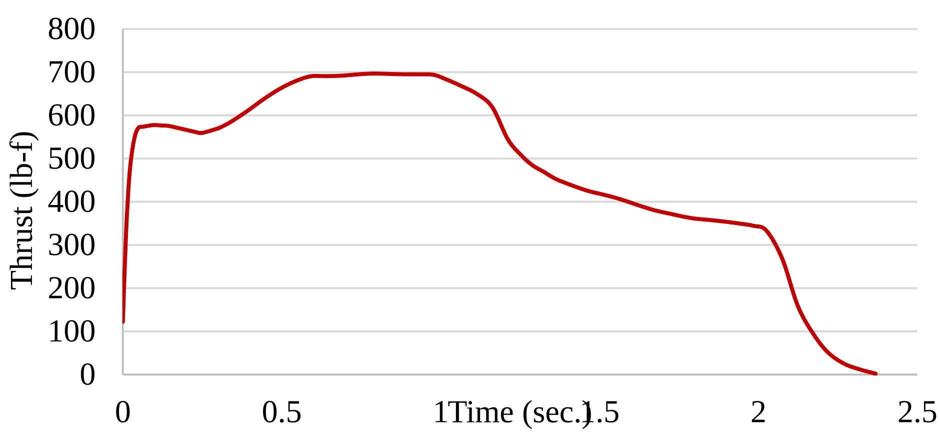

Figure 63: Aerotech L2200 simulated thrust curve used in all simulations.

The maximum expected thrust produced by the motor is approximately 700lbs. To account for the possibility of a centering ring failure, each centering ring was designed to withstand a minimum load of 350 lbs., while maintaining a minimum factor of safety greater than 2.0. Finite Element Analysis (FEA) was performed on each centering ring using ANSYS to verify that they could support the loads generated during motor burn. The results of the FEA were further discussed in 3.3.4.1 and showed that the centering ring design would withstand the forces experienced during motor burn.

3.3.11.2.3. Flight Characteristics

Using a combination of the equations discussed in 3.3.11.1 and the OpenRocket software, several flight related characteristics of the launch vehicle were calculated. Several critical flight characteristics of the vehicle are shown below in Table 28.

Exit Rail Velocity from a 141-inch rail (ft./s)	94.5
Center of Gravity Location at Rail Exit (in. from nose cone tip)	78.46
Center of Pressure Location at Rail Exit (in. from nose cone tip)	92.26
Stability Margin at Rail Exit (cal.)	2.21
Maximum Acceleration (ft./s^2)	456
Maximum Velocity (ft./s)	702
Maximum Thrust to Weight Ratio	14.44
Predicted apogee in 10mph wind (ft.)	5,435
Time to Apogee (sec.)	17.8

Table 28: Critical flight characteristcs of the launch vehicle.

3.3.11.2.4. OpenRocket Simulation Results

Simulations have been conducted with the OpenRocket software to predict the flight of the launch vehicle with an inactive VDS. In the simulation setup, it was specified that the launch vehicle launches from a 144-in. rail. All simulations, unless stated otherwise, were conducted with a wind speed of 10mph, under international standard atmospheric conditions, and launching from Bragg Farms in Toney, AL, at an elevation of 251 meters above sea level. These conditions were chosen for the simulations as they simulate likely conditions on competition launch day. The primary goal of the simulations was to predict the apogee altitude of the launch vehicle. Simulated apogee altitudes achieved by the launch vehicle in varying wind speeds are shown below in Table 29.

Wind Speed (mph)	Apogee Altitude (ft.)
0	5,462
5	5,455
10	5,435
15	5,409
20	5,372

Table 29: Simulated apogee altitudes at varying wind speeds.

The following plots in **Error! Reference source not found.** through **Error! Reference source not found.**, show various OpenRocket simulation results indicating proper motor selection and stability.

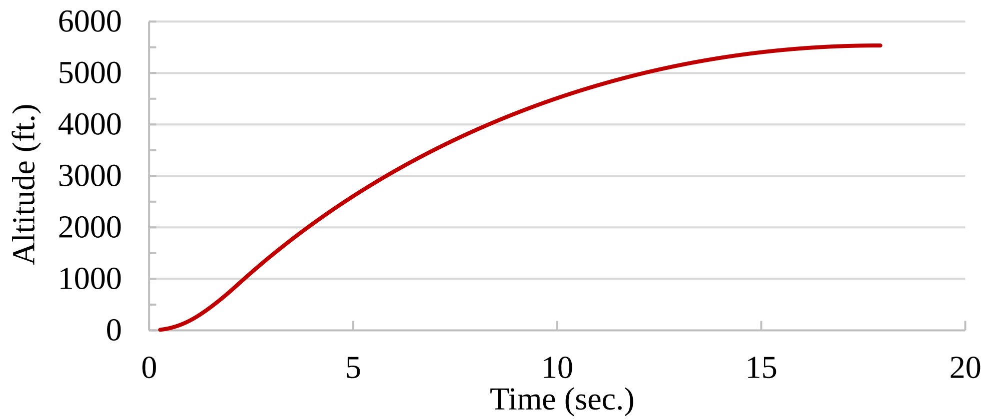

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Figure 64: Altitude vs time with an inactive VDS

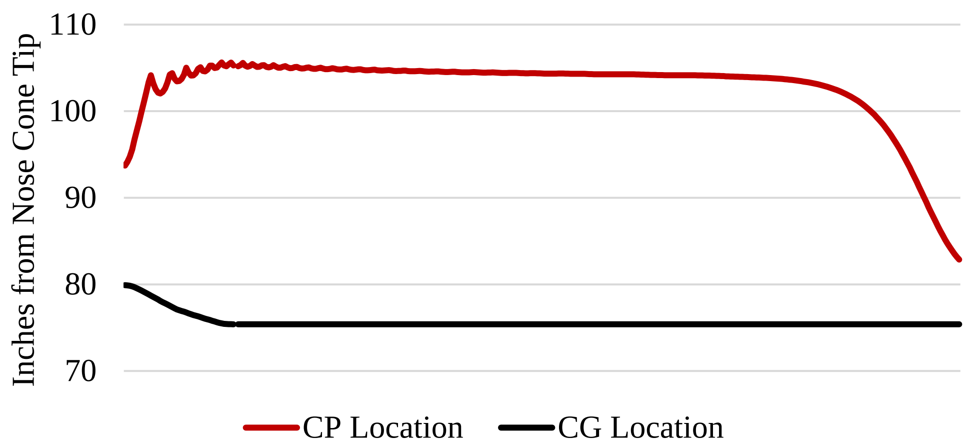


Figure 65: CP and CG location over time.

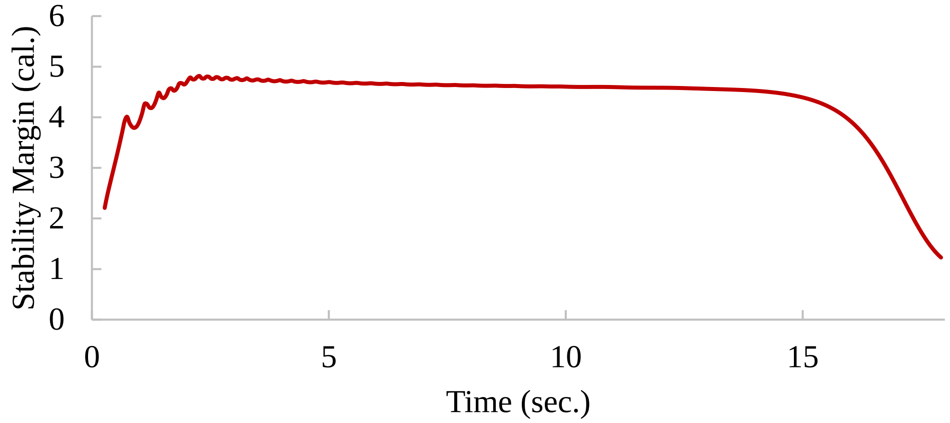


Figure 66: Stability margin vs time.

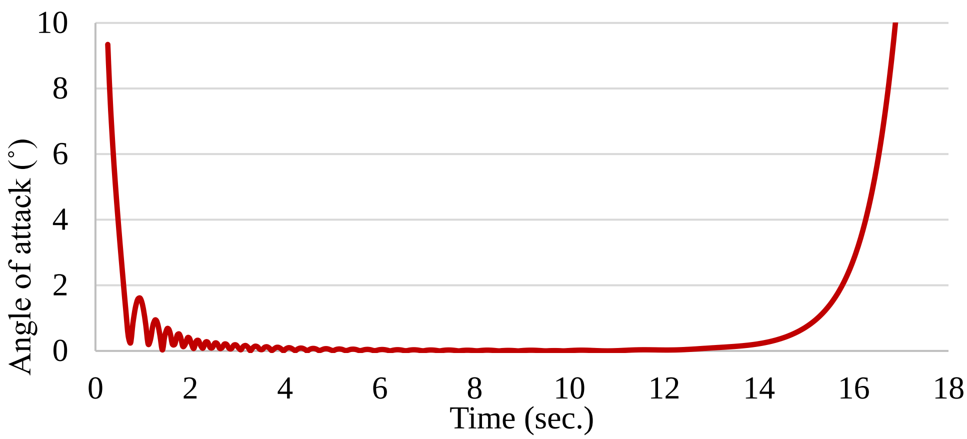


Figure 67: Angle of attack vs time.

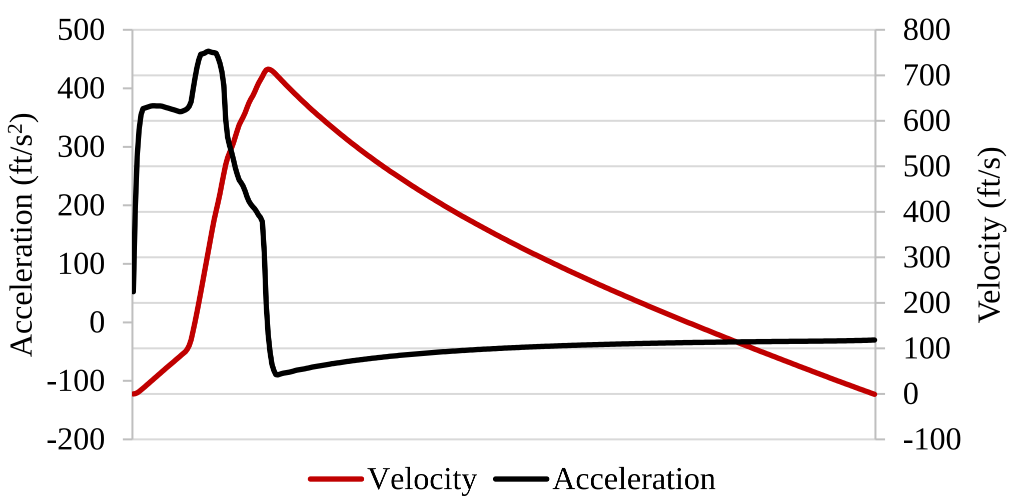


Figure 68: Simulated velocity and acceleration of the launch vehicle during flight.

3.3.11.3. VDS Simulation

To account for drag effects on the launch vehicle, the team has developed a simulation using Matlab. This simulation uses an iterative vector derivation of Newton's second law to model the kinematics of the rocket. The primary equation of this simulation is shown as

$$\bar{a}(i+1) = \frac{((-k * \bar{v}(i)^2) + F - M * g)}{M} \quad \textbf{(31)}$$

where *a* is the acceleration vector, *v* is the velocity vector, *F* is the function of thrust with respect to time (referring to the thrust curve of the L2200 motor), *M* is the function of mass of the rocket with respect to time, and *k* is

$$k = .5 * C_d * \rho * Ar \quad \textbf{(32)}$$

where C_d is the coefficient of drag, ρ is air density, and Ar is the area of the rocket. The simulation uses a Runge-Kutta approach to iterate the following value based on a previous data point. This iteration is shown in Figure 69: Runge-Kutta. This approach is an alternative method to using differential equations to model the rocket's mass to thrust behavior.

```
for i = 1 : length(tsym)-1
    t=tsym(i);

%      %Runge Kutta Approach
  hddot(i+1)=(-k*(hdot(i))^2+feval(Thrust_Func,t)-feval(Mass_Func,t)*g)/feval(Mass_Func,t);
   k1=hddot(i);
   k2=(-k*(hdot(i)+tstp/2*k1)^2+feval(Thrust_Func,t)-feval(Mass_Func,t)*g)/feval(Mass_Func,t);
   k3=(-k*(hdot(i)+tstp/2*k2)^2+feval(Thrust_Func,t)-feval(Mass_Func,t)*g)/feval(Mass_Func,t);
   k4=(-k*(hdot(i)+tstp*k3)^2+feval(Thrust_Func,t)-feval(Mass_Func,t)*g)/feval(Mass_Func,t);
   hdot(i+1) = hdot(i)+tstp/6*(k1+2*k2+2*k3+k4);
```

Figure 69: Runge-Kutta function

The initial simulation did not take into account drag forces from the VDS actuation blades, and only considered the vehicle launching vertically with no weather-cocking. The simulated altitude, velocity, and acceleration graphs are shown in Figure 70 and Figure 71, and utilized a C_d of .59 as predicted via the subscale model.

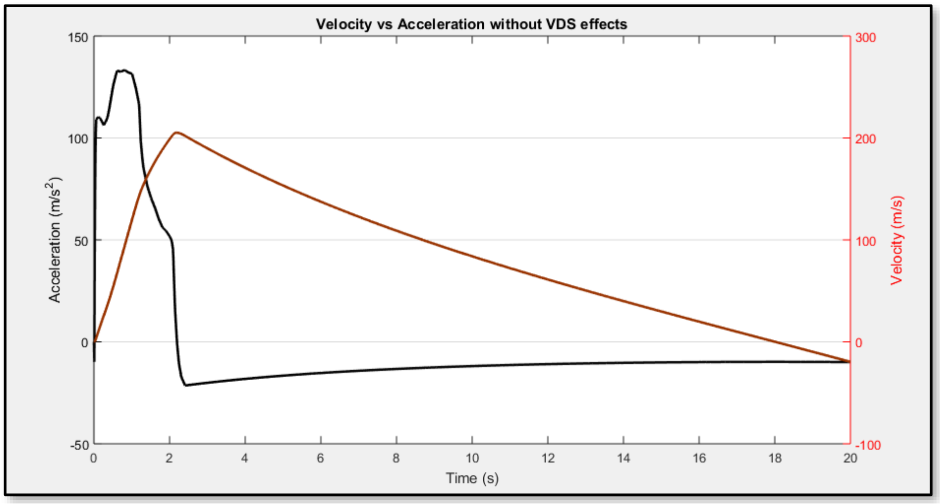


Figure 70: Simulated velocity and acceleration without VDS drag effects.

The purpose for the simulation with no drag effect is to verify the team's simulation equations with the OpenRocket's prediction software, and to model the difference between the movement of the rocket with and without the drag of the blade actuation. Shown in Figure 71: Simulated altitude without VDS drag effects is the simulated altitude of the vehicle, with an apogee of 1671 m. (5482 ft.). This data is similar to the OpenRocket based prediction of 5,462 ft. apogee.

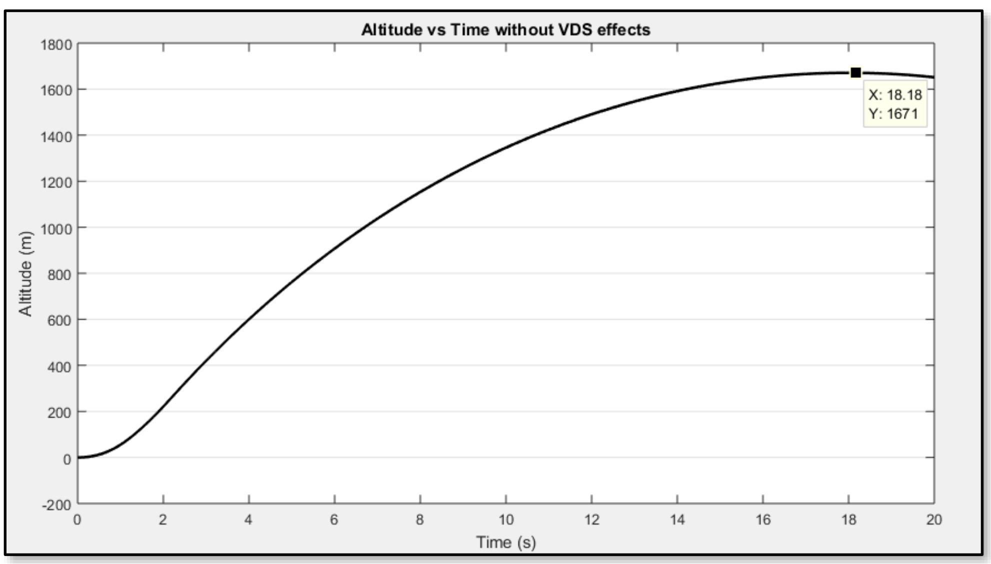


Figure 71: Simulated altitude without VDS drag effects

The simulation was re-run with an increased drag coefficient of .79 to represent a full actuation of the blades. This C_d value was chosen due to the drag blades increasing the overall drag coefficient of the vehicle by 1.35. In this simulation, the drag blades are brought to the fully deployed position as soon as the burn phase has ended around roughly 2.2 seconds. The blades are fully actuated until apogee, at which point they are retracted to ensure a safe recovery. In this instance, the simulated apogee altitude was shown to be 1573 m. (5,160 ft.).

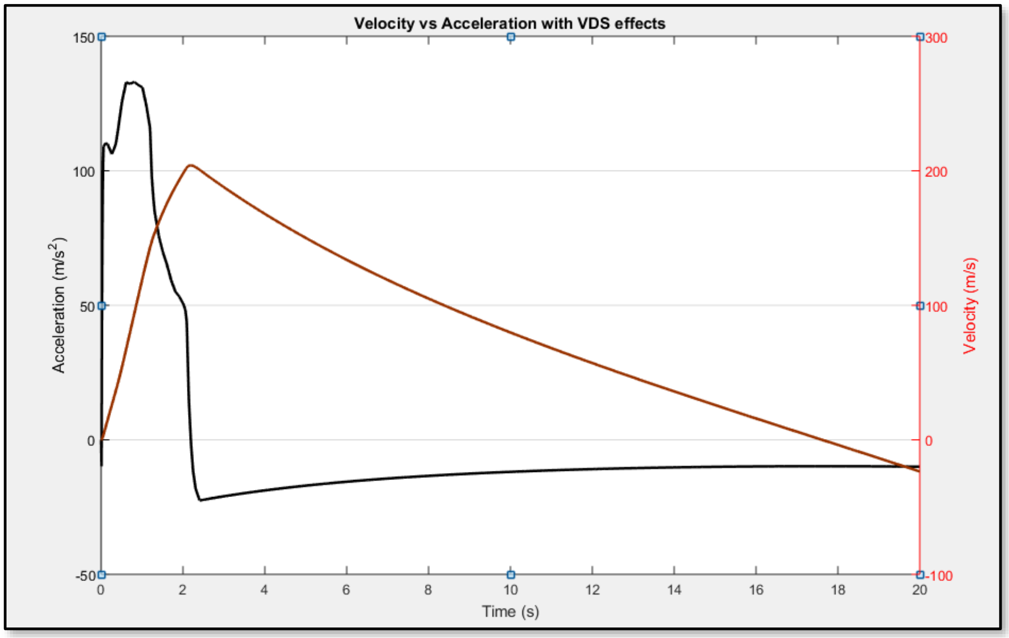


Figure 72: Simulated velocity and acceleration with VDS drag effects

These simulated data graphs are representational of the first and second full scale test flights that will be performed, the first with no actuation of the blades and the second with full actuation. These data sets are not representational of an actual VDS flight, where the computer will be consistently correcting from deviations from the Setpoint path. It is for this reason that an actual flight of the VDS is difficult to simulate, as it is not possible to predict what factors will act on the vehicle in a real flight. It is an average between a flight with no VDS drag, and one with full VDS drag that will bring the rocket to the desired target altitude, and cause the blades to dynamically actuate throughout the flight rather than stay statically deployed or not deployed.

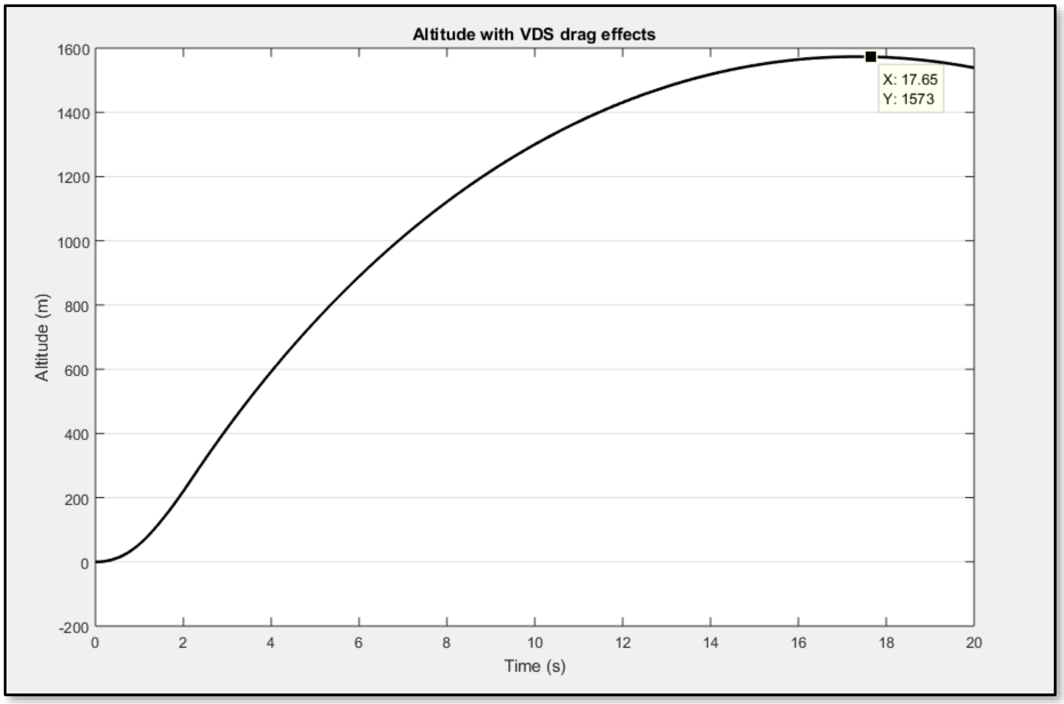


Figure 73: Simulated altitude with VDS drag effects

3.3.11.4. Center of Pressure CFD Analysis

The center of pressure of the launch vehicle discussed in section *2.13.2.3* is the launch vehicle's center of pressure at rail exit. SolidWorks Flow Simulation was used to verify the accuracy of the center of pressure location determined using OpenRocket. The center of pressure at launch rail exit must be known to ensure that the vehicle begins its ascent with an adequate stability margin. The CFD simulation was configured under the parameters presented in Table 30.

Parameter	Value
Airflow Velocity in Axial (Z) Direction (ft/s)	95
Airflow Velocity in Radial (Y) Direction (ft/s)	14.67
Angle of Attack (degrees)	0
Static Pressure (lbf/ft^2)	2116.217
Fluid Temperature (°F)	68.09

Table 30: SolidWorks Flow Simulations CFD Parameters for Center of Pressure Analysis.

The location of the center of pressure was measured from the tip of the launch vehicle's nose cone. A local coordinate system was established in the CFD simulation at the aft end of the launch vehicle. The coordinate system is shown in Figure 74.

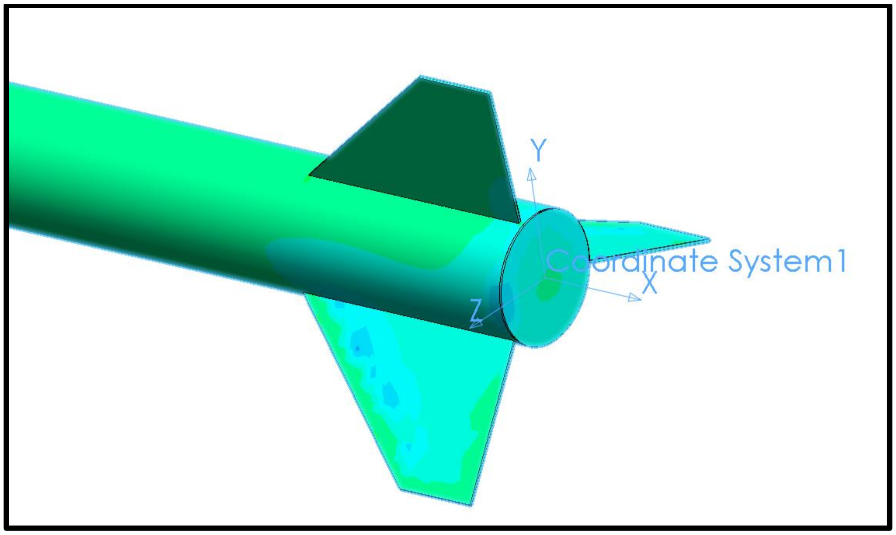


Figure 74: CFD simulation local coordinate system.

A global goal was established to determine the force in the Z direction of the local coordinate system, and the torque in the Y direction of the local coordinate system. The center of pressure was determined by dividing the torque in the Y direction by the force in the Z direction. The surface pressure plot resulting from the CFD simulation are shown below in Figure 75.

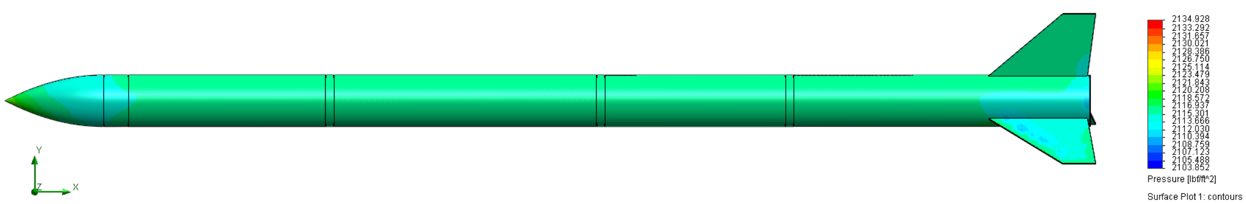


Figure 75: CFD simulation surface pressure plot.

The simulation converged to a solution after 90 iterations. The results of the simulation are compared to the center of pressure determined using OpenRocket in Table 31.

Center of Pressure Location at Rail Exit Measured from Nose Cone Tip		
CFD Simulation CP (in.)	OpenRocket Simulation CP (in.)	Percent Difference (%)
89.28	92.26	3.28

Table 31: CFD simulation and OpenRocket Center of Pressure Analysis Results Comparison.

The center of pressure obtained using OpenRocket simulation software resulted in a higher stability margin higher than the value obtained using SolidWorks Flow Simulation for CFD. The results obtained using SolidWorks Flow Simulation were less consistent than those obtained using OpenRocket, and therefore were only used to verify the value obtained using OpenRocket.

3.3.11.5. Fin Flutter Analysis

3.3.11.5.1. Hand Calculation

To ensure that the fin design can withstand the forces experienced during flight, analysis was conducted to determine if the fins would experience any aeroelastic flutter. Aeroelastic flutter, or “fin flutter”, can occur when the fin design, or the material selection, is not strong enough to endure the maximum velocity experienced. To ensure that fin flutter would not occur on the launch vehicle, the velocity at which the fins would begin to deform was calculated using

$$V_f = a\sqrt{\frac{G}{1.337\left(\frac{b^2}{S}\right)^3 P\left(\frac{c_t}{c_r}+1\right)\Big/ 2\left(\frac{b^2}{S}+2\right)\left(\frac{t}{c_r}\right)^3}} \tag{33}$$

where a is the speed of sound, G is the fin material’s shear modulus, b is the fin’s semi-span, S is the fin’s area, P is the air pressure at the altitude of max velocity, c_t is the fin’s tip chord, c_r is the fin’s root chord, and t is the thickness of the fin material. Using a shear modulus of 15.2 GPa, which is 20% lower than a typical low strength quasi-isotropic carbon fiber laminate, the velocity at which fin flutter would occur was calculated to be 909 ft./sec. This calculated fin flutter velocity is 207 ft./sec greater than the expected maximum velocity of the launch vehicle. These results provide the team with confidence that the fins will not fail during flight.

3.3.11.5.2. AeroFinSim Software Analysis

Fin flutter analysis was also conducted using the software AeroFinSim to verify the hand calculation’s results. The results of the AeroFinSim analysis is shown below in .

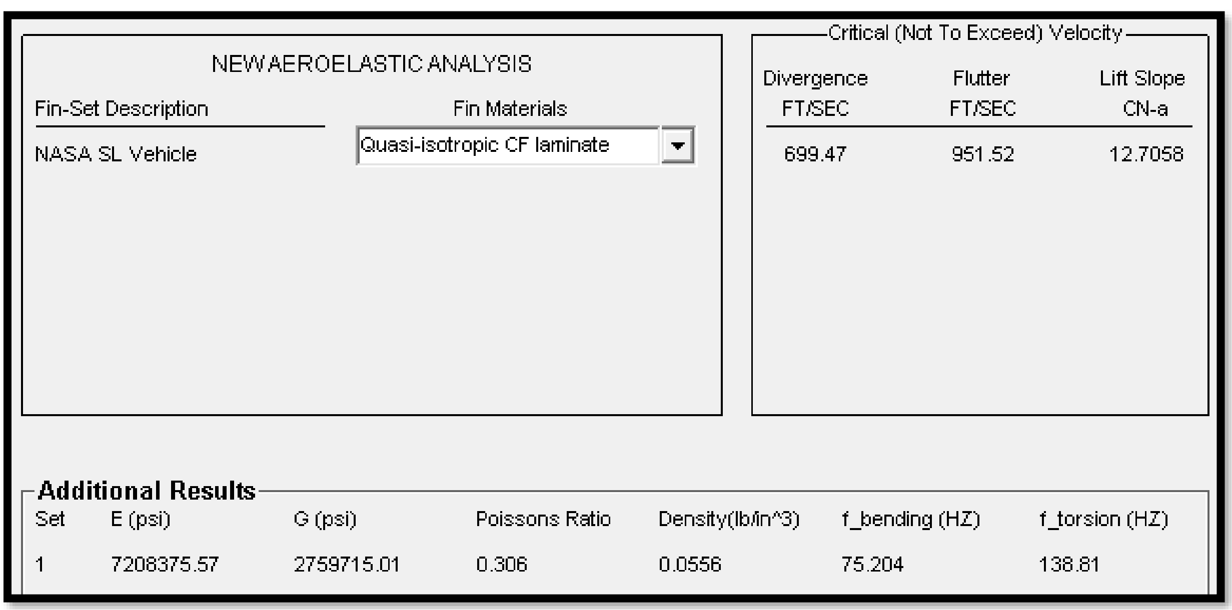


Figure 76.

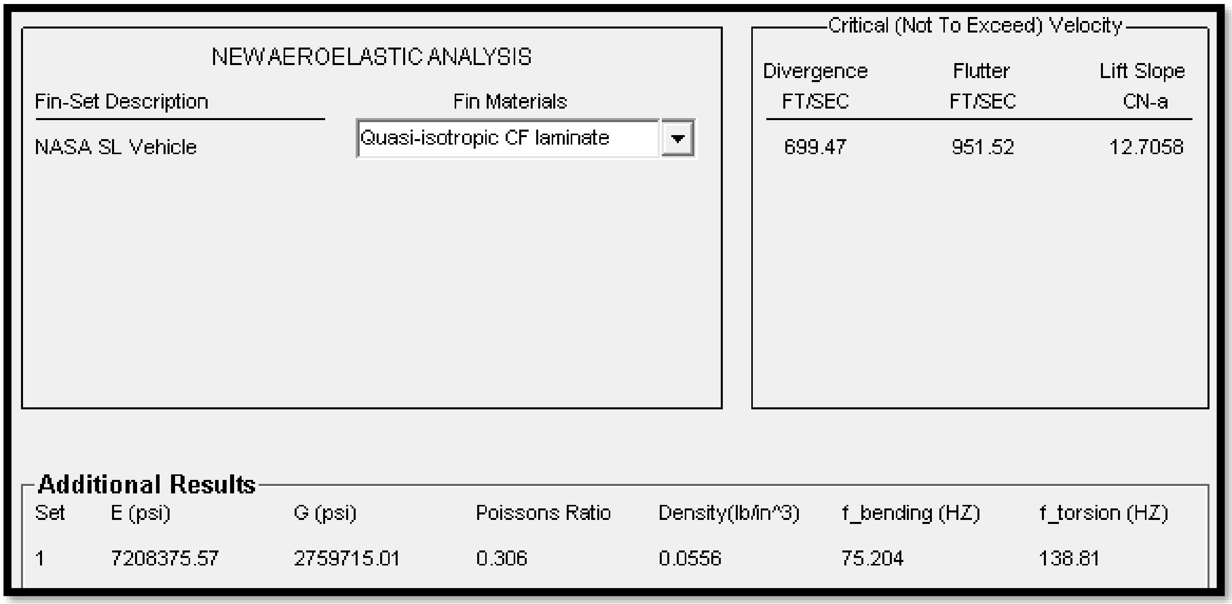


Figure 76: AeroFinSim fin flutter analysis results.

The results of the AeroFinSim fin flutter analysis show a flutter speed of 952 ft/s which is well above the expected maximum velocity of the launch vehicle. This result, combined with the hand calculation result, verify that the launch vehicle's fins will maintain structural integrity during flight.

3.3.11.6. VDS Fin Airflow Interference Analysis

As discussed in section 3.3.4.3, the launch vehicle was designed to implement a three cropped-delta fin design. The Variable Drag System was designed to be integrated into the launch vehicle with minimal interference to the airflow surrounding the vehicle's three stabilizing fins. The three drag inducing blades of the VDS are positioned at a 60-degree offset relative to the fins to eliminate the risk of any interference. The flow trajectory plot from the SolidWorks CFD Simulation referenced in section 3.3.11.4 shown below in Figure 77 verifies that the turbulent airflow induced by the drag blades does not interfere with the fins.

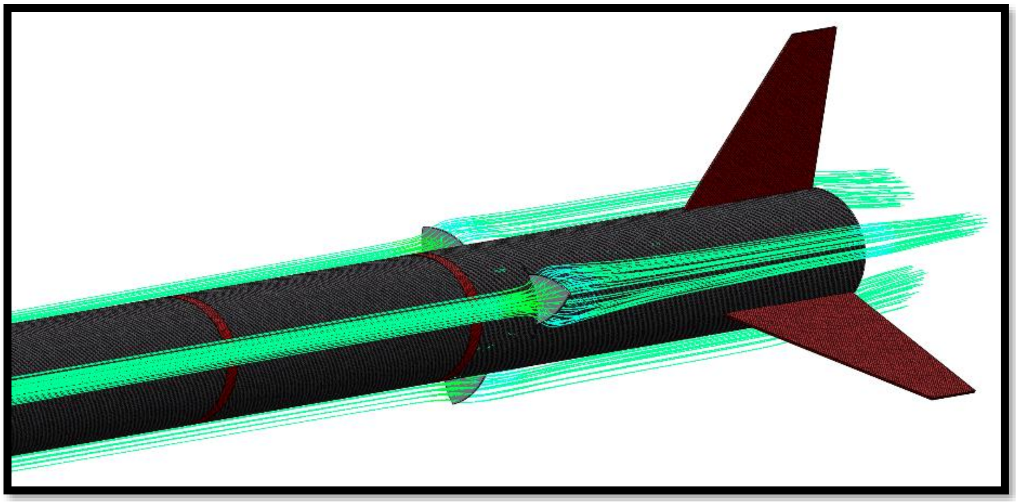


Figure 77: VDS blade CFD flow trajectory plot.

3.3.11.7. Kinetic Energy at Landing

All main and drogue parachutes are designed to land their segments of the launch vehicle at or under 65 ft-lbs of force in a nominal landing. The kinetic energy at landing of each independent section, as well as their terminal velocities at landing are shown below in Table 32.

Section	Section Mass (lbs.)	Ground Hit Velocity (ft/s)	KE (ft-lbs)
Nosecone	3.14	25.6	43.8
Payload	17.08	20.7	65.0
Coupler	1.98	25.6	17.5
Booster	19.29	20.7	65.0

Table 32: Kinetic energy at landing for each independent section.

3.3.11.8. Drift

The team has manufactured custom parachutes to fit each niche of the recovery system. The drogues were manufactured to adhere to the maximum drift specifications dictated in SOW 3.9 of section 6.1.1. The maximum drift value is used to solve for the size of the drogue parachute since they are tailored to specific needs. This process is outlined below and explained further in section 0.

Total drift is calculated by first finding the amount of drift seen while under the main parachute phase in 20 mph winds and subtracting it from the allotted 2500 ft. The main parachute size is dependent upon kinetic energy requirements and is a fixed number in this case. Its terminal velocity can be solved for using

$$V_e = \sqrt{\frac{2mg}{C_D S_o \rho}} \tag{34}$$

Where V_e is terminal velocity, m is the mass of the segment, g is acceleration due to gravity, C_D *is the drag coefficient*, S_o is surface area, and ρ is the density of air at sea level. Using the speed and knowing that each main event occurs at 500 ft., we can solve for the time until ground hit using

$$\frac{500\ ft.}{terminal\ velocity} = time \tag{35}$$

Using this time, we can solve for the distance the main parachute will drift laterally using

$$Windspeed \times time = distance \tag{36}$$

This process is shown in Table 33 below.

Section	Section Mass (lbs.)	Main Size (in.)	Terminal Velocity (ft/s)	Main decent duration (s)	Main Drift (ft.)	Drogue Drift (ft.)
Nosecone	3.14	30	25.6	20.0	573.7	1926.3
Payload	17.08	88	20.7	24.2	709.1	1790.9
Coupler	1.98	30	25.6	20.0	573.7	1926.3
Booster	19.29	99	20.7	24.2	709.1	1790.9

Table 33: Drift during main decent phase

The amount of distance left must be allotted to drogue decent phase. With this distance it is possible to solve for the minimum time at which the launch vehicle can safely descend without crossing these bounds. This equation is shown below.

$$\frac{2{,}500\ ft - main\ drift}{windspeed} = time \tag{37}$$

With the amount of time the vehicle can drift horizontally, we can find the speed at which it should fall vertically in that same time to cover the distance from apogee to main deployment height. The booster segment, which will have the largest drift, will have 53 seconds to fall from apogee to main deployment height before it crosses the bounds.

$$\frac{5280 - deployment\ height}{time} = vertical\ velocity \tag{38}$$

The vertical velocity found for the booster segment was 89.5 ft/s or 61 mph. The surface area of each drogue can then be solved for and simplified into a whole number for manufacturing simplicity, for example 30.3 in. becomes 30 in. The drift is then recalculated using the rounded number. The calculated drift values are shown in Table 34.

Wind Speed (MPH)	Wind Speed (Ft/s)	Booster Drift (Ft.)	Payload Drift (Ft.)	Coupler Drift (Ft.)	Nosecone Drift (Ft.)

5	7.3	569.1	565.8	535.2	532.0
10	14.7	1138.2	1131.6	1070.5	1063.9
15	22.0	1707.2	1697.4	1605.7	1595.9
20	29.3	2276.3	2263.2	2141.0	2127.8

Table 34: Drift values

All drift values are within the dictated boundaries. **Error! Reference source not found.** shows the booster's drift values overlaid on the competition field.

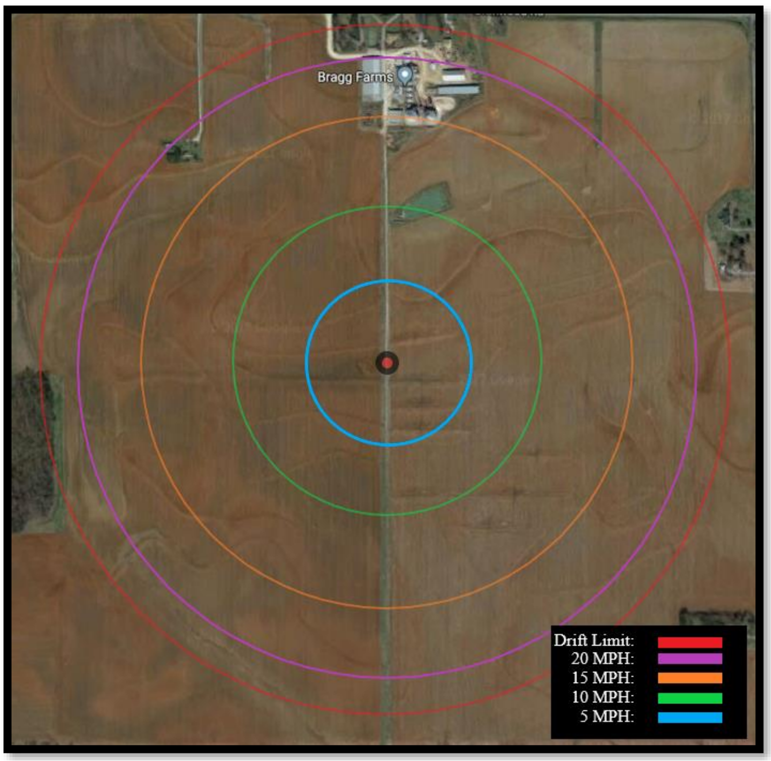


Figure 78: Booster drift visualization

4. Safety

4.1. Safety Requirements

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

4.1.1. Statement of Work Requirements

The Statement of Work requirements were provided by NASA and are shown in Table 35

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

		as new materials are used and processes are conducted.
5.3.4.	Assist in the writing and development of the team's hazard analyses, failure modes analyses, and procedures.	**<u>Demonstration</u>** The Safety Officer will meet with the lead of each subsystem to review and update the hazard and failure analyses and procedures prior to each test flight and major review.
5.4	During test flights, teams will abide by the rules and guidance of the local rocketry club's RSO. The allowance of certain vehicle configurations and/or payloads at the NASA Student Launch Initiative does not give explicit or implicit authority for teams to fly those certain vehicle configurations and/or payloads at other club launches. Teams should communicate their intentions to the local club's President or Prefect and RSO before attending any NAR or TRA launch.	**<u>Demonstration</u>** By agreeing to the Safety Manual, all team members agreed to follow decisions made by the RSO during all launches.
5.5	Teams will abide by all rules set forth by the FAA.	**Demonstration:** By signing the Safety Manual, all team members agreed to follow the regulations defined in FAR 14 CFR.

Table 35: Safety Statement of Work Requirements.

4.2.Safety Manual

The team Safety Manual outlines the specific shop procedures including

- Availability and location of emergency equipment including eyewash stations, fire extinguishers, and PPE.
- The need to understand the MSDS for materials used by the team and where all team MSDS are stored in the team cage.
- Proper waste disposal of hazardous waste like solvent contaminated rags and proper cleaning of machining chip and shavings.
- Required certification by Engineering Garage staff prior to use of the Electronic Bench and soldering irons
- Requirement that team members pass a mandatory safety quiz prior to accessing the Machine Cage and the heavy machining equipment that is stored inside
    - The quiz covers topics from the HSM Shop Safety handbook
    - All team members were warned about the penalty of being barred from the Engineering Garage if they do not conform to the Safety Manual

The following are required to use any equipment in the Machine Cage by Engineering Education Garage management:

Wear Safety Glasses- you must wear safety glasses AT ALL TIMES while in the shop area. You must wear safety goggles over prescription glasses unless your glasses have side shields and are ANSI safety approved.
Use hearing Protection- you will wear hearing protection as instructed during machine training
No jewelry- you will remove all rings, watches, necklaces, bracelets, and dangling earrings before operating any machinery or tools.
Proper Attire- you will wear ankle-length pants, loose hair and clothing are extremely dangerous. You must tuck in your shirt, roll up long sleeves, secure draw strings, tie back hair etc.
Clean up- before leaving the shop area, you must assist in cleaning all messes- metal chips, wood shavings, and splashed coolant) All liquids must be cleaned immediately to avoid slips
Return of tools and parts- all tools, instruments, bits, etc. must be returned to their proper location after use.
You must not operate equipment alone OR that you have not been trained to use. You must follow proper operating procedures detailed in the Job Safety and Sequence Instruction cards that are posted near all machinery.
You must not enter the shop area under the influence of drugs or alcohol, specifically over-the-counter drugs that include warnings against operating machinery. You must not consume alcohol within 8 hours of entering the shop area.
If the machine makes an unusual noise or acts in any suspicious manner, you must stop the machine and inform the Engineering Garage manager immediately.

All of the topics mentioned above were covered in the mandatory Safety Briefing. All members signed the Safety Agreement Form following this briefing, saying that they agreed to follow all things covered in the Safety Manual.

4.3. Safety Checklists and Launch Procedures

The Safety Officer is responsible for writing, enforcing, and maintaining all Safety Checklists and Launch Procedures. These lists are critical to ensure the safety of personnel, spectators, equipment, and success of the team.

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

The Safety Checklists and Launch Procedures are listed in the Appendix.

4.4. Hazard Analysis

The risk hazard tables were updated to better classify the risks before mitigations (RBM) and risks with mitigations (RWM).

4.4.1. Risk Assessment Matrix

By methodically examining each human interaction, environment, rocket system and component, hazards have been identified. However, hazards and risks will continue to be revised through the competition as new components are designed and manufactured. Risk assessment and mitigation are vital to the success of our project and team safety.

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

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

The Severity Criteria was maintained from PDR and is shown in Table 36.

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

Table 36: Severity criteria.

A probability level between A and E has been assigned to each identified hazard with a level of A being most likely. The probability value was determined for each hazard based on an estimated percentage chance that the mishap will occur. The Probability Table is shown in Table 37.

Probability		
Description	**Level**	**Criteria**
Almost Certain	A	Greater than a 90% chance that the mishap will occur
Likely	B	Between 50% and 90% chance that the mishap will occur

Moderate	C	Between 25% and 50% chance that the mishap will occur
Unlikely	D	Between 1% and 25% chance that the mishap will occur
Improbable	E	Less than a 1% chance that mishap will occur

Table 37: Probability criteria.

Through the combination of the severity value and the probability level, an appropriate risk level has been assigned using the risk assessment matrix found in Table 38. The matrix identifies each combination of severity and probability values as either a high, moderate, low, or minimal risk. The team's goal is to have every hazard to a low or minimal risk level by the time of the competition launch. Hazards that are not currently at a low risk level will be readdressed with redesign, additional safety regulations, or other measures as required. Risk levels are also being reduced through verification systems.

Probability Level	**Severity Value**			
	1 - Catastrophic	2 - Critical	3 - Marginal	4 - Negligible
A–Almost Certain	1A	2A	3A	4A
B – Likely	1B	2B	3B	4B
C – Moderate	1C	2C	3C	4C
D – Unlikely	1D	2D	3D	4D
E – Improbable	1E	2E	3E	4E

Table 38: Risk Assessment Matrix.

The Risk Level Matrix was updated to match the Handbook and to better describe the effect of the assigned mitigation on each hazard's risk. A Risk Level Approval Matrix was created based on the risk level of a hazard to show what level of approval is required for each risk level to be acceptable. If the risk levels are not reduced by FRR, the required approvals will be received and documented. The matrix is shown in Table 39.

Risk Level and Approval Matrix	
Risk Level	Level of Approval Required
High Risk	Highly undesirable. Documented approval of NASA SL team, RSO, team sub-team leads, team safety officer, and team co-captains required.
Moderate Risk	Undesirable. Documented approval of all team sub-team leads, team safety officer, and team co-captains required.
Low Risk	Acceptable. Documented approval from team sub-team lead overseeing the component's development.
Minimal Risk	Acceptable. Documented approval is not required. Sub-team lead will ensure that sub-team members are familiar with the hazard.

Table 39: Risk Level and Approval Matrix.

The risks evaluated were grouped initially by hazard to equipment, personnel, and the environment. The equipment hazards have been further divided by sub-teams.

Payload Risk Assessment

The payload will not be permanently fixed in the launch vehicle and will require multiple components to ensure proper deployment. These hazards contain the testing, assembly, and flight concerns associated with the payload. The hazards are listed in Table 77.

Vehicle Assembly Risk Assessment

The hazards outlined in Table 78 are risks that could potentially be encountered throughout the assembly phase and during launch preparations.

Propulsion Risk Assessment

The hazards outlined in are risks associated with stability and propulsion. The team has one member with certifications supporting that he can safely handle motors and design stable rockets the size of the competition rocket. Other members are currently working toward certifications as well. These assessments are listed in Table 79.

Recovery Risk Assessment

The hazards outlined in Table 80 are risks associated with the recovery. Since there are two recovery systems onboard, many of the failure modes and results will apply to all the systems but will be stated only once for conciseness.

VDS Actuation Risk Assessment

The hazards outlined in this section discuss the risks associated with testing and flight of the VDS. The VDS interfaces with the main structure of the vehicle with potential risk from manufacturing, assembly, and installment. The VDS hazards are outlined in Table 81.

Personnel Safety Risk Assessment

Construction and manufacturing of parts for the rocket will be performed in on-campus labs, the Engineering Garage and Sackett Lab. The hazards assessed in associated with machinery, tools, and chemicals in the lab are assessed in Table 82.

Environmental Hazards to Launch Vehicle Risk Assessment

The hazards outlined in Table 83 are risks from the environment that could affect the rocket or a component of the rocket. These hazards are the exception for needing to achieve a low risk level. This is because several of these hazards are out of the team's control, such as the weather. In the case that environmental hazards present themselves on launch day, putting the team at a moderate risk, the launch will be delayed until a low risk level can be achieved. The hazards that the team can control will be mitigated to attain a low risk level.

Launch Vehicle Hazards to Environment Risk Assessment

The hazards outlined in Table 84 are risks that construction, testing or launching of the rocket can pose to the environment.

5. Design of Payload Equipment

5.1.1. Overview

The deployable rover challenge was selected for this year's experimental payload. An orientation correction system utilizing custom made bearings will ensure proper orientation of the rover at landing. The rover will be remotely deployed after landing by a team member after RSO permission to proceed has been granted. The rover will be made to resemble a military tank with a tread style drive system capable of driving the rover a minimum of five feet from the launch vehicle. A custom designed foldable solar array will be autonomously deployed to conclude the payload's primary mission. The final design of the payload in its fully stowed, flight ready configuration and fully deployed, mission complete configuration is shown below in Figure 79.

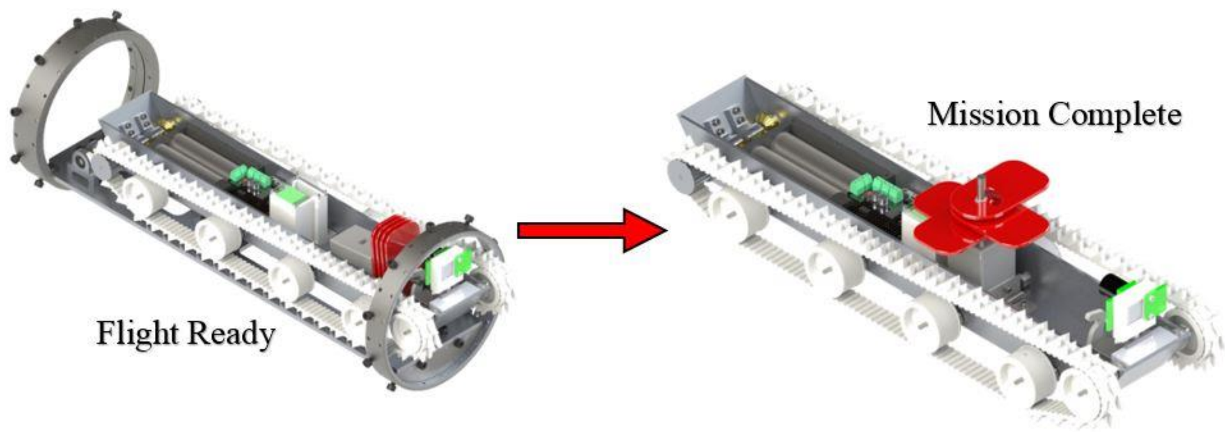


Figure 79: Final Payload design stowed and deployed.

5.1.2. System Overview

The payload has been divided into nine subsystems for division of labor to ensure that the payload be able to successfully achieve all requirements set forth in the NASA Statement of Work and all team derived requirements. These subsystems are listed and briefly described below in Table 40.

Payload Subsystem	Subsystem Overview
Rover Orientation Correction Systems (ROCS)	The ROCS will be responsible for ensuring upright orientation upon landing prior to deployment of the rover.
Rover Locking Mechanism (RLM)	The RLM will be responsible for retaining the rover in the launch vehicle for the entirety of the flight and releasing the rover at the time of deployment.
Deployment Trigger System (DTS)	The DTS will be responsible for sending the deployment signal to the rover upon receiving approval to proceed from the RSO.
Rover Body Structure (RBS)	The RBS will be responsible for providing structural support for the rover and assist in retaining the rover in the launch vehicle during flight.

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

Table 40: Payload subsystems.

5.1.3. Dimensional Overview

The Rover Orientation Correction System, Rover Locking Mechanism, and Deployment Trigger System will remain fixed in the launch vehicle airframe after deployment of the rover and as such, dimensions of the ROCS and RLM will be discussed separate from the other subsystems which will be referred to as the rover for simplicity. The final dimensions and weights of the payload are listed below in Table 41.

ROCS/RLM	
Dimension	**Value**
Diameter	6.000 in.
Length	17.9 x 6.000 in.
Weight	4.78 lbs.
Rover	
Dimension	**Value**
Stowed Length x Width	16.82 x 4.73 x 3.73 in.
Deployed Length x Width	16.82 x 4.73 x 4.05 in.
Weight	3.51 lbs.
Payload	
Total Weight	8.29 lbs.

Table 41: Final dimensions and weights.

5.1.4. Mission Overview

The mission of the payload is outlined below in Table 42.

Mission Step	Mission Step Description
1	The payload will be integrated into the launch vehicle ready for flight.
2	The launch vehicle will carry the payload to the intended apogee and begin recovery events.

3	Upon landing, the ROCS will allow the rover to rotate independently from the airframe allowing it to settle with the rover upright inside the airframe.
4	A unique deployment signal will be sent to the rover after gaining RSO permission to proceed.
5	The rover will perform an orientation check using two high precision gyroscopes.
6	The RLM will release the rover allowing it to begin advancing forward and exit the airframe.
7	The rover will begin to autonomously drive forward to reach a linear distance of at least five feet from the launch vehicle.
8	While driving, the OAS will detect insurmountable objects in the direct path of the rover and indicate to the CES that the rover should be turned to avoid the obstacle.
9	After reaching at least five feet from the launch vehicle, the rover will stop moving and deploy the SAS concluding the payload's primary mission.
10	The rover will use the energy harvested by the solar array to trigger the SIS to begin taking pictures of the rover and its surroundings.
11	Pictures will continue to be taken until the payload is retrieved and powered down.

Table 42: Payload mission overview.

5.1.5. Rover Orientation Correction System (ROCS)

5.1.6. Subsystem Overview

The Rover Orientation Correction System (ROCS) is responsible for ensuring that the rover is upright prior to deployment regardless of the recovery orientation of the payload bay of the launch vehicle. The ROCS has been broken into three main subassemblies that provide a means of integration with the airframe of the launch vehicle and supporting the rover during flight. The three subassemblies and their function are described below in Table 43.

ROCS Subassembly	Subassembly Function
Aft-End Thrust Bearing (AETB)	To absorb critical forces (without yielding) applied during motor burn, airframe separation, airframe landing, allow free rotation of the payload along its central axis, and support AFT end of BSS
Fwd-End Support Bearing (FESB)	To support FWD end of BSS and allow free rotation of the payload along its central axis
Bridging Sled System (BSS)	To support payload for entirety of vehicle flight, to bridge AETB and FESB, and to prevent rover translation along radial axis of airframe

Table 43: ROCS subassemblies and function.

The ROCS is shown fully assembled below in Figure 80.

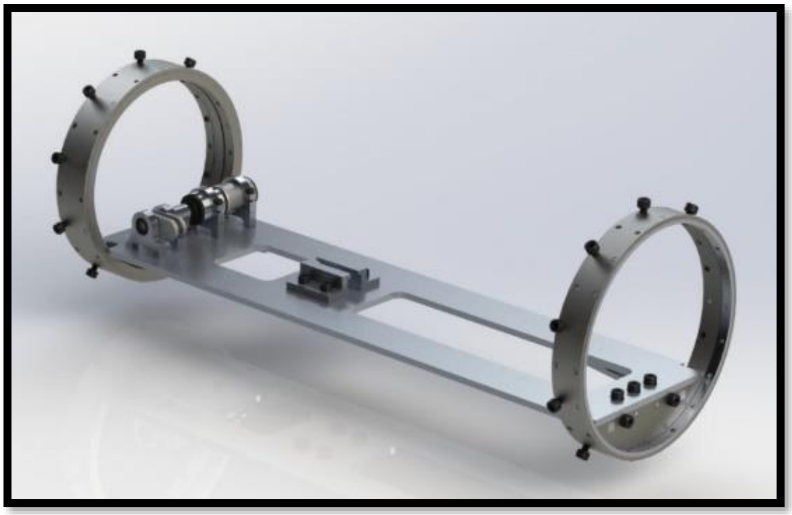
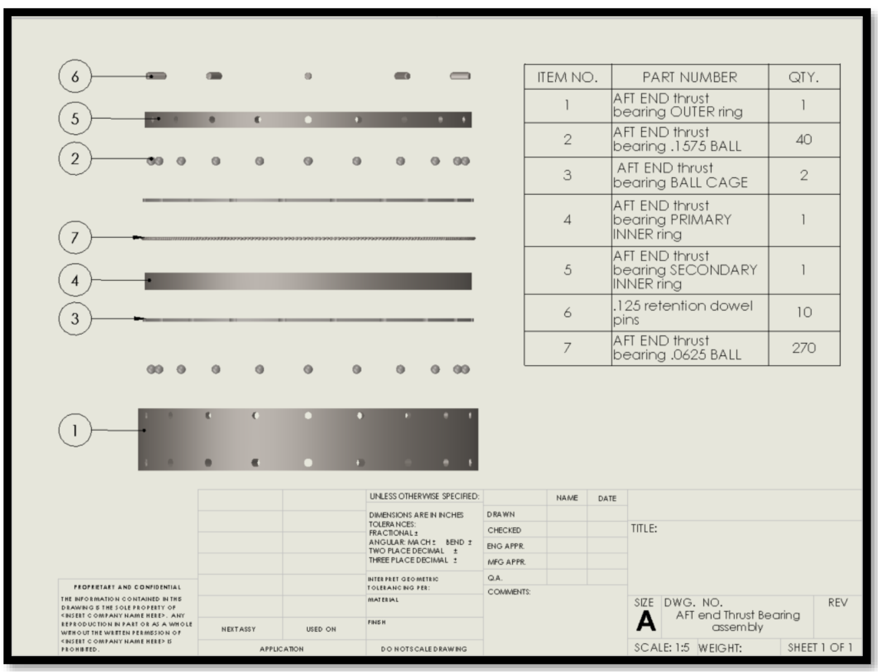


Figure 80: Fully assembled ROCS.

5.1.7. AFT End Thrust Bearing (AETB)

The primary purpose of the AETB is to support and allow free rotation of the payload, while remaining structurally sound under applied loads experienced during vehicle flight. A bill of materials for the AETB is shown below in Figure 81.



ITEM NO.	PART NUMBER	QTY.
1	AFT END thrust bearing OUTER ring	1
2	AFT END thrust bearing .1575 BALL	40
3	AFT END thrust bearing BALL CAGE	2
4	AFT END thrust bearing PRIMARY INNER ring	1
5	AFT END thrust bearing SECONDARY INNER ring	1
6	.125 retention dowel pins	10
7	AFT END thrust bearing .0625 BALL	270

Figure 81: AETB bill of materials.

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

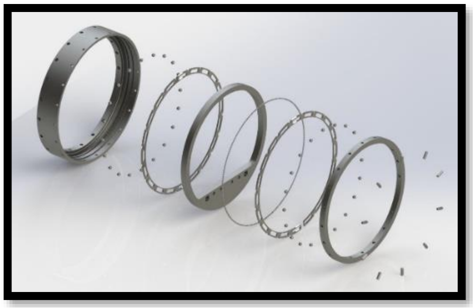


Figure 82: Exploded view of AETB.

Figure 83: Section view of AETB.

5.1.7.1. AFT End Thrust Bearing Design Notes: Material selection, Analysis, Integration

5.1.7.1.1. Material selection

Each item of the assembly is listed below with a material that the item will be made out of and a justification for that material in Table 44.

Item(s)	Material	Justification of Material
Outer Ring, Primary Inner Ring, Secondary Inner Ring	D2 tool steel	D2 tool steel is a high carbon, high chromium tool steel with great wear resistant properties and a yield strength of 55,000 psi in its annealed state. D2 tool steel is primarily used in Rolls, Forming, Dies, and Punches. Typical bearing material is AISI 52100 Chromium Steel and is fully hardened. Due to machine tool and cutting/abrasive tool accessibility limitations, a steel with great wear resistance, sufficient mechanical properties in its annealed state (20-26 HRC), and used in applications similar to bearing contacts, was required. D2 Tool steel was selected to meet wear, yield strength, and application requirements. Another noteworthy characteristic that influenced the selection of D2 tool steel is its ability to remain dimensionally stabile when machined/ground to thin wall thicknesses.
0.1575 in. Ball Bearing	Si_3N_4 Silicon Nitride	Initial selected material was AISI 52100 Chromium Steel but was later replaced by Si_3N_4 Silicon Nitride. Factors for choosing ceramic ball bearings over chrome steel ball bearings include superior surface finish (Grade 5 vs Grade 25), weight (.0038 oz. vs .0091 oz.), and machined surface finish of bearing raceways. The bearing raceways in the AETB are machine finished and Scotch-Brite™ polished as opposed to the superior ground finish in commercial bearings. To partially mitigate friction caused by machining imperfections present on the raceway surface, the ball bearing with the best surface finish were selected; Si_3N_4 Silicon

		Nitride ball bearings. The compressive strength of the Si3N4 Silicon Nitride ball bearings is 435 ksi.
0.0625 in. Ball Bearings	AISI 52100 Chromium Steel	AISI 52100 Chromium Steel ball bearings were selected because of their low cost, light weight, exceptional surface finish, and high yield strength of 295 ksi.
Ball Bearing Retention Ring	316 Stainless Steel	Stainless steel was selected for its ability to retain its flatness when water jetted at a thickness of .060 in., ensuring it fits in the grooves machined into the AETB outer ring.
0.1250 Dowel Pin	Fully Hardened Alloy Steel	Fully hardened alloy steel dowel pins were selected due to their light weight and high single shear strength of 1,845 lbf.

Table 44: AETB material justifications.

5.1.7.1.2. Analysis

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

During liftoff, the L2200 solid rocket motor accelerates the vehicle at approximately 457.24 $\frac{ft}{s^2}$ and during recovery opening, a maximum acceleration of 664 $\frac{ft}{s^2}$ is predicted. A conservative combined weight estimate of eight pounds for the rover and BSS and maximum acceleration of 644 $\frac{ft}{s^2}$ was used to calculate the maximum force of 160 lbf applied to the AETB. During motor burn and recovery opening, the AETB primary inner ring raceway will accelerate into the 0.15748 in. ball bearings and outer ring raceway. The surface contact between the ball bearings and raceways of the AETB primary inner ring and outer ring will be under Hertzian contact stress.

The maximum Hertzian contact stress between one ball bearing and one raceway wall was calculated using

$$p_{max} = \frac{3F}{2\pi a^2} \tag{39}$$

Where the F is the applied load and a is the contact radius between the two surfaces. The contact radius a was derived using

(40)

$$a = \sqrt[3]{\frac{3F}{8} * \frac{\frac{(1-v_1^2)}{E_1} + \frac{(1-v_2^2)}{E_2}}{\frac{1}{d_1} + \frac{1}{d_2}}}$$

Where v is the Poisson's ratio of material n, E is the elastic modulus of material n, and d is the diameter of object n. A maximum Hertzian contact stress of 41.9 ksi was calculated using the data from Table 45 below provided by material supplier.

Object 1: 0.1575 Si3N4 Silicon Nitride ball bearing	
Poisson's ratio	.260
Elastic modulus	43500 ksi
Diameter	.1575 in.
Object 2: D2 Tool Steel raceway	
Poisson's ratio	.285
Elastic modulus	30000 ksi
Diameter	.1585 in.

Table 45: Properties of Si3N4 Silicon Nitride and D2 Tool Steel

The maximum Hertzian contact stress of 41.9 ksi applies to the full load being passed through only one ball bearing and raceway (worst case scenario). In practice, the bulk of the force will be transferred through seven 4mm ball bearings and two raceways, while the remaining thirteen ball bearings absorb the remainder of the force. Computer simulation methods will continue to be researched to gain accurate results.

10-24 socket head cap screws with a rated yield strength of 2,835 lbs. and single shear strength of 3060 lbs. will be used to fasten the AETB to the airframe. The maximum estimated force of 160 lbf. is well under the yield strength of one 10-24 SHCS. Ten 10-24 SHCS will be used to securely fasten the AETB to the airframe and to ensure a minimum of two 10-24 SHCS will remain engaged with the primary inner ring's crescent shaped mounting area whether the payload is stationary or rotating.

5.1.7.1.3. Integration

The outer ring will be placed on a surface plate to begin AETB integration. The first components to be placed into the outer ring will be the .0625 in. ball bearings. A low viscous die grease will be used to retain the .0625 in. ball bearings in outer ring middle groove during integration. Once all the .0625 in. ball bearings are put in place, one ball bearing retaining ring will be placed in the groove closest to the outer ring raceway. Once the ball bearing retaining ring is firmly in place, one .1575 in. ball bearing will be placed into each one of the twenty, ball bearing retaining ring slots. Next, insert the primary inner ring with the ground face up (raceway down). With the primary inner ring inserted, place a separate ball bearing retaining cage into the final internal outer ring groove. Proceed to place one .1575 in. ball bearing into each one of the twenty-ball bearing retaining ring slots. Next place the secondary inner ring into the outer ring with the ground side face up (raceway down). Finally, proceed to pin the secondary inner ring to the outer ring with the .1250 dowel pins. Pin ten equally spaced holes to complete the AETB integration.

5.1.8. FWD End Support Bearing (FESB)

The primary purpose of the FESB is to assist the AETB by supporting and allowing free rotation of the payload while providing clearance for the rover to exit the launch vehicle. A bill of materials for the support bearing is shown below in Figure 84.

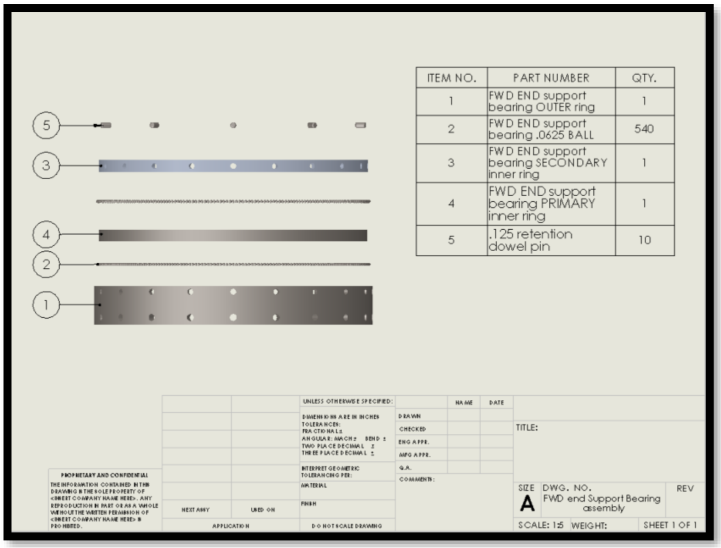


Figure 84: FESB bill of materials.

An exploded view and section view, shown in Figure 85 and Figure 86 respectively, of the FESB final design are shown below to display the individual components and method in which the parts will be integrated with one another.

Figure 85: FESB exploded view.

Figure 86: FESB section view.

5.1.8.1. FWD End Support Bearing Design Notes: Material selection, Analysis, Integration

5.1.8.1.1. Material selection

Each item of the assembly is listed below with a material that the item will be made out of and a justification for that material in Table 46.

Item(s)	Material	Justification of Material

Outer Ring, Primary Inner Ring	D2 Tool Steel	D2 tool steel is a high carbon, high chromium tool steel with great wear resistant properties and a yield strength of 55,000 psi in its annealed state. D2 tool steel is primarily used in Rolls, Forming, Dies, and Punches. Typical bearing material is AISI 52100 Chromium Steel and is fully hardened. Due to machine tool and cutting/abrasive tool accessibility limitations, a steel with great wear resistance, sufficient mechanical properties in its annealed state (20-26 HRC), and used in applications similar to bearing contacts, was required. D2 Tool steel was selected to meet wear, yield strength, and application requirements. Another noteworthy characteristic that influenced the selection of D2 tool steel is its ability to remain dimensionally stabile when machined/ground to thin wall thicknesses.
0.0625 in. Ball Bearing	AISI 52100 Chromium Steel	AISI 52100 Chromium Steel ball bearings were selected because of their low cost, light weight, exceptional surface finish, and high yield strength of 295 ksi.
0.1250 Dowel Pin	Fully Hardened Alloy Steel	Fully hardened alloy steel dowel pins were selected due to their light weight and high single shear strength of 1,845 lbf.
Secondary Inner Ring	6061-T6 Aluminum	6061-T6 Aluminum was chosen for the secondary inner ring for its light weight yet sufficiently high yield strength of 40,000 psi.

Table 46: FESB material justifications.

5.1.8.1.2. Analysis

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

During flight and under the forces induced by liftoff and recovery opening force, the FESB's clearances mitigate much of the mechanical impact that the AETB experiences. The FESB also reduces the deflection of the bridging sled by adding support on the forward end of the sled.

The FESB will absorb no critical forces and only serves as a support to the AETB and BSS.

10-24 socket head cap screws with a rated yield strength of 2,835 lbs. and single shear strength of 3060 lbs. will be used to fasten the FESB to the airframe. The maximum estimated force of 160 lbf. is well under the yield strength of one 10-24 SHCS. Ten 10-24 SHCS will be used to securely fasten the AETB to the airframe and to ensure a minimum of two 10-24 SHCS will remain engaged with the primary inner ring's crescent shaped mounting area whether the payload is stationary or rotating.

5.1.8.1.3. Integration

The outer ring will be placed on a surface plate to begin FESB integration. The first components to be placed into the outer ring will be the .0625 in. ball bearings. A low viscous die grease will be used to retain the .0625 in. ball bearings in outer ring grooves during integration. Perform this step to both internal grooves located in outer ring. Once all .0625 in. ball bearings are in place, insert primary inner ring ground face up. Next insert secondary inner ring and proceed to pin in place. Pin ten equally spaced holes to complete the FESB integration.

5.1.9. Bridging Sled System (BSS)

The primary purpose of the Bridging Sled System (BSS) is to provide a stable surface for the rover to mount to. The BSS also mechanically prevents the rover from translating perpendicular to the central axis of the payload bay. A bill of materials for the Bridging Sled is shown below in Figure 87.

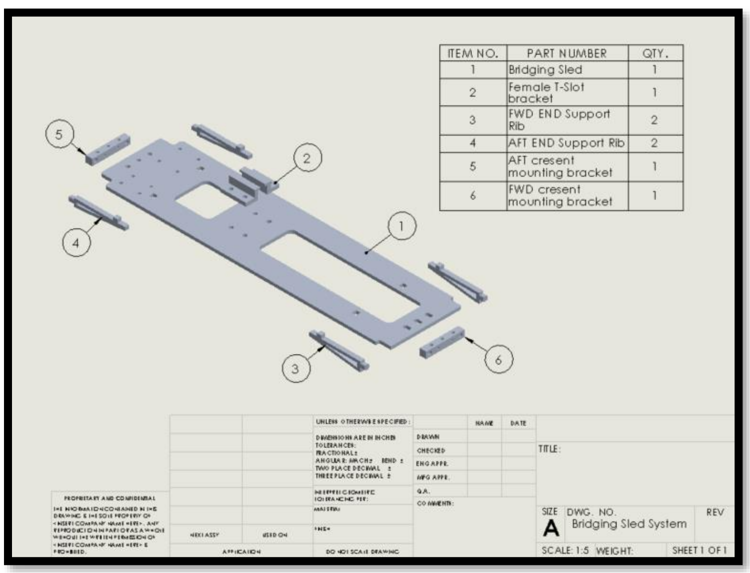


Figure 87: BSS bill of materials.

An exploded view and full assembly model of the BSS final design are shown in Figure 88 and Figure 89 respectively to display the individual components and the method in which the parts will be integrated with one another.

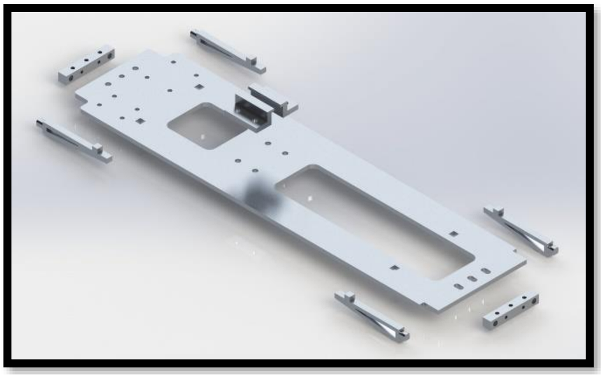


Figure 88: Exploded view of BSS.

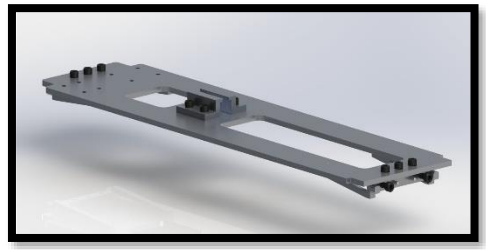


Figure 89: Fully assembled view of BSS.

5.1.9.1. Bridging Sled System Design Notes: Material selection, Analysis, Integration

5.1.9.1.1. Material Selection

Each item of the assembly is listed below with a material that the item will be made out of and a justification for that material in Table 47.

Item(s)	Material	Justification of Material
All BSS components	6061-T6 Aluminum	6061-T6 Aluminum was chosen for all BSS components for its light weight yet sufficiently high yield strength of 40,000 psi.

Table 47: BSS material justification.

5.1.9.1.2. Analysis

BSS analysis will be discussed under RLM Analysis and FEA of BSS and RLM, as simulation results are dependent on each other.

5.1.9.1.3. Integration

The Bridging Sled serves as the chassis of the BSS. The female T-slot bracket will be fastened to the bridging sled via three 10-24 SHCS. Next the FWD and AFT support ribs will be press fit into their respective mounting locations on the FWD and AFT end of the bridging sled. The FWD and AFT end crescent mounting brackets will be attached during full ROCS integration.

5.1.9.2. Support Ribs

The BSS will provide structural support for the rover and provide a rigid surface for the rover's tracks to transfer power to. During vehicle flight, the payload bay will be subjected to forces from multiple directions. While the AETB is designed to absorb the majority of the forces propagating through the central axis of the payload bay, the BSS and RLM is also impacted by the same forces.

To reduce static deflection and deflection caused by forces perpendicular to the central axis of the payload bay, FWD and AFT end support ribs will be used. The support ribs are situated directly under the point of contact between the rover and bridging sled and extend inward towards the center of the bridging sled. In doing so, the distance between the center of gravity of the rover and its nearest "rigid" support is minimized. By reducing the distance between the center of gravity of the rover and its nearest rigid support, single point bending is reduced. An aft end support rib is shown below in Figure 90.

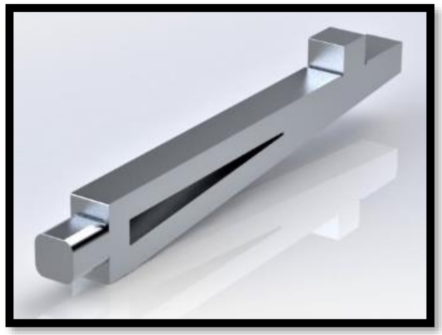


Figure 90: AFT End Support Rib.

5.1.9.3. T-Slot

The female T-slot brackets situated on the bridging sled will work in conjunction with the male T-slot nut fastened to the chassis of the rover. When the male T-slot nut and female T-slot are engaged, translation of the rover in the radial axis direction will be minimized. A front view of the final design of the male T-slot nut engaged with the female T-slot is shown below in Figure 91.

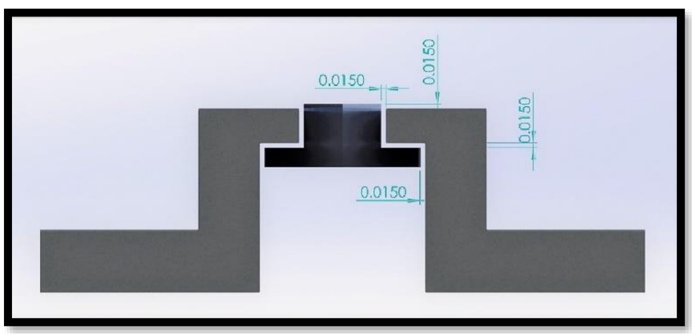


Figure 91: Male T- slot nut and Female T-slot engagement.

The T-slot assembly will aid the RLM by providing a point of contact close to the payloads forward end, where the translation and rotation of the payload along the radial axis is restricted. The T-slot assembly's 0.015 in. clearance will allow the rover to translate along the central axis of the payload once the RLM is disengaged.

5.1.9.4. Crescent Mounting Bracket

The FWD and AFT end Crescent Mounting Brackets will connect the BSS to the AETB and FESB's Primary Inner Ring. Three 10-24 SHCS will be used to mount the BSS onto the mounting bracket from the top and two 10-24 SHCS aligned by two 0.1250 in. Grade 8 Alloy Steel dowel pins will be used to fasten the bracket to the primary inner rings. Each 10-24 socket head cap has a rated yield strength of 2,835 lbs., single shear strength of 3,060 lbs, and a tensile strength of 3,150 lbs. Figure 92 below illustrates the mounting method.

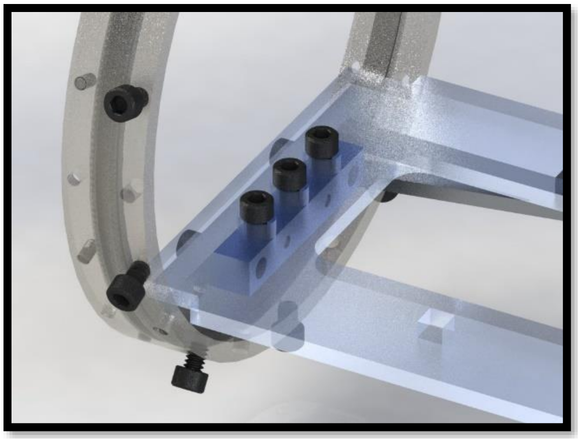
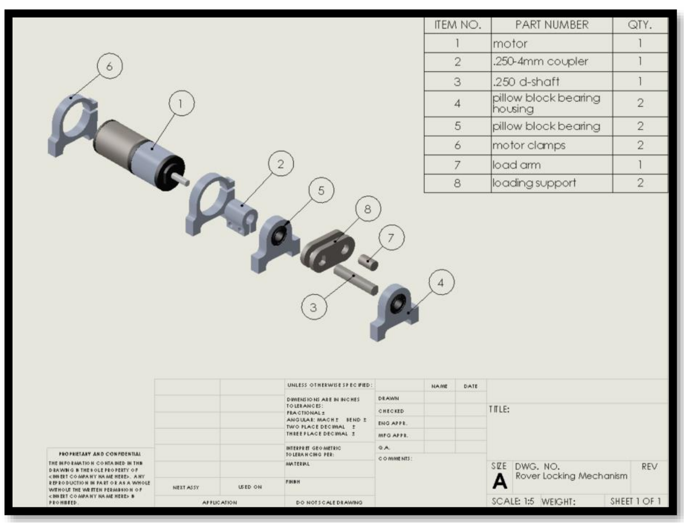


Figure 92: Crescent Mounting Bracket (dark blue) fully fastened and pinned.

5.1.10. Rover Locking Mechanism (RLM)

The primary purpose of the Rover Locking Mechanism (RLM) is to mitigate the translational movement of the rover along the central axis of the payload bay and to absorb critical forces applied, while remaining structurally sound. A bill of materials for the RLM is shown below in Figure 93.



ITEM NO.	PART NUMBER	QTY.
1	motor	1
2	.250-4mm coupler	1
3	.250 d-shaft	1
4	pillow block bearing housing	2
5	pillow block bearing	2
6	motor clamps	2
7	load arm	1
8	loading support	2

Figure 93: RLM Bill of Materials.

The full RLM assembly is displayed in Figure 94 and an exploded view of the RLM in Figure 95.

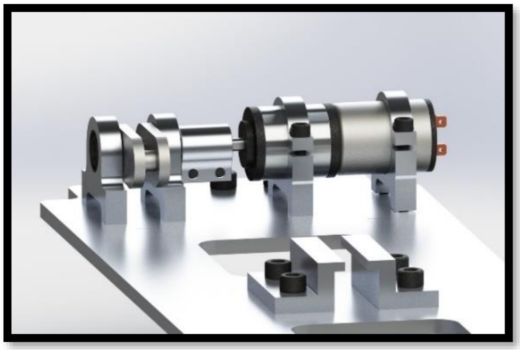


Figure 94: RLM assembly fastened to BSS

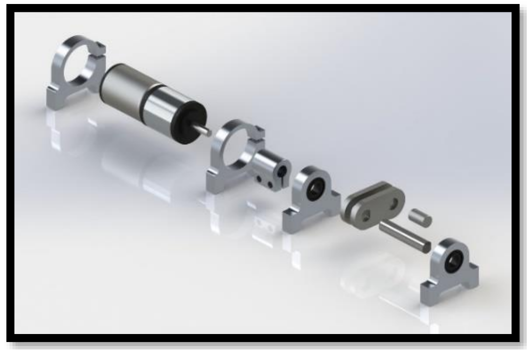


Figure 95: Exploded view of RLM

5.1.11. Rover Locking Mechanism Design Notes: Material/Component Selection, Analysis, Integration

5.1.11.1. Material/Component selection

Each item of the assembly is listed below with a material that the item will be made out of and a justification for that material in Table 48.

Item(s)	Material	Justification of Material
26 RPM Planetary Gear Motor	Various materials	The 26 RPM Planetary Gear Motor was chosen for its light weight and exceptional stall torque of 36.4 lb-in.
Shaft Coupler	Aluminum	The 0.15748 in. to 0.25 in. coupler was selected as a light weight method to couple the 0.25 in. D-shaft to the 4mm motor D-shaft.
Motor Clamps	Aluminum	The motor clamp was selected as a light weight, high strength method for securely retaining the 26 RPM planetary gear motor in place and fastening the motor to the BSS.
Loading Brackets	D2 Tool Steel	D2 tool steel is a high carbon, high chromium tool steel with great wear resistant properties and a yield strength of 55,000 psi in its annealed state (20-26 HRC). D2 Tool steel was selected to meet wear, yield strength, and application requirements. Another noteworthy characteristic that influenced the selection of D2 tool steel is its ability to remain dimensionally stabile when machined/ground to thin wall thicknesses.
Loading Arm	Grade 8 Alloy Steel	A .2500 through hardened (body: HRC 47-58; surface: HRC 60) Alloy Steel dowel pin was selected as the 160lbf main load bearing component. The Grade 8 Alloy Steel dowel pin has a double shear strength of 14720 lbs and a single shear strength of 7360 lbs.
0.25 in. D-shaft	304 Stainless Steel	The ¼ in. Stainless Steel D-shaft was selected as a light weight, small footprint, high strength method of driving the loading/locking bracket and arm. The 304 Stainless Steel material has a yield strength of 40,000 psi.

Pillow Block and Bearing	Aluminum and AISI 52100 Chrome	The pillow block and bearing assembly were selected as a light weight, small footprint, high strength method of reducing the torque on the 4mm motor D-shaft and handling the dynamic load that will be transferred to the ¼ in. D-shaft.

Table 48: RLM material justifications.

The pillow block and pillow block bearing manufacturer specs are listed below in Table 49.

Pillow Block and Pillow Block Bearing	
Product weight	0.48 oz
Material	Aluminum
Bearing ID	0.250 in
Bearing Material	AISI 52100 Chrome Steel
Static Load	84 lbs
Dynamic Load	186 lbs
Max RPM	50,000

Table 49: Pillow Block assembly manufacturer specs.

5.1.11.2. Analysis and FEA of BSS and RLM

Analysis of the BSS and RLM was performed using SolidWorks Simulation. These simulations are shown together as they are dependent on each other in flight configuration.

5.1.11.2.1. Loading Bracket and Load Arm

A static simulation was performed with bonded contact sets and a mesh of .05 inches. A two-component load totaling 160 lbf was applied to the RLM and BSS assembly. During motor burn, as the vehicle translates forward, the rover has the potential to pivot at the load arm. Although the T-slot mechanism restricts the vertical travel to .015 inches, a two-component load is created. Figure 96 below illustrates the angle of 0.96 degrees formed when the rover pivots at the load arm, and the male T-nut bottoms out against the female T-slot.

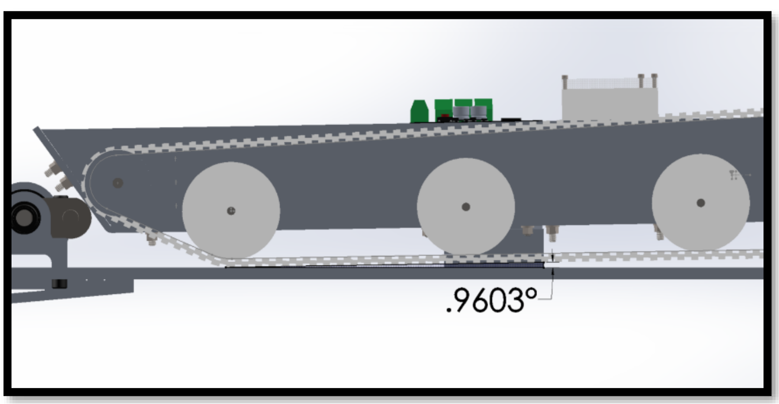


Figure 96: Angle methodology results.

The force component along the central axis of the payload bay of 159.96 lbf was applied to the RLM load arm and the force component normal to the bridging sled of 2.79 lbs was applied to the top of the male T-nut, forcing contact between the T-slot male and female components. The central axis force component of 159.96 lbf was calculated using

$$F_{central\ axis} = (160)lbf * \cos(1) \tag{41}$$

And the normal component of 2.79 lbf was calculated using

$$F_{normal} = (160)lbf * \sin(1) \tag{42}$$

Figure 97 below graphically displays the loading on the Load arm for context.

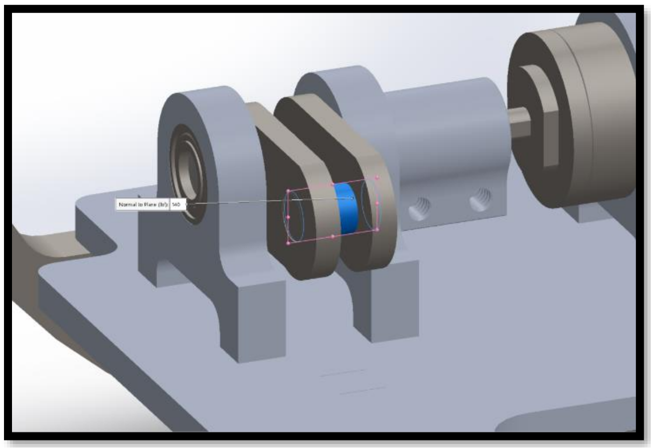


Figure 97: Graphical depiction of component $F_{central\ axis}$ applied to Load Arm

Approximate results were hand derived to gauge the FEA simulation results accuracy. The load arm, under double shear, has a strength of 14,750 lbs was not considered to be a failure point and was not considered. A bearing stress approximation method was used to predict the stress on the loading brackets contact area with the load arm. A rectangular contact area where the height is the diameter of the load arm (.250 inches) and the width is the thickness of the load bracket wall (.150 inches) and a load of 80 lbf (160*.5; double shear) was used to calculate the stress. Using

$$\sigma = \frac{F}{A} \tag{43}$$

where σ is the resultant stress, F is the applied force, and A is the area where the force is applied over, a stress of 2133 psi was calculated. Using Solidworks simulation, results display a stress range between 3000-4000 psi in the contact area between the load bracket and load arm. Figure 98 and Figure 99 below displays the results.

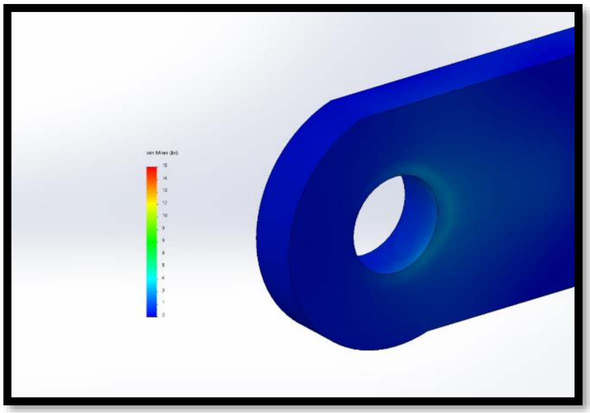


Figure 98: Loading Bracket-Load Arm FEA results

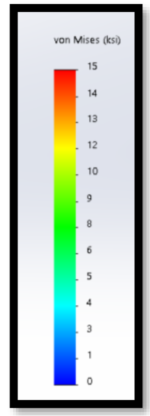


Figure 99: Enlarged stress plot

A hand calculation using the same method to used approximate the stress between the contact load arm and load bracket was used to approximate the stress in the 0.25 in. D-shaft and load bracket contact area. A same load of 80 lbf was used and the wall thickness of 0.150 inches did not change. However, the height of the rectangular area changed from 0.250 in. to 0.1517 in. to account for the D-shaft pad. The stress approximation resulted at 3515 psi. Solidworks simulation, results display a stress range between 3000-5500 psi in the contact area between the load bracket and ¼ inch D-shaft with a max stress concentration of 15,000 psi occurring at the filleted ends of the D-pad contact area in the loading bracket The max stress concentration in the D-pad area of the ¼ in.D-shaft is 13,000 psi. Figure 100 and Figure 101 displays the results.

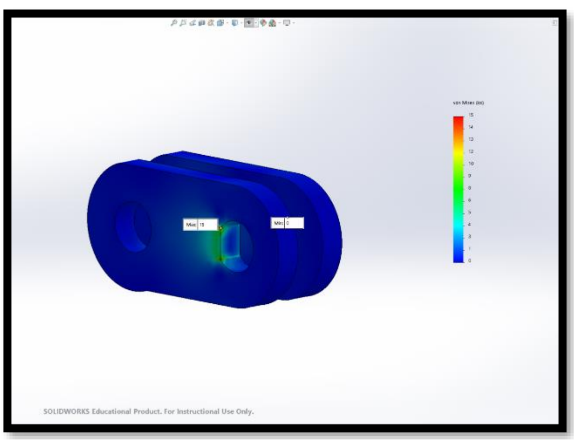


Figure 100: Loading Bracket-1/4 in. D-shaft FEA results

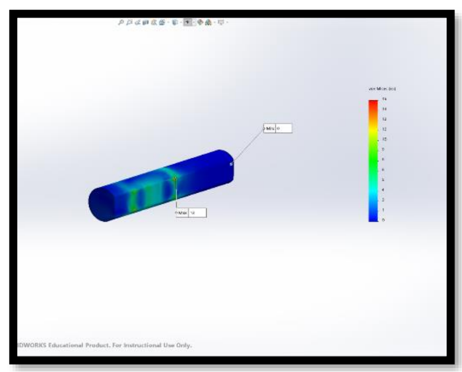


Figure 101: 1/4 in. D-shaft- Loading Bracket FEA results

The D2 tool steel the loading brackets are made of possess a yield strength of 55,000 psi. Using the maximum Von Mises stress of 15,000 psi, the factor of safety for the loading bracket is 3.67. It is possible the 15,000 psi stress concentration at the filleted ends of the D-pad area is a convergence error and the max Von Mises stress is 5,500 psi. 5,500 psi is close to the hand calculated result of 3515 psi. If so, the factor of safety is 10.

5.1.11.2.2. D-Shaft

The 304 Stainless Steel the ¼ in. D-shaft is made of possess a yield strength of 40,000 psi. Using the maximum Von Mises stress of 13,000 psi, the factor of safety for the ¼ in. D-shaft is 3.08. It is possible the 13,000 psi stress concentration at the filleted ends of the D-pad area is a convergence error and the max Von Mises stress is 5,500 psi. 5,500 psi is close to the hand calculated result of 3,515 psi. If so, the factor of safety is 7.27.

5.1.11.2.3. Pillow Blocks

The pillow blocks that support the ¼ in. D-shaft each have a manufactured dynamic load strength of 186 lbs each. With a maximum load of 160 lbf being applied to the RLM for four seconds, the two pillow block configuration has a factor of safety of 2.33.

5.1.11.2.4. 26 RPM Planetary Gear Motor

The 26 RPM Planetary Gear Motor has a stall torque of 36.4 lb-in. The distance of 0.600 inches between the center of the 4mm D-shaft and the center of the Load Arm was used as the moment arm to approximate the minimum torque required to unlock the rover. Using a conservative weight estimate of 7 lbs for the rover, the minimum calculated torque required to unlock the rover once the unlock signal is received is 4.2 lb-in. The approximate factor of safety for the unlocking torque requirement is 8.67. Testing will have to be performed on the RLM motor to verify the motor can overcome frictional forces in the RLM assembly and lock/unlock the rover.

5.1.11.2.5. BSS RLM Interface

The RLM is fastened to the BSS using eight 6-32 SHCS with a rated tensile strength of 1,640 lbs, yield strength of 1,470 lbs, and single shear strength of 1615 lbs. Each motor clamp and pillow block will be fastened through the bottom of the Bridging Sled using two 6-32 SHCS each. Under critical loading, the 6-32 SHCS will be under shear stress and transfer the load to the BSS via fastener and thread contact. Figure 102 below displays a top view of the resultant stress on the BSS and Figure 103 displays the bottom view.

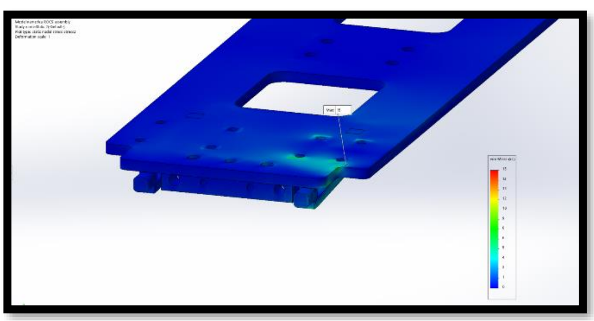


Figure 102: Top view of BSS under 160 lbf load

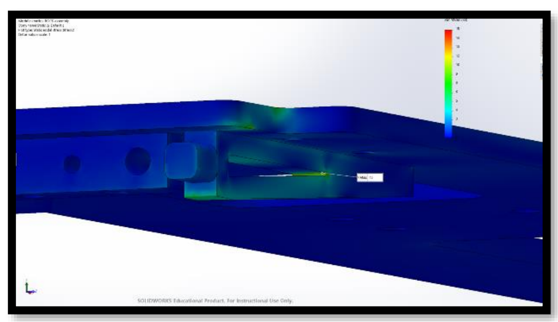


Figure 103: Bottom view of BSS under 160 lbf load

The majority of the stress concentrates over the area occupied by the pillow blocks. The stress in this area of Bridging Sled ranges from 3,000 psi to a max of 10,000 psi located on the outer edge of Bridging Sled. Figure 104 displays the simulation results.

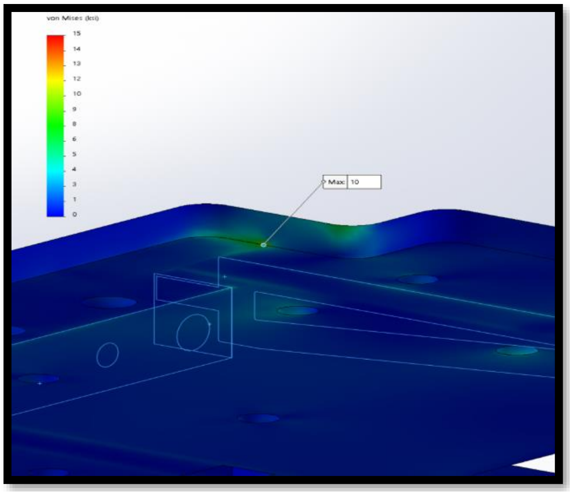


Figure 104: Bridging Sled FEA simulation results

The Bridging Sled is made from 6061-T6 Aluminum with a yield strength of approximately 40,000 psi. With a maximum stress of 10,000 psi, the Bridging Sled has a factor of safety of 4.

5.1.11.2.6. AFT End Support Ribs

Under load, a max stress concentration of 15,000 psi formed on the AFT end support rib located directly under the Load Arm. Aside from the max stress concentration, stresses in the AFT end Support Rib range from 3,000-9,000 psi. Figure 105 below displays the simulation results.

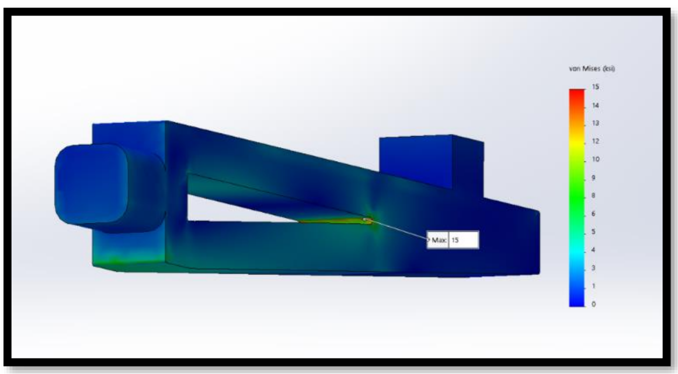


Figure 105: AFT end Support Rib FEA simulation results

The AFT end Support Rib is made from 6061-T6 Aluminum with a yield strength of approximately 40,000 psi. With a maximum stress of 15,000 psi, the AFT end Support Rib has a factor of safety of 2.67.

5.1.11.2.7. AFT Crescent Mounting Bracket

The AFT Crescent Mounting Bracket used to fasten the BSS to the AETB developed a max stress concentration of 5,000 psi in the upper most corner, closest to the load arm. Figure 106 below displays the simulation results.

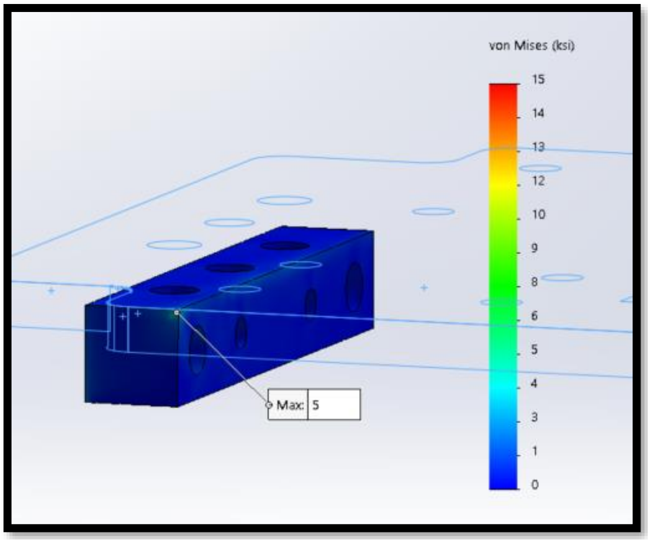


Figure 106: AFT Crescent Mounting Bracket FEA simulation results

The AFT Crescent Mounting Bracket is made from 6061-T6 Aluminum with a yield strength of approximately 40,000 psi. With a maximum stress of 5,000 psi, the AFT Crescent Mounting Bracket has a factor of safety of 8.0.

5.1.11.3. Integration

The motor will be placed inside both motor clamps and tightened via the 10-24 SHCS located on the top of the clamps. Next the 4mm ID side of the coupler will be fitted and tightened onto the 4mm D-shaft on motor. The 0.25 in. D-shaft will then be inserted into the 0.25 in. ID side of the coupler and tightened via the 6-32 SHCS on the coupler. Next, the 0.25 in. D-shaft will be passed through one pillow block assembly. Pin the Load Arm to the Load Brackets and slide the loading assembly onto the 0.25 in. D-shaft. Next the final pillow block assembly will be placed onto the 0.25 in. D-shaft. The final step in the RLM is to fasten the RLM onto the AFT end of the BSS sled via eight 6-32 SHCS through the bottom of the sled. This completes the RLM integration.

5.1.12. Final ROCS Integration

Once the AETB, FESB, BSS, and RLM have been assembled, final integration can proceed. The AFT and FWD Crescent Mounting Brackets will be doweled (two 0.1250 dowel pins) and fastened (two 10-24 SHCS) in place, onto the inner face of their respective primary inner rings. Once complete, the AFT end of the BSS will be fastened onto the AFT Crescent Mounting Bracket via three 10-24 SHCS. The same process will be repeated to fasten the FWD end of the BSS onto the FWD Crescent Mounting Bracket and complete full ROCS integration.

5.1.13. Deployment Trigger System (DTS)

The Deployment Trigger System (DTS) will consist of a receiver module, receiver antenna, transmitter module, and Yagi antenna that will be responsible for allowing the team to deploy the rover after gaining RSO permission to proceed. The receiver module will reside in the payload recovery bay coupler. Wires will pass through the bulk plate into the payload bay using a slip rig flange and connect, via a magnetic pull-apart mechanism, into a software serial port running on the CES control board described in section 5.1.44.1.

5.1.14. Receiver Module

The module that will be used for the DTS is the HC-12 transceiver module. This half-duplex wireless communication module has the ability to communicate on 100 channels in the 433.4-473.0 MHz range. In this range of frequencies 433.4-470 fall under the Unlicensed ISM/SRD bands (FCC Part 15.231; 15.205). The rest of the range 470-473 MHz is allowed for intermittent control signals per (FCC Part 15.231). Therefore, operating the HC-12 will not require any licensing from the FCC verifying requirement DTS-1. The module can operate in full functionality between 3.2V and 5.5V and as such will be powered by the 3.7V controller batter described in section 5.1.44.4. The HC-12 module connects to an external antenna through an IPEX MHFI connector and to the rover control system through three pins for Rx, Tx, and module command. Once operational and connected to the receiving antenna the HC-12 module will allow for the Control Electronics System to receive the deployment signal. The HC-12 is shown below in Figure 107.

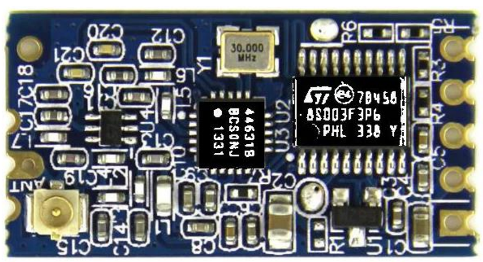


Figure 107: HC-12 transceiver module.

5.1.14.1. Mounting

The HC-12 module will be secured within the payload recovery bay. A 3D printed sled, connected to the coupler via two threaded rods, will secure the module. There will be enough room within the pocket so that the module will be able to connect to the antenna described in 0 via an SMA cable. The overall dimensions of the sled are 4.394 x 0.394 x 1.771 in. A rendering of the module and its sled is shown below in Figure 108.

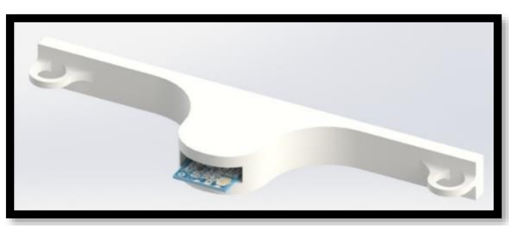


Figure 108: Receiver Module Sled

5.1.14.2. Slip Ring Flange

The five wires that will connect the receiver to the CES control board need to pass through a recovery bulk. The wires on the receiver side of the bulkplate will remain stationary while the wires on the rover side of the bulkplate need to be able to spin freely due to the ROCS. This requires the use of a slip ring flange that will pass the wires through the bulkplate and allow the rover side wires to rotate freely while maintaining connection to the receiver side. The flange is shown below in Figure 109.

Figure 109: Slip ring flange.

5.1.15. Pull-apart Mechanism

Upon receiving the deployment signal and successful orientation check (described in section 5.1.46.1), the rover will begin to drive forward and exit the launch vehicle. The wires connecting the receiver module to the control board will disconnect as they are not needed after the signal is received and would otherwise hinder forward motion of the rover. The pull-apart mechanism will be responsible for ensuring ease of detachment of these wires. This mechanism is shown below in Figure 110. Each of the HC-12 modules five pins will be connected through the pull-apart mechanism.

Figure 110: Magnetic connector.

The leading alternative at PDR of an audio jack was determined to be too resistive to pulling apart for the rover to overcome and as such, the design was changed to use magnetic connectors.

5.1.16. Receiving Antenna

The DTS receiver antenna will be secured to the exterior of the payload bay. The carbon fiber airframe would otherwise restrict the reception of the deployment signal. Mud-flap antennas can receive at 473 MHz, which falls in the operating range of the HC-12 module. This flexible type of antenna is ideal for securing the it to the curvature of the airframe. A simulation model of the antenna and the antenna secured to the airframe with tape are shown below in Figure 111 and Figure 112.

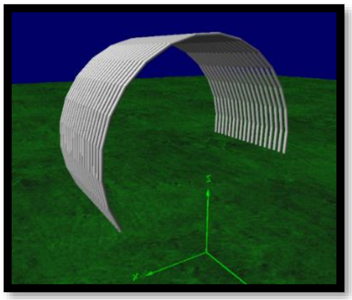


Figure 111: Mud-flap antenna model.

Figure 112: Receiver antenna.

Mud-flap antennas are comprised of multiple sections of receiving wiring in parallel across the entire flap. Each wire connects back to a central point where an outside wire can be added for connecting to F and SMA coaxial cable types. Along with the ability to receive at 473 MHz, the mud-flap antenna can operate on the power supplied by the HC-12 module through the SMA coaxial connection. By testing the mud-flap antenna around a carbon fiber tube that matched the intended rocket body diameter, it was found to have a reliable receiving signal at 3,200 ft. away from the DTS transmitter station. The receiver antenna will be wrapped around the entire circumference of the airframe to achieve reception regardless of the landing orientation of the payload bay.

5.1.17. DTS Transmitter Station

The DTS Transmitter Station will be used to send a deployment signal to the receiver after receiving RSO permission to deploy. The station includes a Yagi high-gain antenna that has the ability to transmit at 473 MHz, thus allowing the receiver and transmitter station to communicate on the same frequency channel. The Yagi is attached to the top of a pole and connected to the transmitter module by a F type coaxial cable. With attaching the Yagi antenna to the top of a pole, line-of-sight can be achieved ensuring good communication. The Transmitter Station is shown below in Figure 113.

Figure 113: DTS Transmitter Station.

5.1.18. Rover Body Structure (RBS)

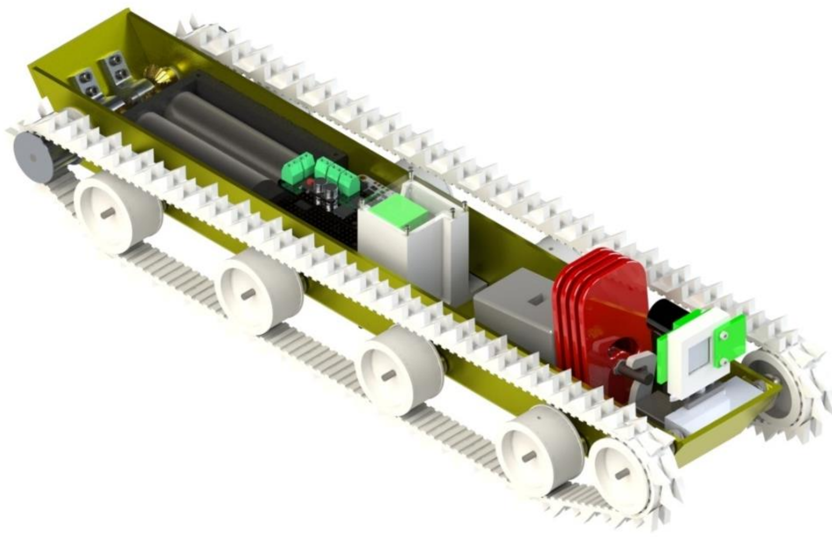


Figure 114: RBS Highlighted in Yellow.

The RBS will be responsible for providing support for all the components in the rover including the Rover Drive System. It will also serve as the electronics bay of the rover. Two slots at the front of the RBS walls will allow for tensioning of the two drive belts. The RBS will be sloped in the front to ensure that the rover will not contact obstacles. The rear of the RBS will also be sloped to ensure that, while the rover is overcoming obstacles, the rear of the rover will not contact the ground. The RBS is shown below in Figure 115.

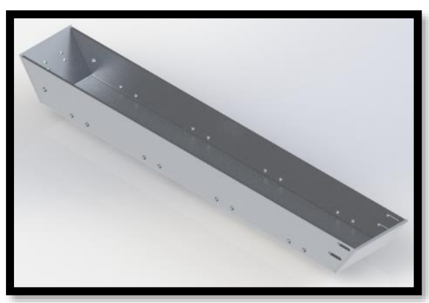


Figure 115: RBS rendering.

5.1.19. Aluminum Rover Body

The rover body will be constructed of a single sheet of 1/10" thick aluminum that is water jetted and bent using a press brake. The sides will then be welded together for strength. Alternative methods of construction would require a jig to be built to align pieces normal to each other and could lead to imprecision. The aluminum sheet metal before forming is shown below in Figure 116.

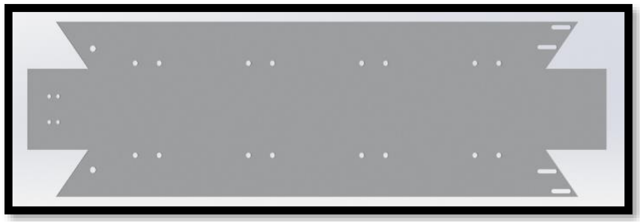


Figure 116: Pre-Bent Aluminum Sheet.

5.1.19.1. Justification of Material

Aluminum was chosen to act as the body structure material of the rover due to its high strength, affordability, accessibility, and ease of integration of other subsystems. This is also ideal as the rover can be constructed of a single sheet of aluminum water jet for high precision and bent to form the internal cavity of the rover.

5.1.20. Electronics Mounts

The electronics controlling the rover will be secured to the RBS by directly fastening them to the body with screws or custom designed mounts. The Control Electronics System printed circuit board has been design to be screwed directly to the RBS with mounting holes through the PCB.

5.1.20.1. Motor Battery and Controller Battery Mount

The battery mount will be 3D printed and will secure the motor battery and the controller battery described in section 5.1.46.3 and 5.1.44.4 respectively. The battery mount will be attached to the RBS by four 4-40 screws. The lid of the battery mount will be secured to the top by five 0-80 screws. A rendering of this mount is shown below in Figure 117.

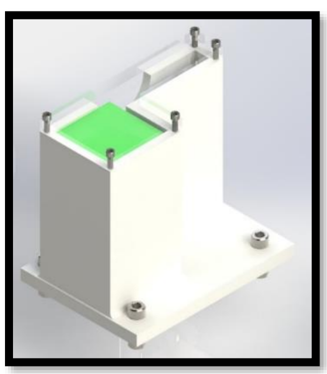


Figure 117: Battery mount.

5.1.21. Rover Drive System (RDS)

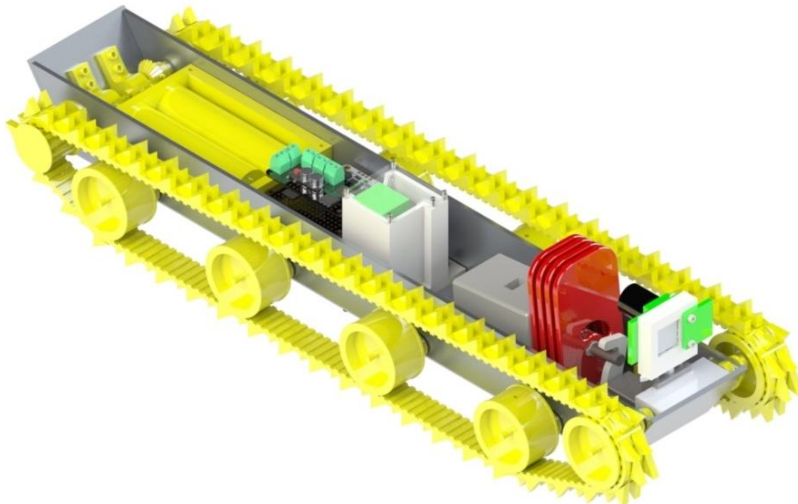
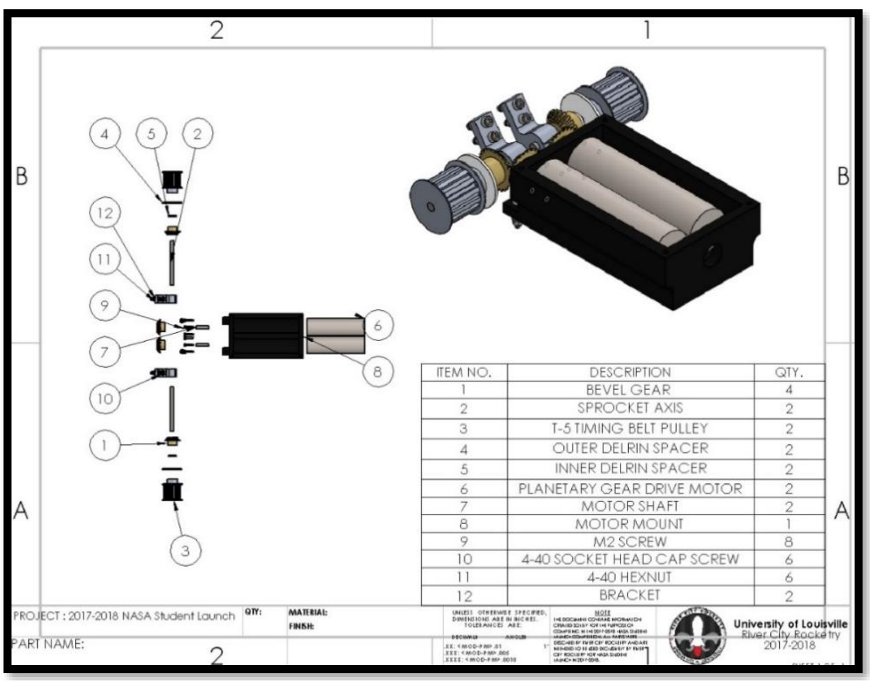


Figure 118: RDS Highlighted in Yellow.

The primary purpose of the RDS is to translate the rover at least five feet from the launch vehicle on any terrain that may be encountered and surmount small obstacle in the rover's path. A bill of materials for the RDS is shown below in Figure 119.



ITEM NO.	DESCRIPTION	QTY.
1	BEVEL GEAR	4
2	SPROCKET AXIS	2
3	T-5 TIMING BELT PULLEY	2
4	OUTER DELRIN SPACER	2
5	INNER DELRIN SPACER	2
6	PLANETARY GEAR DRIVE MOTOR	2
7	MOTOR SHAFT	2
8	MOTOR MOUNT	1
9	M2 SCREW	8
10	4-40 SOCKET HEAD CAP SCREW	6
11	4-40 HEXNUT	6
12	BRACKET	2

Figure 119: Rover Drive System Components.

5.1.22. Belt Drive System

A belt driven system was chosen for the drive train of the rover. This system will utilize pulleys and rubber belts to drive the rover. A belt driven system will provide high traction due to the high contact area of the treads. A belt with a small width will be obtained to maximize clearance between the ROCS and the rover. The decrease in width allows for increased height as the restricting geometry of the ROCS is a circle. This will allow for easy integration into the ROCS. Another advantage of the belt drive system is that it minimizes the amount of moving parts in the system which makes it more simplistic and reliable.

5.1.23. Rover Track Design

The belts that will drive the rover will be T5 timing belts. Table 50 shows the specifications of this timing belt.

Specification	Value
Depth	0.25 in.
Width	0.63 in.
Material	Polyurethane
Coefficient of Friction (on dry concrete)	1.0

Table 50: Timing Belt Specifications.

This belt was chosen because of its small width and customizability. Using a belt with teeth minimizes the possibility of the belt slipping. Having large teeth on the outside of the belt was chosen to improve traction on all terrains and provide a larger clearance from the ground. Rubber was chosen for its high coefficient of friction on most surfaces. The timing belt is shown below in Figure 120.

Figure 120: Rover Track Design.

5.1.24. Pulley Configuration

Pulleys will support the T5 timing belts configured in a manner to maximize the climbing ability of the rover. Each configuration will be composed of six pulleys. The forward most pulley will be elevated to produce a slope that will improve the rover’s ability to overcome obstacles. The rear pulley will be the driven pulley. The length of the timing belt in this configuration is 33.27 in. An illustration of the pully configuration is shown in Figure 121.

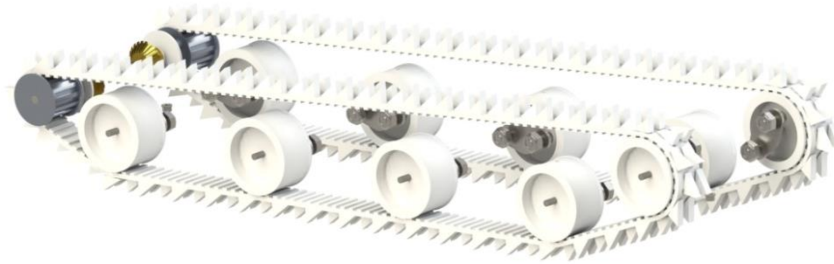


Figure 121: Pulley configuration.

5.1.24.1. Drive Pulley Design

The drive pulleys will be responsible for driving the timing belt. The drive pulleys that were chosen, have a 0.197 in. pitch to match the timing belts. These pulleys are made of machined aluminum and will be fastened to the drive axel via 8-32 set screws. An illustration of the drive pully is shown in Figure 122.

Figure 122: Drive pulley design.

5.1.24.2. Passive Pulley Design and Material Justification

The passive pulleys will be responsible for keeping the timing belts aligned during operation. These pulleys will be machined out of Delrin using a CNC lathe. Delrin was chosen for its toughness and low weight. Other materials that were considered were PLA plastic and aluminum. PLA plastic is weaker than Delrin with inadequate weight saving to justify using PLA. Aluminum would add a great deal of unnecessary weight to the system. These pulleys will be mounted with 4-40 set screws to 1/8 in. shafts that will connect to bearings. An illustration of the passive pully as designed and as manufactured is shown in Figure 123.

Figure 123: Passive Pulley Design.

5.1.24.3. Pulley Bearings

The bearings will be stainless steel two-bolt flange-mounted ball bearings. These bearings will be mounted to the outside of the rover body and will be responsible for allowing the passive pulleys to rotate freely. The shaft connecting the pulleys to the bearings is a 0.125 in. diameter, 0.75 in. long steel shaft and will be press fit to the bearings. The bearings have a dynamic radial load capacity of 125 lbs. An illustration of the bearings is shown in Figure 124.

Figure 124: Pulley Bearing.

5.1.25. Main Drive Motors

The two main drive motors will serve as the source of propulsion for the rover. The motors will be mounted in the rear of the rover to drive the center of gravity of the rover towards the rear. For the purposes of this mission, maintaining forward motion is more valuable than the speed of the rover and therefor a motor with low RPM and high torque should be chosen. The main drive motors have been selected from a line of premium planetary gear motors. The motor is shown below in Figure 125 followed by specifications in Table 51.

Figure 125: Actobotics planetary gear motor.

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

Table 51: Main drive motor specifications.

5.1.26. Bevel Gears

The primary function of the bevel gears is to transfer the rotational motion of the drive motors 90 degrees to the shaft of the drive pulleys. This function will allow for mounting the drive motors parallel to the drive pulley and for a more compact design of the overall rover. The bevel gear configuration is shown in Figure 126.

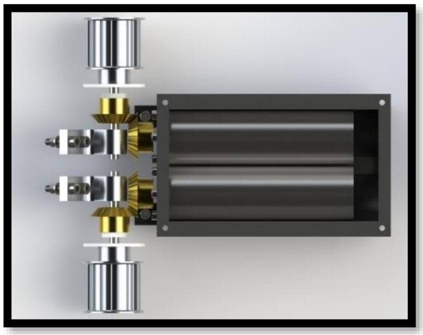


Figure 126: Bevel gear configuration.

5.1.27. Motor Mount

The motor mount will be responsible for retaining the motors during flight and while driving the rover. The motor mount will be 3D printed out of PLA plastic and mounted to the bottom of the RBS by two 4-40 bolts. Each motor will be mounted to the front of the motor mount by four M2 screws. The motor mount is shown as designed and as manufactured below in Figure 127 and Figure 128.

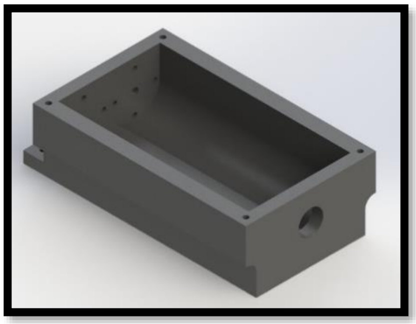


Figure 127: Motor Mount as Designed.

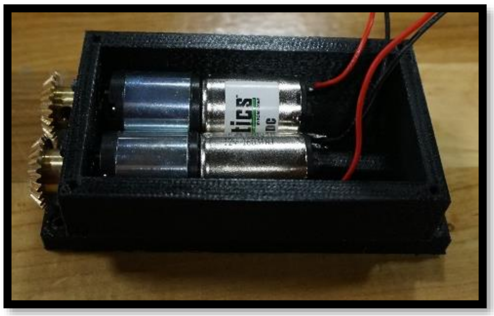


Figure 128: Motor Mount as Made.

5.1.28. Rover Drive System Bracket

The RDS brackets will be responsible for supporting both drive pulley axels. These brackets will be custom machined out of aluminum to ensure minimal wear and high rigidity. The brackets will be mounted to the rear of the rover body via two 4-40 bolts. The bracket design is shown below in Figure 129.

Figure 129: RDS Bracket.

5.1.29. Tipping Analysis

The Control Electronics System will use the angle at which the rover will tip on the roll axis to determine the safe to deploy angles for the orientation check described in section 5.1.46.1. The following equations were used to find the tipping angle.

$$\tan\theta = (Y/X) \tag{44}$$

where θ is the angle between the ground and the line connecting the center of gravity (CG) to the point of tipping, Y is the distance from the ground to the CG, and X is the horizontal distance from the point of tipping to the CG. The maximum angle before tipping (α) is defined by

$$\alpha = 90 - \theta \tag{45}$$

Solidworks was used to calculate CG. An illustration of the variables is shown in Figure 130.

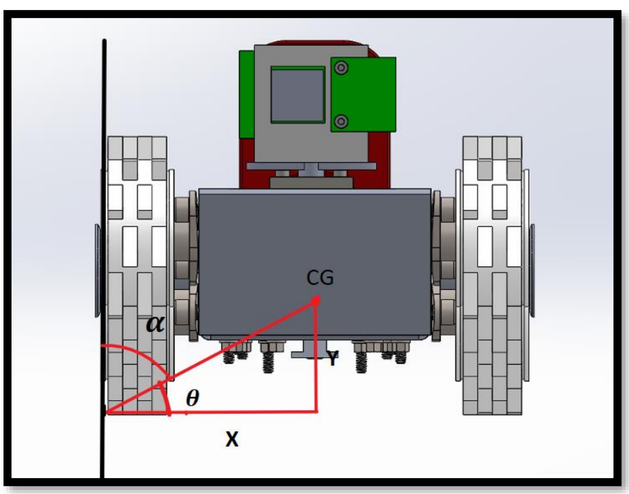


Figure 130: Tipping Analysis.

The result of this analysis is that an angle of 59.7° is the maximum angle of roll before the rover will tip.

5.1.30. Obstacle Avoidance System (OAS)

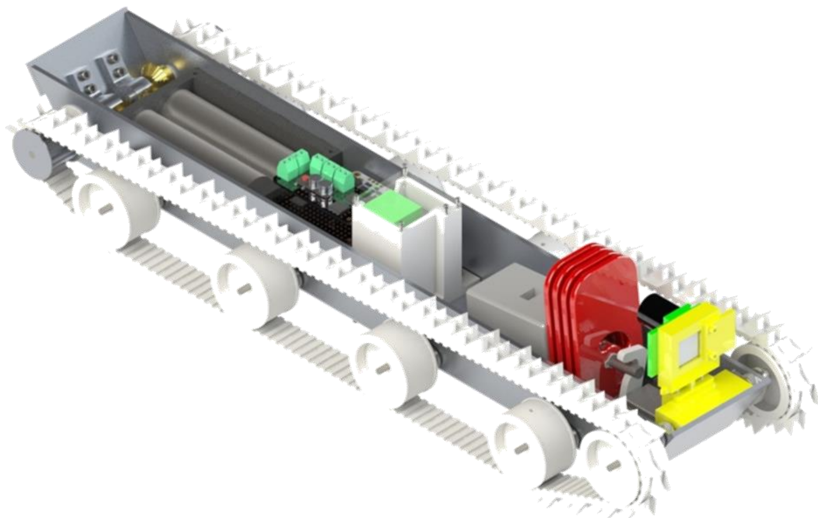


Figure 131: OAS Highlighted in Yellow.

The Obstacle Avoidance System will be responsible for determining if there is an object in the path of the rover that will stop the rover from reaching its destination of five feet away from the payload bay. This will be achieved by using a distance sensor on the front of the rover, sweeping the sensor from left to right giving it a field of view, and analyzing the data to determine if an obstacle taller than the rover is in the immediate path of the rover. If the data concludes that an obstacle is present, the Control Electronics System control scheme will determine a course of action as described in section 5.1.32. This section will outline the operation and interaction of devices for the OAS.

5.1.31. Lidar Sensor

Lidar sensors have very a narrow field of view, but high accuracy for short and long distances in ambient and low light. The data stream fed back from the sensor can be done at a very high rate and no conversions are necessary to determine distance. The VL53L0X has been selected as the distance sensor of the OAS due to its long range, adequate accuracy, and low cost for a significant increase in range. The accuracy of the sensor is documented to be 6% -- 9% outdoors at a range of 1.97 – 47.24 in. which will be verified through testing. This sensor is easily compatible with the CES control board with readily available libraries. The sensor is shown below in Figure 132.

Figure 132: VL53L0X lidar sensor.

5.1.32. Algorithm

The control scheme used by the CES is described in full in the Control Electronics System section 5.1.42. For convenience, the OAS phase of the control scheme is shown below in Figure 133.

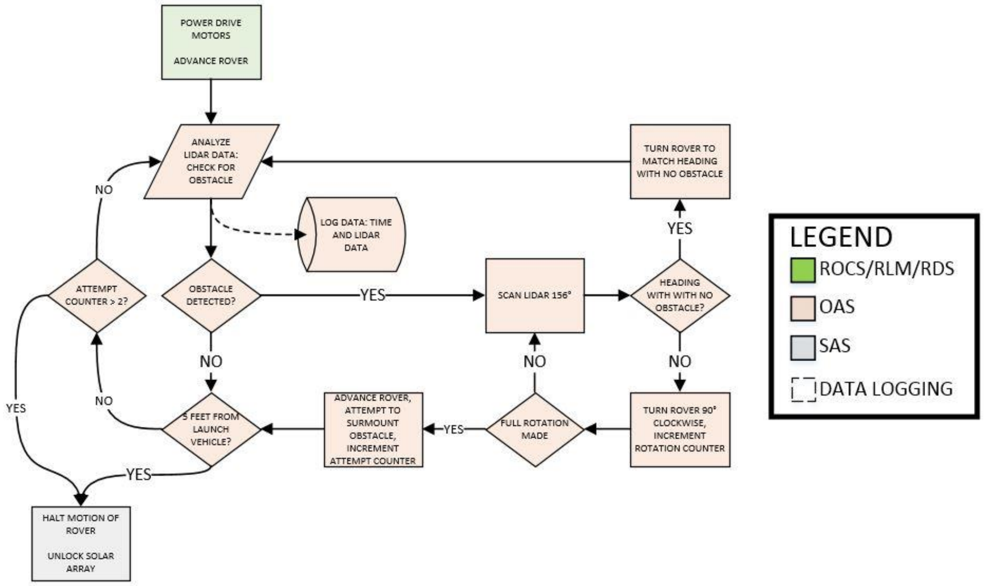


Figure 133: OAS phase of control scheme.

An object will affirm the “Obstacle Detected?” decision if the object is within 20 in. of the lidar sensor. This will halt the rover and begin the panning sequence. A distance value will be taken on each degree of a 156° sweep. A rolling average of 10 data points at a time will be used to mitigate possible outlier data points collected and determine a heading of least obstruction. The degree angle with the highest calculated average will be chosen as the heading and the rover will be turned to match this heading based on the onboard gyroscopes described in section 5.1.46.1.

5.7.3 Field of View

Due to the finely focused laser emitted by the lidar, a wider field of view is desired to maximize the data collected without needing to use the main drive motors to turn the rover for the lidar to see a different heading if an obstacle is detected. To achieve this field of view, the sensor will be mounted on top of a small servo motor that can be controlled by the CES. When the sensor detects an object in the immediate path of the rover, the small servo will rotate the sensor from left to right 156° to determine if a path with no obstacle exists within the sensors field of view. This angle has been chosen as the largest angle that the servo reliably can sweep with consistent rotation. The field of view to be achieved is displayed in blue below in Figure 134.

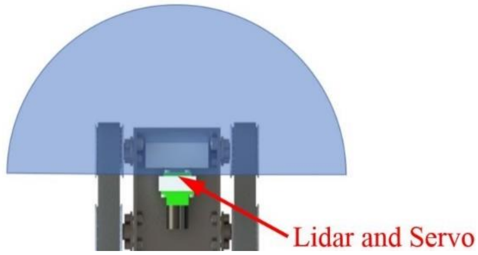


Figure 134: OAS field of view.

5.7.3.1 OAS servo

The OAS servo motor will be mounted to the front of the rover with sufficient clearance from other rover structures allowing it to rotate the sensor freely. TowerPro servos have both a small package and are easily integrated with the CES control board described in section 5.1.44.1 using readily available libraries. The SG92R has been selected as the OAS servo motor. This motor will provide sufficient torque and rotation while maintaining a low footprint onboard the rover and low cost. The motor is shown below in Figure 135.

Figure 135: SG92R servo motor.

5.7.4 Mounting

The SG92R servo motor will be secured to the front of the rover inside of a 3D printed mount. This 3D printed mount will provide a flat surface for the motor to be vertically flush against, as the front of the rover is slanted. Another custom designed mount will be attached to the shaft of the motor. The OAS lidar sensor will be secured to this mount by two M2 bolts. The overall dimensions of the lidar mount are 1.225 x 1.262 x 0.35 in. The design of the assembly of the lidar sensor (green), the electronics mount (white), and the servo (grey) is shown below in Figure 136.

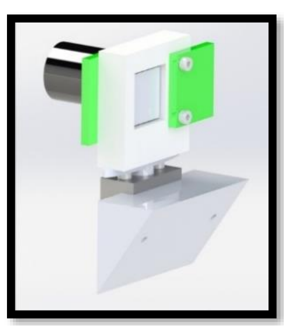


Figure 136: Lidar sensor mount assembly.

5.1.33. Solar Array System (SAS)

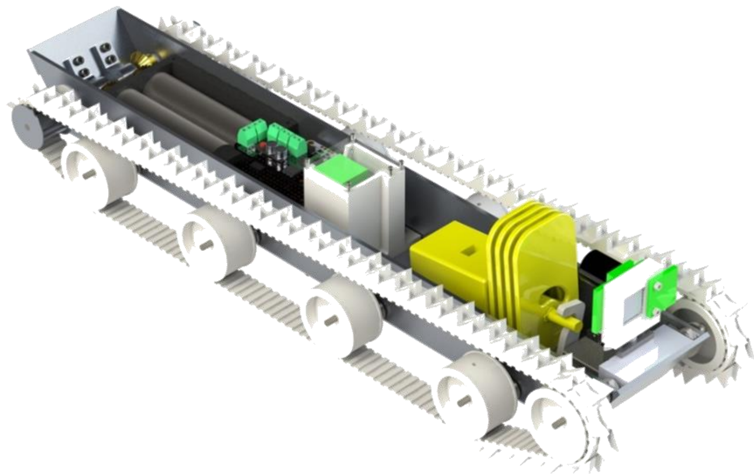


Figure 137: SAS Highlighted in Yellow.

The Solar Array System will be responsible for satisfying requirement 4.5.4 of the SOW. This system consists of four thin film solar panels mounted to support arms that are actuated by means of a tower assembly including a deployment motor, locking motor, and spring hinge. The operation and completed design of this system is discussed in this section. Statement_of_Work

5.1.34. Solar Panels

The solar cell panels will be responsible for harvesting solar energy and using the power generated as an input to the Control Electronics System. Thicker, rigid solar panels were not considered for this design of the SAS to save space, weight, and allow flexibility of the panels. The efficiency of the panels is also considered low priority as the panels will not be used to power any systems directly. For these reasons, the PowerFilm Solar MPT3.6-150 solar panels were chosen for their ultra-thin profile of 0.00787 in., large solar cell surface area of 17.11 in.^{2}, low weight of 0.0068 lbs., and high flexibility while maintaining a wattage of 360 mW. The MPT3.6-150 solar panels are shown below in Figure 138.

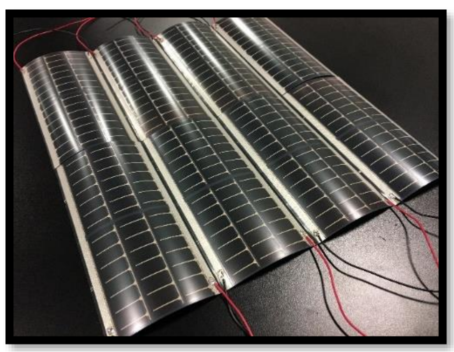


Figure 138: PowerFilm Solar MPT3.6-150 solar panels.

5.1.35. Deployment Motor

The deployment motor will be responsible for unfolding the solar array after the tower assembly, described below in section 0, actuates fully from its stowed position. At this time, the Control Electronics System will power a small motor inside the tower assembly housing that will unfold the solar panels from each other. The deployment motor is shown inside the tower assembly housing below in Figure 139.

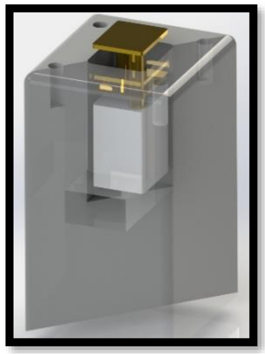


Figure 139: Deployment motor inside tower base.

A single shaft, high torque, minimum dimension motor is desirable for this design to ensure the motor will fit inside the tower base and provide sufficient torque to unfold the solar panels. This motor will be driven by the same 11.1V LiPo battery, described in section 5.1.46.3, that will power the main drive motors and as such, only nominal 12V motors were considered.

The Pololu Micro Metal Gearmotors line of motors provides small dimension motors ideal for this system. This system calls for a high torque, low RPM motor to unfold the panels without damaging them or any other nearby system. For this reason, the 32 RPM, 0.65 ft-lb Micro Metal Gearmotor was chosen for its small dimensions and ideal torque and RPM characteristics. The chosen motor is shown below in Figure 140.

Figure 140: Pololu 1000:1 Micro Metal Gearmotor.

5.1.35.1. Shaft Extension

The 0.354 in. long shaft of the motor must be extended to drive the uppermost panel support arm, described in 5.1.36.4. The 0.118 in. diameter shaft will be coupled to the 0.236 in. diameter D shaft of the deployment motor with a set screw shaft coupler, shown below in Figure 141. The larger diameter will give more contact area to distribute the panel support arm actuation load.

Figure 141: Set screw shaft coupler.

Acrylic washers of 0.1 in. thickness will be used as spacers and will be pressed onto the coupler once the extension shaft is installed to allow for secure component mounting. Acrylic was chosen due to its low friction and ability to cut the washers using a laser cutter for a high accuracy press fit. The spacers are shown below in Figure 142.

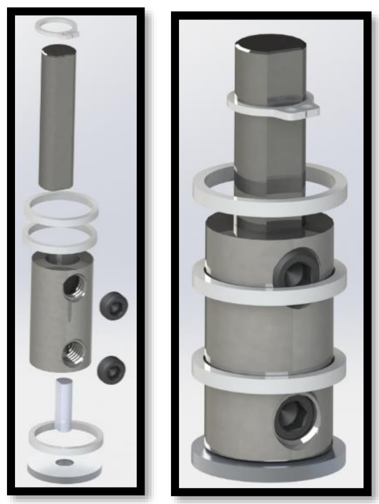


Figure 142: Motor shaft extension and spacers.

5.1.36. Tower Assembly

The tower assembly will rotate about a hinge and then deploy the solar array after the rover reaches its final destination. Array deployment requires the panels to be above the walls of the rover and clear of any other system for proper deployment. However, the diameter of the airframe would restrict the number of panels that can be deployed if the array were to remain fixed in the upright configuration. The actuating design allows for scalability to any number of panels. The assembly is shown in **Figure** 143.

Figure 143: Actuated and stowed solar tower assembly.

5.1.36.1. Tower Base

The tower base will be the deployment motor housing as well as the solar array mount. The deployment motor will fit into the slot in the top of the tower base and the motor power wires will exit the tower base through an aft-facing hole. The tower base is shown in Figure 144.

Figure 144: Tower base with wire chanel facing forward.

5.1.36.2. Spring Hinge

The spring hinge is an assembly of two hinge plates, a torsion spring, and a pin. The tower base will mount to the bottom of the rover with the spring hinge. This hinge will both raise the tower from its stowed flight position and keep the tower in the upright position. The assembly is shown below in Figure 145.

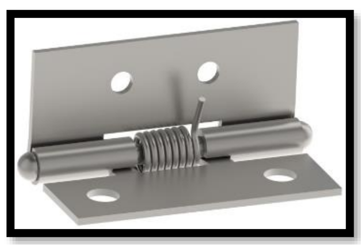


Figure 145: Spring hinge assembly.

5.1.36.3. Tower Locking Motor

The tower locking motor will be the same motor used for the deployment motor. The locking motor will be mounted to the rover using the custom designed mounting bracket shown below in Figure 146.

Figure 146: Locking motor mount bracket.

An L-shaped attachment will be mounted onto the locking motor's shaft with retaining rings. The attachment will hook around the end of the deployment motor shaft extension, locking the base in the stowed configuration until the rover has reached its final destination. The attachment will rotate away from the shaft extension when the solar array is ready to deploy, allowing the torsion spring will then actuate the tower into place.

5.1.36.4. Panel Support Arm

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

Each arm will consist of a central shaft hole, a peg slot, and a towing peg on one face. The peg slots will be oriented at different angels for each arm, as shown in Figure 147.

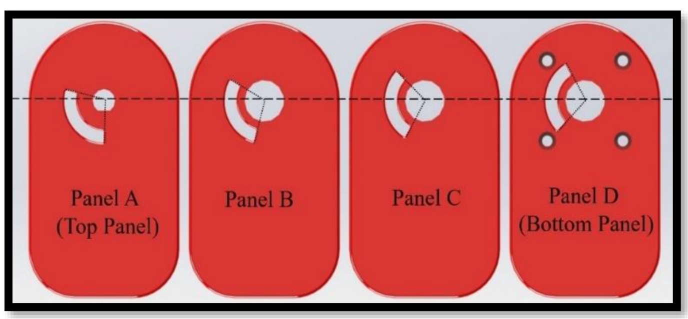


Figure 147: Panel Support Arm peg slots.

The arms will stack vertically and rest on the spacers that are pressed onto the shaft extension coupler. The top panel will be secured with a retaining ring. The arms will stow in the orientation shown in Figure 148. The peg locations were outlined to clarify their position and insertion depth.

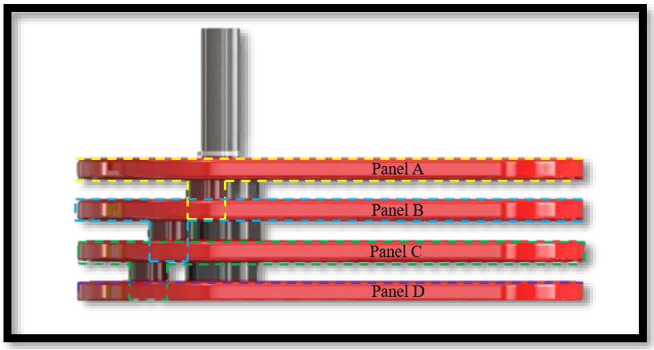


Figure 148: Panel support arms stacked and peg locations outlined.

The change in the peg slot angle will allow each towing peg to protrude into the peg slot of the support arm below it. A towing peg is shown in Figure 149.

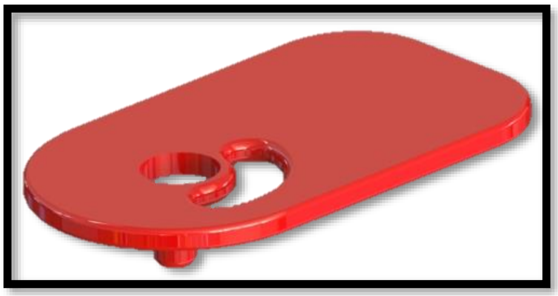


Figure 149: Isometric view of the arm and towing peg.

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

As the top panel, panel A, is driven by the motor, the towing peg will follow the peg slot of panel B until the peg contacts the end of the slot. After the towing peg of panel A hits the end of panel B's peg slot, panel A will cause panel B to rotate. When panel B's towing peg contacts the end of panel C's slot, it will cause panel C to rotate until it contacts the end of panel D's slot. AT this point, each arm will be 90° apart from each other exposing all 4 solar panels fully.

Figure 150 shows the order of arm actuation, beginning with the top driven panel, panel A. The arms have been lengthened to make the deployment order clear.

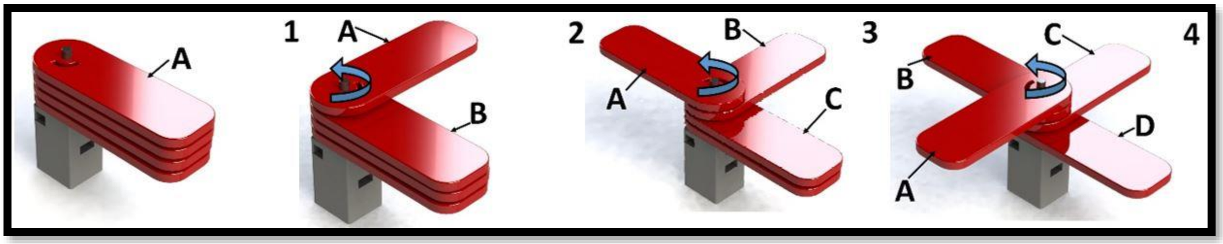


Figure 150: Panel support arm actuation order.

Figure 151 and Figure 152 show the towing peg locations of two adjacent panels before and after rotation.

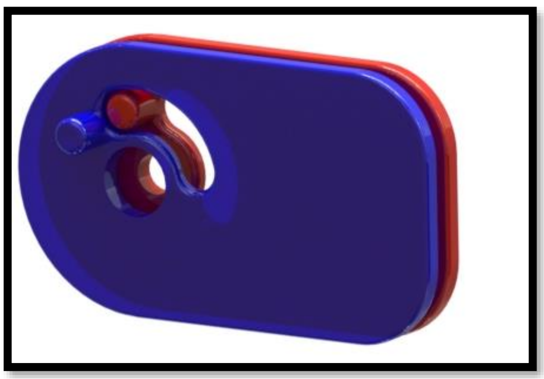


Figure 151: Two panel support arms in the stowed position.

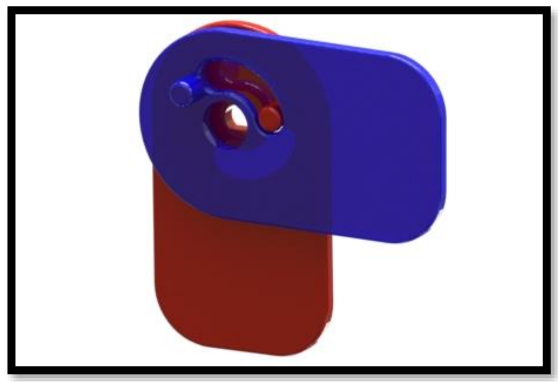


Figure 152:Two panel support arms in the deployed position.

5.1.37. Interface with the CES

The four solar panels will be connected in parallel and the output of the panels connected directly to an input of the Control Electronics System control board discussed in section 5.1.44.1. The power level of the solar power generated will be read by the CES control board and a threshold set using a scaling factor in order to trigger the Surface Imaging System's camera module which is discussed below in section 5.1.39. The panels will be connected in parallel to allow the trigger to be set at a value lower than the maximum possible wattage of all four panels to account for weather conditions during the mission.

5.1.38. Surface Imaging System (SIS)

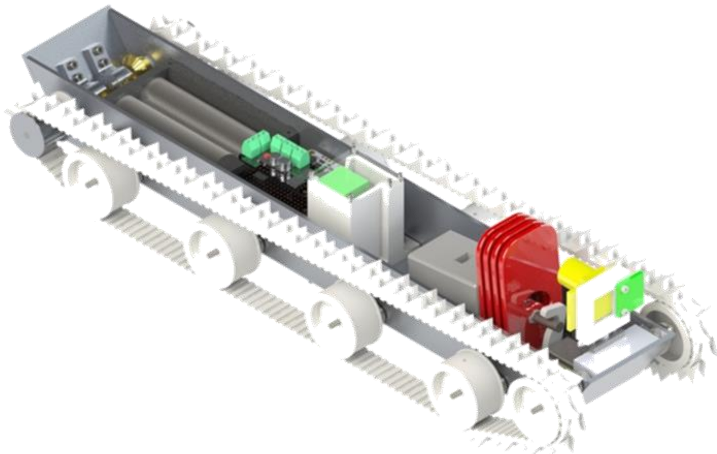


Figure 153: SIS Highlighted in Yellow.

The Surface Imaging System has been added to the payload as a secondary mission. This mission will not affect in any way the ability of the payload to complete its primary mission. The system has been added to embrace the mindset of the rover challenge of deploying an autonomous rover on another planet to collect data about that planet, of which images provide great scientific value. The SIS will be responsible for using the power harvested by the SAS as a trigger to take images of the rover and surrounding ground area and storing the images on the CES data logging board's microSD card, described in section 5.1.44.2, for analysis after retrieval of the payload.

5.1.39. Camera Module

The camera module will be responsible for taking the images and relaying them to the microSD card. The ArduCAM Mini 5MP OV5642 has been selected as the SIS camera module due to its high-resolution images. This will allow the images to be much clearer providing more information as scientific data. On a foreign planet, high resolution image data would be vital to understanding the characteristics of the surface of the planet. The camera will be configured to take pictures at 1280x920 full HD resolution. This is not the highest resolution the camera is capable of but has been chosen due to its lower power consumption.

This module was also chosen for its compatibility with the CES control board. The camera sensors are controlled over I2C and the data is transmitted to and from the master over SPI, both of which are accessible on the CES control board described in section 5.1.44.1. The camera module and a sample image taken using the camera are shown below in Figure 154.

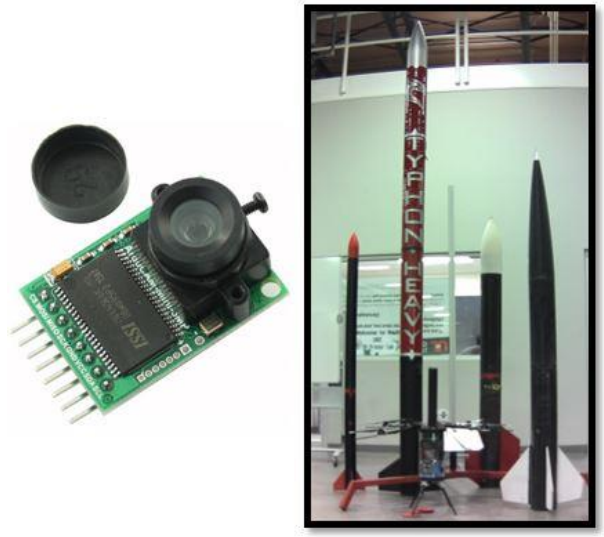


Figure 154: OV5642 camera and sample picture.

5.1.40. Field of View Extension

The field of view, and thus data collected, will be increased by mounting the camera module on the back side of the OAS lidar sensor mount. By mounting the camera in the front of the rover with the lens facing the rear, the camera will provide data on both the rover's state after completion of its primary mission and the ground in the surrounding area of the rover. By mounting the camera to the same OAS servo motor in section Figure 155, a single motor can provide a drastically increased field of view for both the lidar sensor and camera. The servo will pan the camera 180 degrees allowing it to take images of a much wider range of the rover and surrounding ground. The increased field of view is represented in blue below in Figure 155.

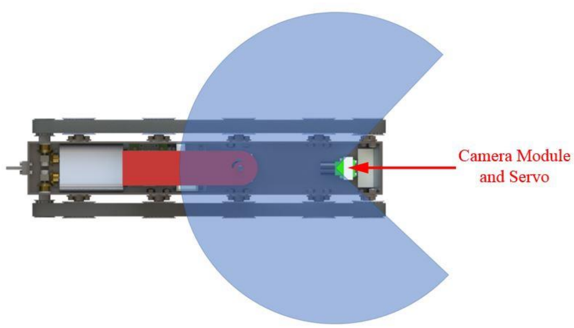


Figure 155: SIS field of view.

5.1.40.1. Mounting

The camera will be secured to the same mount that the lidar distance sensor will be secured to, as described above in 0. Two 2-56 socket-head screws will secure the camera to the mount with a hole in the mount for the camera's heat sink to sit in. The camera will be mounted sideways to prevent the camera's wires from interfering with the rotation of the mount assembly. The design of the assembly of the camera module (green), the electronics mount (white), and the servo (grey) is shown below in Figure 156.

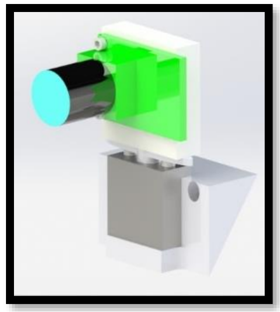


Figure 156: Camera module mount assembly.

5.1.41. Interface with CES

The SIS camera module requires a 3-5V supply voltage and draws 390mA peak during operation. The "BAT" pin of the CES control board is directly taken from the controller battery which is 3.7V and can be used to power the camera module. This negates the need for a voltage regulator to power the camera.

The power generated by the solar panels will be fed into the control board as an input that will only be high if the panels have been deployed successfully as described in section 5.1.37. This input will be read by the CES control board as a voltage level and scaled to be within the range of 0 to 100,000 where 100,000 indicates all four panels exposed to intense, direct sunlight. A trigger will be set at 90,000 (to account for weather conditions) to initiate the SIS phase of the control scheme described in section 5.1.43. An example of the trigger being surpassed is shown below in Figure 157.

```
Voltage = 87988.28 LOW
Voltage = 88183.59 LOW
Voltage = 87890.63 LOW
Voltage = 87792.97 LOW
Voltage = 91503.91 HIGH
Voltage = 99902.34 HIGH
Voltage = 99902.34 HIGH
Voltage = 99902.34 HIGH
```

Figure 157: Power level trigger.

5.1.42. Control Electronics System (CES)

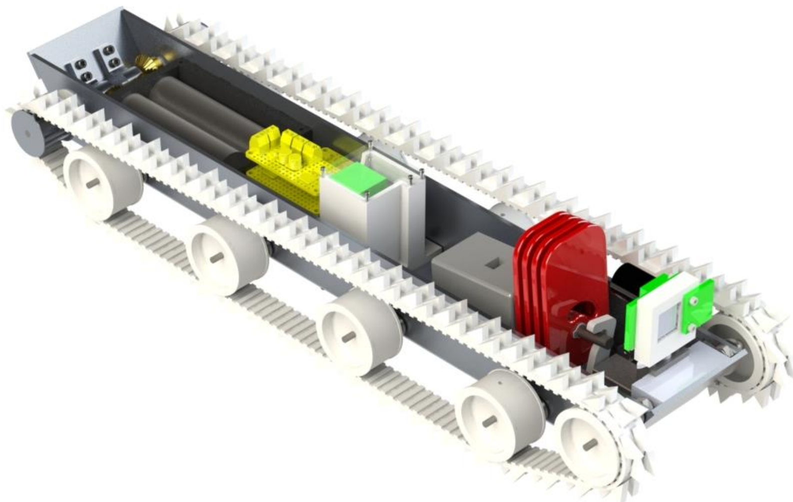


Figure 158: CES Highlighted in Yellow.

The Control Electronics System will be contained in the body of the rover and perform the master control scheme for the payload. This section will detail the operation of the CES during each phase of the mission.

5.1.43. Control Scheme

The control scheme software will be constructed with safety as the highest priority. All systems will be in a safe configuration while unpowered. In the event of an abort mission at any point, the CES will shut down, effectively locking the system in a safe state. A process diagram for the control scheme is shown below in Figure 159.

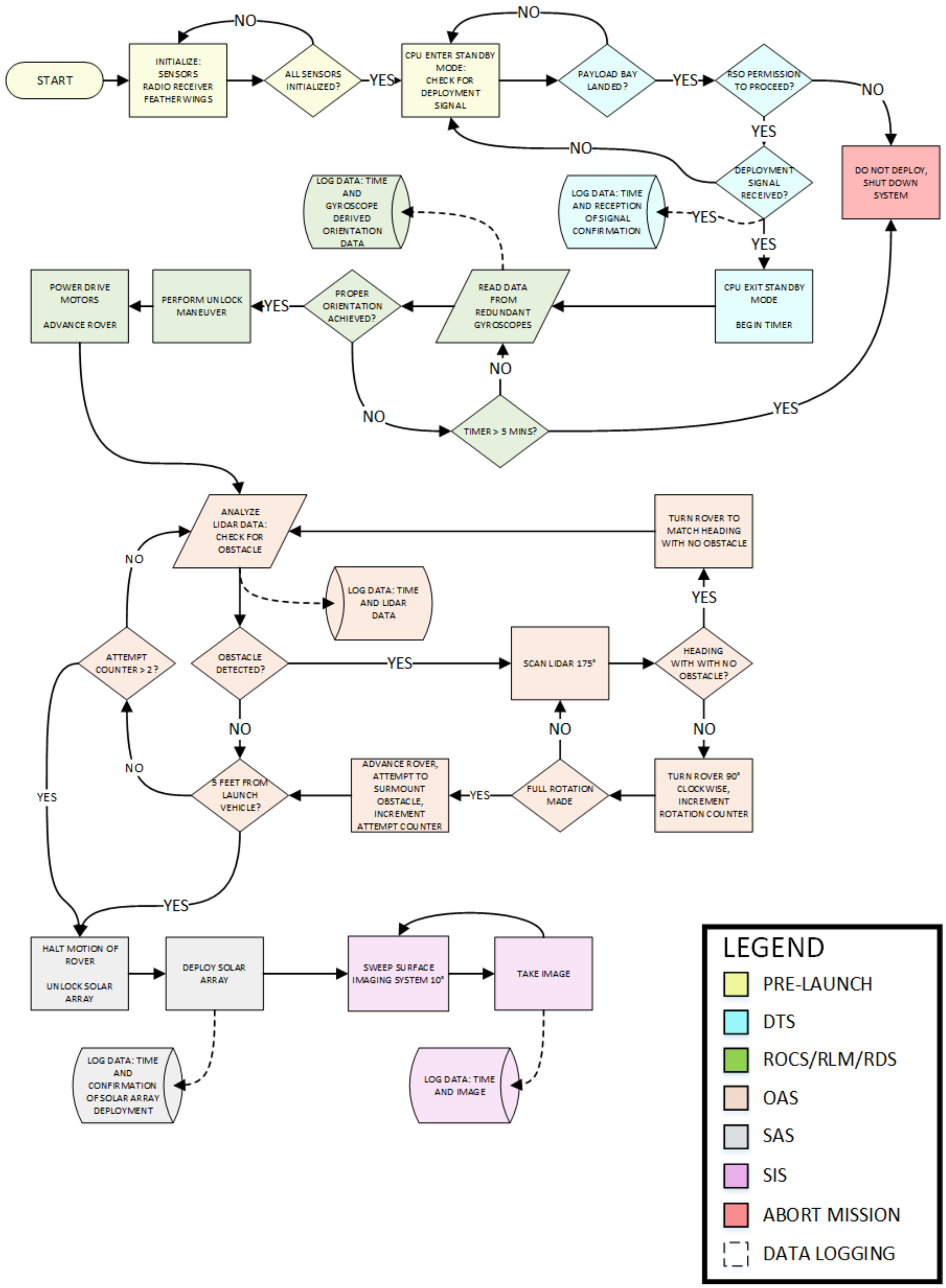


Figure 159: CES control scheme process diagram.

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

5.1.44. Pre-Launch Phase

The pre-launch phase of the CES mission will consist of final software and hardware checks before integration with the launch vehicle. All batteries used for pre-launch testing will be replaced by fully-charged batteries as the last operation before integration. This will ensure that the lifetime of the control electronics will exceed the duration of the mission.

5.1.44.1. Control Board

The CES control board will be powered on at this point. The control board refers to the microcontroller chosen to run the control scheme for the payload. The Adafruit Feather M0 Bluefruit LE has been selected for the control board due to its low footprint, number of available GPIO, and Bluetooth control capability to simplify testing and provide a more realistic hands-off environment while testing. The microcontroller is shown below in Figure 160 followed by specifications in Figure 160.

Figure 160: Adafruit Feather M0 Bluefruit LE.

5.1.44.2. Data logging Board

The selection of the Feather M0 Bluefruit as the control board requires a FeatherWing stackable add on to be chosen for the purposes of logging data as the control board does not have a mircoSD card onboard. The data logging board will be initialized and configured by the control board during this phase of the mission. The data logging board will be responsible for storing all data and images collected through the mission on a single microSD card. The Adafruit FeatherWing Adalogger has been selected for its stackable compatibility with the control board minimizing footprint and data logging capability with a battery backed real time clock for accurate data timestamps. The board is shown below in Figure 161.

Figure 161: Adafruit FeatherWing Adalogger.

5.1.44.3. Motor Driver Board

A motor driver board will be initialized and configured by the control board during this phase of the mission. The motor drive board will be responsible for driving the 4 motors of the payload: two main drive motors, SAS deployment, and RLM locking motor. The DC Motor + Stepper FeatherWing is also stack compatible with the Feather M0 and Adalogger saving space. The board is shown stacked on top of a Feather microcontroller below in Figure 162.

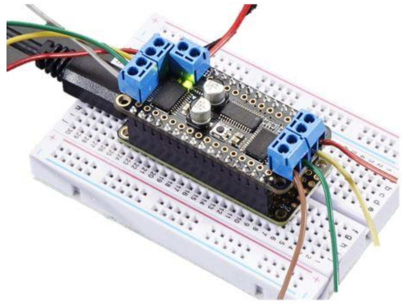


Figure 162: Adafruit FeatherWing motor driver board.

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

5.1.44.4. Controller Battery

The controller battery will be responsible for supplying power to the control board throughout the duration of the flight and payload mission. A 3.7V Lithium Polymer (LiPo) battery has been chosen dictated by the necessary 3.7V input for the Feather M0 Bluefruit microcontroller. The minimum required lifetime for the mission has been determined to be three hours accounting for pad time, flight, and payload mission times. Minimum dimensions and weights are desirable. These factors have led to the selection of the 500mAh LiPo for the controller battery with a projected lifetime of 3.8 hours given by the equation

$$Battery\ Life = \frac{Battery\ Capacity}{Device\ Consumption} * 0.7 \tag{46}$$

where the battery capacity is given in milliamp-hours, device consumption is given in milliamps, and the resultant battery life is given in hours. The battery is shown below in Figure 163.

Figure 163: 3.7V 500mAh controller battery.

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

5.1.44.5. Initialization

The conclusion of the pre-launch phase will be final integration of the payload into the launch vehicle and start-up of the control board. Start-up will be controlled by a switch mounted to the front of the rover. At this time, the control scheme will be initiated and the control board will initialize sensors and FeatherWings. The control board will then fall into standby mode to conserve power.

5.1.45. DTS Phase

The DTS phase of the CES mission extends throughout the flight of the launch vehicle. The Deployment Trigger System receiver module will continuously loop waiting for the deployment signal to be sent from the transmitter on the ground. The unique signal will ensure that no premature deployment signal is recognized. Permission from the RSO to continue the mission is required for the deployment signal to be sent. After receiving of the deployment signal, the control board will exit standby mode and continue the control scheme.

5.1.46. ROCS/RLM/RDS Phase

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

5.1.46.1. ROCS Orientation Check

Two onboard gyroscopes will be reading orientation data during this phase of the mission. This data will be stored on the microSD card and checked by the control board to ensure that upright orientation has been achieved prior to unlocking the rover as a redundant check. The deployment signal and upright orientation readings from both gyroscopes will need to be attained prior to unlocking the rover. The BNO055 9-DOF Inertial Measurement Unit has been selected for the pair of sensors for the orientation check due to its high accuracy of 0.05% and cost reduction. The sensor is shown below in Figure 164.

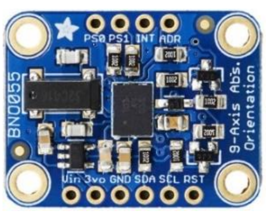


Figure 164: BNO055 9-DOF IMU.

5.1.46.2. RLM Unlocking

After receiving both the deployment signal and confirmation of upright orientation from the two gyroscopes, the control board will power the RLM motor to unlock the rover. This will allow the rover to move forward freely and continue its mission.

5.1.46.3. RDS Advance

After unlocking the RLM, the CES will power the two main drive motors of the rover advancing the rover forward out of the payload bay. The motors will be driven through the FeatherWing motor driver described above in section 5.1.44.3 allowing bi-directional control. Selection of the main drive motors is described in section 5.1.25. These motors have a 12V nominal operating voltage and 210mA no load, 4.9A stall current draw. These motors will have the largest current consumption under load compared to all other systems on the rover and therefor dictates the selection of a battery.

The battery has been selected from multiple batteries with similar performance characteristics, but different dimensions. The Storm 11.1V 400mAh 50C LiPo Battery Pack has been selected as it has minimum dimensions for its performance specifications and is rechargeable. The capacity has been selected to be 400mAh which is projected to allow the motors to be powered in worst case scenario stall for approximately 3.5 minutes. The battery is shown below in Figure 165.

Figure 165: Storm 11.1V 400mAh 50C LiPo battery.

The rover is expected to exceed the five foot minimum distance away from the launch vehicle in order to take into account possible obstacles encountered during the missions runtime.

5.1.47. OAS Phase

The OAS phase of the CES mission will consist of a continuous check for insurmountable obstacles in front of the rover and course adjustment accordingly. The CES will store the data from the OAS on the microSD card of the FeatehrWing Adalogger and analyze that data to determine if the rover needs to turn to avoid an obstacle. Selection of the sensor and interface with the CES is described in section 5.1.30.

5.1.48. SAS Phase

The SAS phase of the CES mission will consist of the unlocking and deployment of the foldable solar array. The control board will halt the main drive motors of the rover followed by unlocking of the SAS. After the panels have been unlocked, the CES will power the deployment motor through the FeatherWing motor driver to deploy the solar array. Selection of components and interface with the CES is described in section 5.1.33. The conclusion of the SAS phase of the CES mission will be achieving full deployment of the Solar Array System. This is also the conclusion of the payload's primary mission.

5.1.49. SIS Phase

The SIS phase of the CES mission will consist of harvesting solar power by means of the SAS and using it to trigger the Surface Imaging System's camera module. The SIS will take images of the rover and surrounding ground and store them on the FeatherWing Adalogger's microSD card for analysis after retrieval of the payload. Selection of components and interface with the CES is described in section 5.1.38. This phase will continue to run until a team member is allowed to retrieve the rover. This marks the completion of the control scheme and the payload's secondary mission.

5.1.50. Bluetooth Control

The Feather M0 Bluefruit LE microcontroller is Bluetooth compatible. This is desirable for ease of testing and creating a hands-off environment while testing to most accurately simulate mission conditions while still maintaining full control of the rover if need be. The interface used to control the payload's systems consists of 8 pushbuttons and a terminal window. An example of the controller is shown below in Figure 166.

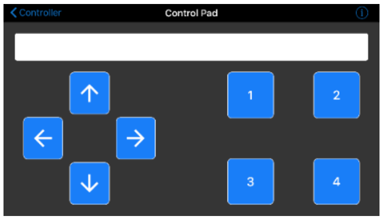


Figure 166: Bluetooth control panel.

5.1.51. CES Electrical Design

A custom designed printed circuit board (PCB) will be used to interface all electronics to the control board and to mount the control board to the rover. The PCB utilizes header pins to jumper exterior GPIO from sensors and components to the I/O pins of the control board reducing the amount of circuitry needed on the PCB and thus its overall dimensions. The Feather M0 and two FeatherWings will be stacked on the left as well as the two BNO055s stacked on the right. This drastically reduces the electronics footprint. The schematic and board layout for the PCB is shown below in Figure 167.

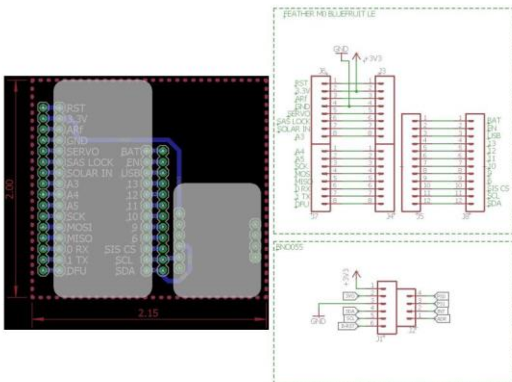


Figure 167: CES printed circuit board design.

5.1.52. CES Control Software

The control software will be written in the Arduino IDE as it is easily compatible with the Feather M0 Bluefruit LE microcontroller. This IDE will be used to write test scripts for requirement verifications and troubleshooting controls. The flight ready software is currently being constructed and will employ multiple levels of safety measures to ensure that the payload is safe to fly and will not prematurely deploy. A snippet of the controls code in the Arduino IDE used to control data logging and motors through the motor driver board.

6. Project Plan

6.1. Testing

6.1.1. VDS Test Campaign

Further testing is necessary to verify the battery life and the current draw requirements of the system. The test will consist of actuating the full hardware setup under an extended period of time. A full hardware setup is defined as all components connected on the printed circuit board (PCB). Data collection and motor actuation will verify that the components are functional and that the manufacturing process was successful. The current draw and voltage of both batteries will be monitored during extended operation to verify system requirements referenced in Table 52

Test	Test Description	Requirements Verified	Status
Xbee 1 mile test –Urban	The Xbee was tested in a 1 mile line of sight urban setting, and the data transmission was tested.	V.1.2, V.1.4	Complete, mixed results. Secondary test scheduled for January 17th.
Xbee 2 mile test - Rural	The Xbee will be further tested at even further extended distances in a rural setting as opposed to an urban setting. Data will be taken on the difference in performance.	V1.2, V 1.4	Incomplete, scheduled for Late January.
Flight configuration test	Externally from the vehicle, the VDS will be assembled in full configuration with electronics running and blades being tested for actuation. Power consumption from li-po batteries as well as BeagleBone operation performance will be monitored, and updates will be made if needed.	V.1.1.4, V.1.1.5, V.1.1.6	Incomplete, scheduled for late January
Full scale flight - control launch.	VDS will be present inside of full scale rocket launch #1 will electronics powered one and with full weight. The purpose of this test is to acquire data on the flight without use of the drag blades to learn what the true apogee of the vehicle is, and determine the level at which the VDS needs to operate to bring the vehicle to the desired height.	2.2, 2.6, 2.9, 2.19, 2.19.1, 2.19.2, V.1.1	Incomplete, scheduled for February 10th.
Full scale flight control launch – full break.	VDS will be present inside of full scale rocket launch #2 will electronics powered one and with full weight. The purpose of this flight is to acquire data on the flight to learn what the absolute lowest apogee the VDS is able to bring	2.2, 2.6, 2.9, 2.19, 2.19.1, 2.19.2, V.1.1	Incomplete, scheduled for February 17th.

	the vehicle to. This data will be used for calculations and simulations in determining the VDS's true flight model.		
Full scale flight - full actuation	VDS will be present inside of full scale rocket launch #2 will electronics powered one and with full weight. The VDS will perform at full functionality. Adjustments will be made if needed.	2.2, 2.6, 2.9, 2.19, 2.19.1, 2.19.2, V.1.1	Incomplete, scheduled for February24th.

Table 52: VDS test plan.

6.1.1.1. Xbee distance testing

Test to gauge battery longevity as well as verifying signal strength within a miles distance.

Items to be tested

- Current consumption and power draw from Xbee transmitter
- Signal strength and reliability at varying differences

Pass/Fail Criteria

Test Name	Requirement(s) to be Verified	Pass/Fail Criteria
Xbee distance testing	V1.2, V 1.4	This test will be considered passed if the battery has negligible power loss after extended use, and is able to transmit data at distances up to two miles.

Table 53: Pass/Fail Critera

Pre-Test

Setup

- Assembly of antenna and circuitry on test board
- Set up MAC address of Xbee units with XCTU software
- Verify connection
- Place receiving unit at static point, gradually move transmitting unit further at a constant rate

Equipment

- 2 Xbee Pro SX units
- 2 full sized breadboards
- Breadboard wiring
- BeagleBone Computer
- Li-Po battery
- 5V and 3.3V voltage regulators

6.1.1.2. Flight Configuration Test

External VDS preliminary testing.

Items to be tested

- Blade Actuation and limit switch functionality
- Program and electronics check
- Validate data intake from sensors

Pass/Fail Criteria

Test Name	Requirement(s) to be Verified	Pass/Fail Criteria
Flight Configuration Test	V.1.1.4, V.1.1.5, V.1.1.6	This test will be considered passed if the blades are able to actuate fully, and retract autonomously with the use of the limit switches.

Table 54: Pass/Fail criteria.

Pre-Test

Setup

- Assemble blade configuration with wiring into electronics sled.
- Validate that electronics are operational by sending test commands

Safety Notes

All parties will be aware of hot power lines and will keep at least 1 foot from moving drag blades to prevent pinching or blocking.

6.1.1.3. Full scale flight – control launch

This test will serve as a benchmark to show what apogee altitude the vehicle can achieve with an inactive VDS.

Items to be tested

- Vehicle max apogee without drag effects from the VDS
- Sensors on the VDS electronics

Pass/Fail Criteria

Test Name	Requirement(s) to be Verified	Pass/Fail Criteria
Control Vehicle Flight Test	2.2, 2.6, 2.9, 2.19, 2.19.1, 2.19.2	The launch vehicle control test flight shall be considered a pass if the vehicle ascends stably, does not exceed 5,600ft. AGL, is +/- 10% of the expected apogee altitude, takes adequate data to use in future simulations, and recovers safely within the predicted drift radius.

Table 55: Pass/Fail criteria.

Equipment

- Full scale launch vehicle
- Four PerfectFlite Stratologger CF altimeters
- Steel ballasts to simulate the weight of the payload
- VDS and VDS electronics
- Ejection charges
- AeroTech L2200-G motor
- 12-foot rail and launch pad
- Launch system
- ARRD
- Booster Main and Drogue
- Booster Main and Drogue shock cords
- Fins
- Multimeter

Safety Notes

All parties will be respectful of all flight safety procedures as outlined in the safety handbook. All electrical team members will be aware of hot power lines and will be aware of any potential hazards associated with operation the VDS and its electronics.

6.1.1.4. Full scale flight – full break

This test will serve as a benchmark to show what apogee altitude the vehicle can achieve with a fully deployed VDS. This flight will serve as the absolute minimum apogee altitude for the rocket.

Items to be tested

- Vehicle minimum height with full drag effects from the VDS
- Sensors on the VDS electronics

Pass/Fail Criteria

Test Name	Requirement(s) to be Verified	Pass/Fail Criteria
Full Break test	2.2, 2.6, 2.9, 2.19, 2.19.1, 2.19.2, V.1.1	This test will be considered passed if the vehicle ascends stably, does not exceed 5,200ft. AGL, is +/- 10% of the expected apogee altitude, takes adequate data to use in future simulations, and recovers safely within the predicted drift radius.

Table 56: Pass/Fail criteria.

Equipment

- Full scale launch vehicle
- Four PerfectFlite Stratologger CF altimeters
- Steel ballasts to simulate the weight of the payload
- VDS and VDS electronics
- Ejection charges
- AeroTech L2200-G motor
- 12-foot rail and launch pad
- Launch system
- ARRD
- Booster Main and Drogue
- Booster Main and Drogue shock cords
- Fins
- Multimeter

Safety Notes

All parties will be respectful of all flight safety procedures as outlined in the safety handbook. All electrical team members will be aware of hot power lines and will be aware of any potential hazards associated with operation the VDS and its electronics.

6.1.1.5. Full scale flight – full actuation

This test will serve as the first 'full run' of the VDS with full use of altitude correction software and telemetry system.

Items to be tested

- VDS drag blade actuation ability
- Sensors on the VDS electronics
- Telemetry system
- Accuracy of the apogee altitude of the vehicle

Test Name	Requirement(s) to be Verified	Pass/Fail Criteria
Full Actuation test	2.2, 2.6, 2.9, 2.19, 2.19.1, 2.19.2, V.1.1	This test will be considered passed if the vehicle ascends stably, does not exceed 5,300ft. AGL, is +/- 10% of the expected apogee altitude, takes adequate data to use in future simulations, and recovers safely within the predicted drift radius.

Table 57: Pass/Fail Critera

Equipment

- Full scale launch vehicle
- Four PerfectFlite Stratologger CF altimeters
- Steel ballasts to simulate the weight of the payload
- VDS and VDS electronics
- Ejection charges
- AeroTech L2200-G motor
- 12-foot rail and launch pad
- Launch system
- ARRD
- Booster Main and Drogue
- Booster Main and Drogue shock cords
- Fins
- Multimeter

Safety Notes

All parties will be respectful of all flight safety procedures as outlined in the safety handbook. All electrical team members will be aware of hot power lines and will be aware of any potential hazards associated with operation the VDS and its electronics.

6.1.2. Vehicle Test Campaign

Test	Test Description	Requirements Verified	Status

Subscale Vehicle Separation Test	The separation of the subscale vehicle will be tested by igniting pyrotechnic charges inside the vehicle on the ground to ensure proper separation of the vehicle to allow successful recovery of the vehicle.	2.24	Completed 11/10/17 and 12/1/17. Outcome: Pass
Subscale Vehicle November 11th Flight	A subscale vehicle flight will be used to estimate the coefficient of drag of the full-scale launch vehicle, verify construction techniques, and confirm simulation accuracy.	2.18, 2.18.2, 2.18.2.1	Completed 11/11/17. Outcome: Fail
Subscale Vehicle December 2nd Flight	A subscale vehicle flight will be used to estimate the coefficient of drag of the full-scale launch vehicle, verify construction techniques, and confirm simulation accuracy.	2.18, 2.18.2, 2.18.2.1	Completed 12/2/17. Outcome: Pass
Nose Cone Drop Test	A nose cone drop test will occur to verify that the additively manufactured nose cone can structurally withstand hitting the ground while falling under its parachute.	2.23	Completed 12/21/17. Outcome: Pass
Parachute Drop Test	A main parachute drop test will occur to verify opening force, time, stability, landing force and nominal drag force under freefall. A drogue parachute drop test will occur to verify opening force, vertical velocity, and stability.	3.3	Completed 12/26/17 Outcome: Pass
Reefing Ring Drop Test	A main parachute drop test with reefing ring will occur to verify deployment time retardation.	3.3.1	Completed 12/26/17 Outcome: Pass
CO_2 Mock Payload Bay Separation Test	A separation test will occur using a previously manufactured airframe tube modified to simulate the payload bay in size and weight. This test serves to verify the size CO_2 charge needed to separate the payload bay during flight	2.23	Completed 1/3/18. Outcome: Fail
Payload Bay Black Powder Containment Test	To verify that a black powder separation test will not damage the rover payload, a test will be conducted with a mock payload and black powder separation setup.	2.23	Completed 1/6/18 Outcome: Pass
Bulkplate Assembly Test	The bulkplate assemblies will be placed at least twice the maximum forces applied by the parachutes during descent to verify they will not fail.	2.22, 2.23	Incomplete. Scheduled for late January.

Fin Material Tensile Test	A tensile test will be used to determine material properties of the quasi-isotropic carbon fiber sheet the team manufactured for use in the fins.	2.23	Incomplete. Scheduled for late January.
Vehicle Separation Test	The separation of the vehicle will be tested by igniting pyrotechnic charges inside the vehicle on the ground to ensure proper separation of the vehicle to allow successful recovery of the vehicle.	2.24	Incomplete. Scheduled for early February.
Control Vehicle Flight Test	A full-scale launch vehicle test flight will be used to test the stability of the vehicle and the integrity of the mechanical design of the vehicle. The VDS will be inactive during this flight.	2.2, 2.6, 2.9, 2.19, 2.19.1, 2.19.2	Incomplete. Scheduled for February.

6.1.2.1. Subscale Vehicle Separation Test

This test will demonstrate the system's ability to separate sections of the subscale vehicle to allow the recovery equipment to deploy during flight.

Items to be tested

- Ejection charges are properly sized to successfully separate the subscale launch vehicle during recovery.

Pass/Fail Criteria

Test Name	Requirement(s) to be Verified	Pass/Fail Criteria
Subscale Vehicle Separation Test	2.6, 2.24	This test will be considered a pass if all sections of the subscale launch vehicle separate flawlessly. The nose cone must separate from the recovery bay and deploy the drogue parachute. The ARRD must release its pin and deploy the main parachute. If any section does not separate flawlessly the test is considered a failure.

Table 58: Pass/Fail criteria.

Pre-Test

Equipment

- The subscale launch vehicle shall be utilized to ensure proper volume of the recovery bay.
- An electronic ignition station shall be utilized in order to ignite the ejection charge from a safe distance.
- Prepared black powder ejection charges shall be utilized in order to separate the section.

Setup

Ejection charges shall be inserted into the recovery bay. The recovery bay shall be attached to the corresponding sections of the electronics bay in accordance with the launch vehicle.

Safety Notes

All spectators and testers shall be a minimum of 12 feet from the subscale launch vehicle during testing. No person or object shall be directly in front of or behind the subscale launch vehicle during ejection charge testing. All spectators shall wear safety goggles during preparation of charges and during separation testing.

Procedure

1) Prepare ejection charge using the specified amount of black powder measured using the black powder measuring kit located in the explosives box.
2) Connect the prepared ejection charge to the drogue terminal block.
3) Assemble the electronics bay and recovery bay using the #4-40 UNC nylon shear pins.
4) Connect the electronic ignition station to the terminal block.
5) Ensure the area is clear around the subscale launch vehicle.
6) Fire the ejection charge using the electronic ignition station.
7) After nose cone separation, repeat steps 4-6 for the ARRD.

Results

After completion of the subscale launch vehicle separation test, the test resulted in a pass. The nose cone separated from the recovery bay and deployed the drogue. The ARRD released its pin and allowed the main parachute to be deployed.

6.1.2.2. Subscale Vehicle November 11th Test Flight

To comply with Statement of Work Requirement 2.18, a subscale model of the full-scale launch vehicle was designed, manufactured, and flown. The subscale model was designed to resemble and perform as similarly as possible to the full-scale design.

Items to be tested

- The subscale vehicle shall be utilized to ensure proper volume of the recovery bay.
- An electronic ignition station shall be utilized in order to ignite the ejection charge from a safe distance.
- Prepared black powder ejection charges shall be utilized in order to separate the section.
- StratoLogger altimeters will be used on the subscale to verify their validity for use on the full-scale launch vehicle.

Items not tested

- The VDS, Payload, and GPS devices will not be tested due to the size constraints of the subscale vehicle.

Pass/Fail Criteria

The following table describes the requirements that are being tested and the specifications to be able to accept tested requirements as a verified requirement.

Test Name	Requirement(s) to be Verified	Pass/Fail Criteria

Subscale Vehicle November 11th Flight	2.18, 2.18.2, 2.18.2.1	The launch of the subscale vehicle is considered a success if the exit rail velocity and the achieved apogee altitude is within 10% of the simulated apogee altitude. The recovery of the subscale vehicle is considered a success if the subscale vehicle is undamaged upon recovery.

Table 59: Pass/Fail criteria.

Pre-Test

The following sections describe information about the setup and approach being used for this test.

Equipment

- Subscale vehicle
- Two PerfectFlite StratoLogger Altimeters
- Two Duracell 9 volt batteries
- Altimeter Sled
- ARRD assembly
- Launch Controls
- Aerotech I300 Motor
- Ejection charge equipment
- Cruciform Drogue
- Toroidal Main
- Shock Chord

Setup

The recovery equipment for the subscale vehicle was inserted into a single recovery bay with a separation point at the avionics bay located in the center of the vehicle.

Safety Notes

All spectators and launch attendees shall be at the appropriate distance from the launch vehicle as outlined by the NAR Safety Code.

Procedure

Subscale Vehicle Preparation

1. Program the two PerfectFlite Stratologger CF altimeters to ignite an ejection charge at apogee and at a lower specific altitude.
2. Mount altimeters to altimeter sled and connect each altimeter to a Duracell 9 volt battery.
3. Create two ejection charges by inserting an appropriate amount of black powder into an ejection charge canister with a e-match and seal them off with electrical tape.
4. Check continuity between terminal barrier blocks and altimeters.

Subscale Recovery Preparation

1. Inspect shroud lines, panels, and stitching for compromising damage.
2. Fold drogue arms under center overlap of panels. Neatly fold shroud lines and wrap assembly in nomex.
3. Fold main parachute panels sequentially in half on top of each other.
4. Fold shroud lines into S folds.

5. Stow main parachute in deployment bag with S folded shroud lines tucked neatly inside bag.
6. Inspect and clean ARRD of any black powder residue.
7. Install redundant E-matches through hole in black canister.
8. Fill black powder canister to line with black powder and cover with retaining sticker.
9. Install shackle pin, ball bearings, spring, and piston into red ARRD body.
10. While depressing piston just past threads and holding shackle in place, screw black powder canister into ARRD body. Test integrity by pulling and twisting shackle.
11. Install ARRD onto bulkplate.
12. Connect drogue and main to their respective shock cords.
13. Carefully slide recovery bay tube over assembly for transportation to launch field.

Launch Site Subscale Preparation

1. Insert motor into motor mount and secure using the motor retainer.
2. Insert altimeter sled into the avionics coupler and secure to propulsion bay by using three 6-32 SCHS fasteners.
3. Seal avionics coupler with bulkplate, which holds the terminal barrier blocks, ARRD, and eye bolt for attaching recovery equipment.
4. Connect ejection charge and ARRD to its respective terminal barrier block.
5. Connect main shock cord to eyebolt. Connect drogue shock cord to ARRD shackle.
6. Connect main deployment bag tether to ARRD shackle.
7. Insert dog barf under main bag and slide airframe over coupler.
8. Connect nose cone to vehicle via a friction fit.
9. Setup launch pad 100 feet away from spectators. For more detail, reference see the NAR Safety Code.
10. Attach 12 foot rail to launch pad and prepare launch system.
11. Transport subscale vehicle to launch pad location and attach subscale vehicle to launch rail by sliding rail buttons into the rail.
12. Arm each altimeter by turning each screw switch to the on position, which are accessible via the vent hole in the avionics bay.
13. Insert the igniter into the motor, ensuring that the igniter tip is inserted far enough into the motor.
14. Connect to the igniter to the launch system via alligator clips.
15. Check continuity between the igniter and the launch system.
16. Launch subscale vehicle.

Results

After conducting the subscale flight test on November 11th, the outcome resulted in a failed test. The full details on the results of this test can be found in 3.3.10.3.2.

6.1.2.3. Subscale Vehicle December 2nd Test Flight

To comply with Statement of Work Requirement 2.18, a subscale model of the full-scale launch vehicle was designed, manufactured, and flown. The subscale model was designed to resemble and perform as similarly as possible to the full-scale design. This test served to test new recovery subsystem designs implemented after the failed test launch on November 11th.

Items to be tested

- The subscale vehicle shall be utilized to ensure proper volume of the recovery bay.
- An electronic ignition station shall be utilized in order to ignite the ejection charge from a safe distance.
- Prepared black powder ejection charges shall be utilized in order to separate the section.
- StratoLogger altimeters will be used on the subscale to verify their validity for use on the full-scale launch vehicle.

Items not tested

- The VDS, Payload, and GPS devices will not be tested due to the size constraints of the subscale vehicle.

Pass/Fail Criteria

The following table describes the requirements that are being tested and the specifications to be able to accept tested requirements as a verified requirement.

Test Name	Requirement(s) to be Verified	Pass/Fail Criteria
Subscale Vehicle December 2nd Flight	2.18, 2.18.2, 2.18.2.1	The launch of the subscale vehicle is considered a success if the exit rail velocity and the achieved apogee altitude is within 10% of the simulated apogee altitude. The recovery of the subscale vehicle is considered a success if the subscale vehicle is undamaged upon recovery.

Table 60: Pass/Fail criteria.

Pre-Test

The following sections describe information about the setup and approach being used for this test.

Equipment

- Subscale vehicle
- Two PerfectFlite StratoLogger Altimeters
- Two Duracell 9 volt batteries
- Altimeter Sled
- ARRD assembly
- Launch Controls
- Aerotech I300 Motor
- Ejection charge equipment
- Cruiciform Drogue
- Toroidal Main
- Shock Chord

Setup

The recovery equipment for the subscale vehicle was inserted into a single recovery bay with a separation point at the avionics bay located in the center of the vehicle.

<u>Safety Notes</u>

All spectators and launch attendees shall be at the appropriate distance from the launch vehicle as outlined by the NAR Safety Code.

Procedure

Subscale Vehicle Preparation

1. Program the two PerfectFlite Stratologger CF altimeters to ignite an ejection charge at apogee and at a lower specific altitude.
2. Mount altimeters to altimeter sled and connect each altimeter to a Duracell 9 volt battery.
3. Create two ejection charges by inserting an appropriate amount of black powder into an ejection charge canister with a e-match and seal them off with electrical tape.
4. Check continuity between terminal barrier blocks and altimeters.

Subscale Recovery Preparation

1. Inspect shroud lines, panels, and stitching for compromising damage.
2. Fold drogue arms under center overlap of panels. Neatly fold shroud lines and wrap assembly in nomex.
3. Fold main parachute panels sequentially in half on top of each other.
4. Fold shroud lines into S folds.
5. Stow main parachute in deployment bag with S folded shroud lines tucked neatly inside bag.
6. Inspect and clean ARRD of any black powder residue.
7. Install redundant E-matches through hole in black canister.
8. Fill black powder canister to line with black powder and cover with retaining sticker.
9. Install shackle pin, ball bearings, spring, and piston into red ARRD body.
10. While depressing piston just past threads and holding shackle in place, screw black powder canister into ARRD body. Test integrity by pulling and twisting shackle.
11. Install ARRD onto bulkplate.
12. Connect drogue and main to their respective shock cords.
13. Carefully slide recovery bay tube over assembly for transportation to launch field.

Launch Site Subscale Preparation

1. Insert motor into motor mount and secure using the motor retainer.
2. Insert altimeter sled into the avionics coupler and secure to propulsion bay by using three 6-32 SCHS fasteners.
3. Seal avionics coupler with bulkplate, which holds the terminal barrier blocks, ARRD, and eye bolt for attaching recovery equipment.
4. Connect ejection charge and ARRD to its respective terminal barrier block.

5. Connect main shock cord to eyebolt. Connect drogue shock cord to ARRD shackle.
6. Connect main deployment bag tether to ARRD shackle.
7. Insert dog barf under main bag and slide airframe over coupler.
8. Connect nose cone to vehicle via a friction fit.
9. Setup launch pad 100 feet away from spectators. For more detail, reference see the NAR Safety Code.
10. Attach 12-foot rail to launch pad and prepare launch system.
11. Transport subscale vehicle to launch pad location and attach subscale vehicle to launch rail by sliding rail buttons into the rail.
12. Arm each altimeter by turning each screw switch to the on position, which are accessible via the vent hole in the avionics bay.
13. Insert the igniter into the motor, ensuring that the igniter tip is inserted far enough into the motor.
14. Connect to the igniter to the launch system via alligator clips.
15. Check continuity between the igniter and the launch system.
16. Launch subscale vehicle.

Results

After conducting the subscale flight test on December 2nd, the outcome resulted in a passed test. The full details on the results of this test can be found in 3.3.10.3.3.

6.1.2.4. Nose Cone Drop Test

This test will demonstrate the ability of the additively manufactured nose cone to structurally withstand the impact force from hitting the ground while descending under parachute. The nose cone design is further discussed in 3.3.7.

Items to be tested

- Full scale nose cone 3D printed at the University of Louisville Rapid Prototyping Center.

Pass/Fail Criteria

The following table describes the requirements that are being tested and the specifications to be able to classify the results of the tested requirement as a "pass".

Test Name	Requirement(s) to be Verified	Pass/Fail Criteria
Nose Cone Drop Test	2.23	The force that the nose cone experiences impacting the ground after descending under fully deployed parachute must not cause any structural damage to the nose cone.

Table 61: Pass/Fail criteria.

Pre-Test

The following sections describe information about the setup and approach being used for the test.

Equipment

- The nose cone to be used on the full-scale launch vehicle will be utilized to ensure that the test results do not deviate from the system's actual performance.
- A parachute with the exact dimensions to be used for the nose cone section on the full-scale launch vehicle shall be utilized to ensure accurate descent velocity.
- A 6 in. diameter carbon fiber coupler will be utilized to simulate the mass of the nose cone avionics coupler, and to serve as point of attachment for the parachute.

Setup

The nose cone avionics coupler will be temporarily fastened to the nose cone. The parachute shall be securely fastened to a bulk-plate at the base of the nose cone avionics coupler.

Safety Notes

All spectators must be a minimum of 100 ft. from the drift radius of the nose cone at ground level.

Procedure

1) Assemble the coupler and bulk plates using ¼"-20 all-thread rods and a single U-bolt.
2) Fasten the nose cone and coupler together.
3) Secure the parachute's shock cord to the U-bolt in the center of the coupler bulk plate.
4) Ensure that the drop zone area surrounding the elevated platform is clear for a safe test.
5) Drop the nose cone assembly.

Results

After conducting the nose cone drop test, the test resulted in a pass. The additively manufactured nose cone will be used on the full-scale launch vehicle and is discussed in detail in 3.3.7.

6.1.2.5. Parachute Drop Test

This test will demonstrate the ability of the laser cut parachutes to fully deploy, verifying opening force, vertical velocity, and stability.

Items to be tested

- Main Toroidal Parachute Booster
- Main Toroidal Parachute Payload
- Drogue Cruciform Parachute Booster
- Drogue Cruciform Parachute Payload

Pass/Fail Criteria

The following table describes the requirements that are being tested and the specifications to be able to classify the results of the tested requirement as a "pass".

Test Name	Requirement(s) to be Verified	Pass/Fail Criteria
Parachute Drop Test	3.3	Opening force will not exceed structural capabilities of the bulk plate. Parachute will fully deploy and shroud lines will not twist. Maintain appropriate terminal velocity.

Table 30: Pass/Fail criteria.

Pre-Test

The following sections describe information about the setup and approach being used for the test.

Equipment

- AIM XTRA accelerometer
- 5 lb Ballast
- Shock cord
- Parachute

Setup

The parachute to be tested was attached to the 5lb ballast with shock cord.

Safety Notes

All spectators must be a minimum of 100 ft. from the drift radius of the ballast at ground level.

Procedure

1) Attach the parachute to the ballast
2) Engage accelerometer
3) Ensure that the drop zone area surrounding the elevated platform is clear for a safe test.
4) Drop the parachute assembly.
5) Repeat for all parachutes

Results

After conducting the parachute drop test, the test resulted in a pass. Both the toroidal and cruciform parachutes will be used on the full-scale launch vehicle and is discussed in detail in 2.9.

6.1.2.6. Reefing Ring Test

This test will demonstrate the ability of the reefing ring to retard the deployment duration of the main parachute.

Items to be tested

- Main Toroidal Parachute Reefing Ring

Pass/Fail Criteria

The following table describes the requirements that are being tested and the specifications to be able to classify the results of the tested requirement as a “pass”.

Test Name	Requirement(s) to be Verified	Pass/Fail Criteria
Reefing Ring Test	3.3.2	Opening force will not exceed structural capabilities of the bulk plate. Parachute will fully deploy over a 20-30% greater period of time than without a reefing ring

Table 30: Pass/Fail criteria.

Pre-Test

The following sections describe information about the setup and approach being used for the test.

Equipment

- AIM XTRA accelerometer
- 5 lb Ballast
- Shock cord
- Parachute
- Reefing Ring

Setup

The parachute to be tested was attached to the 5lb ballast with shock cord.

Safety Notes

All spectators must be a minimum of 100 ft. from the drift radius of the ballast at ground level.

Procedure

1) Attach the parachute with reefing ring to the ballast
2) Engage accelerometer
3) Ensure that the drop zone area surrounding the elevated platform is clear for a safe test.
4) Drop the parachute assembly.

Results

After conducting the reefing ring test, the test resulted in a pass. The reefing ring will be used on the full-scale launch vehicle and is discussed in detail in 2.9.

6.1.2.7. CO_2 Mock Payload Bay Separation Test

This test will determine if CO_2 is a viable option for separating the payload bay from the rest of the launch vehicle.

Items to be tested

- Raptor CO_2 ejection devices

Pass/Fail Criteria

Test Name	Requirement(s) to be Verified	Pass/Fail Criteria
CO2 Mock Payload Bay Separation Test	2.23	This test is considered a pass if the coupler separates from the payload bay cleanly and with no issues.

Pre-Test

Equipment

- Raptor CO2 ejection devices
- Payload coupler
- 23g CO2 cartridge
- 35g CO2 cartridge
- E-match and igniter

Setup

1. Epoxy bulkplate into payload recovery coupler.
2. Install Raptor CO2 ejection device into the bulkplate.

Procedure

1. Secure the coupler to the payload bay using 3 #4-40 nylon shear pins.
2. Set payload bay and payload recovery coupler on a stand outdoors oriented with the coupler facing slightly up.
3. Initiate the CO2 ejection devices.

Results

After conducting the CO2 mock payload bay separation test, the test resulted in a failed test. The CO2 devices failed to produce enough pressure to separate the payload recovery coupler from the payload bay. Due to the results of the test, the CO2 separation method has been ruled unacceptable for this section and a traditional black powder separation method will be pursued.

6.1.2.8. Payload Bay Black Powder Containment Test

This test will determine an effective way to shield the rover payload from the black powder charge needed to separate the payload bay from the payload coupler. The black powder charge will be wrapped in a custom sewn nomex charge well that will direct the charge away from the rover.

Items to be tested

- Nomex charge well
- Black powder charge

Pass/Fail Criteria

Test Name	Requirement(s) to be Verified	Pass/Fail Criteria
Payload Bay Black Powder Containment Test	2.6	This test will be considered a pass if no excessive amount of black powder residue or particulate is expelled from the nomex charge well. This test will be considered a failure if there is an excessive amount of residue, or if the nomex charge well is burned excessively.

Table 62: Pass/Fail criteria.

Pre-Test

Equipment

- Nomex
- Black powder
- Black powder charge canister
- Sewing machine
- E-match and igniter

Setup

- Place precisely 4.5 grams of black powder into a black powder charge canister
- Insert e-match into black powder charge canister
- Place black powder charge into nomex charge well

Safety Notes

All spectators must stand at least 15 ft. from the black powder charge while wearing safety glasses.

Procedure

1. Ignite black powder charge within nomex charge well.
2. Inspect area surrounding charge well for excess residue.
3. Inspect charge well for excess burning/damage.

Results

After completing the payload bay black powder charge containment test, the test resulted in a pass. The nomex charge well contained the black powder successfully and suffered minimal burning. The nomex charge well design will be tweaked and tested further in late January.

6.1.2.9. Bulkplate Assembly Test

The bulkplate assembly test will prove the integrity of the design with the expected load from the opening force of the recovery equipment. This test will verify that the carbon fiber plate, wood plate, and U-bolt together can withstand 412 lbs, the maximum opening force of the parachute during decent.

Items to be tested.

- Carbon Fiber Plate
- Wood Plate
- U-Bolt and Washer

Pass/Fail Criteria

Test Name	Requirement(s) to be Verified	Pass/Fail Criteria
Bulkplate Assembly Test	**2.22** 2.23	The bulkplate assembly must not permanently deform in any way under loading of 412 lbs.

Table 63: Pass/Fail criteria.

Pre-Test

Equipment

- Bulkplate Assembly
- Test Fixture
- Test Load

Setup

1. The wooden plate is placed on top of the carbon fiber plate.
2. The u-bolt is inserted, from the bottom, through the carbon fiber plate and wooden plate, with the washers and nuts installed on top of the wooden plate.
3. The assembly is then suspended, carbon fiber side down, from the all thread on the test fixture with nuts and washers installed on the carbon fiber side.

Safety Notes

Keep all body parts clear of test fixture and potential falling objects.

Procedure

1) Assemble bulkplate.
2) Install bulkplate in test fixture, with carbon fiber side down.
3) Check that u-bolt and all nuts are properly secured.
4) Suspend load from bulkplate.
5) Gradually increase load to 412 lbs, checking for signs of failure.

6.1.2.10. Fin Material Tensile Test

This test will determine ultimate tensile strength of the fin material.

Items to be tested.

- Fin Material

Pass/Fail Criteria

The following table describes the requirements that are being tested and the specifications to be able to accept tested requirement as a passed requirement.

Test Name	**Requirement(s) to be Verified**	**Pass/Fail Criteria**
Fin Material Tensile Test	**2.22** 2.23	Test data shall provide the team with a clear understanding of the material properties of the fin material.

Table 64: Pass/Fail criteria.

Pre-Test

Equipment

20,000 Lbs MTS
Laptop with MTS software
1"x12" fin material sample

Setup

Airframe Samples

1. Cut 1"x12" strips from fin material.
2. Sample strips shall be cut.
3. Reinforce 1" from both ends of sample strips, front and back, to distribute clamping force.

Procedure

1) Connect laptop to MTS.
2) Enter sample specs into MTS program.
3) Set MTS gap to match sample length.
4) Secure sample in MTS clamp.
5) Run program/tensile test.
6) Record material properties.

6.1.2.11. Full-Scale Vehicle Separation Test

This test will demonstrate the system's ability to separate sections of the vehicle to allow the recovery equipment to deploy during flight.

Items to be tested

- Ejection charges are properly sized to successfully separate the launch vehicle during recovery.

Pass/Fail Criteria

Test Name	Requirement(s) to be Verified	Pass/Fail Criteria
Full-Scale Vehicle	2.6, 2.24	This test will be considered a pass if all sections of the launch vehicle separate flawlessly. The payload bay must separate from the payload coupler and deploy the drogue parachute. The

Separation Test		nose cone must separate from the payload recovery bay and deploy the drogue parachute. The payload coupler must separate from the booster and deploy the main parachute. The ARRD must release its pin and deploy the main parachute. If any section does not separate flawlessly the test is considered a failure.

Table 65: Pass/Fail criteria.

Pre-Test

Equipment

- The full scale launch vehicle shall be utilized to ensure proper volume of the booster recovery bay and payload recovery bay.
- An electronic ignition station shall be utilized in order to ignite the ejection charge from a safe distance.
- Prepared black powder ejection charges shall be utilized in order to separate the section.

Setup

Ejection charges shall be inserted into the appropriate terminal block of their associated recovery bay. Recovery bays shall be attached to the corresponding sections of the electronics bay in accordance with the launch vehicle.

Safety Notes

All spectators and testers shall be a minimum of 12 feet from the launch vehicle during testing. No person or object shall be directly in front of or behind the launch vehicle during ejection charge testing. All spectators shall wear safety goggles during preparation of charges and during separation testing.

Procedure

1) Prepare ejection using the specified amount of black powder measured using the black powder measuring kit located in the explosives box.
2) Connect the prepared ejection charge to the drogue terminal block.
3) Assemble the electronics bay and recovery bay using the #4-40 UNC nylon shear pins.
4) Connect the electronic ignition station to the terminal block.
5) Ensure the area is clear around the launch vehicle.
6) Fire the ejection charge using the electronic ignition station.
7) After nose cone separation, repeat steps one through six for the main deployment instead of the drogue deployment.

6.1.2.12. Control Vehicle Flight Test

This test will demonstrate the flight characteristics, recovery, and structural integrity of the full-scale vehicle. It will also serve as a benchmark to show what apogee altitude the vehicle can achieve with an inactive VDS.

Items to be tested

- The full-scale launch vehicle shall be utilized to ensure proper volume of the recovery bay.
- An electronic ignition station shall be utilized in order to ignite the ejection charge from a safe distance.
- Prepared black powder ejection charges shall be utilized in order to separate the section.

Items not tested

- The payload will not be tested and instead a payload mass simulator shall be used in its place.

Pass/Fail Criteria

The following table describes the requirements that are being tested and the specifications to be able to accept tested requirement as a passed requirement.

Test Name	Requirement(s) to be Verified	Pass/Fail Criteria
Control Vehicle Flight Test	2.2, 2.6, 2.9, 2.19, 2.19.1, 2.19.2	The launch vehicle control test flight shall be considered a pass if the vehicle ascends stably, does not exceed 5,600ft. AGL, is +/- 10% of the expected apogee altitude, and recovers safely within the predicted drift radius.

Table 66: Pass/Fail criteria.

Pre-Test

Equipment

- Full scale launch vehicle
- Four PerfectFlite Stratologger CF altimeters
- Steel ballasts to simulate the weight of the payload
- VDS and VDS electronics
- Ejection charges
- AeroTech L2200-G motor
- 12-foot rail and launch pad
- Launch system
- ARRD
- Booster Main and Drogue
- Booster Main and Drogue shock cords
- Fins
- Multimeter

6.1.3. Payload Master Test Plans

This section will describe the tests required to prove the integrity of the final payload design. Each test will be conducted to verify requirements intended to confirm the performance of a designated system of the designed payload and confirm flight readiness of the entire payload. Tests to be performed, requirements each test will verify, and the scheduled date of the test is shown below in Table 67.

Test	Requirement to be Verified	Scheduled date
Rover Performance Test	4.5.3 of the NASA SOW	February 23rd, 2018
ROCS Roll Test	ROCS-3	February 10th, 2018
DTS 50 foot Radius Test	DTS-4	January 18th, 2018
RDS Sloped Driving Test	RDS-3	January 27th, 2018
OAS Accuracy Test	OAS-2	January 13th, 2018
CES Orientation Accuracy Test	CES-1	January 31st, 2018
CES Autonomous Control Testing Series	4.5.3 of the NASA SOW CES-2 CES-3 CES-4 CES-5 CES-6	February 18th – February 23rd, 2018
Battery Life Testing Series	CES-8 CES-9	February 25th, 2018
Flight Loads Testing Series	ROCS-4 RLM-4 RBS-3 SAS-5	February 17th February 24th
Full Flight Performance Testing Series	DTS-6 CES-7	February 10th (DTS-6) February 24th (CES-7)

Table 67: Payload testing plan.

6.1.3.1. **Rover Performance Test**

Objective

The objective of this test is to ensure that the payload can successfully complete its primary mission. This will be the final performance test of the rover to determine any changes to interaction between systems and mechanical adjustment. The requirement to be verified by this test is requirement 4.5.3 of the NASA Statement of Work.

Items/Variable to be Tested

- Distance of travel after exiting the airframe
- Interaction between all systems in flight configuration

Methodology

This test will be an acceptance level test conducted to analyze the system's performance as it is intended to be used and affirm its capability to perform the intended tasks. The environment of the test will simulate the conditions in which the system is expected to perform during the mission by integrating the payload into the payload bay of the launch vehicle, running flight ready software, and deploying the rover as will be done during flight. The pretest setup, test procedure, post-test operations, and suspension criteria are described below.

Pretest Setup

- Load flight ready software onto Feather M0 Bluefruit LE microcontroller
- Integrate electronics into the rover
- Integrate the rover into the ROCS
- Integrate the ROCS into the payload bay of the launch vehicle and secure with set screws
- Ensure that the DTS antenna is installed on exterior of payload bay
- Configure transmitter electronics and mount on Yagi antenna

Test Procedure

____ 1.) Inform all bystanders of testing
____ 2.) Place payload bay section on the ground in an open area
____ 3.) Power up electronics
____ 4.) Walk within a 50 foot radius of the payload bay
____ 5.) Trigger deployment of the rover using the transmitter module
____ 6.) Inspect rover during autonomous operation
____ 7.) Allow rover to stop operating
____ 8.) Power down electronics

Post-test Operations

- Measure linear distance from rover to payload bay
- Log distance measurement and compare to 5 foot minimum required

Suspension Criteria

- Any safety risk not accounted for is encountered
- Rover does not reach 5 foot minimum distance from payload bay
- Rover does not deploy
- Loss of power to electronics

Success Criteria

The test will be considered successful only if the rover deploys from the payload bay after the deployment signal has been sent and comes to rest at least 5 feet from the payload bay from which it deployed. This test does not require the rover to deploy the solar array to be considered successful. The test will be considered a failure if the rover does not deploy or is not capable of reaching at least 5 feet away from the payload bay.

Safety Considerations

Proper safety precautions will be taken to avoid injury. Point points will be identified prior to any work being done. All bystanders will be cleared prior to testing to avoid collision and tripping hazards. The testing area will be cleared to avoid collision. No person will be allowed within 3 feet of the rotating parts while they are operating. All circuits will be continuity checked for shorts prior to any power being applied to avoid harm to personnel or components.

Results

This test requires full construction of the payload and is scheduled to occur February 23rd, 2018.

6.1.3.2. ROCS Roll Test

Objective

The objective of this test is to ensure that the Rover Orientation Correction System will reliably bring the rover to rest upright inside the payload bay regardless of the landing orientation of the launch vehicle. The results of this test will determine if any changes need to be done to the ROCS before flight. The requirement to be verified by this test is ROCS-3.

Items/Variable to be Tested

- Angle of inclination along the roll axis of the rover
- Consistency of performance of the system

Methodology

This test will be a system level test conducted to analyze the system's performance as it is intended to be used. The environment of the test will simulate the conditions in which the system is expected to perform during the mission. The pretest setup, test procedure, post-test operations, and suspension criteria are described below.

Pretest Setup

- Construct ramp out of wood with a 20° angle of inclination to provide consistent test platform
- Place ramp in large, flat, open area with no obstructions in front of the slope
- Load test script onto Feather M0 Bluefruit LE microcontroller
- Integrate electronics into the rover
- Integrate the rover into the ROCS
- Integrate the ROCS into the payload bay of the launch vehicle and secure with set screws

Test Procedure

____ 1.) Inform all bystanders of testing
____ 2.) Place payload bay section at the peak of the ramp
____ 3.) Power up electronics
____ 4.) Visually confirm orientation data is being collected by inspecting the
____ LEDs

_____ 5.) Perform countdown prior to releasing the payload bay
_____ 6.) Release payload bay
_____ 7.) Allow payload bay to come to rest
_____ 8.) Inspect and log the color of the LEDs as either green or red
_____ 9.) Power down electronics
10.) This concludes a trial. Repeat procedure to obtain 10 trials

Post-test Operations

- Remove set screws
- Remove ROCS from payload bay
- Remove rover from ROCS
- Remove electronics from the rover
- Extract gyroscope data from all 10 trials from microSD card
- Plot roll axis data and 50° upper limit vs. time
- Compare steady-state gyroscope data to 50° upper limit to confirm LED inspection results

Suspension Criteria

- Any safety risk not accounted for is encountered
- Either LED is red after the payload bay comes to rest
- Loss of power to electronics
- Failure of a trial

Success Criteria

Each trial will be considered successful only if both LEDs are seen to be green after the payload bay comes to rest. The test will be considered successful only if 10 consecutive trials yield successful results and graphical data is consistent with the LED data. Any deviation from these criteria will render the trial and test a failure at which point the test will be suspended, a solution will be determined and implemented, and the test will be conducted again.

Safety Considerations

Proper safety precautions will be taken to avoid injury. Point points will be identified prior to any work being done. Power tool safety will be adhered to while constructing the ramp. All bystanders will be cleared prior to testing to avoid collision and tripping hazards. The testing area will be cleared to avoid collision. All circuits will be continuity checked for shorts prior to any power being applied to avoid harm to personnel or components.

Results

This test is scheduled to occur February 10th, 2018.

6.1.3.3. **DTS 50 Foot Radius Test**

Objective

The objective of this test is to ensure that the complex RF emission pattern of the receiver module backed by a rounded conductive body is capable of maintaining communication with the transmitter module anywhere within close proximity. The requirement to be verified by this test is DTS-4.

Items/Variable to be Tested

- Integrity of communication at short range
- Communication rate

Methodology

This test will be a system level test conducted to analyze the system's performance as it is intended to be used. The environment of the test will simulate the conditions in which the system is expected to perform during the mission. The pretest setup, test procedure, post-test operations, and suspension criteria are described below.

Pretest Setup

- Integrate receiver antenna with receiver module
- Acquire 6 in. carbon fiber airframe section
- Mount receiver antenna to exterior of airframe section
- Integrate transmitter module with Yagi antenna
- Establish communication output to computer terminal

Test Procedure

_____ 1.) Inform all bystanders of testing
_____ 2.) Place airframe section on the ground in open area
_____ 3.) Power up electronics
_____ 4.) Stand 50 feet away from receiver with transmitter in hand
_____ 5.) Confirm communication link on computer terminal
_____ 6.) Walk transmitter slowly directly towards airframe section
_____ 7.) Monitor communication link at all times
_____ 8.) Walk transmitter directly over and past airframe section until reaching
_____ 50 feet away
_____ 9.) Walk transmitter 90° along perimeter of 50 foot radius centered on
_____ airframe section
10.) Walk transmitter toward, over, and past airframe section until reaching 50 feet away
11.) Power down electronics

Post-test Operations

- Collect airframe section and electronics
- Analyze communication rate and integrity based on timestamps on each data point

Suspension Criteria

- Any safety risk not accounted for is encountered
- Communication downlink occurs at any point
- Loss of power to electronics

Success Criteria

The test will be considered successful only if no downlink in communication occurs at any point during the test. The communication must maintain a consistent rate measured by the time stamps on each data point to be considered successful. The test will be considered a failure if there is any loss of communication or inconsistent, lagging communication occurs.

Safety Considerations

Proper safety precautions will be taken to avoid injury. The transmitter module and computer will be operated by separate team members. Each team member will keep their eyes on the ground in front of them while walking to avoid tripping hazards. All circuits will be continuity checked for shorts prior to any power being applied to avoid harm to personnel or components.

Results

This test is scheduled to occur January 18th, 2018.

6.1.3.4. **RDS Sloped Driving Test**

Objective

The objective of this test is to evaluate the Rover Drive System design's ability to maintain forward motion of the rover up an inclined terrain. The requirement to be verified by this test is RDS-3.

Items/Variable to be Tested

- Maximum incline surmountable by RDS
- Rate at which the RDS allows the rover to climb each slope angle

Methodology

This test will be a system level test conducted to analyze the system's performance in extreme conditions and determine limit of design. After meeting the success criteria, the test will continue to analyze the system in extreme conditions to expose any apparent design issues. The pretest setup, test procedure, post-test operations, and suspension criteria are described below.

Pretest Setup

- Assemble RDS system onto the body structure of the rover
- Build ramp with adjustable incline

- Place ramp on the ground in an open area
- Mark preset angles on adjustable ramp
- Mark one foot increments of distance up the ramp
- Set initial incline of ramp at 20°
- Load test script on Feather M0 Bluefruit LE microcontroller
- Confirm functionality of test script on flat ground

Test Procedure

____ 1.) Inform all bystanders of testing
____ 2.) Place rover at base of ramp facing upward
____ 3.) Power on electronics
____ 4.) Begin driving rover forward using Bluetooth controls and begin timer
____ 5.) Once rover has reached one foot marker, stop driving and stop timer
____ 6.) Remove rover from ramp
____ 7.) Log time taken to reach one foot and incline of ramp
____ 8.) Increment ramp angle
____ 9.) Repeat trial

Post-test Operations

- Plot inclination angle vs. time to climb one foot
- Determine upper inclination angle limit of system's capability

Suspension Criteria

- Any safety risk not accounted for is encountered
- Rover is not capable of reaching one foot up the ramp
- Ramp angle is not accurate
- Rover has not reached one foot up the ramp within one minute of driving
- Loss of power to electronics

Success Criteria

The test will be considered successful if the rover is able to climb at least one foot up the inclined surface of the ramp while the ramp is set at 20°. The rover must be able to reach this marker within one minute of driving. The test will be considered a failure if the rover is not capable of reaching at least one foot up the ramp within one minute at which point the design will be evaluated and a solution determined.

Safety Considerations

Proper safety precautions will be taken to avoid injury. Point points will be identified prior to any work being done. Power tool safety will be adhered to while constructing the ramp. All bystanders will be cleared prior to testing to avoid collision and tripping hazards. The testing area will be cleared to avoid collision. All circuits will be continuity checked for shorts prior to any power being applied to avoid harm to personnel or components.

Results

This test is scheduled to occur January 27th, 2018.

6.1.3.5. **OAS Accuracy Test**

Objective

The objective of this test is to evaluate the range finding accuracy of the VL53L0X lidar sensor of the Obstacle Avoidance System and ensure that the minimum accuracy meets the minimum requirement. The requirement to be verified by this test is OAS-2.

Items/Variable to be Tested

- Accuracy of the VL53L0X Time of Flight lidar sensor

Methodology

This test will be a unit/component level test conducted to analyze the sensor's accuracy the low and high end of the documented range of the sensor. The results of this test will determine the range of reliable operation of the lidar sensor to use for the control scheme. The pretest setup, test procedure, post-test operations, and suspension criteria are described below.

Pretest Setup

- Acquire object large enough to be placed in front of the sensor
- Load test script on Feather M0 Bluefruit LE microcontroller
- Place ruler below lidar sensor with zero mark in line with the sensor
- Confirm distance data is being displayed on computer terminal

Test Procedure

____ 1.) Inform all bystanders of testing
____ 2.) Power on electronics
____ 3.) Place object 5 in. away from sensor
____ 4.) Record trial number, location data from lidar, and location data from
____ ruler
____ 5.) Move object away from sensor by 2 in.
____ 6.) Repeat steps 4 and 5 until 5 trials have been completed
____ 7.) Move object 40 in. away from sensor
____ 8.) Record trial number, location data from lidar, and location data from
____ ruler
____ 9.) Move object closer to sensor by 2 in.
10.) Repeat steps 8 and 9 until 10 trials have been completed

11.) Power down electronics

Post-test Operations

- Analyze data to determine accuracy of every trial
- Determine average accuracy over low end of the range
- Determine average accuracy over high end of the range
- Determine average accuracy over the whole range

Suspension Criteria

- Any safety risk not accounted for is encountered
- Lidar does not sense object in front of sensor
- Loss of power to electronics

Success Criteria

The test will be considered successful only if the lidar sensor distance data is within 1 in. of the measured ruler data for all of the 10 data points taken. The test will be considered a failure if any data point displays an accuracy outside of +/- 1 in. even in the case of an average accuracy being below +/- 1 in. In the event of a failed test, a different object will be chosen to evaluate the effect of object color, geometry, and size on the readings.

Safety Considerations

Proper safety precautions will be taken to avoid injury and damage to electronics. Grounding mats will be used to avoid electrostatic discharge when handling electronics. All circuits will be continuity checked for shorts prior to any power being applied to avoid harm to personnel or components.

Results

This test is scheduled to occur January 13th, 2018.

6.1.3.6. **CES Orientation Accuracy Test**

Objective

The objective of this test is to validate the orientation check as a high reliability risk mitigation for premature deployment. The requirement to be verified by this test is CES-1.

Items/Variable to be Tested

- Accuracy of the BNO055 9DOF IMUs
- The drift of the BNO055 9DOF IMUs

Methodology

This test will be a unit/component level test conducted to analyze the sensor's accuracy and drift over long duration. The results of this test will be used to set the orientation angles determined to be safe to deploy the rover. The pretest setup, test procedure, post-test operations, and suspension criteria are described below.

Pretest Setup

- Acquire section of 6 in. airframe
- Load test script on Feather M0 Bluefruit LE microcontroller
- Confirm orientation data is being logged on FeatherWing Adalogger's microSD card
- Confirm LEDs are indicating orientation of the sensors properly
- Breadboard 2 BNO055 sensors, 1 microcontroller, and 1 data logger
- Tape breadboard to interior of the 6 in. airframe

Test Procedure

____ 1.) Inform all bystanders of testing
____ 2.) Ensure sensors are flat to the surface
____ 3.) Power up electronics
____ 4.) Log all orientation angles at startup
____ 5.) Roll airframe section one complete revolution while watching changes
____ in LEDs color
____ 6.) Log angles at each color change
____ 7.) Log orientation angles once the full revolution is complete
____ 8.) Repeat steps 5, 6, and 7 until 10 trials have been completed
9.) Power down electronics

Post-test Operations

- Remove breadboard from airframe
- Compare angles set to change the LEDs' color with measured angles when LEDs changed color
- Analyze data to determine sensor accuracy
- Compare angles measured at the end of each full revolution to the angles measured at startup to determine drift of the sensor

Suspension Criteria

- Any safety risk not accounted for is encountered
- Either LED does not change color when breadboard passes set threshold
- Data accuracy exceeds +/- 1°
- Loss of power to electronics

Success Criteria

The test will be considered successful only if the two BNO055s maintain an accuracy of +/- 0.1°, the LEDs change color appropriately after crossing the threshold, and the drift stays within 1° over the testing period of 10 trials. The test will be considered a failure if any of these criteria are not met at which point a solution will be determined to ensure accuracy and reliability.

Safety Considerations

Proper safety precautions will be taken to avoid injury and damage to electronics. Grounding mats will be used to avoid electrostatic discharge when handling electronics. All circuits will be continuity checked for shorts prior to any power being applied to avoid harm to personnel or components.

Results

This test is scheduled to occur January 31st, 2018.

6.1.3.7. CES Autonomous Control Testing Series

Objective

The objective of this testing series is to validate the autonomous controls of each system of the rover and refine the control scheme to achieve mission success. The requirements to be verified by this testing series are the following which are linked to their respective description: 4.5.3, CES-2, CES-3, CES-4, CES-5, CES-6.

Items/Variable to be Tested

- Autonomous control scheme
- Precision of desired operation

Methodology

Each test will be a system level test conducted to validate the autonomous control scheme. The results of these tests will determine changes to the autonomous controls of each system needed before integration of all autonomous controls into one program. The pretest setup, post-test operations, and suspension criteria common among all tests are described below. The test procedure for each test is described individually in detail.

Pretest Setup

- Integrate subsystem or component to be tested with Control Electronics System
- Load flight ready autonomous control for that subsystem or component onto Feather M0 Bluefruit LE microcontroller
- Confirm Bluetooth link in case of override
- Clear testing area of anything not necessary for testing

Test Procedure for SOW Verification (4.5.3)

____ 1.) Inform all bystanders of testing

____ 2.) Integrate Rover into ROCS
____ 3.) Integrate ROCS into payload bay of launch vehicle
____ 4.) Integrate DTS into launch vehicle
____ 5.) Power up electronics
____ 6.) Send deployment signal to rover using DTS
____ 7.) Measure distance traveled of the rover
____ 8.) Note distance at which the rover stops moving
____ 9.) Power down electronics

Test Procedure for RLM Control (CES-2)

____ 1.) Inform all bystanders of testing
____ 2.) Integrate CES into rover
____ 3.) Integrate Rover with RLM
____ 4.) Confirm Bluetooth link with Feather M0 Bluefruit LE
____ 5.) Send lock trigger over Bluetooth using cellphone
____ 6.) Inspect state of the locking mechanism
____ 7.) Send unlock trigger over Bluetooth using cellpnone
____ 8.) Inspect state of the locking mechanism
____ 9.) Repeat steps 5, 6, 7, and 8 until 10 trials have been completed
____ 10.) Power down electronics

Test Procedure for SAS Locking Motor Control (CES-3)

____ 1.) Inform all bystanders of testing
____ 2.) Power on electronics
____ 3.) Confirm Bluetooth link with Feather M0 Bluefruit LE
____ 4.) Send lock trigger over Bluetooth using cellphone
____ 5.) Inspect the state of the SAS Locking Motor
____ 6.) Send unlock trigger over Bluetooth using cellphone
____ 7.) Inspect the state of the SAS Locking Motor
____ 8.) Repeat Steps 4, 5, 6, and 7 until 10 trials have been completed
____ 9.) Power down electronics

Test Procedure for SAS Deployment Motor Control (CES-4)

____ 1.) Inform all bystanders of testing
____ 2.) Power on electronics
____ 3.) Confirm Bluetooth link with Feather M0 Bluefruit LE
____ 4.) Send unfold trigger over Bluetooth using cellphone
____ 5.) Inspect the state of the Solar Panel Support Arms
____ 6.) Send fold trigger over Bluetooth using cellphone
____ 7.) Inspect the state of the Solar Panel Support Arms

____ 8.) Repeat Steps 4, 5, 6, and 7 until 10 trials have been completed
____ 9.) Power down electronics

Test Procedure for OAS Control (CES-5)

____ 1.) Inform all bystanders of testing
____ 2.) Place objects of at least 4 in. tall at random locations in front of the rover
____ 3.) Power up electronics
____ 4.) Confirm Bluetooth link with Feather M0 Bluefruit LE
____ 5.) Place rover in front of field of objects
____ 6.) Send "Begin test" command using cellphone
____ 7.) Note performance as rover drives around objects
____ 8.) After the rover is at least 5 feet from any object, send "Stop test" command using cellphone
____ 9.) Power down electronics
____ 10.) Repeat steps 2 through 9 until 5 trials have been completed

Test Procedure for RDS Control (CES-6)

____ 1.) Inform all bystanders of testing
____ 2.) Mark setpoint locations on the ground
____ 3.) Place rover at start location
____ 4.) Power up electronics
____ 5.) Confirm Bluetooth link with Feather M0 Bluefruit LE
____ 6.) Send "Begin test" command using cellphone
____ 7.) Note the location of the rover at each point the rover attempts to change direction vs the location of the setpoints
____ 8.) Once the rover has reached the end setpoint, send "Stop test" command
____ using cellphone
____ 9.) Power down electronics
10.) Repeat steps 3 through 9 until 5 trials have been completed

Post-test Operations

- Analyze data from test to determine precision of control scheme
- Use data to refine control scheme if necessary
- Store data in secure locations for referencing

Suspension Criteria

- Any safety risk not accounted for is encountered
- Loss of power to electronics
- Error encountered in software leads to test abort

Success Criteria

Each test will be considered successful if the end of the test is reached without any intervention from team members and data collected is consistent with expected performance characteristics. Failure of any test will require a solution to be determined before restarting the test.

Safety Considerations

Proper safety precautions will be taken to avoid injury and damage to electronics. Grounding mats will be used to avoid electrostatic discharge when handling electronics. Pinch points will be identified prior to any work being done. No person will be allowed within 3 feet of rotating parts while they are operating. Tripping hazards will be identified and mitigated. All circuits will be continuity checked for shorts prior to any power being applied to avoid harm to personnel or components.

Results

The schedule for this testing series is listed below in Table 68

Test	**Scheduled Date**
SOW Verification 4.5.3	February 23rd, 2018
RLM Control	February 18th, 2018
SAS Locking Motor Control	February 19th, 2018
SAS Deployment Motor Control	February 20th, 2018
OAS Control	February 22nd, 2018
RDS Control	February 23rd, 2018

Table 68: Control testing schedule.

6.1.3.8. Battery Life Testing Series

Objective

The objective of this testing series is to ensure that the capacity of the chosen batteries matches the runtime calculated based on current draw from electronics. The requirements to be verified by this testing series are the following which are linked to their respective description: CES-8, CES-9.

Items/Variable to be Tested

- Controller Battery Lifetime
- Motor Batter Lifetime

Methodology

These tests will be unit/component level tests conducted analyze power consumption of the payload electronics. The results of these test will determine if larger or smaller capacity batteries are necessary for the mission. The pretest setup, post-test operations, and suspension criteria common among both tests are described below. The test procedure for each test is described individually in detail.

Pretest Setup

- Integrate all payload electronics with CES
- Load flight ready software on to the Feather M0 Bluefruit LE microcontroller
- Acquire timer

Test Procedure for Controller Battery Lifetime (CES-8)

____ 1.) Inform all bystanders of testing
____ 2.) Take voltage of battery using voltage monitor
____ 3.) Power up electronics and begin timer
____ 4.) Record voltage level on 10 minute increments
____ 5.) After 3 hours of continuous runtime, confirm electronics are still powered
____ 6.) Take voltage of battery using voltage monitor
____ 7.) Power down electronics

Test Procedure for Motor Battery Lifetime (CES-9)

____ 1.) Inform all bystanders of testing
____ 2.) Plug voltage monitor into battery and take initial reading
____ 3.) Power up electronics and start timer
____ 4.) Record voltage level on 15 second increments
____ 5.) After 5 minutes of total motor runtime, confirm motors are still being
____ powered
____ 6.) Power down electronics
7.) Record final voltage level using voltage monitor

Post-test Operations

- Plot data point of each voltage reading vs time at which reading was taken
- Analyze battery life trend

Suspension Criteria

- Any safety risk not accounted for is encountered
- Loss of power to electronics

Success Criteria

The tests will be considered successful only if the battery being tested outlasts the time frame allotted for its runtime. If at the completion of the controller battery test, the voltage has not dropped below 3.7V and is still powering the electronics, the test will be successful. If at the completion of the motor battery test, the voltage has not dropped below 6V and is still powering the motors, the test will be successful. The test will be considered a failure if power loss occurs at which point a larger capacity battery will be obtained and the test restarted.

Safety Considerations

Proper safety precautions will be taken to avoid injury and damage to electronics. Grounding mats will be used to avoid electrostatic discharge when handling electronics. All circuits will be continuity checked for shorts prior to any power being applied to avoid harm to personnel or components.

Results

These tests are scheduled to occur on February 25th, 2018.

6.1.3.9. **Flight Loads Testing Series**

Objective

The objective of this testing series is to ensure that all load bearing systems of the rover are capable of sustaining high loads experienced during flight of the full-scale launch vehicle. The requirements to be verified by this testing series are the following which are linked to their respective description: ROCS-4, RLM-4, RBS-3, SAS-5.

Items/Variable to be Tested

- Fidelity of Rover Orientation Correction System assembly under flight loads
- Fidelity of Rover Locking Mechanism assembly under flight loads
- Fidelity of Rover Body System assembly under flight loads
- Fidelity of Solar Array System solar panel support arms under flight loads

Methodology

These tests will be system level tests conducted corroborate analysis that has been done on systems and confirm the load bearing capabilities of these systems under full-scale flight loads. The results of this testing series will determine if any design changes need to be made in the interest of safety. The pretest setup, post-test operations, test procedure and suspension criteria common among all tests are described below.

Pretest Setup

- See payload preflight checklist in launch procedures

Test Procedure

____ 1.) Follow payload launch procedures

Post-test Operations

- Inspect systems for any deformation or damage

Suspension Criteria

- N/A

Success Criteria

The tests will be considered successful only if no deformation or impedance to performance of a system is seen after full-scale flight and recovery of the launch vehicle. A failure of these tests will indicate a safety risk and a necessary change to the design of the system to mitigate the risk.

Safety Considerations

Proper safety precautions will be taken to avoid injury. Adhere to all risk mitigation and safety procedures associated with full-scale flights. All circuits will be continuity checked for shorts prior to any power being applied to avoid harm to personnel or components.

Results

These tests are scheduled to occur on February 17th, 2018 for ROCS-4 and February 24th, 2018 for the rest during full-scale flights.

6.1.3.10. **Flight Full Performance Testing Series**

Objective

The objective of this testing series is to validate the flight performance of the DTS receiver antenna and the data logging capability of the CES. The requirements to be verified by this testing series are the following which are linked to their respective description: DTS-6, CES-7.

Items/Variable to be Tested

- Integrity of DTS receiver antenna
- Data logging capacity of the CES

Methodology

These tests will be unit/component level tests conducted verify a robust design has been implemented. The pretest setup, post-test operations, and suspension criteria common among both tests are described below. The test procedure for each test is described individually in detail.

Pretest Setup

- See payload preflight checklist in launch procedures

Test Procedure for DTS Receiver Antenna Fidelity (DTS-6)

____ 1.) Follow payload launch procedures
____ 2.) At landing, send deployment signal to receiver module
____ 3.) Inspect color of LED indicator to determine functionality of receiver antenna

Test Procedure for CES Data logging (CES-7)

____ 1.) Follow payload launch procedures
____ 2.) At landing, deploy rover

____ 3.) After rover has completed its mission, recover all data from CES microSD card

Post-test Operations

- Analyze data gathered to determine success of test

Suspension Criteria

- N/A

Success Criteria

The tests will be considered successful only if the DTS receiver recognizes the deployment signal for DTS-6 and the CES microSD card successfully logged data from the DTS deployment signal, two gyroscopes, OAS lidar sensor, autonomous drive commands, solar power generation levels, and SIS images for CES-7. A failure of DTS-6 will require minor modification to the exterior antenna. Failure of CES-7 will require minor changes to software.

Safety Considerations

Proper safety precautions will be taken to avoid injury. Adhere to all risk mitigation and safety procedures associated with full-scale flights. All circuits will be continuity checked for shorts prior to any power being applied to avoid harm to personnel or components.

Results

These tests are scheduled to occur on February 10th, 2018 for DTS-6 and February 24th, 2018 for CES-7 during full-scale flights.

6.1.4. Requirements Compliance

6.1.5. Launch Vehicle Requirements

In the following tables, the statement of work launch vehicle requirements are listed along with team derived requirements. Following each table are the derivations of each team derived requirement discussed in the table.

6.1.5.1. Requirement 2.1

Requirement Number	Requirement Description	Method of Verification	Status
2.1	The vehicle will deliver the payload to an apogee altitude of 5,280 feet above ground level (AGL).	**Analysis:** The launch vehicle shall be designed to reach an apogee altitude of 5,280 feet AGL. Several OpenRocket simulations as well as hand calculations will be performed to ensure the ideal motor is selected. The VDS will be tested to ensure an accurate altitude is achieved.	Complete

2.1.1	The vehicle will reach an apogee of 5,500ft AGL with an inactive VDS.	**Analysis:** The launch vehicle shall be designed using lightweight materials and the proper motor to overshoot the target altitude with an inactive VDS.	Complete
2.1.1.1	The launch vehicle's mass will not exceed 50 pounds.	**Analysis:** The launch vehicle shall be designed to use lightweight materials and the masses of each component shall be recorded during the design phase.	Complete
2.1.1.2	Hand calculations must be performed to verify OpenRocket simulations are accurate.	**Analysis:** Hand calculations shall be performed prior to CDR to ensure the OpenRocket simulations are accurate.	Complete
2.1.1.3	The ascent of the launch vehicle shall be safe and stable.	**Analysis:** The launch vehicle shall be designed to ascend safely. Safety checklists shall be written to ensure that no assembly steps will be missed.	Complete
2.1.1.4	The launch vehicle's overall coefficient of drag shall not exceed 0.50.	**Analysis:** CFD simulations shall simulate flight conditions and compute the coefficient of drag of the entire launch vehicle	Incomplete: Scheduled for late January 2018

6.1.5.1.1. Derivation of Requirement 2.1.1

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

6.1.5.1.2. Derivation of Requirement 2.1.1.1

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

6.1.5.1.3. Derivation of Requirement 2.1.1.2

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

6.1.5.1.4. Derivation of Requirement 2.1.1.3

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

6.1.5.1.5. Derivation of Requirement 2.1.1.4

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

6.1.5.2. Requirement 2.2

Requirement Number	Requirement Description	Method of Verification	Status
2.2	The vehicle will carry one commercially available, barometric altimeter for recording the official altitude used in determining the altitude award winner.	**Inspection:** A PerfectFlite StratoLogger CF altimeter will be used to record the official apogee altitude for the competition flight.	Incomplete: Scheduled for February 2018
2.2.1	All separation events will have two altimeters for redundancy.	**Inspection:** Two PerfectFlite StratoLogger CF altimeters will be used to initiate separation events during recovery. The altimeters will be programmed to be delayed relative to each other to prevent over pressurization of the airframe.	Incomplete: Scheduled for February 2018
2.2.2	All altimeters will be secured in the launch vehicle via a 3D printed sled.	**Analysis**: The launch vehicle shall be designed to accept 3D printed sleds for holding each altimeter. All 3D printed sleds will be designed prior to CDR.	Complete

6.1.5.2.1. Derivation of Requirement 2.2.1

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

6.1.5.2.2. Derivation of Requirement 2.2.2

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

6.1.5.3. Requirement 2.4

Requirement Number	Requirement Description	Method of Verification	Status
2.4	Each altimeter will have a dedicated power supply.	**Inspection:** Each Stratologger altimeter shall be powered by a dedicated 9-Volt battery.	Incomplete: Scheduled for February 2018
2.4.1	All altimeters will be powered by a Duracell 9-volt battery.	**Inspection:** Each altimeter shall be powered by a Duracell brand 9-Volt battery. Due to their internally welded contact points, Duracell brand batteries are well suited for the high accelerations experienced during flight.	Incomplete: Scheduled for February 2018
2.4.1.1	All batteries will be brand new and the voltage will be measured to be greater than 9.3 volts.	**Test:** Each battery will be new and measured with a digital multi-meter to assure that the voltage is greater than 9.3 volts.	Incomplete: Scheduled for February 2018
2.4.2	All batteries will be secured onto their 3D printed sled.	**Inspection:** Each battery will be secured to its 3D printed sled using a cover secured to the sled via four screws.	Incomplete: Scheduled for February 2018

6.1.5.3.1. Derivation of Requirement 2.4.1

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

6.1.5.3.2. Derivation of Requirement 2.4.1.1

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

6.1.5.3.3. Derivation of Requirement 2.4.2

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

6.1.5.4. Requirement 2.6

Requirement Number	Requirement Description	Method of Verification	Status

2.6	The launch vehicle will be designed to be recoverable and reusable. Reusable is defined as being able to launch again on the same day without repairs or modifications.	**Analysis:** The launch vehicle shall be designed to be fully recoverable and reusable. Several test launches will be conducted before competition to verify recoverability and reusability of the launch vehicle.	Complete
2.6.1	The launch vehicle will utilize the Removable Fin System (RFS).	**Analysis:** The launch vehicle will be designed to accommodate the RFS and to remove and install fins from the launch vehicle.	Complete
2.6.1.1	There will be two extra fins available for use on launch day if any fins are broken during transportation or launch of the launch vehicle.	**Inspection**: The team will prepare a minimum of two extra fins before launches and account for the added expense in the team budget. This will be in conjunction with requirement 2.6.1.	Incomplete: Scheduled for February 2018

6.1.5.4.1. Derivation of Requirement 2.6.1

For the launch vehicle to be fully reusable, the ability to replace broken fins becomes a necessity. Without the RFS, a broken fin would require complete reconstruction of the booster section. This would be a setback to the team's schedule and budget. For this reason, the RFS is required to be used on the launch vehicle to prepare for the possibility of a fin becoming damaged during a test flight.

6.1.5.4.2. Derivation of Requirement 2.6.1.1

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

6.1.5.5. Requirement 2.9

Requirement Number	Requirement Description	Method of Verification	Status
2.9	The launch vehicle will be capable of being prepared for flight at the launch site within 3 hours of the time the Federal Aviation Administration flight waiver opens.	**Demonstration:** Assembly of the launch vehicle shall take less than 3 hours. A comprehensive checklist will be prepared by the team to assist in accurate and expedited vehicle assembly while preparing for flight.	Incomplete: Scheduled for February 2018
2.9.1	The launch vehicle will be capable of being	**Demonstration:** The launch vehicle shall be designed to be	Incomplete: Scheduled

	assembled in two hours independent of the payload.	assembled in less than two hours. A comprehensive checklist will be prepared by the team to assist in accurate and expedited vehicle assembly while preparing for flight. The team shall practice assembling the launch vehicle prior to competition to ensure Requirement 2.9.1 is met. This will be in accordance with Requirement 2.9.	for February 2018
2.9.2	The payload subsystem will be capable of being assembled independent of the launch vehicle.	**Analysis:** The launch vehicle shall be designed so that the rover payload can be assembled separate of the launch vehicle.	Complete
2.9.2.1	The payload subsystem will be capable of being integrated into the launch vehicle within 30 minutes.	**Demonstration:** The launch vehicle shall be designed so that the rover payload can be integrated into the launch vehicle within 30 minutes. The team shall practice integrating the payload into the launch vehicle to verify Requirement 2.9.2.1 is met.	Incomplete: Scheduled for February 2018
2.9.3	The launch vehicle will utilize witness triangles.	**Inspection:** The launch vehicle shall utilize witness triangles on each coupler.	Incomplete: Scheduled for February 2018
2.9.4	The motor will be packed at least 24 hours prior to the launch.	**Demonstration:** A minimum of 24 hours prior to launch, the launch vehicle's motor shall be packed.	Incomplete: Scheduled for February 2018

6.1.5.5.1. Derivation of Requirement 2.9.1
To prepare for launch day and ensure that the launch vehicle meets Requirement 2.9, the launch vehicle is required to be capable of being assembled in two hours. This provides a buffer of one hour in the event of unexpected delays during assembly.

6.1.5.5.2. Derivation of Requirement 2.9.2
To prepare for launch day and ensure that the launch vehicle meets Requirement 2.9, the payload subsystem is required to be capable of being assembled independent of the launch vehicle. This requirement allows for the payload and vehicle sub-teams to work independently on launch day. This will result in an overall more efficient assembly process and ensure that no issues with the payload subsystem can affect the assembly time of the launch vehicle.

6.1.5.5.3. Derivation of Requirement 2.9.2.1
To prepare for launch day, and ensure that the launch vehicle meets Requirement 2.9, the payload subsystem is required to be capable of being integrated into the launch vehicle within 30 minutes.

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

6.1.5.5.4. Derivation of Requirement 2.9.3

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

6.1.5.5.5. Derivation of Requirement 2.9.4

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

6.1.5.6. Requirement 2.10

Requirement Number	Requirement Description	Method of Verification	Status
2.10	The launch vehicle will be capable of remaining in launch-ready configuration on the pad for a minimum of 1 hour without losing the functionality of any critical on-board components.	**Demonstration:** The power supplies for the VDS and recovery electronics shall demonstrate the capability to exceed Requirement 2.10.	Incomplete: Scheduled for February 2018
2.10.1	All electronics onboard the launch vehicle will be capable of remaining in launch-ready configuration at the pad for a minimum of 1.5 hours without losing functionality of any on-board components.	**Demonstration:** The power supplies for the VDS and recovery electronics shall demonstrate the capability to exceed Requirement 2.10.	Incomplete: Scheduled for February 2018
2.10.2	All electronics onboard the launch vehicle will be capable of remaining in launch-ready configuration at temperatures of 120°F.	**Analysis:** The components chosen for the systems shall be considered for their ability to withstand high temperatures while still retaining functionality.	Complete

6.1.5.6.1. Derivation of Requirement 2.10.1

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

6.1.5.6.2. Derivation of Requirement 2.10.2

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

6.1.5.7. Requirement 2.11

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

6.1.5.7.1. Derivation of Requirement 2.11.1

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

6.1.5.8. Requirement 2.12

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

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

6.1.5.8.1. Derivation of Requirement 2.12.1

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

6.1.5.8.2. Derivation of Requirement 2.12.2

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

6.1.5.9. Requirement 2.16

Requirement Number	Requirement Description	Method of Verification	Status
2.16	The launch vehicle will have a minimum static stability margin of 2.0 at the point of rail exit. Rail exit is defined at the point where the forward rail button loses contact with the rail.	**Analysis:** The launch vehicle shall be designed to have a stability of 2.0 at the point of rail exit. OpenRocket simulations shall be used to design the launch vehicle to the stability margin required and hand calculations shall be used to verify the stability.	Complete
2.16.1	The launch vehicle will have a stability margin at rail exit of 2.2.	**Analysis:** The launch vehicle shall be designed to have a stability margin of 2.2 upon rail exit.	Complete

6.1.5.9.1. Derivation of Requirement 2.16.1

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

6.1.5.10. Requirement 2.17

Requirement Number	Requirement Description	Method of Verification	Status

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

6.1.5.10.1. Derivation of Requirement 2.17.1

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

6.1.5.11. Requirement 2.18

Requirement Number	Requirement Description	Method of Verification	Status
2.18	All teams will successfully launch and recover a subscale model of their rocket prior to CDR. Subscales are not required to be high power rockets.	**Test:** The team shall design, build, and test a subscale model of the rocket prior to CDR.	Complete
2.18.1	The subscale model should resemble and perform as similarly as possible to the full-scale model, however, the full-scale will not be used as the subscale model.	**Analysis:** The subscale model of the rocket shall be designed to closely follow the design of the full-scale model to ensure similar performance and flight characteristics.	Complete
2.18.1.1	The subscale model will be a half-scale replica to the full-scale launch vehicle.	**Analysis:** The subscale model shall be built as a 1:2 scale to be in order to test recovery design decisions and to identify potential design obstacles.	Complete
2.18.2	The subscale model will carry an altimeter capable of reporting the model's apogee altitude.	**Inspection:** Stratologger CF altimeters shall be mounted in the subscale model in order to report the model's apogee altitude.	Complete

2.18.2.1	The subscale model will carry two Stratologger altimeters powered by 9-volt Duracell batteries.	**Inspection:** The subscale model shall carry two Stratologger altimeters powered independently by new 9-volt Duracell batteries for redundancy.	Complete

6.1.5.11.1. Derivation of Requirement 2.18.1.1

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

6.1.5.11.2. Derivation of Requirement 2.18.2.1

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

6.1.5.12. Requirement 2.19

Requirement Number	Requirement Description	Method of Verification	Status
2.19	All teams will successfully launch and recover their full-scale rocket prior to FRR in its final flight configuration.	**Test:** The team will conduct a test launch prior to FRR to verify the success of all launch vehicle systems.	Incomplete: Scheduled for February 2018
2.19.2.2	The payload will successfully complete its mission on at least one occasion prior to competition.	**Test:** The rover will be tested to drive five feet and deploy its solar panels in a realistic environment prior to competition.	Incomplete: Scheduled for March 2018
2.19.8	The launch vehicle will undergo at least four full-scale test flights prior to competition.	**Test:** Several full-scale launches will be conducted to identify potential problems and to verify repeatability of successful launch vehicle systems.	Incomplete: Scheduled for February-March 2018

6.1.5.12.1. Derivation of Requirement 2.19.2.2

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

6.1.5.12.2. Derivation of Requirement 2.19.8

To be prepared for competition, the launch vehicle will be required to undergo at least four full-scale test flights. The first of which will serve as a control launch with an inactive VDS, the second will serve as a full-brake test for the VDS, and the third and fourth will serve as performance tests for the VDS and payload subsystems. These test flights will also ensure that any issues with recovery can be solved by competition.

6.1.5.13. Requirement 2.22

Requirement Number	**Requirement Description**	**Method of Verification**	**Status**
2.22	All structural components of the launch vehicle will undergo Finite Element Analysis to ensure an efficient and structurally sound design.	**Analysis:** During the design phase of the launch vehicle, each structural component will undergo Finite Element Analysis.	Complete

6.1.5.13.1. Derivation of Requirement 2.22

To ensure that all structural components have been designed to be as efficient and safe as possible, all structural part models will be analyzed using finite element analysis software.

6.1.5.14. Requirement 2.23

Requirement Number	Requirement Description	Method of Verification	**Status**
2.23	All structural components of the launch vehicle will undergo structural testing to ensure an efficient and structurally sound design.	**Test:** During the construction phase of the launch vehicle, each structural component will undergo structural testing.	Incomplete. Scheduled for February 2018

6.1.5.14.1. Derivation of Requirement 2.23

To verify that the results of the finite element analysis are correct, all structural components of the launch vehicle will undergo structural testing. This will give the team confidence that the launch vehicle will experience no structural failures during flight.

6.1.5.15. Requirement 2.24

Requirement Number	Requirement Description	Method of Verification	**Status**
2.24	Prior to each launch, the launch vehicle shall be fully assembled and undergo black powder separation testing.	**Test:** The launch vehicle shall undergo separation testing with the calculated amount of black powder needed to separate each section.	Incomplete: Scheduled for February 2018

6.1.5.15.1. Derivation of Requirement 2.24

To verify correct black powder quantity calculations and shear pin strengths, the launch vehicle will undergo black powder separation testing prior to each launch. The testing will give the team confidence that the launch vehicle will separate as designed during flight and allow for a successful recovery. Under no circumstances shall the launch vehicle be launched without undergoing black powder separation tests.

6.1.5.16. SOW Verifications

Requirement Number	Requirement Description	Method of Verification
3.1	The launch vehicle will stage the deployment of its recovery devices, where a drogue parachute is deployed at apogee and a main parachute is deployed at a lower altitude. Tumble or streamer recovery from apogee to main parachute deployment is also permissible, provided that kinetic energy during drogue-stage descent is reasonable, as deemed by the RSO.	**Analysis:** The StratologgerCFs shall be programed such that the payload section and booster section drogue parachutes will be deployed at apogee and both main parachutes will be deployed at 500 ft.
3.2	Each team must perform a successful ground ejection test for both the drogue and main parachutes. This must be done prior to the initial subscale and full-scale launches.	**Demonstration:** Black powder tests shall be repeated until nominal effects are observed a minimum of two times during initial testing and a minimum of one time before each flight of the launch vehicle.
3.3	At landing, each independent sections of the launch vehicle will have a maximum kinetic energy of 75 ft-lbf.	**Analysis:** All parachute sizes shall be calculated according to the kinetic energy requirement such that all sections of the launch vehicle shall land with a kinetic energy under 75 ft-lb.
3.4	The recovery system electrical circuits will be completely independent of any payload electrical circuits.	**Inspection:** All StratologgerCFs will be independent of any surrounding electrical circuits and will be individually powered by Duracell 9V batteries.
3.5	All recovery electronics will be powered by commercially available batteries.	**Inspection:** Each StratologgerCF shall be powered by a new Duracell 9-Volt battery. Duracell batteries are chosen because of their internally soldered leads which makes them highly reliable.
3.6	The recovery system will contain redundant, commercially available altimeters. The term "altimeters" includes both simple altimeters and more sophisticated flight computers.	**Inspection:** The recovery system shall contain four PerfectFlite StratologgerCF altimeters. Two StratologgerCFs will be included in each recovery bay for redundancy.
3.7	Motor ejection is not a permissible form of primary or secondary deployment.	**Inspection:** The launch vehicle shall use an AeroTech L2200 which does not include an ejection charge after the burnout of the propellant grain.
3.8	Removable shear pins will be used for both the main parachute compartment and the drogue parachute compartment.	**Inspection:** Three nylon 4-40 socket head cap screw shear pins shall be used for each recovery bay intended to separate.

3.9	Recovery area will be limited to a 2500 ft. radius from the launch pads.	**Analysis:** Drift calculations will be done to ensure that all components of the launch vehicle shall be recovered within 2500 ft. of the launch rail.
3.10	An electronic tracking device will be installed in the launch vehicle and will transmit the position of the tethered vehicle or any independent section to a ground receiver.	**Inspection:** Each independent section shall include a Trackimo GPS tracking device.
3.10.1	Any rocket section, or payload component, which lands untethered to the launch vehicle, will also carry an active electronic tracking device.	**Inspection:** A Trackimo GPS tracking device shall be secured in each independent section of the launch vehicle, and will transmit the position of the launch vehicle to a ground receiver. The payload shall also carry a GPS tracking device.
3.10.2	The electronic tracking device will be fully functional during the official flight on launch day.	**Demonstration:** The electronic tracking devices shall be tested and monitored during all test flights prior to the official flight to ensure that they are fully functional and used on the official launch day.
3.11	The recovery system electronics will not be adversely affected by any other on-board electronic devices during flight (from launch until landing).	**Analysis:** All on-board electronic devices shall be designed not to interfere or affect the recovery system electronics.
3.11.1	The recovery system altimeters will be physically located in a separate compartment within the vehicle from any other radio frequency transmitting device and/or magnetic wave producing device.	**Inspection:** All StratologgerCFs shall be placed in a separate section of the launch vehicle from any other radio frequency transmitting device and/or magnetic wave producing device.
3.11.2	The recovery system electronics will be shielded from all onboard transmitting devices, to avoid inadvertent excitation of the recovery system electronics.	**Analysis:** The recovery devices shall be designed to be shielded from all transmitting devices on the launch vehicle.
3.11.3	The recovery system electronics will be shielded from all onboard devices which may generate magnetic waves (such as generators, solenoid valves, and Tesla coils) to avoid inadvertent excitation of the recovery system.	**Analysis:** The recovery devices shall be designed to be shielded from all transmitting devices on the launch vehicle that generate magnetic waves.

3.11.4	The recovery system electronics will be shielded from any other onboard devices which may adversely affect the proper operation of the recovery system electronics.	**Demonstration:** The recovery system electronics will be tested during test flights of the launch vehicle to ensure that any other onboard electronic devices to not interfere with the performance of the recovery system electronics.

Table 69: SOW requirement verification

6.1.5.17. Team Derived Requirements

In addition to the Requirements dictated in the SOW outlined in Table , self-derived requirements have been made to ensure a safe recovery for the launch vehicle, the onboard electronics, and the crowd. These requirements will be named as TDR.x.x to show they are self-derived and separate from the SOW requirements. These requirements are shown in Figure 168 below.

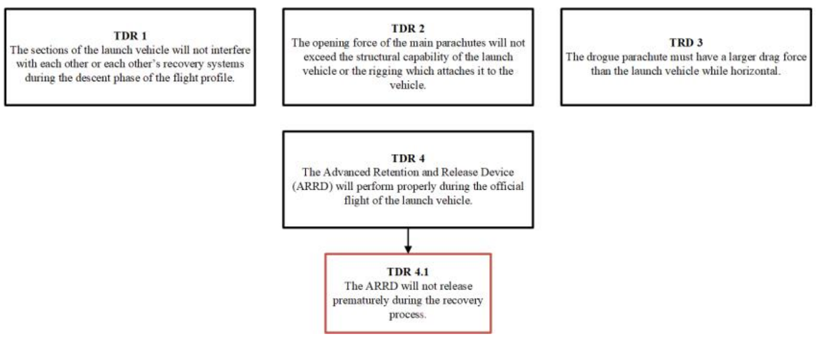


Figure 168: Self derived requirement flowchart

These requirements and verification are detailed below in Table 70.

Requirement Number	Requirement Description	Method of Verification
TDR 1	The sections of the launch vehicle will not interfere with each other or each other's recovery systems during the descent phase of the flight profile.	**Demonstration:** The recovery system shall be designed so that the booster drogue parachute will be deployed at apogee, and after a two second delay, the payload section drogue parachute will be deployed. This will ensure that the payload and booster sections of the launch vehicle will descend at different altitudes to prevent interaction. This system shall be proven during test flights of the launch vehicle prior to the official competition flight.

TDR 2	The opening force of the main parachutes will not exceed the structural capability of the launch vehicle or the rigging which attaches it to the vehicle.	**Test:** The main parachutes shall each have an opening force reduction ring which will create friction with the shroud lines and reduce the opening forces. This system shall be proven during test flights of the launch vehicle and data will be collected via sensors on-board the payload and booster sections of the launch vehicle. The data will be analyzed, and the size of the opening force reduction ring will be adjusted to ensure that the opening force of the main parachutes will not exceed 500 ft-lb, maintaining a factor of safety larger than two.
TDR 3	The drogue parachute must have a larger drag force than the launch vehicle while horizontal.	**Analysis:** The drogue parachute shall be designed to have a larger drag force than the launch vehicle while horizontal. ANSYS fluid simulation will be used to verify calculations.
TDR 4	The Advanced Retention and Release Device (ARRD) will perform properly during the official flight of the launch vehicle.	**Demonstration:** Ground tests shall be done on the ARRD to ensure that it will perform properly during all flights of the launch vehicle. Test flights prior to the competition flight shall also test the reliability of the ARRD to ensure that it will perform as planned during the competition flight.
TDR 4.1	The ARRD will not release prematurely during the recovery process.	**Demonstration:** The ARRD shall be load tested to at least 400 lbs to ensure that it will not release. If release occurs, the ARRD shall be reassembled and tested again to ensure that it will not release prematurely during the recovery process.

Table 70: Team derived requirement verifications

6.1.6. VDS Team Requirements Derivation and Verification

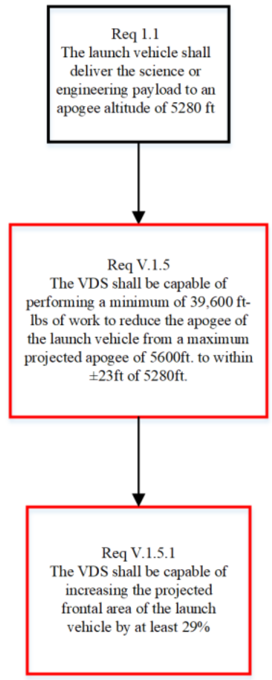
Requirement number	Requirement Description	Method of Verification	Status
V.1.1	The VDS will autonomously actuate its drag blades, and alter the drag of the rocket to achieve an altitude of ±23 feet.	**Test:** The actuation method will be tested independently of the launch vehicle, as well as through several test launches to verify the system.	Incomplete. Scheduled for February 24th. This Requirement will be satisfied before the first full scale launch. 6.1.1.5

V.1.1.1	The VDS must actuation method must provide continuous control over actuation and retraction of the drag blades.	**Demonstration:** Three drag inducing blades set with radial gear teeth that mesh with a central spur gear will allow for continuous control by a single DC motor.	Incomplete. Scheduled for February 24th. This Requirement will be satisfied before the first full scale launch.
V.1.1.2	The drag inducing blades must fully actuate in less than 0.5 seconds.	**Test:** An DC motor with a torque ratio which provides the fastest actuation speed with	Complete December 4th. DC motor Andymark Neverest40 Motors chosen meet requirements.6.1.1.2
V.1.1.3	The DC motor shall not experience a torque greater than 388 oz-in during the actuation of the drag blades.	**Analysis:** The friction force between the drag blades and support plates will be calculated to determine the required torque to actuate the drag blades.	Complete December 4th. DC motor Andymark Neverest40 Motors chosen meet requirements.
V.1.1.3.1	The coefficient of friction between the drag blades and the support surface must be lower than 0.5 to provide a bearing surface for the drag blades.	**Demonstration:** Delrin Acetyl Resin was chosen for the bearing surface because it has a coefficient of friction of approximately 0.3 with Aluminum.	Complete November 20th. Calculations completed on friction of materials chosen.
V.1.1.4	The actuation method must provide simultaneous actuation of all three drag blades.	**Test:** A prototype of the VDS V3 gear assembly will be manufactured to verify that the design will provide a simultaneous actuation.	Incomplete. Scheduled for February 17th. 6.1.1.4 .
V.1.1.5	The drag blades shall not over-actuate beyond their mechanical limit.	**Demonstration:** Two limit switches will be fastened to the top VDS aluminum support plate, and communicate with the DC motor to prevent over actuation.	Incomplete. Scheduled for January 25th. This Requirement will be satisfied before the first full scale launch.
V.1.1.6	No component of the actuation device may be damaged as a result of the actuation.	**Analysis:** Finite Element Analysis was performed using ANSYS Workbench to verify the structural	Complete on November 28th. Finite element analysis was completed on all components.

		integrity of the gear mesh and drag blades.	

Requirement number	Requirement	Method of Verification	Status
V.1.2	The VDS will telemetrically communicate its current state to a ground station during flight, without altering the path of the vehicle.	**Test:** This requirement will be verified in sub scale flight testing. The tests will verify whether the range and data transmission rates are of acceptable standards for the needs of the VDS.	Semi-Complete. Test completed on January 4th, but further testing is expected to verify antenna fidelity on January 18th. 6.1.1.1
V.1.3	The VDS shall have access to power controls externally from the vehicle.	**Demonstration:** The VDS bay will contain a port which grants access to the power source of the VDS in order to prevent power depletion during integration.	Incomplete. Will be completed on January 25th.
V.1.4	The VDS shall be capable of determining the state of the vehicle (i.e. altitude and velocity) with noise limits of no more than ± 5.0 m and ± 5.0 m/s respectively.	**Demonstration:** The VDS will take data points and employ the use of a built in sensory Kalman filter in order to reduce noise in data intake.	Incomplete. Will be completed upon first full scale launch on February 10th..

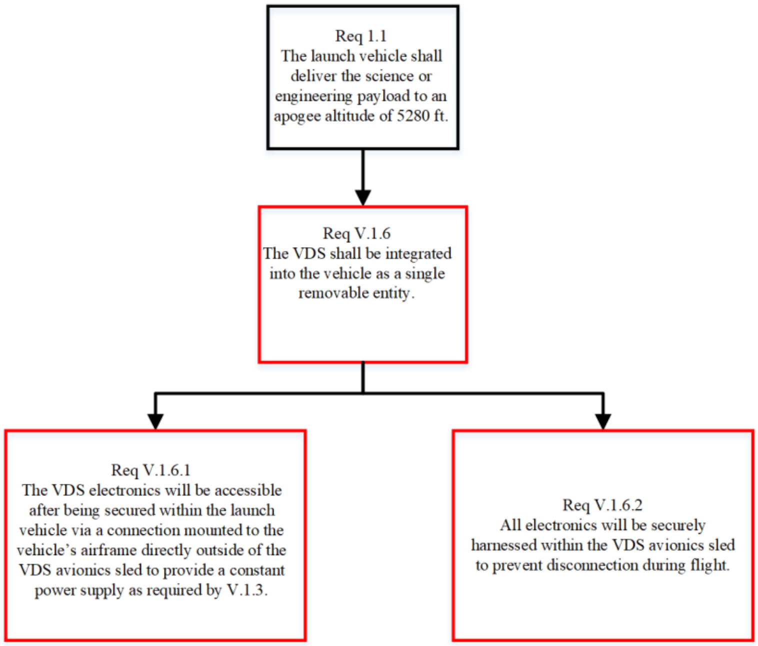
6.1.6.1. Verification of Braking Power Requirements



Requirement number	Requirement	Method of Verification	Status
V.1.5	The VDS shall be capable of performing a minimum of 39,600 ft-lbs of work to reduce the apogee of the launch vehicle from a maximum projected apogee of 5600ft. to within ±23ft of 5280ft.	**Test:** Multiple launches will be conducted to verify that the VDS can perform the required amount of work on the launch vehicle.	Complete. Calculations and simulations have been completed to verify numerical expectations of VDS mechanics. 6.1.1.5
V.1.5.1	The VDS shall be capable of increasing the projected frontal area of the launch vehicle by at least 29%	**Inspection:** Computer aided design software was used to verify that the VDS V3 will increase the projected area of the rocket by at least 29%	Complete. Calculations and simulations have been completed to verify numerical expectations of VDS mechanics.

Derivation of requirement V.1.5

6.1.6.2. Verification of VDS Integration Method



Requirement number	Requirement	Method of Verification	Status
V.1.6	The VDS shall be integrated into the vehicle as a single removable entity.	**Demonstration:** CAD software will be used to ensure that all components of the VDS fit within a single 6 in. x 12 in. carbon fiber coupler.	Incomplete. Will be completed upon integration into fully manufactured full scale vehicle.
V.1.6.1	The VDS electronics will be accessible after being secured within the launch vehicle via a connection mounted to the vehicle's airframe directly outside of the VDS avionics sled to provide a constant power supply as required by V.1.3.	**Test:** The BeagleBone Green computer will be connected via USB cable and will confirm connection through PC.	Incomplete. Will be completed upon integration into fully manufactured full scale vehicle. 6.1.1.2, **Error! Reference source not found.**
V.1.6.2	All electronics will be securely harnessed within the VDS avionics sled to prevent disconnection during flight.	**Inspection:** All components will be visually inspected to ensure that all fasteners have been set in place before inserting the VDS into the launch vehicle.	Complete. Solidworks model of avionics sled has been created and wiring plan has been made.

6.1.7. NASA Requirements Verification Payload

Req. ID	Requirement	Verification Method	Status
4.1	Each team will choose one design option from the following list.	Deployable rover has been chosen from the provided list.	Complete
4.2	Additional experiments (limit of 1) are allowed, and may be flown, but will not contribute to scoring.	No additional experiments will be flown.	Complete
4.3	If the team chooses to fly additional experiments, they will provide the appropriate documentation in all design reports, so experiments may be reviewed for flight safety.	N/A	N/A
4.5.1	Teams will design a custom rover that will deploy from the internal structure of the launch vehicle.	Inspection The team will design a custom rover vehicle capable of deploying from the internal structure of the launch vehicle and all necessary subsystems to accomplish this task using the computer aided design (CAD) software SolidWorks.	Complete
4.5.2	At landing, the team will remotely activate a trigger to deploy the rover from the rocket.	Demonstration The rover and the rest of the payload will be integrated into the launch vehicle in full flight configuration. An external transmitter module will be held by a team member. The launch vehicle will be placed on the ground simulating landing. The team member will trigger the deployment signal to be sent to the payload. After the rover is seen to begin deployment only after the deployment signal has been sent, verification will be complete and considered successful.	Incomplete - Scheduled for February 26th, 2018
4.5.3	After deployment, the rover will autonomously move at least 5 ft. (in any direction) from the launch vehicle.	Test The rover and the rest of the payload will be integrated into the launch vehicle in full flight configuration. The rover's control system will be running flight ready software. The deployment signal will be sent to the payload by a team member using the transmitter module. Upon receiving the deployment signal, the rover will exit the launch vehicle and continue driving at least 5 ft. with no	Incomplete - Scheduled for February 23rd, 2018

		external intervention. After reaching at least 5 ft., verification will be complete and considered successful. See [Rover Performance Test Plan.]	
4.5.4	Once the rover has reached its final destination, it will deploy a set of foldable solar cell panels.	Demonstration The rover has been designed with an onboard actuating foldable solar array. The rover will be fully constructed, assembled and configured running software representative of the rover having begun to exit the launch vehicle. The rover will drive at least 5 ft. at which point it will stop and deploy a set of foldable solar cell panels with no external intervention. After visual confirmation of an increase in surface area of exposed solar cells, verification will be complete and considered successful.	Incomplete - Scheduled for February 24th, 2018

6.1.8. Payload Team Requirements Derivation and Verification

Subsystem	**Requirement**	**Verification Method**	**Status**
Payload ROCS-1	The ROCS assembly shall maintain a factor of safety of two at minimum.	Analysis SolidWorks will be used to generated CAD models of each subassembly of the ROCS. Analysis will be performed using SolidWorks FEA tools on each subassembly and the system as a whole under high load conditions representative of those expected during flight. The maximum loads allowable before yielding will then be determined and compared to the results of the analysis done at the expected flight loads to determine factors of safety. After analysis confirms a minimum factor of safety of two for the ROCS assembly and each subassembly, the verification will be complete and considered successful.	Complete

Payload ROCS-2	The entire ROCS assembly shall be capable of being integrated and removed from the airframe within five minutes.	<u>Demonstration</u> The ROCS will be manufactured and fully assembled. The payload bay of the launch vehicle will be placed on a table next to the ROCS. A team member will attempt to integrate the ROCS into the payload bay and subsequently remove it. A stopwatch will be started at the moment that the team member touches the ROCS and stopped at the moment that the ROCS is fully removed and the team member need not touch it anymore. If at any point, a design issue is encountered, the trial will be halted and the issue will be assessed at that time. After three consecutive successful trials, the verification will be complete and considered successful.	Incomplete - Scheduled for February 5th, 2018
Payload ROCS-3	After being rolled, The ROCS shall come to rest in such a way that the bridging sled is nearest to the ground and the rover is oriented within 50° of perfectly upright.	<u>Test</u> The ROCS will be manufactured and fully assembled. The rover will be fully assembled and integrated onto the ROCS bridging sled as intended for its flight configuration. The full assembly will be integrated into the payload bay of the launch vehicle. Two BNO055 gyroscopes will be collecting and logging gyroscopic data onboard the rover. The Feather M0 Bluefruit LE microcontroller and FeatherWing Adalogger will be used to analyze and collect the sensor data. Two RGB LEDs will indicate the status of each gyroscopes readings. If the orientation is within 50° of upright, the LED will be turned green and if the orientation is greater than 50° the LED will be turned red. The payload bay will be placed on an inclined surface and released. After coming to rest, the color of the two LEDs will be noted and the data stored on the Adalogger will be graphed and analyzed. A trial will be considered successful if both LEDs show green after the payload bay has come to rest. After 10 consecutive successful trials, the verification will be complete and considered successful. See ROCS Roll Test Plan.	Incomplete - Scheduled for February 10th, 2018

Payload ROCS-4	The ROCS shall sustain high loads experienced during liftoff, opening of the payload bay main parachute, and landing.	<u>Test</u> The ROCS will be manufactured and fully assembled. The systems will be integrated into the launch vehicle prior to a full-scale test launch. After landing of the payload bay, the ROCS will be removed and thoroughly inspected. After no significant deformation or damage is seen on the ROCS or payload bay, the verification will be complete and considered successful. See Flight Loads Testing Series	Incomplete - Scheduled for February 17^{th}, 2018
Payload RLM-1	The RLM shall maintain a factor of safety of two at minimum.	<u>Analysis</u> SolidWorks will be used to generated CAD models of each subassembly of the RLM. Analysis will be performed using SolidWorks FEA tools on each subassembly and the system as a whole under high load conditions representative of those expected during flight. The maximum loads allowable before yielding will then be determined and compared to the results of the analysis done at the expected flight loads to determine factors of safety. After analysis confirms a minimum factor of safety of two for the RLM assembly and each subassembly, the verification will be complete and considered successful.	Complete. See section 5.1.11.2
Payload RLM-2	The RLM shall keep the rover fixed to the ROCS and immobile relative to the bridging sled regardless of orientation.	<u>Demonstration</u> The ROCS, RLM, and RBS will be manufactured and fully assembled. The RBS will be integrated into the ROCS via the RLM and the system will be placed in the locked configuration. The entire assembly will then be oriented in all possible ways. If any movement of the RBS is seen, the trial will be halted and note will be made of the movement, the location of the slack that allowed movement, and the orientation of the assembly when the movement was seen. A minimum of three people will observe each demonstration. After five consecutive trials in which no movement was detected, the verification will be complete and considered successful.	Incomplete - Scheduled for February 6th, 2018

Payload RLM-3	The RLM motor shall fully release the rover after the deployment signal has been sent allowing the rover to move under its own power.	Demonstration The RLM and RBS will be manufactured and fully assembled. The rover will be integrated into the RLM and shown to be locked in place. The RLM motor will be powered using 11.1V such that the motor releases the locking mechanism. If the mechanism does not immediately unlock the rover, the demonstration will be halted and a cause will be determined and attended to. After five consecutive successful releases, the verification will be complete and considered successful.	Incomplete - Scheduled for February 6th, 2018
Payload RLM-4	The RLM shall retain the rover inside the airframe of the launch vehicle throughout the duration of the flight and recovery.	Test The payload will be manufactured and fully assembled. The payload will then be integrated into the payload bay of the launch vehicle prior to a full-scale test launch. Cameras will be mounted inside of the payload bay. After landing, the payload and footage from the payload bay cameras will be inspected. After confirming that the rover has been retained inside the launch vehicle at all stages of flight, the verification will be complete and considered successful. See Flight Loads Testing Series.	Incomplete - Scheduled for February 24th, 2018
Payload DTS-1	The DTS transmitter and receiver modules shall transmit at a frequency and power level allowable without a license.	Demonstration Two HC-12 Wireless Serial Modules will be obtained and configured for communication at 433.4 MHz and transmission power less than 100 mW. Each module will be connected to a Feather M0 Bluefuit LE microcontroller running software intended to test the communication between he wireless modules. After successful communication is achieved at the specified frequency and power level, the verification will be complete and considered successful.	Complete

Payload DTS-2	The DTS slip ring flange shall allow the communication and power wires to pass through the upper payload bay recovery bulkplate without tangling during any rotation of the rover on the ROCS.	<u>Demonstration</u> The DTS will be configured to maintain communication and the received data will be viewed in real-time on a computer terminal window. The communication and power wires from the microcontroller to the receiver module will be connected through the slip ring flange. The receiver microcontroller will be rotated clockwise and counter-clockwise 10 times each. If at any point, communication or power is lost, the trial will be halted and a solution will be determined. After 10 consecutive successful trials, the verification will be complete and considered successful.	Incomplete - Scheduled for January 17th, 2018
Payload DTS-3	The DTS power and communication lines from the CES will detach by means of the rover driving forward.	<u>Demonstration</u> The RDS and RBS will be manufactured and fully assembled. The Feather M0 Bluefruit LE microcontroller and FeatherWing Motor Shield will be used to control the two main drive motors of the RDS via Bluetooth from a team member's cellphone. This is to represent as accurately as possible the situation of autonomous mission performance while still maintaining full control of the rover. The power and communication wires for the DTS will be connected using breadboard jumper wires. The rover will be driven forward while the receiver module is held in place. At full extension, the wires will disconnect without hindering the forward motion of the rover. After five consecutive disconnect trials, the verification will be complete and considered successful.	Incomplete - Scheduled for January 17th, 2018
Payload DTS-4	The DTS receiver and transmitter modules shall maintain communication within a distance	<u>Test</u> The DTS will be configured to maintain communication and the received data will be viewed in real-time on a computer terminal window. The receiver module will be connected to a flexible antenna wrapped around the exterior of a section of 6 in. carbon fiber airframe representative of the payload bay. The carbon fiber section will be placed on the ground 50 linear feet from the transmitter module. A team member will carry the	Incomplete - Scheduled for January 18th, 2018

	of 50 linear feet of each other.	transmitter along two perpendicular diameters of a 50 foot radius circle centered on the receiver. After five trials with no downlink in communication, the verification will be complete and considered successful. See DTS 50 Foot Radius Test.	
Payload DTS-5	The DTS receiver module shall receive the unique deployment signal data packet and relay the information to the CES.	Demonstration An HC-12 Wireless Serial Module will act as the receiver and be connected to a Feather M0 Bluefuit LE microcontroller. A second HC-12 will act as the transmitter and be connected to a second Feather M0 and a yagi antenna designed to operate at 433.4 Mhz transmission frequency. A single RGB LED will be connected to the receiver side Feather M0. Software on the transmitter side will be configured to send multiple different data streams representing outside signals not sent to deploy the rover as well as a unique packet sent to deploy the rover. Software for the receiver side will be configured to look for the unique packet of data. When this packet is received, the LED will be turned green. Otherwise, the LED will be turned red. After five consecutive trials of successful reception and recognition of the unique packet and rejection of all other signals, verification will be complete and considered successful.	Incomplete - Scheduled for January 19th, 2018
Payload DTS-6	The DTS receiver antenna shall remain functional after flight and landing of the launch vehicle.	Test The DTS will be assembled in full flight configuration and flight ready software will be loaded onto the transmitter and receiver microcontrollers. The DTS will be integrated into the launch vehicle prior to a full-scale test launch. After landing, a team member will walk to within 50 feet of the payload bay and transmit a packet of data to the receiver module. Upon receiving the packet of data, the receiver module microcontroller will turn an RGB LED green indicating reception of the data. After successful acquisition of the signal, the verification will be complete and considered successful. See Full Flight Performance Testing Series.	Incomplete - Scheduled for February 10th, 2018

Payload RBS-1	The RBS shall maintain a minimum ground clearance of 0.125 in. while driving.	Inspection The RBS and RDS will be manufactured and assembled. The Feather M0 Bluefruit LE microcontroller and FeatherWing Motor Shield will be used to control the two main drive motors of the RDS via Bluetooth from a team member's cellphone. This is to represent as accurately as possible the situation of autonomous mission performance while still maintaining full control of the rover. An object of 0.125 in. in height will be placed in front of the lowest point of the rover, the male T-Slot. The rover will be driven over the object. After clearing the object without touching the rover, the verification will be complete and considered successful.	Incomplete - Scheduled for January 25th, 2018
Payload RBS-2	The RBS shall house all rover electronics and mechanical systems in a secure and easily accessible manner.	Inspection The RBS will be manufactured and fully assembled with all electronics and rover systems mounted in flight configuration. All parts will be inspected and shown to be accessible without the need to remove or alter in any way, any other component in the RBS. The RBS will then be turned upside-down to ensure that all components are secure. After accessibility and secure mounting are confirmed, the verification will be complete and considered successful.	Incomplete - Scheduled for February 15th, 2018
Payload RBS-3	The RBS shall sustain high loads experienced during liftoff, opening of the payload bay main parachute, and landing.	Test The RBS, RLM, and ROCS will be manufactured and fully assembled. The systems will be integrated into the launch vehicle prior to a full-scale test launch. After landing of the payload bay, the payload systems will be removed and the RBS will be thoroughly inspected. After no significant deformation or damage is seen on the RBS, the verification will be complete and considered successful. See Flight Loads Testing Series.	Incomplete - Scheduled for February 24th, 2018
Payload RDS-1	The RDS main drive motors shall provide a torque capable of advancing the	Demonstration The RBS and RDS will be manufactured and assembled using two Actobotics 52 RPM Planetary Gear Motors as the main drive motors. The Feather M0 Bluefruit LE microcontroller and FeatherWing Motor Shield will be used to control the two main drive motors of the RDS via Bluetooth from a team member's cellphone. This is to represent as	Incomplete - Scheduled for January 25th, 2018

	rover in all directions.	accurately as possible the situation of autonomous mission performance while still maintaining full control of the rover. Weight will be added to the rover to simulate the full flight configuration weight of the rover. The rover will be driven forward, backward, left, and right. After demonstrating the ability of the drive motors to advance the rover in all directions, the verification will be complete and considered successful.	
Payload RDS-2	The RDS shall allow the rover to surmount a vertical step of minimum height 1 in.	Demonstration The RBS and RDS will be manufactured and assembled. The Feather M0 Bluefruit LE microcontroller and FeatherWing Motor Shield will be used to control the two main drive motors of the RDS via Bluetooth from a team member's cellphone. This is to represent as accurately as possible the situation of autonomous mission performance while still maintaining full control of the rover. An object with a vertical step height of 1 in. will be placed in front of the rover. The rover will be driven over the object. After five consecutive trials of clearing the obstacle, the verification will be complete and considered successful.	Incomplete - Scheduled for January 25th, 2018
Payload RDS-3	The RDS shall allow the rover to continue forward motion on a sloped terrain of a minimum incline of 20°.	Test The RBS and RDS will be manufactured and assembled. The Feather M0 Bluefruit LE microcontroller and FeatherWing Motor Shield will be used to control the two main drive motors of the RDS via Bluetooth from a team member's cellphone. This is to represent as accurately as possible the situation of autonomous mission performance while still maintaining full control of the rover. The rover will be placed on an adjustable incline ramp set at 20°. The rover will be driven one foot up the incline. After five consecutive trials in which the rover reaches the one foot mark driving up the incline, the verification will be complete and considered successful. The incline will then be incremented and the test repeated to determine the maximum incline possible. See RDS Sloped Driving Test.	Incomplete - Scheduled for January 27th, 2018

Payload RDS-4	The RDS drive belts shall maintain traction without misalignment on wet and dry concrete, grass, gravel, loose dirt, compact dirt, and sand.	<u>Demonstration</u> The RBS and RDS will be manufactured and assembled. The Feather M0 Bluefruit LE microcontroller and FeatherWing Motor Shield will be used to control the two main drive motors of the RDS via Bluetooth from a team member's cellphone. This is to represent as accurately as possible the situation of autonomous mission performance while still maintaining full control of the rover. The rover will be placed on dry and wet concrete, grass, gravel, loose dirt, compact dirt, and sand and driven forward, backward, left, and right for two minutes on each. If at any point, the treads loose traction or become misaligned, the demonstration will be halted and a solution determined. After performance has been confirmed on all terrains, the verification will be complete and considered successful.	Incomplete - Scheduled for January 28th, 2018
Payload OAS-1	The OAS lidar sensor shall detect an object directly in front of the sensor within a distance range of 5 to 45 in.	<u>Demonstration</u> The VL53L0X Time of Flight lidar sensor will be connected to the Feather M0 Bluefuit LE microcontroller. The sensor will be configured to relay the distance of any object in front it to the serial monitor of the Arduino IDE. An object will be placed in front of the sensor 5 in. away from the sensor. The object will be moved within the sensors line-of-sight to 45 in. away from the sensor. If at any time, the sensor fails to recognize the object, the trial will be halted. After five consecutive trials of no loss of recognition of the object within the range, the verification will be complete and considered successful.	Complete
Payload OAS-2	The OAS lidar sensor shall relay the distance between the sensor and an object to the CES with an accuracy of +/- 1 in.	<u>Test</u> The VL53L0X Time of Flight lidar sensor will be connected to the Feather M0 Bluefuit LE microcontroller. The sensor will be configured to relay the distance of any object in front it to the serial monitor of the Arduino IDE. An object will be placed in front of the sensor at various locations within the range of 1.97 to 47.24 in. The object distance will be measured with a ruler and compared to the distance data collected by the sensor. Five readings will be taken at the low end of the range (5 to 15 in.) and five readings will be taken at the high end of the range (30 to 40 in.). After 10 consecutive readings within +/- 1 in. of the ruler distance, the	Incomplete - Scheduled for January 13th, 2018

		verification will be complete and considered successful. See OAS Accuracy Test.	
Payload OAS-3	The OAS servo motor shall pan the lidar sensor along a 156° field of view.	Demonstration The VL53L0X Time of Flight lidar sensor and SG92R servo motor will be connected to the Feather M0 Bluefuit LE microcontroller. The sensor will be configured to relay the distance of any object in front it to the serial monitor of the Arduino IDE. Two objects will be placed 156° apart from each other. The servo will pan the lidar sensor starting from one object and ending on the other. The lidar will confirm recognition of the two objects based on the distance data sent to the Arduino IDE. After five consecutive trials of recognition of both objects, the verification will be complete and considered successful.	Incomplete - Scheduled for January 14th, 2018
Payload SAS-1	The SAS tower assembly shall actuate from the stowed position via the springe hinge and remain upright under its own power.	Inspection The SAS will be manufactured and assembled. The tower will be held by a team member in the stowed configuration. The tower will then be released and allowed to actuate via the spring hinge. Upon reaching full actuation, the tower shall remain upright under its own power. After five consecutive demonstrations of the tower being supported upright under its own power, the verification will be complete and considered successful.	Incomplete - Scheduled for February 1st, 2018
Payload SAS-2	The SAS shall unfold the solar panels such that the exposed solar panel surface area increases to four times that of the folded configuration.	Demonstration The SAS tower and solar panel support arms will be manufactured and fully assembled. The deployment motor will be controlled by the Feather M0 Bluefruit LE and FeatherWing Motor Controller Via Bluetooth from a team member's cellphone. This is done to avoid possibility of damage to the support arms and solar panels during testing by maintaining full control of the deployment motor. The solar panels will be in their folded configuration where only one panel is exposed to light. The deployment motor will be activated unfolding all of the solar panels. After all four solar panels are exposed, the motor will be stopped. A team member will visually confirm that all four solar panels are fully exposed. After five consecutive demonstrations, the verification will be complete and considered successful.	Incomplete - Scheduled for February 3rd, 2018

Payload SAS-3	The SAS solar panels shall harvest energy from the sun and pass a continuous minimum power of 4 mW to the CES.	Demonstration The SAS will be manufactured and assembled. The SAS will be put in its fully deployed configuration. The solar panels will be connected in parallel to the Feather M0 Bluefruit LE microcontroller. An RGB LED will be illuminated red when the power level is below 4 mW and green when the power is above 4 mW. Once the LED is green, a stop watch will begin counting up. If at any point the light turns red, the trial will be halted and restarted. After the LED is shown to remain stable green for more than 30 seconds representing continuous power being delivered, the verification will be complete and considered successful.	Incomplete - Scheduled for February 4th, 2018
Payload SAS-4	The SAS shall raise the solar panel support arms from a stowed flight configuration to achieve a minimum clearance of 0.5 vertical inches from any other rover component.	Demonstration The rover will be manufactured and fully assembled. The SAS locking motor and deployment motor will be connected to the Feather M0 Bluefruit LE microcontroller and FeatherWing Motor Shield. The locking motor will be commanded to release the tower assembly. Once the tower assembly is fully upright, the deployment motor will be activated. Once the panels are fully deployed, the vertical distance from each solar panel support arm and the next closest component to it will be measured. After confirming the smallest distance is greater than 0.5 in., the verification will be complete and considered successful.	Incomplete - Scheduled for February 16th, 2018
Payload SAS-5	The SAS shall support the four solar panels throughout the duration of the launch vehicle's flight and the payload's mission.	Test The payload will be manufactured and fully assembled. The payload will then be integrated into the payload bay of the launch vehicle prior to a full-scale test launch. After landing, a team member will walk out to the payload bay and deploy the rover. A team member will carefully inspect the SAS support arms and solar panels while and after the payload performs its mission. If at any point, a solar panel is damaged or dislodged from its support arm, note will be taken to determine a solution. After all four solar panels are deployed and undamaged, the verification will be complete and considered successful. See Flight Loads Testing Series.	Incomplete - Scheduled for February 24th, 2018

Payload SIS-1	The SIS camera module shall take images and store them on the CES SD card at a minimum rate of six images per minute.	Demonstration The SIS ArduCAM OV5642 camera module will be connected to the Feather M0 Bluefruit LE. The software will be configured to loop taking images at a rate of six images per minute. This will represent mission configuration. Images will be stored on the FeatherWing Adalogger's microSD card. Images will be inspected after allowing the camera to take pictures for a total of five minutes. If less than 30 images had been taken, the software will be changed accordingly trial will be restarted. After a minimum of 30 images is taken, the verification will be complete and considered successful.	Incomplete - Scheduled for January 22nd, 2018
Payload SIS-2	The OAS servo motor shall pan the SIS camera module to achieve a minimum field of view of 156°.	Demonstration The SIS ArduCAM OV5642 camera module and SG92R servo motor will be connected to the Feather M0 Bluefuit LE microcontroller. The camera and microcontroller software will be configured to store images taken at the starting angle and at the final angle on the FeatherWing Adalogger's MicroSD card. Two objects will be placed 156° apart from each other. The servo will pan the camera starting from one object and ending on the other. The images will be inspected to ensure that both objects were photographed. After confirmation of capturing both objects, the verification will be complete and considered successful.	Incomplete - Scheduled for January 23rd, 2018
Payload SIS-3	The SIS shall be triggered to begin taking images by the amount of energy produced by three of the four SAS solar panels being exposed to full and direct sunlight.	Demonstration The SIS ArduCAM OV5642 camera module and four SAS solar panels will be connecetd to the Feather M0 Bluefruit LE microcontroller. Software will be configured to begin taking images with the camera once the power level of the input from the solar panels exceeds 4 mW which is experimentally the case when three of the four panels are fully exposed in direct light. The Solar panels will begin covered and one by one will be fully uncovered in a well-lit place. After the third panel is fully revealed the camera will be commanded to begin taking pictures by the microcontroller. If this does not occur, the trigger power level will be adjusted and the trial restarted. After five consecutive trials without need for adjustment, the verification will be complete and considered successful.	Incomplete - Scheduled for January 24th, 2018

Payload CES-1	The CES shall obtain the 3-axis orientation of the rover with an minimum accuracy of +/- 0.1°.	Test Two BNO055 9DOF IMUs with documented accuracy of +/- 0.05° will be connected to the Feather M0 Bluefruit LE microcontroller. Software will be configured to receive 3-axis gyroscope data from the two IMUs. An RGB LED will be illuminated green if the sensor data reflects less than 50° of inclination in the pitch and roll directions with a base point of the sensor being flat on the surface. The LED will be turned red if the inclination exceeds 50°. The electronics will be fixed inside a tube and rolled in all possible pitch and roll angles. After 10 consecutive trials of the LED correctly indicating the angle of inclination within 0.1° error and with a drift of less than 1° over the testing period, the verification will be complete and considered successful. See CES Orientation Accuracy Test.	Incomplete - Scheduled for January 31st, 2018
Payload CES-2	The CES shall autonomously control motor of the RLM to both lock and release the rover to the ROCS.	Test The payload will be manufactured and fully assembled. The rover will be integrated with the RLM and ROCS. Software will be configured to release and lock the RLM after trigger commands have been sent from a team member's cellphone via Bluetooth representing the deployment signal being received. The state of the RLM will be inspected after each command is sent. If the RLM is not either fully locked or fully released, the trial will be halted and a solution determined. After 10 consecutive commands have been given, the verification will be complete and considered successful. See CESAutonomousControlTests.	Incomplete - Scheduled for February 18th, 2018
Payload CES-3	The CES shall autonomously control the SAS locking motor to both release and lock the tower assembly.	Test The rover will be manufactured and fully assembled. Software will be configured to release and lock the SAS locking motor after trigger commands have been sent from a team member's cellphone via Bluetooth representing the SAS deployment stage of the mission being reached. The state of the SAS locking motor will be inspected after each command is sent. If the SAS is not either fully locked or fully released, the trial will be halted and a solution determined. After 10 consecutive commands have been given, the verification will be complete and considered successful. See CES Autonomous Control Testing Series.	Incomplete - Scheduled for February 19th, 2018

Payload CES-4	The CES shall autonomously control the SAS deployment motor to both fold and unfold the solar array.	<u>Test</u> The rover will be manufactured and fully assembled. Software will be configured to fold and unfold SAS solar panel support arms after trigger commands have been sent from a team member's cellphone via Bluetooth representing the SAS deployment stage of the mission being reached. The state of the SAS solar panel support arms will be inspected after each command is sent. If the SAS is not either fully folded or fully deployed, the trial will be halted and a solution determined. After 10 consecutive commands have been given, the verification will be complete and considered successful. See CES Autonomous Control Testing Series.	Incomplete - Scheduled for February 20th, 2018
Payload CES-5	The CES shall autonomously analyze the data collected by the OAS lidar sensor in real-time to determine the optimal travel path for mission success.	<u>Test</u> The rover will be manufactured and fully assembled. Objects of greater than 4 in. tall will be placed in front of and around the rover. The CES will command the rover to drive in a straight line until an object is detected within 10 in. of the OAS sensor. The rover will be halted, the OAS servo motor will pan the lidar sensor 175°, and data points will be collected at each degree of the sweep. The data will be stored on the CES microSD card. The CES will analyze the data and turn the rover to avoid the object. If the rover touches an object at any point, the trial will be halted and a solution determined. After no objects are within five feet of the rover, the trial will be completed. After five consecutive trials have been completed, the verification will be complete and considered successful. See CES Autonomous Control Testing Series.	Incomplete - Scheduled for February 22nd, 2018
Payload CES-6	The CES shall autonomously control the two main drive motors of the RDS and all forward motion and maneuvering of the rover via the drive motors.	<u>Test</u> The rover will be manufactured and fully assembled. The Feather M0 Bluefruit LE microcontroller and FeatherWing Motor Shield will be connected to the main drive motors. Software will be configured to autonomously drive the rover along a path consisting of four forward, one reverse, one left turn, and one right turn commands. Markers will indicate the intended location of each setpoint. If at any point, the intended setpoint location is not reached, the trial will be halted and a solution determined. After the rover reaches and stops on the end marker five times	Incomplete - Scheduled for February 23rd, 2018

		consecutively, the verification will be complete and considered successful. See CES Autonomous Control Testing Series.	
Payload CES-7	The CES shall log all deployment signal, gyroscope, lidar, drive controls, solar power harvesting, and images acquired throughout the mission on a microSD card.	<u>Test</u> The payload will be manufactured and assembled. The payload will be integrated into the launch vehicle prior to a full-scale test launch. Flight ready software will be loaded onto the CES prior to launch. After the full mission is completed, data collected on the microSD of the CES FeatherWing Adalogger will be analyzed. After confirmation of data collected from the DTS deployment, two gyroscopes, OAS lidar sensor, autonomous drive commands, solar power harvest levels, and SIS images is achieved, the verification will be complete and considered successful. See Full Flight Performance Testing Series.	Incomplete - Scheduled for February 24th, 2018
Payload CES-8	The CES controller battery lifetime shall exceed a minimum of three hours running flight ready software.	<u>Test</u> The rover will be manufactured, assembled, and integrated with the DTS. The 500 mAh controller battery will be recharged fully prior to beginning the test. The rover will be switched to internal power and left to run flight ready software for three hours. Controller battery levels will be monitored at 10 minute increments. If at any point the battery level reduces below the required level of 3.7V to power the rover's electronics, the trial will be halted and a larger capacity battery obtained. After three hours of continuous runtime has been reached, the verification will be complete and considered successful. See Battery Life Testing Series.	Incomplete - Scheduled for February 25th, 2018

Payload CES-9	The CES motor battery lifetime shall exceed a minimum of 5 minutes of motor runtime on a single full charge.	Test The rover will be manufactured and assembled. The 400 mAh motor battery will be fully recharged prior to beginning the test. Software will be running on the CES Feather M0 Bluefruit LE microcontroller to run the main drive motors at full power for 30 seconds at a time with 15 seconds of rest between each interval. Battery levels will be monitored every 15 seconds. If at any point the battery voltage drops below 6V, the trial will be halted and a battery of larger capacity will be obtained. After five minutes worth of motor runtime has been achieved, the verification will be complete and considered successful. See Battery Life Testing Series.	Incomplete - Scheduled for February 25th, 2018

6.1.8.1. Derivation of Requirement ROCS-1

This requirement has been imposed in the interest of safety of all bystanders during launches to prevent the rover from prematurely exiting the payload bay.

6.1.8.2. Derivation of Requirement ROCS-2

The payload must be entirely enclosed in the airframe of the launch vehicle per SOW requirement 4.5.1. This requirement has been imposed to drive a design that will minimize time to integrate the payload into the launch vehicle. This also provides ease of access to the entire payload for servicing and analysis of flight effects on the ROCS.

6.1.8.3. Derivation of Requirement ROCS-3

The final design of the rover requires that the rover be within 50° of perfectly upright prior to the rover beginning its mission to avoid tipping over resulting in mission failure. This requirement has been imposed based on the tipping analysis described in SECTION# with added precautionary buffer to account for any uncontrollable factors on launch day such as wind.

6.1.8.4. Derivation of Requirement ROCS-4

This requirement has been imposed in the interest of safety of all bystanders during launches. The ROCS will be confirmed safe to fly and confirm analysis performed on the system.

6.1.8.5. Derivation of Requirement RLM-1

This requirement has been imposed in the interest of safety of all bystanders during launches to prevent the rover from prematurely exiting the payload bay.

6.1.8.6. Derivation of Requirement RLM-2

This requirement has been imposed with the primary intent of ensuring safety of all bystanders during launches. The secondary intent is to minimize the possibility of damage to the rover during flight that could cause mission failure.

6.1.8.7. Derivation of Requirement RLM-3

This requirement has been imposed to drive a design that can both fully restrict and fully allow free motion of the rover that can be controlled autonomously by the rover. This increases safety during flight and increases chance of success after landing.

6.1.8.8. Derivation of Requirement RLM-4

This requirement has been imposed in the interest of safety of all bystanders during launches to prevent the rover from prematurely exiting the payload bay. Verification of this requirement is proof positive that the payload is safe to fly without presenting serious risk to bystanders.

6.1.8.9. Derivation of Requirement DTS-1

This requirement has been imposed to ensure the selection and configuration of transmitter and receiver modules that remain within the legal bounds of use of RF without the need for permission or license.

6.1.8.10. Derivation of Requirement DTS-2

This requirement has been imposed to drive a design that will allow the receiver antenna to remain fixed to the exterior of the airframe after deployment of the rover. This also allows the ROCS to provide independent rotation of the rover in the airframe without losing connection to the fixed receiver.

6.1.8.11. Derivation of Requirement DTS-3

This requirement has been imposed to drive a design that will allow the receiver antenna to remain fixed to the exterior of the airframe after deployment of the rover without restricting the ability of the rover to perform its mission.

6.1.8.12. Derivation of Requirement DTS-4

This requirement has been imposed after receiving word that a team member will be allowed to be in close vicinity to the payload bay to send the deployment signal. The minimum 50 foot radius range is to ensure that the signal reception be sound from all angles to the receiver antenna at a distance that accounts for the close proximity.

6.1.8.13. Derivation of Requirement DTS-5

This requirement has been imposed in the interest of safety of all bystanders during launches. This requirement is necessary to prevent premature deployment due to other teams using RF and any other sources. Only the signal sent by a River City Rocketry team member will deploy the rover.

6.1.8.14. Derivation of Requirement DTS-6

This requirement has been imposed to drive antenna design selection that is robust enough to sustain functionality after flight of the launch vehicle and as a means of verifying functionality in full flight configuration.

6.1.8.15. Derivation of Requirement RBS-1

This requirement has been imposed to drive a design that is capable of traversing uneven terrain without bottoming out causing failure of the mission.

6.1.8.16. Derivation of Requirement RBS-2

This requirement has been imposed to drive a design that will allow for quick integration into the launch vehicle and easy accessibility for any minor changes that need to be performed on the payload without need for deconstruction. This will also provide a means of quick replacement if a component is shown to be dysfunctional.

6.1.8.17. Derivation of Requirement RBS-3

This requirement has been imposed in the interest of safety of all bystanders. This is also done to drive a robust design capable of completing its mission after flight of the launch vehicle.

6.1.8.18. Derivation of Requirement RDS-1

This requirement has been imposed to ensure that requirement 4.5.3 of the SOW be met. This will also aid in the justification of a selected motor to act as the main drive motors.

6.1.8.19. Derivation of Requirement RDS-2

This requirement has been imposed to ensure that the rover be capable of traversing on uneven and unpredictable terrain. This will drive the optimization of the drive system design.

6.1.8.20. Derivation of Requirement RDS-3

This requirement has been imposed to ensure that the rover be capable of traversing on uneven and unpredictable terrain. This will drive the selection of drive treads and tread design.

6.1.8.21. Derivation of Requirement RDS-4

With the mindset behind the challenge being that the rover be deployed on a foreign planet's surface, the rover must be capable of driving on any terrain encountered. This requirement has been imposed to account for any terrain that may be encountered at the time of the rover's mission.

6.1.8.22. Derivation of Requirement OAS-1

This requirement has been imposed to achieve a rover design that minimizes the possibility of damage and mission failure due to obstacles in the immediate path. The lower limit of the range has been determined as the minimum distance from an object required to turn the rover with no possibility of touching the object. The upper limit has been determined as the maximum distance from the sensor to an object for highly accurate readings in all light levels and object colors.

6.1.8.23. Derivation of Requirement OAS-2

This requirement has been imposed to ensure that the sensor data collected be transmittable to the CES microcontroller running the autonomous control software with high enough precision to remove possibility of damage or failure of mission due to inaccurate distance measuring.

6.1.8.24. Derivation of Requirement OAS-3

This requirement has been imposed to increase the efficiency of the rover's mission thus expanding its capabilities within the same runtime by decreasing the amount of time taken to determine of an optimal path of least obstruction. The sweep angle has been determined as the maximum reliable sweep angle capability of the servo motor.

6.1.8.25. Derivation of Requirement SAS-1

This requirement has been imposed to ensure that the solar panels remain fully exposed to the sun after deployment to harvest as much energy as possible.

6.1.8.26. Derivation of Requirement SAS-2

This requirement has been imposed to drive a design capable of significantly multiplying the area of exposed solar cell panels and increase energy harvesting rates drastically after unfolding while maintaining a small footprint prior to deployment. This requirement will aid in satisfying SOW requirement 4.5.4.

6.1.8.27. Derivation of Requirement SAS-3

This requirement has been set to design a rover that will make use of the energy that will be harvested by the solar panels after deployment. The minimum power requirement has been determined as 80% of the maximum power generated by four MPT3.6-150 solar panels. The requirement calls for continuous power to be delivered to ensure that the solar array is a useful continuous power source.

6.1.8.28. Derivation of Requirement SAS-4

This requirement has been imposed to drive a design that can be stowed for flight and drastically increase its footprint after exiting the launch vehicle to harvest as much energy as possible. The vertical clearance has been imposed to ensure that no other component impede the deployment of the solar panels.

6.1.8.29. Derivation of Requirement SAS-5

This requirement has been imposed to drive a robust design capable of protecting the solar cell panels throughout the flight of the launch vehicle to maximize their efficiency of energy harvesting during the payload mission.

6.1.8.30. Derivation of Requirement SIS-1

This requirement has been imposed as a secondary mission of the payload to embrace the mindset of the challenge being that the rover is being deployed on a foreign planet. This will drive a design capable of providing scientific data in the form of images about the state of the vehicle after mission completion and the ground area around the rover. On a foreign planet, this data would be extremely valuable for further optimizing the design of ther rover and determining characteristics of the foreign planet.

6.1.8.31. Derivation of Requirement SIS-2

This requirement has been imposed to increase the field of view of the camera thus increasing the amount of scientific data collected by the rover. The sweep angle has been determined as the maximum reliable sweep angle capability of the servo motor.

6.1.8.32. Derivation of Requirement SIS-3

This requirement has been imposed to design a rover that will make use of the energy that will be harvested by the panels after deployment. The trigger set to three of the four panels being fully exposed experimentally generates 80% of the maximum power capable of being generated by the system. This accounts for potentially lower light levels during operation still being capable of triggering the secondary mission of the payload.

6.1.8.33. Derivation of Requirement CES-1

This requirement has been imposed with primary interest of the safety of all bystanders. This will be used by the CES as another level of safety to prevent premature deployment of the rover. The minimum accuracy rating will aid in selection of a high accuracy sensor to further mitigate the potential risk. This requirement has been imposed with the secondary interest of allowing the CES to turn the rover a specific number of degrees at a time autonomously.

6.1.8.34. Derivation of Requirement CES-2

This requirement has been imposed to prove autonomous control over the RLM subsystem. This is part of proving full autonomy of the rover that would be necessary on a foreign planet.

6.1.8.35. Derivation of Requirement CES-3

This requirement has been imposed to prove autonomous control over part of the SAS. This is part of proving full autonomy of the rover that would be necessary on a foreign planet. This also aids in satisfying requirement 4.5.4 of the SOW.

6.1.8.36. Derivation of Requirement CES-4

This requirement has been imposed to prove autonomous control over part of the SAS not proven by requirement CES.5. This is part of proving full autonomy of the rover that would be necessary on a foreign planet. This also aids in satisfying requirement 4.5.4 of the SOW.

6.1.8.37. Derivation of Requirement CES-5

This requirement has been imposed to prove autonomous control over the OAS. This is part of proving full autonomy of the rover that would be necessary on a foreign planet. This also drives software design that increases the efficiency of the rover thus increasing its capabilities in the runtime of the mission.

6.1.8.38. Derivation of Requirement CES-6

This requirement has been imposed to prove autonomous control over the RDS. This is part of proving full autonomy of the rover that would be necessary on a foreign planet.

6.1.8.39. Derivation of Requirement CES-7

This requirement has been imposed to provide scientific data throughout the flight of the launch vehicle and the duration of the rover's mission that is retrievable and able to be analyzed. The secondary intent of this requirement is to allow for quick and accurate determination of any problem with the performance of the rover and of solutions to those problems. The tertiary intent of this requirement is to provide physical proof of payload performance.

6.1.8.40. Derivation of Requirement CES-8

This requirement has been imposed to account for the possibility of two hours of pad time prior to launch followed by a maximum of one hour between launch of the launch vehicle and recovery of the payload after completion of its mission. This will account for launching at any position in a volley.

6.1.8.41. Derivation of Requirement CES-9

This requirement has been imposed to account for the total runtime of the motors required to complete the payload's mission. The five minute minimum has been chosen to be well above the expected duration of the payload's primary mission of one minute.

6.2. Budgeting and Timeline

6.2.1. Budget

For the 2017-2018 season River City Rocketry budget has been projected based on anticipated expenses and income for the season. For this season the team anticipates having $50,300 from income and carryover and approximately $35,000 in expenses. This allows us about $15,000 carryover for summer research and to carry us into next year. More details on this year's budget can be viewed in the tables below. Itemized lists of total costs for the 2017-2018 season can be found in PDR.

BUDGET	TOTAL
VDS	$ 2,268.56
VEHICLE	$6,542.18
RECOVERY	$1,453.00
PAYLOAD	$4,406.80
OUTREACH	$ 2,318.41
TRAVEL	$ 4,350.00
MERCHANDISING	$ 1,885.00
TEAM IMPROVEMENTS	$ 6,200.00
TOTAL PROJECTED EXPENSES	$ 29,423.95
TOTAL PROJECTED CARRYOVER	$ 20,876.05
TOTAL PROJECTED INCOME	$ 50,300.00
TOTAL RECEIVED INCOME	$ 45,300.00

Table 71: Budget Overview.

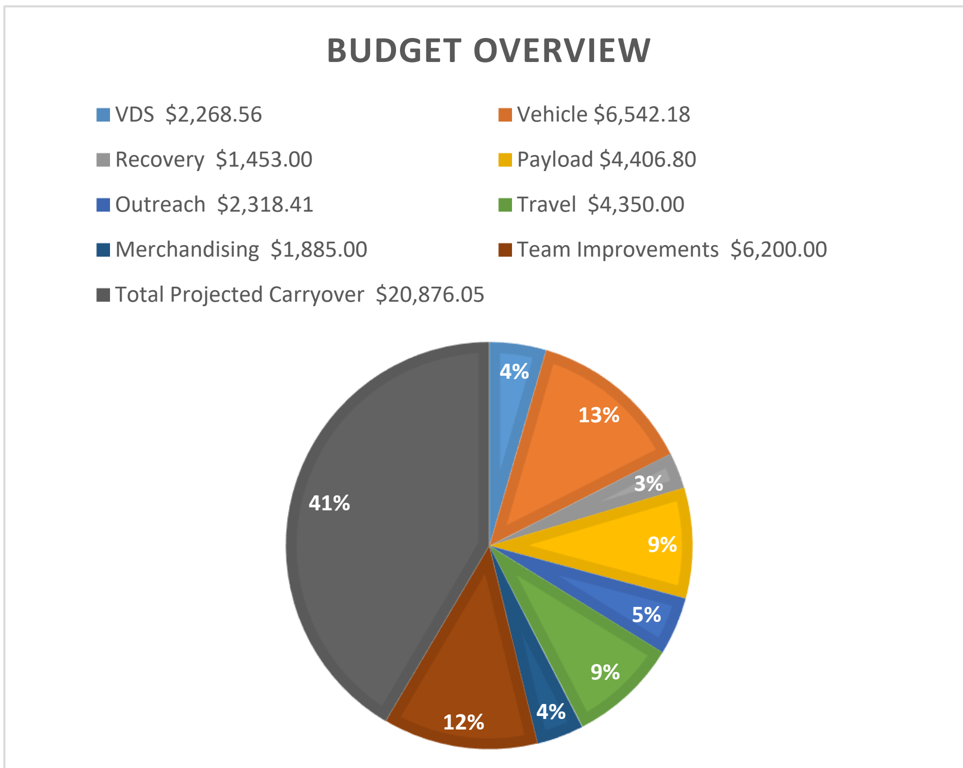


Figure 169: Budget overview analysis.

EXPENSES	
PAYLOAD EXPENSES	$ 1,199.22
VEHICLE EXPENSES	$ 2,614.68
VDS EXPENSES	$ 816.73
GENERAL TEAM EXPENSES	$ 1,085.83
RECOVERY EXPENSES	$ 1,113.84
TOTAL EXPENSES	$ 6,830.30

Table 72: Expense report

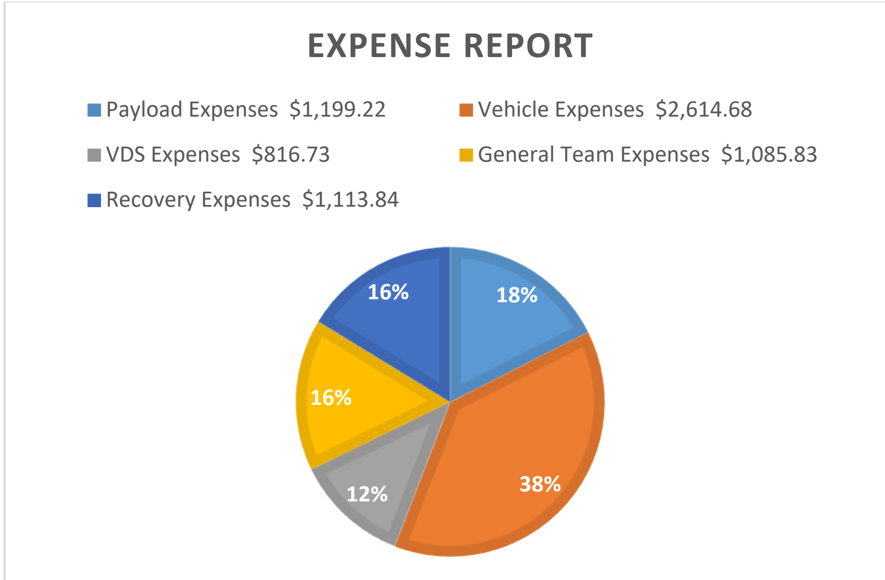


Figure 170: Expense report analysis

SOURCE	AMOUNT
REMAINING BALANCE	$ 12,300.00
DR. KELLY	$ 20,000.00
NASA PRIZE MONEY	$ 5,000.00
SPEED SCHOOL MONEY	$ 5,000.00
PENDING GE GRANT	$ 5,000.00
RAYTHEON	$ 1,000.00
MISC. DONATIONS	$ 2,000.00
TOTAL	**$ 50,300.00**

Table 73: Income report

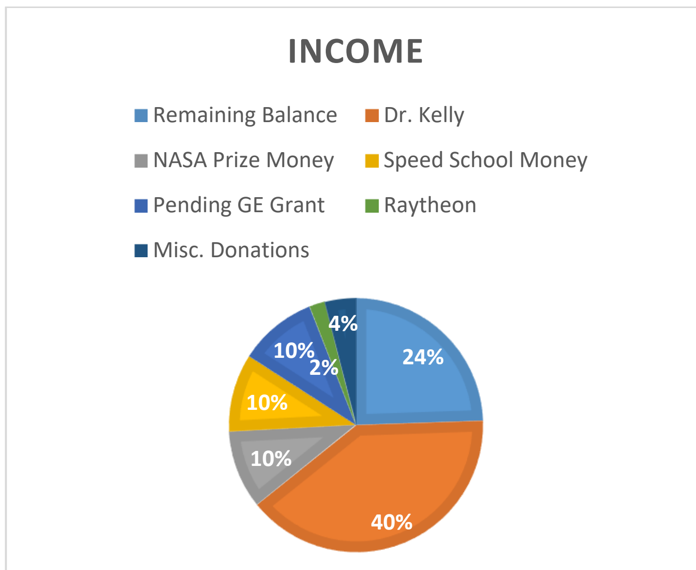


Table 74: Income report analysis

6.2.2. Timeline

For the 2017-2018 season River City Rocketry the team has outlined a timeline which is shown as a Gantt chart in Figure 71 below. This timeline is then broken down into several smaller timelines based on the team identified sub teams where critical events that the team has identified as essential deadlines the team must hit to ensure the project can stay on schedule is highlighted at the top.

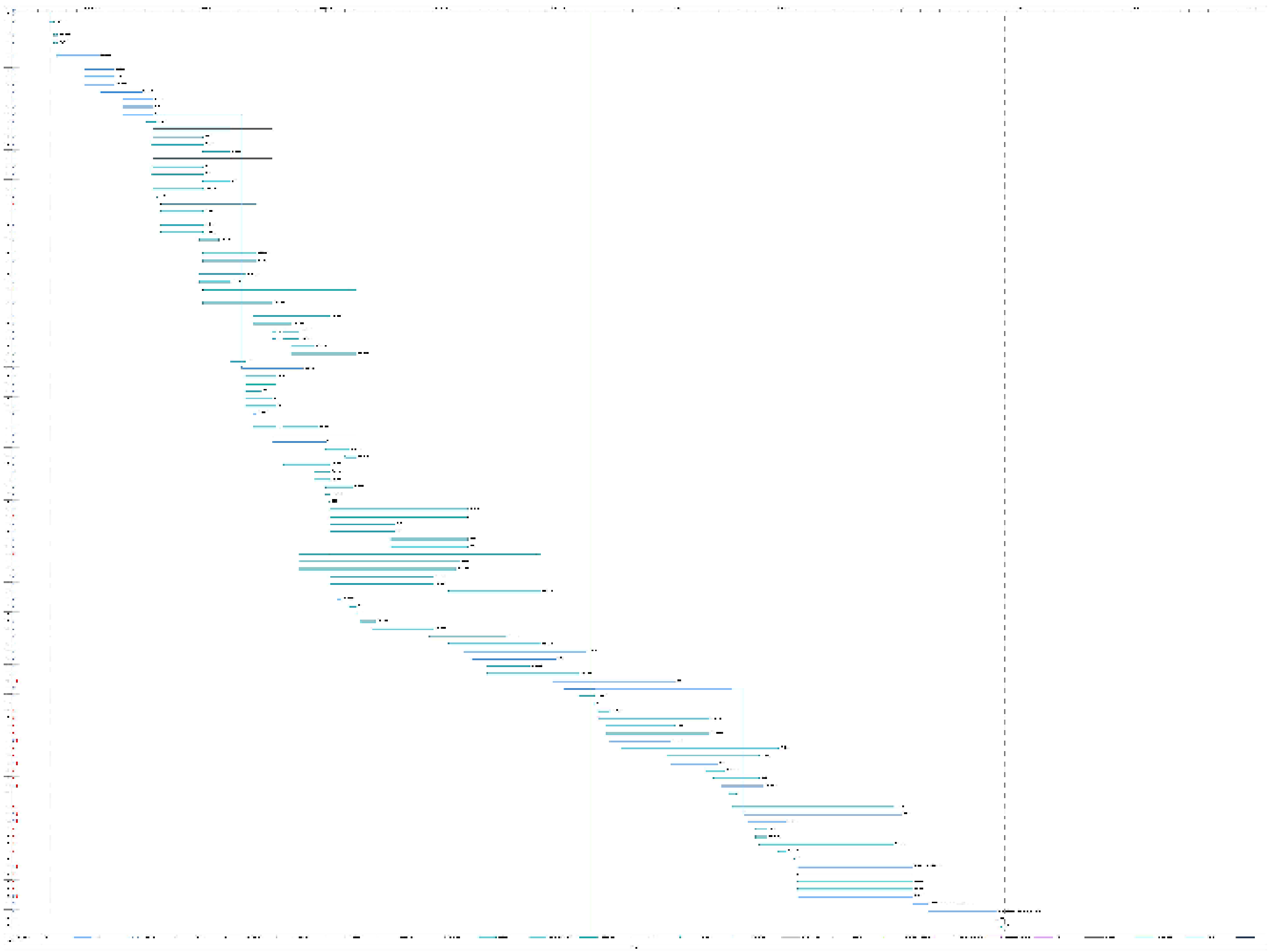


Table 75: Team Project Gantt Chart

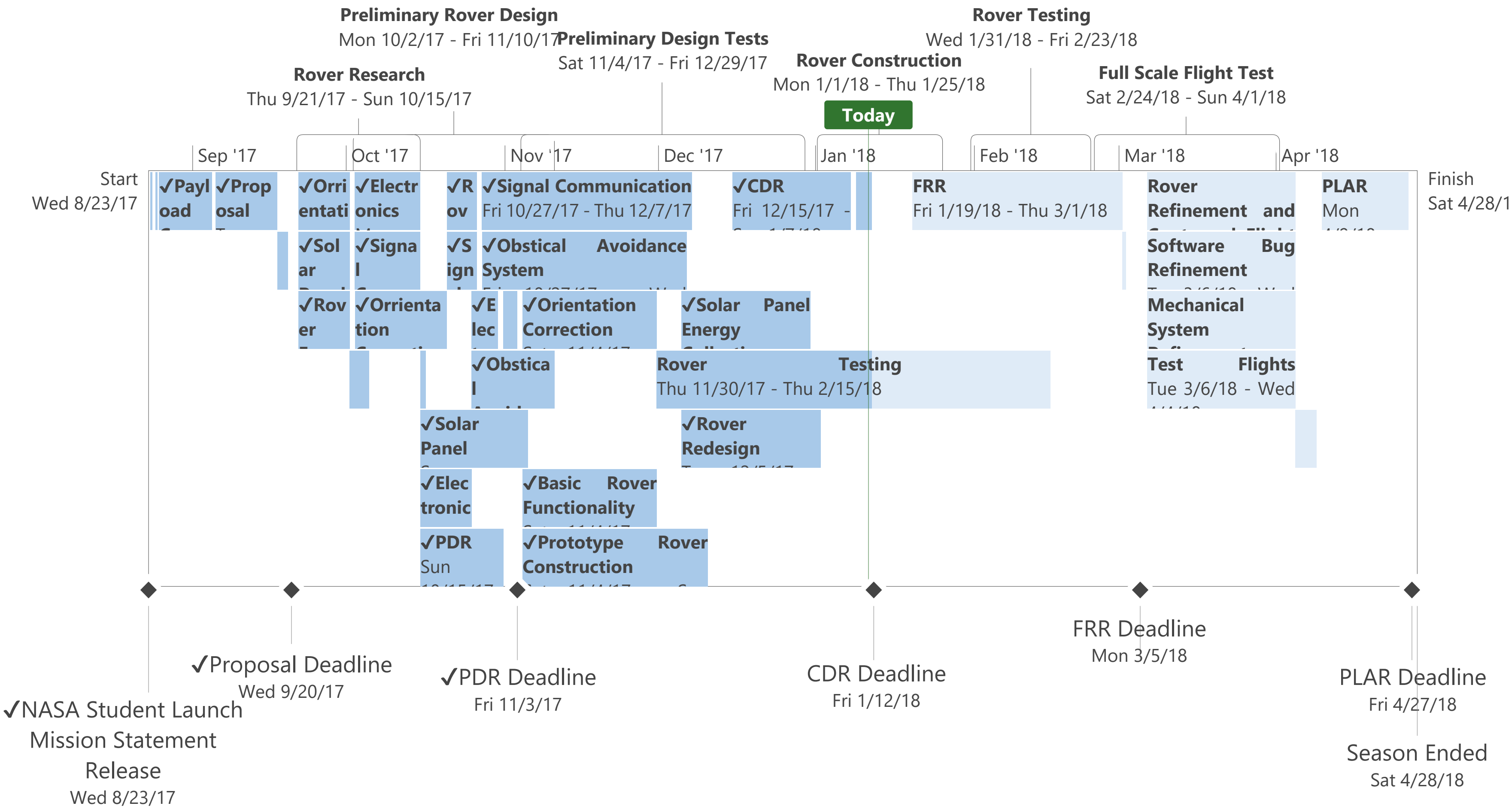


Figure 171: Payload Timeline

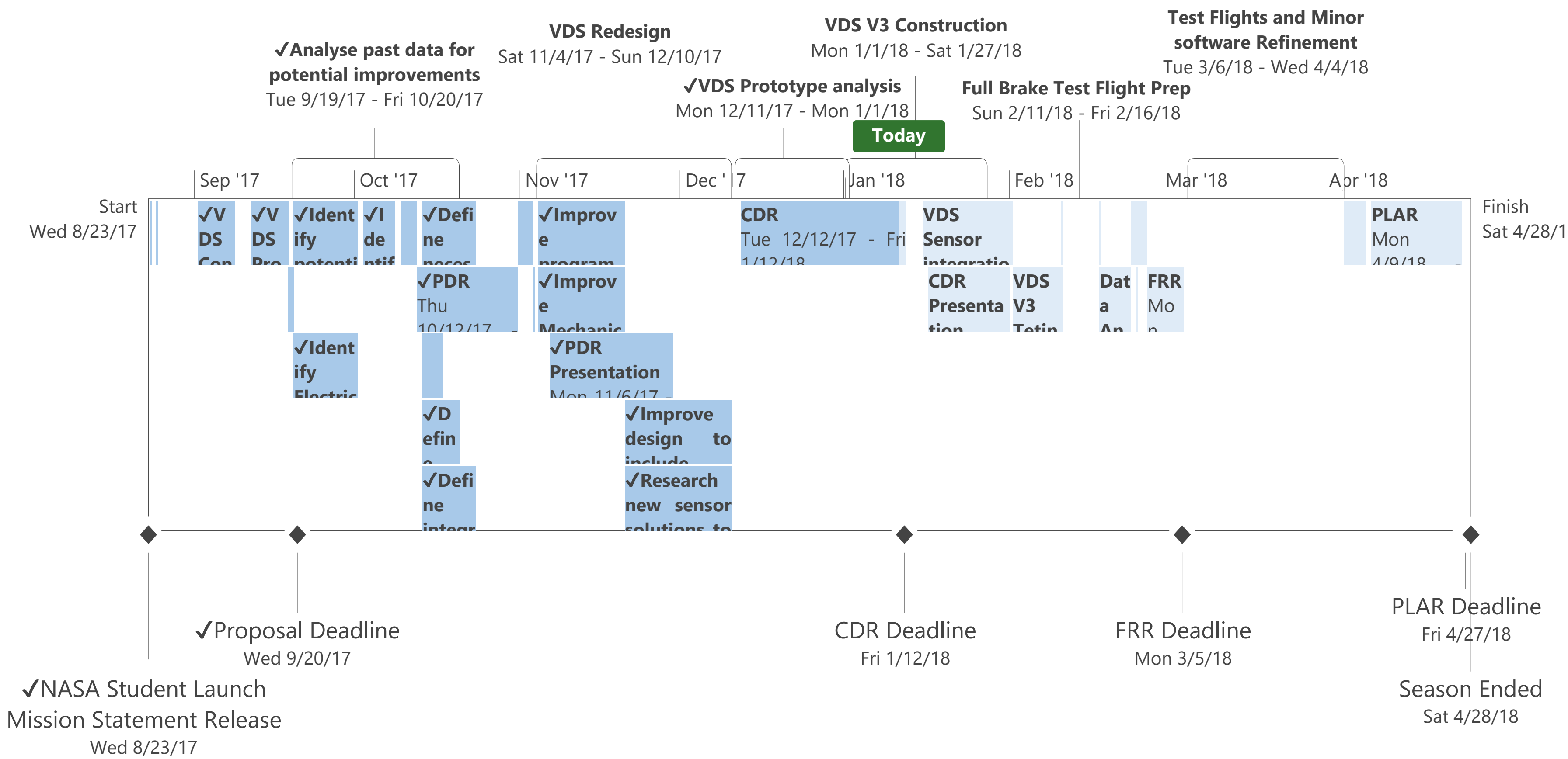


Figure 172: VDS Timeline

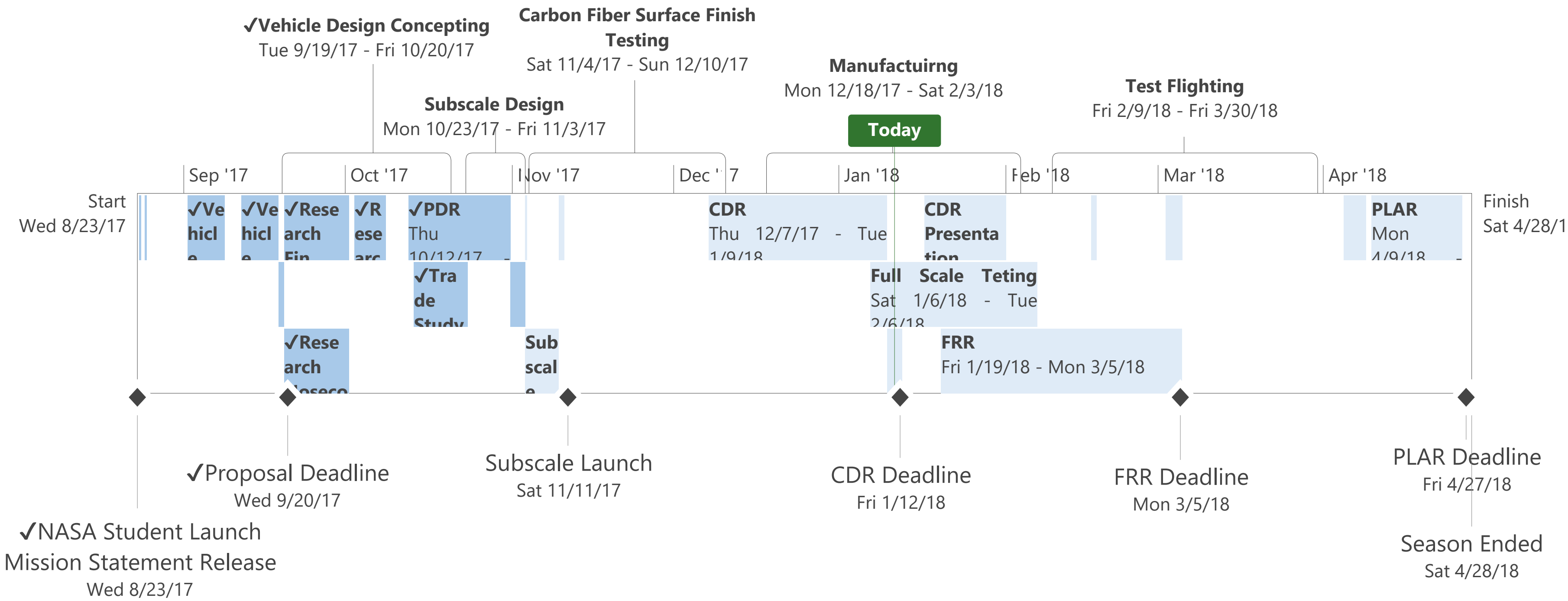


Figure 173: Vehicle Timeline

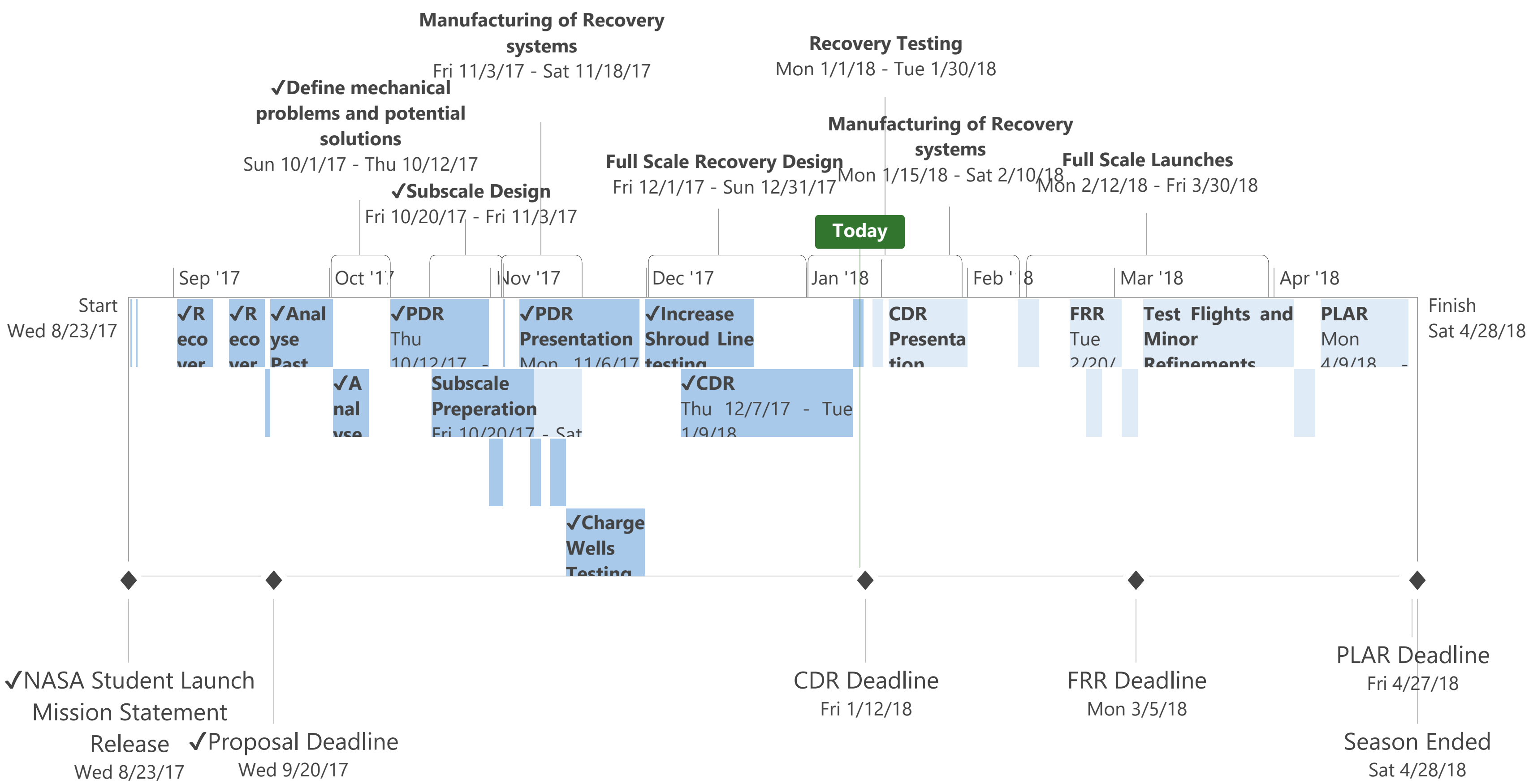


Figure 174: Recovery Timeline

6.2.3. Outreach

6.2.4. Educational Engagement

Throughout the last five years the team has had the opportunity the University of Louisville's River City Rocketry has had the opportunity to engage with over 8,000 students and adults in our local community. The team's outreach allows us to give back to the state of Kentucky by teaching the youth about engineering, math, technology, logical thinking and rocketry. River City Rocketry continues to develop new relationships through outlets like our website, all while further developing the relationships the team has built up over the years by striving to constantly improve the quality of education brought to every individual we engage with in our community.

Through these relationships built with the local community River City Rocketry has outreached to over 2,000 students so far, this year. A breakdown of these events can be seen in Table 76.

Outreach Event	Number
First Lego League	25
Louisville Area Math Circle	21
MiniMaker Faire	200
Cardinal Preview Day	30
MathMovesU	18
Farmer Elementary STEM Expo	100
Cochran Elementary Science Expo	155
Blast off the Noon year	1488
Total	2037

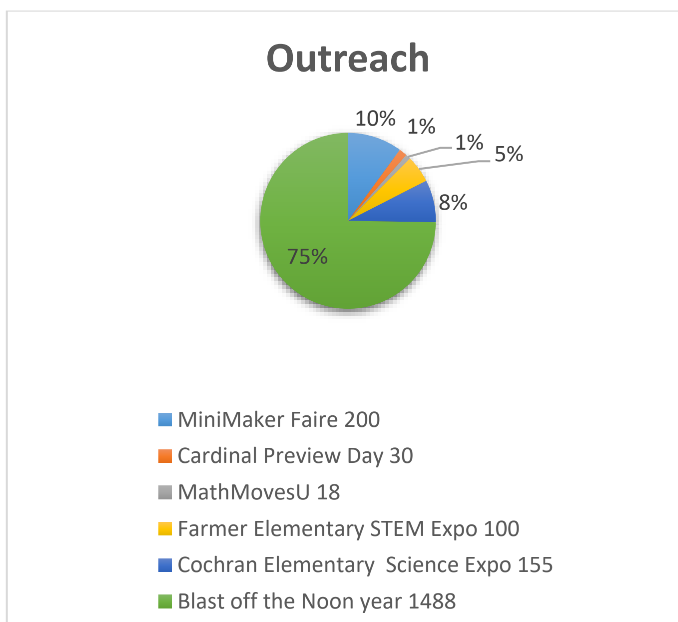


Table 76: Outreach event.

6.2.4.1. Classroom Curriculum

The University of Louisville's River City Rocketry team has developed a variety of educational programs that will be incorporated into this year's outreach program. This list includes many of the different activities in which the team has participated in the past and will continue to do this year.

6.2.4.1.1. Raytheon MathMovesU

Over the past few years, River City Rocketry has partnered with Raytheon Missile Systems and the University of Louisville in order to put on MathMovesU. This event will take place at the Engineering Education Garage, and the intramural fileds. This event is outlined in the following steps below

Step 1: Introduction from RCR and Raytheon

Following check in, a brief presentation about River City Rocketry, what the team does along with going over the agenda for the afternoon. Following the team's presentation, Raytheon Missile systems will give a brief presentation on what they do and how STEM classes are immensely beneficial to building rockets in the future.

Step 2 Construction and Lesson of Estes Kits:

During this time the Team will go over the basics of how rockets operate and the science behind it, along with instructing the students on how to construct their rockets and the functionality of each component on the rocket. During this construction phase volunteers will be stationed throughout the room to assist with construction while a single member directs the students through the process with the aid of a PowerPoint.

Step 3 Safety Briefing and Launching of Estes kits.

Following the construction of the rockets the teams safety officer will perform a safety briefing for students and adults to ensure that all personal, volunteers, students and parents are safe throughout the event. While rockets are being launched everyone that is within the bounds of the intermural field will be required to wear safety glasses and be restricted to remain behind a pre-defined white line located over 100 ft away from the launch stations.

As each rocket is set up on the launch stand, one student per stand at a time will be allowed past the white line, where that student will work with the Range Safety Officer (RSO) to set up the rocket on the stand. Following the setup of the rocket and returning to behind the white line the RSO will give the student permission to insert the safety key. A countdown from 5 to 0 from the student will follow the insertion of the key. Following the student countdown, the students will ignite their rockets into the sky.

The steps and safety precautions used during the MathMovesU event by River City Rocketry is constant with all major rocketry building outreach events.

6.2.4.1.2. STEM night

This will be the second year River City Rocketry will be participating in STEM night at Farmer Elementary. The team will have a table on display of your past rockets. During this event last year, the team used bristle bots. During this event last year students got to choose from racing a prebuilt bot to learn how they operate and test their reaction time or to build their own bristle bot from scratch and learn about how small circuitry works and how it integrates into mechanical systems. This year the team plans to design a similar booth that will be both engaging and educational for the kids.

6.2.4.1.3. Mazeek Middle School team Partnership

During the 2017-2018 season the team has partnered with Mazeek Middle school's rocket team. Mazeek's team is in its first year and we believe partnering with them will dramatically help their success by being a resource for both rocket knowledge along with assistance with organization and recruiting strategies. River City Rocketry hopes that this partnership helps build a successful team and inspires more students to go into STEM career paths.

6.2.4.2. Outreach Opportunities

6.2.4.2.1. Engineering Exposition (E-Expo)

Since 2006, the J.B. Speed School of Engineering Student Council has hosted the largest student-run event on the University of Louisville's campus called the Engineering Exposition. The event is geared towards celebrating strides in engineering as well as getting the local youth interested in the field. During the event, the professional engineering societies on U of L's campus set up educational activities and scientific demonstrations for the elementary and middle school students to participate in.

The University of Louisville River City Rocketry Team will host its sixth annual bottle rocket competition for middle school students. Teams from local middle schools can participate in teams of up to three students to design and build their own water bottle rockets out of two-liter bottles and other allowable materials. Workshops will be held with schools interested to teach the students about the components of a rocket and aerodynamics in preparation for the competition. The students will get to show off their rockets at the E-Expo event and will conclude the day with the competition. Teams will compete for awards in highest altitude, best constructed rocket, and landing closest to the launch pad. This event has been a huge success in the past and many schools have voiced interest in continuing their involvement so we are looking for our best turnout yet this year. A previous paper rocket E-Expo event is shown in **Figure 152**.

Figure 152: Team member, Denny, building rockets with students at E-Expo 2016.

In addition to the water rocket competition, the team will host a paper rocket station for people of all ages. This has been the most popular station at the exposition in the past and we are looking to continue to build up that reputation.

6.2.4.2.2. Boy Scouts and Cub Scouts:

In the past, the University of Louisville River City Rocketry Team has worked with local Boy Scout and Cub Scout troops to assist the earning of the Space Exploration merit badge. The team has assisted in developing a program that meets the requirements to earn the merit badge. The scouts get to learn about the history of space, current space endeavors, and build and launch an Estes rocket. The team has plans to continue to work with these groups throughout the year with one event already scheduled for September 30^{th}, 2017.

While cub scouts are not eligible to earn their merit badge, we still enjoy getting to teach them about rocketry. We have had the pleasure of working with scout troops in educating the kids about the fundamentals of rocketry, while also giving them the opportunity to build and launch their own paper rockets.

6.2.4.2.3. Big Brothers Big Sisters Partnership

Big Brothers Big Sisters is active in the Louisville community and is constantly striving to bring opportunities to underprivileged kids. The team recently put on a program at the Back to School Event for kids that had not yet been paired with a mentor through the program, shown in Figure 153. This is the second year in a row that the team has participated in this event. Both years, this event has been a huge success in brining STEM to under-privileged kids.

Figure 153: Big Brothers Big Sisters Back to School Event (2017).

6.2.4.2.4. Louisville Mini-Maker Faire

Annually, Louisville hosts a Mini-Maker Faire. The team always participates by taking the previous year's project out to show off to anyone attending the event. A variety of people attend this event ranging from small children to adults with experience in the field. This gives the team an opportunity to talk to the community about our project and what it does. This is an informal setting which is perfect for interacting with visitors and answering their questions about the project, what the team does, and about rocketry in general.

6.2.4.2.5. Kentucky Science Center

During the 2015-2016 season, the team first came in contact with Andrew Spence, manager of public programs and events, that assisted in several events in the Louisville area. For this season the team will participate in the Youth Science Summit, Advanced Manufacturing, Noon Year's Science Celebration, and National Engineers' Week at Kentucky Science Center. The team will be able to reach out to hundreds of young rocketeers and teach them about rocketry, engineering, and skills needed to succeed as an engineer.

During this current season, the team will build an exhibit at the science center, this exhibit will consist of a paper rocket building station, the rocket from the 2016-2017 season and informational plats about NSL, River City Rocketry and the science behind rocketry.

7. Appendix

7.1. Safety Checklists and Launch Procedures

Full Scale Launch Safety Checklists

The following checklists were written to prepare the team for a safe and successful launch. Each checklist includes the following features to ensure that assemblers are well equipped, safe, and able to recognize all existing hazards:

- Required hardware, equipment, and PPE for each process
- Labels to indicate explicit safety precautions:

CAUTION -label used to identify where PPE must be used

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

DANGER -label to signal the use of explosives and to indicate specific steps that should be taken to ensure safety

7.1.1. General Material Safety Checklist

To be checked and signed by a River City Rocketry team member and co-captain.		
1. ______________________________	2. ________________________________	
Prior to leaving for launch site		
Equipment to Pack		
☐ 5-minute epoxy	☐ Drill	☐ Electrical tape
☐ Red supply tackle box (x2)	☐ Drill bit set	☐ Black Gorilla Tape
☐ Clear black powder capsules (x4)	☐ Scissors	☐ Paper towels
☐ E-matches (x4)	☐ Garbage bag (x2)	☐ Hot glue gun
☐ Black powder kit	☐ Large tarp	☐ Allen wrench set
☐ Black powder measuring	☐ Folding chair (x3)	☐ Tent (x2)
☐ Paper towels	☐ Dremel	☐ Folding table (x3)
☐ Hot glue gun	☐ Assorted zip tie container	☐ Dremel bit kit
Personal Protective Equipment to Pack		
☐ Safety glasses (x25)	☐ Respirator (x2)	☐ Box of nitrile gloves

7.1.1.1. Recovery Safety Checklist

To be checked and signed by Recovery Lead and team member when *all* steps are completed, indicating that Recovery team is **PREPARED FOR LAUNCH**.		
1. ________________________	2. ________________________	
Prior to leaving for launch site		
Equipment to Pack		
☐ ARRD (x2) ☐ Shock cord (x6) ☐ Booster main parachute ☐ Payload bay main parachute ☐ E-Match (x4)	☐ Booster drogue parachute ☐ Payload bay drogue parachute ☐ Deployment bag (x2) ☐ No Burn fire retardant spray ☐ Parachute packing hook	☐ Masking tape ☐ Nomex cloth (x3) ☐ Quick link (x10) ☐ Recovery insulation (Dog barf) ☐ Opening force reduction ring (x4)
Main Parachute Packing		

The same packing checklist is to be used for both Booster main and Payload main parachutes.		
Booster	Payload	
_____	_____	1. Inspect canopies, shroud lines, and shock cords for any cuts, burns, fraying, loose stitching and any other visible damage.
Note:		*If any damage is identified, immediately inform each co-captain and the safety officer. The launch ehicle will be deemed safe to fly or a corrective action will be decided upon and implemented.*
_____	_____	2. Spray shroud lines with No Burn fire retardant spray.
_____	_____	3. Lay parachute canopy out flat.
_____	_____	4. Ensure shroud lines are taut and evenly spaced and not tangled.
_____	_____	5. Slide opening force reduction ring to where the shroud lines meet the canopy.
_____	_____	6. Attach quick link to shroud lines.
_____	_____	7. Fold parachute per the folding procedures.
_____	_____	8. Daisy chain shroud lines and place folded parachute(s) into respective zip lock bags.
_____	_____	9. Attach quick links to shock cord via bowline knot.
_____	_____	10. Verify all recovery equipment is packed.
To be checked and signed by Recovery Lead and team member when all *prior to leaving* steps are completed, indicating that Recovery team is PREPARED TO DEPART FOR LAUNCH		
1. ______________________________ 2. ______________________________		

Launch day procedures

Parachute Assembly

The same packing checklist is to be used for both Booster main and Payload main parachutes through step 5.		
CAUTION		Safety glasses must be worn while handling black powder
__________		1. Properly assemble the ARRD and perform a force test with a minimum of 400 lbs.
DANGER		E-matches and black powder are explosive. The cartridges and leads of the ARRD must be kept clear of batteries, sparks, and open flames to avoid accidental firing.
Note:		*If any damage is identified, immediately inform each co-captain and the safety officer. The launch vehicle will be deemed safe to fly or a corrective action will be decided upon and implemented.*
__________		2. Secure ARRD to payload bay coupler.
_____ Booster	_____ Payload	3. Properly place folded booster main parachute in deployment bag with shroud lines coming directly out of the bag. Ensure that the shroud lines are not wrapped around the parachute inside the deployment bag.
WARNING *Note* :		If the shroud lines are not wrapped around the parachute, the parachute getting stuck in the deployment bag. Verify that the parachute fits loosely in the deployment bag.
_____ Booster	_____ Payload	4. Secure deployment flaps using shroud lines. Use hook to assist in securing extra length of shroud lines through loops stitched in deployment bag. Continue this pattern in the same direction around the deployment bag to prevent tangling.
__________		5. Repeat steps 3 and 4 for payload bay main parachute
__________		6. Properly fold drogue parachute and insert in respective Nomex cloth. Use masking tape to secure the Nomex cloth.
__________		7. Inspect the inside of the vehicle for carbon fiber splinters or corners that could cause parachutes To be caught during separation.
WARNING		If edges are not identified prior to packing, the parachute may not deploy and could cause to a total mission failure.

___________	8. Attach shock cord to payload bay coupler and main parachute.
___________	9. Secure shock cord slack using masking tape.
___________	10. Tether deployment bag to payload bay ARRD.
___________	11. Connect shock cord from drogue parachute to ARRD and nosecone.
___________	12. Attach shock cord to booster coupler and main parachute.
___________	13. Secure shock cord slack using masking tape.
___________	14. Tether deployment bag to payload bay coupler.

7.1.1.2. Vehicle Safety Checklist

To be checked and signed by Vehicle Lead and team member when *all* steps are completed, indicating that Vehicle team is PREPARED FOR LAUNCH.		
1. ______________________________ 2. ______________________________		
Prior to leaving for launch site: Vehicle		
Equipment to Pack or Prepare		
☐ Precision flathead screwdriver ☐ Standard Philips head screwdriver ☐ Duracell 9V battery (x4) ☐ Eggfinder sled ☐ Trackimo outside of bulk plate	☐ Multimeter ☐ Skytraq ☐ AIMXTRA nose cone ☐4-40 shear pins (x24)	☐ Nosecone altimeter sled ☐ StratoLogger altimeter (x4) ☐Battery clips (x2) ☐Battery holster cover ☐ PVC rocket stands

Vehicle Inspection Prior to Launch	
_____	1. Place airframe on rocket stand.
WARNING	If rocket stands are not used, the airframe could roll off the work surface.
_____	2. Inspect all fins for damage.
_____	3. Inspect nosecone for damage.
_____	4. Inspect airframe for damage.
_____	5. Inspect permanent internal airframe components for damage.
_____	6. Inspect all electronic sleds for damage.
_____	7. Inspect all couplers and shear pin holes, and threads for damage.
_____	8. Inspect motor centering ring epoxy and retainer for damage.
_____	9. Inspect carbon fiber components for signs of delamination or structural weakness.
_____	10. Ensure that all sections and screw holes align without forcing them together. There should also not be extra room between the coupler and airframe.
WARNING	If sections are too tight, they may not be able to separate with black powder. Sections that are too Loose may cause an unstable flight.
WARNING	Any damage must be reported to Vehicle lead and Safety officer to mitigate possible impact on later launches.
Black powder charge preparation	
DANGER	E-matches are explosive and must be kept clear of batteries, sparks, and open flames to avoid premature firing. No open flames are allowed within 25 feet of black powder.
CAUTION	Safety glasses must be worn when using a drill and while handling e-matches.
____	1. Drill a 1/8” hole in the bottom of each black powder capsule.
_____	2. Unwind an e-match and remove the protective cap from the pyrotechnic end.
_____	3. Feed the wire end of the e-match through the top of the capsule and through the hole in the base of the capsule until the pyrotechnic end is against the bottom of the inside of the capsule.
	4. Secure the e-match to the capsule and close the hole with Gorilla tape.
WARNING	Ensure that black powder does not leak out of the capsule. Leakage could cause a failed vehicle separation or failed recovery, leading to a catastrophic vehicle failure. Any spilled powder must be cleaned up with painter’s tape, lint roller, or other non-sparking/non-static producing tools.
____	5. Two people are required to measure and handle black powder

⚠WARNING	Two people are needed to ensure the intended amounts of black powder are used in charges and black powder should only be handled outdoors to provide proper ventilation.
____	5. Fill each capsule with the calculated number of cc's of black powder.
____	6. Fill excess capsule space with cellulose insulation to ensure that the pyrotechnic end of the e-match is submerged in the powder regardless of capsule orientation.
⚠WARNING	An unsubmerged e-match may not ignite the black powder and could cause a failed vehicle separation or failed recovery, leading to a catastrophic vehicle failure.
____	7. Close the capsule with the lid, checking that there is a tight fit.
____	8. Repeat steps 1 through 7 for each required charge.
	9. Unwind 2 e-matches for each ARRD and remove the protective caps from the pyrotechnic ends.
	10. Insert 2 modified e-matches into the ARRDs.
____	11. Assemble the ARRDs according to the manufacturer's instructions.
	12. Store all created charges and ARRDs in the explosive's box.
Vehicle Avionics Preparation	
____	1. Charge all 4 GPS units prior to the day of launch.
____	2. Verify proper avionic shielding.
⚠WARNING	Ensure that the entire inside of the avionics bay is properly shielded to protect from Interference. In the incident that interference occurs, pyrotechnic devices may be actuated Prematurely, causing potential harm to personnel and/or mission failure.
____	3. Verify StratoLogger CF altimeters are properly programmed.
____	4. Identify altimeter that will be used to record an official apogee measurement.
Note:	A NASA official will mark the official competition altimeter during LRR.
____	5. Verify 9V battery has a minimum charge of 8.7 V.
____	6. Mount StratoLoggers onto standoffs on sustainer altimeter sled using #4-40 shear pins
____	7. Attach batteries to battery clips and install into holster.
____	8. Attach battery holster cover using 4, #4-40 shear pins
____	9. Ensure screw switches are turned off and wire screw switches to switch terminal on the StratoLogger.
⚠ DANGER	Altimeters must remain in the OFF position until the vehicle is upright on the launch pad. Skipping this step could lead to premature black powder detonation and serious danger to all personnel.
____	10. Wire battery to +/- terminal on StratoLogger.
____	11. Wire main and drogue terminals on StratoLogger to terminal blocks on the nosecone.
____	12. Install altimeter sled into avionics bay
⚠WARNING	Ensure that the sled is fixed properly to eliminate damage that could prevent recovery from being properly deployed.
____	13. Verify proper screw switch alignment with access hole
⚠WARNING	If the alignment hole and screw switch do not align, the altimeters will not be able to be engaged at the launch pad, possibly delaying the launch.
____	14. Install AIM XTRA into 3D printed sled.
____	15. Install the AIM XTRA sled onto the nosecone all thread and tighten the sled down with nuts.
____	16. Install the Eggfinder into 3D printed sled.
____	17. Install the Eggfinder sled into the Payload recovery coupler.
____	18. Mount the Trackimo in the Payload coupler.
Prior to leaving for launch site: Propulsion	
Equipment to Pack or Prepare	

☐ Gorilla Glue ☐ Grease	☐ Booster bay ☐ Booster stand	☐ AeroTech L2200-G motor ☐ Motor retainer ☐ #10-32 shoulder bolt (x3)

⚠ CAUTION	Protective gloves must be worn while applying grease to the motor.
____	1. The vehicle lead, Matthew Cosgrove, will be responsible for fully preparing the motor and

	installing it within the casing. Only members with Level 2 certification may assist.
____	2. Slide motor casing fully into the motor mount tube.
____	3. Attach the motor retainer to the fin retainer via #10-32 shoulder bolts
____	4. Set the completely assembled bay on the stand, ensure that the bay is not resting on the fins
Note:	If any damage is identified, immediately inform each co-captain and the safety officer. The launch vehicle will be deemed safe to fly or a corrective action will be decided upon.

7.1.1.3. Payload Safety Checklist

To be checked and signed by Payload Lead and team member when *all* steps are completed, indicating that Payload team is PREPARED FOR LAUNCH.

1. ______________________ 2. ______________________

Prior to leaving for launch site

Equipment to Pack or Prepare

☐ ROCS Assembly	☐ DTS Antenna	☐ Extra CES Components	☐ MicroSD Card (x2)
☐ Socket Head Cap Screws (x30)	☐ DTS Receiver Module	☐ Allen Key Set	☐ M-2 Screws (x10)
☐ Extra Controller Batteries (x2)	☐ DTS Transmitter Module	☐ Multimeter	☐ 4-40 Screws (x30)
☐ LiPo Voltage Indicator	☐ DTS Transmitter Yagi	☐ Electrical Wire	☐ 8-32 Screws (x30)
☐ LiPo Battery Charger	☐ Extra Motor Battery	☐ Wire Stripper	☒ Pliers
☐ Rover	☐ Loctite	☐ Laptop	

Rover Orientation Correction System (ROCS)	
____	1. Inspect ROCS assembly for damage or loss of material
Note:	*If any damage is identified, inform Payload Lead to determine degree of damage and if corrective action is necessary.*
____	2. Roll ROCS assembly to verify functionality
Rover Locking Mechanism (RLM)	
____	1. Verify functionality of RLM by applying 12V to RLM motor using power supply
____	2. Fit test rover with the RLM
Deployment Trigger System (DTS)	
____	1. Verify transmitter power source is fully charged
____	2. Wire receiver module through slip ring flange and confirm free range of motion.
WARNING	If the slip ring flange does not rotate freely, the DTS wire can disconnect, leading to an impossible rover deployment and failed payload mission.
____	3. Attach receiver antenna to exterior of airframe
Rover Body Structures (RBS)	
____	1. Install battery mount.
____	2. Install motor mount.
____	3. Install OAS/SIS mount.
____	4. Verify locking bracket is secured to rear of rover.
WARNING	*Failure to verify locking bracket is secure could lead to premature deployment of the rover.*
Rover Drive System (RDS)	
____	1. Verify functionality of main drive motors by applying 12V to each motor using power supply
WARNING	*Bevel gears rotating can cause injury to personnel and should be avoided*
____	2. Mount all wheel bearings with shafts to the RBS
____	3. Mount idler and drive wheels to shafts
____	4. Install tread onto wheels
Obstacle Avoidance System (OAS)	
____	1. Secure lidar to mount and fully tighten all screws
____	2. Install lidar mount onto servo motor
____	3. Install servo motor onto mount inside the RBS
Solar Array System (SAS)	
____	1. Mount spring hinge to tower assembly

_____	2. Install assembly into RBS by screwing spring hinge into RBS and fully tightening all screws
_____	3. Verify spring hinge functionality by allowing it to actuate the tower assembly
_____	4. Verify deployment motor functionality by applying 12V to the motor using a power supply
_____	5. Verify locking motor functionality by applying 3.3V to the motor using a power supply
Surface Imaging System (SIS)	
_____	1. Secure camera module to mount and fully tighten all screws
Control Electronics System (CES)	
_____	1. Load flight ready software onto Feather M0 Bluefruit LE
_____	2. Install PCB into RBS
_____	3. Route lidar sensor, camera module, servo motor, SAS locking motor, and solar panel wires to indicated header pins on CES PCB
Note:	*Use tape to secure wires and ensure no tangling can occur. Cut excess wire to ensure wires cannot be crossed*
_____	4. Route main drive motor (x2) and SAS deployment motor to FeatherWing motor driver and screw down terminal blocks
Note:	*Use tape to secure wires and ensure no tangling can occur. Cut excess wire to ensure wires cannot be crossed. Connect wires in polarity according to colors (Red–Positive, Black–Negative)*
_____	5. Install switches for motor and controller batteries
_____	6. Verify full charge of motor and controller batteries
WARNING	*Failure to fully charge batteries can cause loss of power to electronics and failure of the mission*
Final Assembly	
_____	1. Integrate rover with ROCS/RLM assembly.
_____	2. Connect RLM motor to FeatherWing motor driver and screw down terminal blocks.
_____	3. Route slip ring flange wires for DTS receiver module to indicated header on CES PCB.
_____	4. Press payload into payload bay of the launch vehicle aligning ROCS with drilled screw holes.
_____	5. Screw all 20 Socket Head Caps Screws into ROCS securing the payload inside the airframe.
_____	6. Loctite all screws and bolts.
WARNING	Loose screws or bolts could further loosen during flight, falling out of the payload themselves or allowing another component to fall off the rover during flight.
Launch Site Procedure	
_____	1. Flip motor battery switch to on position
_____	2. Flip controller battery switch to on position
_____	3. Flip DTS transmitter power source to on position
_____	4. Hand-off payload to vehicle team for final launch vehicle assembly
To be checked and signed by Payload Lead and team member when *all* steps are completed, indicating that Payload team is PREPARED FOR LAUNCH.	
1. ______________________________ 2. ______________________________	

7.1.1.4. Variable Drag System (VDS) Safety Checklist

To be checked and signed by VDS Lead and team member when *all* steps are completed, indicating that VDS team is PREPARED FOR LAUNCH.		
1. ______________________________ 2. ______________________________		
Prior to leaving for launch site		
Equipment to Pack or Prepare		
☐ Precision flathead screwdriver ☐ Standard Philips head screwdriver ☐ Duracell 9V battery (x4) ☐4-40 shear pins (x24)	☐ Fuse shunt ☐ Electronics Assembly ☐ Extra 22 AWG wires ☐ Teensy 3.6 ☐ Neverrest40 DC motor ☐ Encoder cable	☐ Nosecone altimeter sled ☐ StratoLogger altimeter (x4) ☐Battery clips (x2) ☐Battery holster cover ☐ SD Card ☐ Wire Cutters/Strippers

☐ Multimeter ☐ Black Tool box ☐ VDS connector cable ☐ SD adapter	☐ Laptop ☐ Banana Plug Cables ☐ USB micro B cable ☐ Electronics Enclosure	☐ 7.4V LIPO battery (stored in insulated battery bag) ☐ 11.1V LIPO battery (stored in insulated battery bag) ☐ Fuse

Prior to leaving for launch site	
_____	1.Before handling VDS electronics, ensure that you are properly grounded. This can be done at Launch by touching the chassis of a car and removing extra layers of clothing.
_____	2. Check battery connector for frayed or loose wires
_____	3. Flip power switches to 'off'
_	4. Insert microSD card into its slot
_____	5. Inspect Board for loose wires or connections
_____	6. Perform continuity checks for active signal
_____	7. Turn off debugging
_____	8. Put software in test mode and perform unit test
_____	9. Ensure the software IS NOT in test mode
_____	10. Check mass parameters and other physical constants
_____	11. Upload this software and verify these things
_____	12. Zip up software and put in documents/flights/2017-2018/VDS/VDS V3/Launch Data/[Launch Date]
At launch Site Procedures: VDS	
_____	1. Install Teensy onto the electronics assembly
_____	2. Plug in USB B into Teensy and computer
_____	3. Open putty (or any serial monitor)
_____	4. Press button on Teensy to reset
_____	5. Connect to COM port(check device manager to learn what yours is)
_____	6. Verify that the VDS V3 title screen prints out. If not type 'S'.
_____	7. Type 'S' to run a system check. Verify that the SD card, BMP280 and BNO055 initialized correctly.
_____	8. Type 'B' to verify that the BMP280 is reading nominally. Type anything to exit the test.
_____	9. Type 'A' to verify that the BNO055 is reading nominally. Type anything to exit the test.
_____	10. When ready, type 'F' to enter flight mode.
_____	11. Verify that the software has entered flight mode.
_____	12. Verify that there is no connection for the 7.4v and the 11.1v batteries by checking power switches.
_____	13. Plug in 7.4v LIPO to its clip. NOTE BATTERY POLARITY.
_____	14. Insert 7.4v into slot.
_____	15. Flip 7.4v power on.
_____	16. Plug in 11.1v LIPO. NOTE BATTERY POLARITY.
_____	17. Insert 11.1v power on.
_____	18. Perform motor check by pressing 'M'
_____	19. Verify motor actuates.
_____	20. Unplug USB. Make sure LED on Teensy remains on.
_____	21. Fix lid on enclosure.
_____	22. Give to vehicle team to install.

	After Recovery
____	1. Remove cover from sled.
____	2. Shut down motor power.
____	3. Plug USB B into Teensy.
____	4. Launch Putty.
____	5. Press 'E' (or any character other than 'F') to end flight mode.
____	6. Remove SD card.
____	7. Shut down power to board
____	8. Store electronics in ESD-safe bag

7.1.1.5. Launch Pad Safety Checklist

To be checked and signed by Vehicle Lead and Safety Officer when *all* steps are completed, indicating that the rocket is a GO FOR LAUNCH.		
1. ______________________ 2. ______________________		
Prior to launch setup		
Equipment Required		
☐ TPL-72 Tank launch pad ☐ 1010 launch rail	☐ Multimeter ☐ Wind vane	☐ Wind vane display ☐ Zip ties
Launch Pad Setup		
____	1. Assemble the launch pad with the 1010 launch rail.	
____	2. Position the assembled launch pad a minimum of 1,500 feet away from any occupied building And 300 feet away from any personnel.	
⚠WARNING	Ensure that the tripod launch pad does not tilt more than 20° from vertical. If the launch pad is not stable, the vehicle flight may be negatively impacted.	
____	3. Clear dry grass within 100 feet of the launch pad	
	4. Fix the wind vane to the launch pad	
____	5. Confirm that the wind vane display shows the wind speed.	
Vehicle Setup on Launch Pad		
Equipment to Pack or Prepare		
☐ Assembled vehicle ☐ Philips head screw driver ☐ Level	☐ Rocket perch ☐ Launch box ☐ Pencil	☐ Checklist ☐ Level 2 certification card
____	1. Verify flight card has been properly filled out and launch permission has been granted by RSO.	
⚠CAUTION	Safety glasses are required to be worn at the launch pad.	
____	2. Select 3 team members to transport the assembled vehicle from the preparation area to the launch pad	
⚠WARNING	Confirm the members can comfortably carry the vehicle together to eliminate the risk of a dropped vehicle.	
Note:	Only required personnel should accompany the rocket to the launch pad to eliminate distraction	
____	2. Slide the vehicle onto the rail. If the section of airframe does not slide freely up and down the entire length of the launch rail, see the trouble shooting section.	
⚠WARNING	This step cannot be considered completed until the airframe moves freely. If the airframe is caught on the launch rail, the vehicle will experience too much friction, jeopardizing successful flight.	
____	3. Ensure that the vehicle does not rest on the fins. Install a rocket perch if required.	
____	4. Tilt and rotate the launch pad as directed by RSO. Use the level to ensure desired angle.	
____	5. Arm all electronics in the following order:	
	___1. Payload ___2. Cameras	___3. Altimeters ____ StratoLoggers in nosecone

		____ StratoLoggers in lower airframe
_____	6. Check for continuity between the launch box and launch pad.	
_____	7. Clear launch pad area and do not return until range has been reopened by the RSO.	
Igniter Installation		
Equipment to Pack or Prepare		
☐ Vehicle ☐ Igniter	☐ Masking tape ☐ Modified nozzle cap	
CAUTION	Safety glasses are required during igniter installation.	
_____	1. Insert the coated end of igniter through motor nozzle throat until it stops against the smoke charge element.	
DANGER	Igniters are explosive and must be kept clear of batteries, sparks, and open flames to avoid accidental firing.	
_____	2. Secure the igniter to the nozzle with a piece of masking tape or the modified nozzle cap.	

7.1.1.6. Troubleshooting Safety Checklist

Problem: The motor does not ignite after engaging launch button for 5 seconds	
Immediate steps:	
_____	1. Disengage safety interlock
_____	2. Do not begin to approach vehicle for a minimum of 60 seconds.
WARNING	The rocket may still launch in the case of a hang fire. Prematurely approaching the vehicle may endanger the team member.
Identify Situation	
Igniter no longer installed connected	
Definition: The igniter is no longer properly inserted against the smoke charge element of the motor.	
Identifier: The igniter is visibly outside of the motor.	
Igniter clips are no longer connected	
Definition: The igniter is no longer connected to the igniter clips and continuity is lost	
Identifier: Igniter is visibly disconnected from the igniter clips.	
Misfire or Hang Fire	
Definition: A misfired motor never ignites while a hang fire ignites after a substantial delay.	
Identifier: The igniter is visually intact and connected to the igniter clips but the motor never ignited	
Steps to Follow	
CAUTION	Safety glasses are required at the launch pad.
_____	3. Disconnect the igniter clip from the igniter.
WARNING	Do not place fingers or hands underneath the vehicle or any part of your body in front of the Vehicle. In the event of a hang fire, exhaust may still exit the motor nozzle.
	4. Remove the vehicle from the launch rail.
WARNING	Keep the motor nozzle pointed away from all handler faces and bodies
_____	5. Remove igniter
DANGER	Igniters are explosive and must be kept clear of batteries, sparks, and open flames to avoid accidental firing.
	6. Repeat motor preparation, igniter installation, and vehicle setup on launch pad.

7.1.1.7. Post Flight Inspection Safety Checklist

To be checked and signed by Vehicle Lead, Recovery Lead, Payload Lead, and Safety Officer when *all* steps are completed, indicating that the rocket has been completely EVALUATED POST FLIGHT.	
1. ______________________________	2. ______________________________
3. ______________________________	4. ______________________________
Altitude Achieved: ______________________________	
Motor Used: __________________________________	

Launch Location: ______________________________	
Launch Time: _________________________________	
Ground Wind Speed: ___________________________	
Estimated Shear Wind Speed: ____________________	
Post Launch Recovery Assessment	
2 Recovery team members must inspect the following components following every launch.	
Requirements	
☐ Vehicle	☐ 2 Recovery team members
_____	1. Inspect all shroud lines for any damage or burn marks.
Damage: Y / N	Notes: ____
_____	2. Inspect all shroud attachment points for damage.
Damage: Y / N	Notes: ____
_____	3. Inspect *booster drogue* canopy for damage like holes, burns, or stretching.
Damage: Y / N	Notes: ____
_____	4. Inspect *booster main* canopy for damage like holes, burns, or stretching.
Damage: Y / N	Notes: ____
_____	5. Inspect *payload bay drogue* canopy for damage like holes, burns, or stretching.
Damage: Y / N	Notes: ____
_____	6. Inspect *payload bay main* canopy for damage like holes, burns, or stretching.
Damage: Y / N	Notes: ____
_____	7. Inspect *booster main* deployment bag for torn fabric that would indicate a snag.
Damage: Y / N	Notes: ____
_____	8. Inspect *payload bay main* deployment bag for torn fabric that would indicate a snag.
Damage: Y / N	Notes: ____
_____	9. Inspect ARRDs and altimeters for signs of damage or unacceptable wear.
Damage: Y / N	Notes: ____
WARNING	Any damage must be reported to Recovery lead and Safety officer to mitigate possible impact on later launches.
Repair Plan: ____ ____ ____	
Recovery team signatures for post launch inspection completion.	
1. ______________________________________ 2. ______________________________________	
Post Launch Vehicle Assessment	
2 Vehicle team members must inspect the following components following every launch.	
Equipment Required	
☐ Vehicle ☐ Sealable bag	☐ Acetone ☐ 2 Vehicle team members
_____	1. All fins for damage.
Damage: Y / N	Notes: ____
_____	2. Inspect nosecone for damage.
Damage: Y / N	Notes: ____
_____	3. Inspect airframe for damage.
Damage: Y / N	Notes: ____
_____	4. Inspect permanent internal airframe components for damage.
Damage: Y / N	Notes: ____
_____	5. Inspect all electronic sleds for damage.
Damage: Y / N	Notes: ____
_____	6. Inspect all couplers and shear pin holes for damage.
Damage: Y / N	Notes: ____
_____	7. Inspect motor centering ring epoxy and retainer for damage.

Damage: Y / N	Notes: ____
_____	8. Inspect carbon fiber components for signs of delamination or structural weakness
Damage: Y / N	Notes: ____
_____	9. Inspect all bolts and threads for signs of galling or deformation.
Damage: Y / N	Notes: ____
_____	9. Inspect all epoxied joints for cracks or edge peeling.
Damage: Y / N	Notes: ____
WARNING	Any damage must be reported to Vehicle lead and Safety officer to mitigate possible impact on later launches.
Repair Plan: ____ ____ ____	
Vehicle team signatures for post launch inspection completion.	
1. ____ 2. ____	
Motor casing cleaning and inspection	
_____	1. Carefully disassemble the motor casing.
WARNING	Ensure that no casing components are dropped or lost. The threaded components could be significantly damaged if dropped, requiring a new casing for a safe launch. Also ensure that the O-ring seal is not thrown away during disassembly.
_____	2. Remove used motor grains from casing.
_____	3. Place grains in a sealable back and place in an inert garbage can.
_____	4. Soak the forward seal in acetone.
_____	5. Dip a paper towel in acetone and wipe down the rest of the motor casing
_____	5. Inspect all components for damage following cleaning.
WARNING	Any damage must be reported to Vehicle lead and Safety officer to mitigate possible impact on later launches.
Vehicle team signatures for post launch motor cleaning and inspection.	
1. ____ 2. ____	

Payload Post Launch Assessment	
1. Walk DTS transmitter to within 50 feet of payload bay	
2. Query RSO for permission to initiate payload mission	
3. Send deployment trigger to the payload	
4. Inspect each phase of the rover's mission for unexpected performance characteristics	
5. Allow SIS to take pictures for 2 minutes	
6. Flip all switches to off position, powering down all electronics	
7. Remove and inspect ROCS/RLM assembly	
Damage: Y / N	Notes: ____
8. Remove microSD card from FeatherWing Adalogger	
9. Log all data from microSD card for analysis and controls modifications	
Payload team signatures confirming post launch assessment completion.	
1. ____ 2. ____	

7.2. Risk Assessment Matrices

7.2.1. Payload Equipment Hazard Risk Assessment

Hazard	Cause	Effect	RBM	Mitigation	Verifications	RWM
Pinched, cut, or disconnected wire.	The DTS receiver wires could be twisted as the ROCS spins during the vehicle flight and recovery.	Damaged wire could short an electrical system, the payload would not be able to receive the signal to exit the payload bay, resulting in a failed payload mission.	2A	The receiver wires will pass through a slip ring flange that will be mounted through the bulk plate.	The payload launch checklist will require confirmation that the slip ring flange rotates freely and prevents wire twisting. 7.1.1.3	2D
Payload bay does not self-orient correctly.	Unexpected contact between ROCS component(s) and airframe prevents the free rotation of the ROCS	The rover would be unable to exit the airframe due to unacceptable gyroscopic orientation readings, causing the payload to fail the mission.	2B	The SAS locking motor will hold the SAS tower assembly in the stowed position until the rover reaches its final destination, preventing it from contacting the airframe. The rover has been designed so that all other components are housed within the RBS.	The free rotation of the ROCS will be tested in requirement ROCS-3 and the integrity of the ROCS and RBS will be tested in requirement ROCS-4 and requirement RBS-3 to withstand all loads experienced during launch. 6.1.8	2D
Black powder residue on light sensitive rover components.	The rover is housed in the bay where the booster separation charge will detonate, expelling black powder residue on exposed surfaces.	The lidar sensor or solar panels may not get adequate light to function properly.	4A	The black powder charge capsule will be covered by a piece of fire retardant cloth that will retain some of the residue.	The current design is a Nomex blast cone that will be tested in a full-scale launch.	4C
Component falls out of payload bay during recovery.	A component could be loosened or broken due to large launch or parachute opening forces, concussive black powder charge explosions, or vibration.	A lost component is a risk to team members or spectators. The payload may not perform as intended, possibly causing a failed payload mission.	1C	All components will be properly attached to the rover body with a sufficient number of mechanical fasteners and Loctite.	Electronic and mechanical systems will be verified secure through requirement RBS-2. Requirement RBS-3 will verify that the rover can withstand high loads during flight. Launch procedures will require confirmation of Loctite on all payload fasteners prior to final assembly. 6.1.8 7.1.1.3	1E
Premature deployment.	Extraneous signal not transmitted by the team releases the RLM prior to the bay landing safely or mechanical failure of the RLM due to unexpected recovery loads.	The rover would no longer be axially fixed to the ROCS and it would fall out of the open end of the payload bay, becoming a risk to team members or spectators. The payload may sustain extensive damage upon landing, preventing it from performing as intended, causing a failed payload mission.	1C	A unique deployment signal will be sent by a team member after gaining RSO permission. After the unique signal is received and confirmed, the gyroscope will check that the payload bay is appropriately oriented with respect to vertical, the RLM will unlock. The RLM is in the locked configuration by default.	Team derived requirement DTS-5 will demonstrate the unique deployment signal reception and requirement CES-1 will test the accuracy of the gyroscope measurement. Requirement RLM-2 will demonstrate the RLM ability to keep the rover fixed in a as integrated locked state. 6.1.8	1D
ROCS socket head cap screws shear.	1. Bearing misalignment causes uneven load sharing across socket head cap screws 2. Unexpected take off loads 3. Unexpected main parachute opening force	ROCS is unable to properly orient the rover for deployment, causing gyroscopic measurements to prevent the release of the RLM. The rover would not be able to exit the airframe, leading to a failed payload mission.	2C	20 socket head cap screws will be used to properly align and secure the ROCS to the payload bay. The screws will share the loads that are experienced during flight.	The strength of the socket head cap screws will be tested by a successful full-scale launch per requirement ROCS-4. 6.1.8	2E

Obstructed rover path.	Field debris, launch vehicle, or rough terrain prevent the rover from being able to drive in a straight line.	The rover will not be able to drive 5 feet away from the vehicle resulting in a failed payload mission.	2B	The OAS will select the least obstructed path within the 156° field of view and the RDS will be designed to transverse different terrains and inclines of at least 20 degrees from horizontal.	Team derived OAS requirements, requirement CES-5, and requirement CES-6 will respectively test and demonstrate the selection of the least obstructed path. 6.1.8	2D
SAS panel support arms are not able to fully deploy.	Panel support arm(s) or tower assembly are damaged by contacting the airframe during flight or contacting launch field debris following rover deployment.	Rover is unable to deploy solar panels, leading to a failed payload mission.	2C	The panel support arms will be locked in a stowed position until the rover reaches its final destination. The deployed panel support arms were designed to fit inside the footprint of the RDS to avoid contact with debris at any point during panel deployment.	Requirement SAS-1 and requirement SAS-4 will verify complete SAS tower actuation from the stowed configuration through inspection and demonstration. Actuation will also be checked in the payload launch procedure checklist. 6.1.8	2E
Rover drive belt becomes misaligned.	The passive pulley bearings were inaccurately manufactured with a sloped surface or the pulley bearings were improperly mounted. Large amounts of friction from terrain may pull the drive belt away from the RBS.	Drive belt begins to walk off or even fall off the RDS. They payload may not be able to drive 5 feet if one or both drive belts are lost, leading to a failed payload mission.	2B	The passive pulleys were designed to maintain tread alignment with the addition of lips along each edge. The rear drive pulleys have taller lips that extend past the top of the tread.	Team derived requirement RDS-4 will demonstrate the proper alignment and traction required to transverse over various terrains. 6.1.8	2D
Batteries are not fully charged.	Batteries were installed prior to full charge, lose charge during assembly or launch setup, or are impacted and damaged during flight.	Insufficient power to motor driver and control board, leaving the rover locked to the RBS unable to deploy.	2B	Voltage indicators will be used to ensure full charge before installation and final assembly. The batteries will be contained in a mount inside the rover body to avoid damage.	Launch procedures will require the batteries to be checked prior to integration. The Payload batteries will be verified secure through team derived requirement RBS-2. Requirement CES-8 will test controller battery and requirement CES-9 will test motor battery. 6.1.8	2E
Electrostatic discharge to sensor or control electronics.	Electrostatic build up on team member.	Shorts and potential component failure.	2D	Grounding mats and wrist straps must be used when testing electronics.	Test procedures, like that required for team derived requirement OAS-2, all team derived CES requirements, and SOW 4.5.3 verification, require the use of grounding mats and wrist bands. Payload electronics test procedures will require the use of grounding mats and wrist bands. 6.1.8	2E

Table 77: Payload Equipment Hazard Risk Assessment.

7.2.2. Vehicle Equipment Hazard Risk Assessment

Hazard	Cause	Effect	RBM	Mitigation	Verifications	RWM
Rocket drop (INERT).	Mishandling of the rocket during transportation or rocket supports were not used to hold the rocket horizontally.	Damage to fins and electrical components if installed. There may be minimal damage and scratches to vehicle airframe.	2C	The rocket has been designed to be durable to survive loads encountered during flight and upon landing. Careful handling will be practiced while transporting the rocket and storing the rocket with custom made PVC rocket supports.	Vehicle launch procedures require airframe bays to be placed on rocket supports during vehicle inspection and assembly. 7.1.1.2	2E

Rocket drop (LIVE).	Mishandling of the rocket during transportation or rocket supports were not used to hold the rocket horizontally.	If charges do not detonate, damage to fins and electrical components if installed. There may be minimal damage and scratches to vehicle airframe. If charges go off, there would be a serious safety threat to personnel in the area and possibly significant damage to the rocket.	1C	The rocket has been designed to endure all loads encountered during flight and upon landing. Careful handling should be practiced while transporting the rocket.	Vehicle launch procedures require bays to be placed on rocket supports during vehicle assembly. Vehicle launch procedures require 3 identified team members to transport the live rocket to the launch pad. 7.1.1.5	1E
Black powder charges go off prematurely.	The altimeters send a false reading or an open flame sets off the charge or avionics are not properly shielded.	A serious safety threat to personnel in the area and possibly significant damage to the rocket could result..	1C	All electronics will be kept in their OFF state until the latest time they can be enabled. Altimeters are not to be armed until the rocket is in the launch pad. Open flames and other heat sources are prohibited in the area.	Vehicle launch procedures require altimeters remain OFF until the vehicle is upright on the launch pad and that avionics are properly shielded. Vehicle launch procedures also prevent black powder charges or charge preparation within 25 feet of an open flame. 7.1.1.2	1E
Seized nut or bolt due to galling or cross threading.	Repetitive uninstalling and reinstalling of parts made of materials prone to galling or excessive friction caused by poor alignment	Component becomes unusable, potentially ruining expensive, custom machined parts. Rework may be required and would depend on the location of the affected component.	2D	If there is resistance to proper alignment, the sections will not be forced together to fit. The cause for the poor fit will be evaluated and corrected.	Vehicle launch procedures require all threads to be checked and evaluated for damage prior to launch and Threads will also be evaluated following launch. Vehicle launch procedures require easy fit and alignment of screws and bolts without excess force or friction. 7.1.1.2	2E
Screw switches to arm electronics are inaccessible.	Avionics sleds are not properly aligned with access holes.	The electronics will not be able to be armed on the pad or will require additional access holes to be cut.	1D	Proper altimeter orientation will be identified on the airframe.	Vehicle launch procedures will require all altimeter access holes to be properly aligned before the vehicle is cleared to leave for the launch pad. 7.1.1.2	1E
Bays are improperly aligned.	Bay symmetry makes it difficult to identify how sections align.	Shear pins may not fit or thread into holes easily, causing them to experience more loading than intended, possibly failing during flight	1C	Witness triangles of varying sizes and shapes will be used to allow for the rocket to align in one orientation.	Vehicle launch procedures require easy fit and alignment of screws and bolts without excess force or friction. 7.1.1.2	1E
Lost GPS signal.	GPS unit damage or power loss.	Potential to lose location of the rocket temporarily or permanently.	2D	GPS units shall be charged prior to the day of launch and will be securely mounted inside the vehicle.	Vehicle launch procedures require GPS units to be fully charged prior to packing and secure mounting. 7.1.1.2	2E
Unstable flight.	Shifted center of gravity due to changes in component design and weight.	Rocket does not fly in anticipated path, possibly causing it to follow a ballistic path endangering all team members and launch spectators.	1B	Ballast will be added if components like the payload are missing from the rocket to maintain calculated center of gravity and stability.	Simulations have been run in OpenRocket to identify the centers of pressure and gravity. Vehicle launch procedures will require ballast to be added to maintain the locations in accordance with SOW 2.19.2.1 and 2.12.2.1.1.	1E
Fin flutter..	Inadequate material strength or damage caused by previous flight.	Fins detach from vehicle causing the rocket to become unstable and fly in an unanticipated path. The rocket could CATO, follow a ballistic path, and endanger all team members and launch spectators.	1B	Fin flutter calculations and simulations will be conducted with fins made of carbon fiber.	Fin flutter calculations and AeroFinSim analysis shows that the carbon fiber fins will not be impacted by fin flutter. Vehicle launch procedures require inspection of fins prior to launch and inspection of fins following launch. 7.1.1.7	1E

Important component left at build site.	Packing list omitted an item.	Temporarily delayed launch, possible risking full-scale launch completion	1C	Packing lists will be thorough and will include all components needed to assemble the vehicle as well as those needed to make small repairs. All sub-team leads will agree to packing lists and add items as necessary. Additional fasteners are brought to the launch site for backup.	Launch checklists detail general equipment and supplies that are needed for each sub-team. Launch checklists must be signed by two members as being completed prior to departing for launch. 7.1.1.2	1E
Zippering.	Recover lines catch and tear through airframe.	Damage to edges of airframe, reducing the flight and/or airframe stability.	2C	The airframe will be made of wound carbon fiber, a strong material. The airframe will be inspected after each recovery to identify any signs of zippering.	Launch checklists require thorough inspection of the vehicle following each recovery.	2D
Carbon fiber delamination.	Excessive heat or fire.	Substantial weakening of the airframe.	1D	The launch vehicle will be stored away from all heating elements. In the event of fire, the entire vehicle will be inspected for damages.	Vehicle launch checklists require all carbon fiber components to be inspected for damage after recovery.	1E
Damage to airframe structure.	Flawed recovery or impact with field debris during recovery.	Substantial weakening of the airframe, possibly leading to buckling or shearing of airframe during next flight.	1C	The vehicle will be made of strong carbon fiber and it will be inspected following each flight.	Vehicle launch checklists require thorough inspection after each launch and recovery.	1E
Unexpected friction between vehicle and launch rail.	Misalignment or ill-fitting launch buttons.	Unstable flight or CATO.	1D	The airframe must slide onto the launch rail without any resistance. If section of airframe does not slide freely up and down the entire length of the launch rail, the vehicle will not be allowed to launch.	Vehicle launch checklists will require the airframe to slide freely up and down the entire length of the launch rail prior to launch. 7.1.1.5	1E

Table 78: Vehicle Equipment Hazard Risk Assessment

7.2.3. Propulsion Equipment Hazard Risk Assessment

Hazard	Cause	Effect	RBM	Mitigation	Verifications	RWM
Catastrophe at take-off (CATO).	Defective or improperly packed motor, defect in the motor casing, or improper cleaning of motor casing following launch.	Extensive damage or delamination of carbon fiber airframe, bulk plates may fail, recovery may not deploy or may not fully deploy to be effective.	1C	2 new motor casings were purchased for the season, eliminating possible damage from previous launches. The motor will only be packed by certified members and all motors will be purchased from Chris' Rocket Supplies, a certified provider.	Vehicle launch procedures require the Level 2 certified vehicle lead to pack the motor in strict accordance with the manufacturer's instructions. 7.1.1.2	1D
Propellant does not ignite.	Incorrect motor packing or igniter installation.	Launch must be delayed or postponed. The vehicle must be removed from the launch pad and a new motor would have to be assembled and installed.	2D	The motor will only be packed by certified members and the igniter has specific installation steps.	Vehicle launch procedures require the Level 2 certified vehicle lead to pack the motor, igniter installation steps, and detail troubleshooting for a motor that does not ignite. 7.1.1.2	2E
Premature propellant burnout.	Incorrect motor packing or faulty motor grain.	The vehicle may not reach the intended altitude.	2D	The motor will only be packed by certified members and all motors will be purchased from Chris' Rocket Supplies, a certified provider. Because the vehicle will overshoot the intended apogee and will be slowed down with the VDS, a slightly premature burnout may still allow the vehicle to reach the intended altitude.	Vehicle launch procedures require the Level 2 certified vehicle lead to pack the motor. The VDS simulation has shown the system's ability to reduce the vehicle's apogee. 7.1.1.2	2E

Improper assembly of motor.	Incorrect order of motor grain installation, casing threads or O-rings were not greased as instructed by the motor manufacturer.	The motor may misfire or hang fire or could result in a CATO.	1D	The motor will only be packed by certified members with strict adherence to the manufacturer's instructions.	Vehicle launch procedures require the Level 2 certified vehicle lead to pack the motor. 7.1.1.2	1E
Motor retainer fails.	Excessive opening forces experienced during recovery.	The motor is no longer held in the vehicle and could fall near team members or launch spectators.	1C	Analysis shows the motor retainer has a factor of safety of 3.65.	Motor retainer FEA was conducted to show the integrity of the design. Vehicle launch procedures require inspection of motor casing following post launch cleaning. 7.1.1.7	1E
Centering ring epoxy failure.	Excessive opening forces or insufficient epoxy levels.	Could result in excessive loads on the remaining centering rings. If all centering reals fail, the motor would shoot up the center of the vehicle.	1D	There are 3 centering rings in the vehicle design for redundancy. Each ring is rated to carry the loads with a factor of safety greater than 2.0.	FEA was conducted on the centering ring. Vehicle launch procedures require epoxy joints to be inspected following every launch. 7.1.1.7	1E
Propellant burns through casing.	Incorrect motor packing, motor casing threads or O-rings were not greased as instructed by the motor manufacturer, or motor casing defects.	The propellant could catch the airframe epoxy on fire, causing structural carbon fiber to delaminate, severely weakening the strength of the vehicle and causing a CATO. The path of the vehicle could also be significantly affected due to abnormal thrust.	1D	2 new motor casings were purchased for the season, eliminating possible damage from previous launches. The motor will only be packed by certified members with strict adherence to the manufacturer's instructions. The casing will also be inspected following launch to identify any damage.	Vehicle launch procedures require the Level 2 certified vehicle lead to pack the motor in strict accordance with the manufacturer's instructions. Vehicle launch procedures require inspection of motor casing following post launch cleaning. 7.1.1.2	1E
Motor is misaligned.	Centering rings were epoxied to the motor mount tube at an angle or too close together to eliminate angled alignment.	Forces transferred to the centering rings will not be shared as expected, possibly causing a ring to yield, allowing the motor to shoot up the center of the rocket.	1D	3 centering rings will be epoxied to the motor mount tube with the use of a laser cut jig to ensure proper alignment, eliminating the possibility for all rings to fail. Each ring is designed to carry the takeoff loads individually with a minimum factor of safety of 2.0. The 3 centering rings are also spaced evenly along the tube to further ensure proper alignment.	Vehicle launch procedures inspection of motor casing following post launch cleaning. 7.1.1.7	1E
Motor igniter fails.	Incorrect igniter installation.	Launch must be delayed and the vehicle must be removed from the launch pad.	4C	The motor will only be packed by certified members and the igniter has specific installation steps.	Vehicle launch procedures require the Level 2 certified vehicle lead to pack the motor, igniter installation steps, and detail troubleshooting for a motor that does not ignite. 7.1.1.2	4E

Table 79: Propulsion Equipment Hazard Risk Assessment

7.2.4. Recovery Equipment Hazard Risk Assessment

Hazard	Cause	Effect	RBM	Mitigation	Verifications	RWM
The vehicle does not separate between the booster and payload bays or between the payload bay and the nosecone.	There was insufficient pressure generated by the black powder charge to shear the pins or the fit between the couplers was too tight.	Recovery sequence will be unsuccessful, causing the rocket to follow a ballistic path, endangering all team members and launch spectators.	1C	The required pressure will be achieved by using rocket epoxy to seal bulk plates and by using accurate dimensions to calculate black powder amounts. Couplers will be sanded down to give an acceptable seal.	Vehicle separation tests will be conducted as described in to verify all sections separate successfully. 7.1.1.2	1D

Altimeter or e-match failure.	The altimeter or e-match may have damaged component.	If altimeter or e-match fails, the rocket will not separate causing recovery to not deploy. The rocket would follow a ballistic path and endanger all team members and launch spectators.	1C	Multiple and e-matches are included in systems for redundancy. Altimeters will be securely installed to prevent damage.	Vehicle launch procedures require avionics to be securely mounted. 7.1.1.2	1E
Recovery does not exit airframe.	Parachutes, shock cord, or shroud lines get caught on a component on the inside of the airframe.	Parachutes or lines may be torn. Recovery sequence will be unsuccessful, causing the rocket to follow a ballistic path, endangering all team members and launch spectators.	1B	internal airframe components will be well shielded to keep recovery materials from being caught on them.	Recovery launch procedures require recovery bays to be inspected for exposed carbon fiber shards and corners. 7.1.1.7	1D
Parachute gets stuck in the deployment bag.	The shock cord or shroud lines could tangle during recovery preparations or they could be wrapped around the parachute prior to packing the deployment bag.	Parachutes or lines may be torn. Recovery sequence will be unsuccessful, causing the rocket to follow a ballistic path, endangering all team members and launch spectators.	1C	Deployment bags are specially made for each parachute and all recovery team members will be taught how to properly pack them.	Recovery launch procedures require the parachute to be easily pulled from the deployment bag and explicitly state to not wrap the cords around the parachute prior to packing it into the deployment bag. .7.1.1.1	1D
Rocket descends too quickly.	Parachute is improperly sized.	The rocket will fall with a greater kinetic energy than designed for, possibly causing damage to the vehicle or onboard systems and causing the rocket to follow a ballistic path, endangering all team members and launch spectators.	2B	The main and drogue parachute designs were each carefully selected and sized to safely recover their sections of the rocket while meeting the kinetic energy limit.	The parachute selection will be tested during a full-scale flight. 6.1.2.12	2D
Rocket descends too slowly.	Parachute is improperly sized.	The rocket will drift farther than calculated, potentially exiting the launch field and the allowed range for rover deployment. The vehicle may also face unexpected environmental obstacles like busy roads or water.	2B	The main and drogue parachute designs were each carefully selected and sized to safely recover their sections of the rocket without exiting the allowed drift radius.	The parachute selection will be tested during a full-scale flight. 6.1.2.12	2D
Parachute has a tear or ripped seam.	Parachute is less effective or completely ineffective depending on the severity of the damage.	The rocket falls with a greater kinetic energy than designed for, causing components of the rocket to be damaged.	2C	The parachutes will be made of rip stop nylon to prevent tears from propagating easily if they form. In the incident that a small tear occurs during flight, the parachute will not completely fail.	Recovery launch checklists require thorough inspection of parachute canopies by 2 recovery team members after each launch 7.1.1.7	2D
Parachute or rigging burns.	Parachute is less effective or completely ineffective depending on the severity of the damage.	The rocket falls with a greater kinetic energy than designed for, causing components of the rocket to be damaged.	2B	Parachutes will all be packed in custom-made fire retardant Nomex deployment bags. All lines will be treated with fire retardant spray.	Recovery launch checklists require careful packing of parachutes in deployment bags and treatment of exposed lines with fire retardant spray. 7.1.1.1	2E
Entire recovery system separates from the rocket.	The bulkhead breaks out of the rocket, the U-bolt parachute connection breaks, or the U-bolt itself breaks.	The vehicle will fall without parachute and will follow a ballistic path and endanger all team members and launch spectators.	1B	The bulk plate factors of safety will be evaluated through FEA and only forged U-bolts will be used to connect parachutes to bulk plates. Parachute lines will be thoroughly inspected before and after flight.	The bulk plate FEA results verify the strength of the bulk plates as well as the bulk plate test procedure. Recovery launch procedures require inspection of parachute connection points and all lines prior to packing. No structural failure will be caused by opening forces as dictated by TDR 2. .7.1.1.1	1E

Lines in parachutes parachute become tangled during deployment.	Incorrect packing or asymmetry in parachute construction may cause rotation of nosecone or deployment bags that could wrap and choke parachutes.	The drogue parachute could be choked by the tangle, reducing drag force which would prevent the main parachute from opening. The main parachute may tangle due to incorrect packing. In both cases, the rocket would then fall with a greater kinetic energy than designed for, possibly causing damage to the vehicle or onboard systems and causing the rocket to follow a ballistic path, endangering all team members and launch spectators.	1B	Parachute panels will be laser cut to ensure size, accuracy, and symmetry. Main parachutes will have shock cord daisy chained so the cord stays organized until it is pulled by the opening force. Custom deployment bags will be used to further prevent tangling.	Drogue parachute symmetry will be tested in drop tests as listed in recovery testing. Recovery launch procedures describe the packing process. The deployment bags will be tested as outlined in separation testing.	1D
Premature main parachute deployment.	The altimeters misfires or the ARRD fails due to damage or incorrect assembly.	The rocket will drift farther than calculated, potentially exiting the launch field and the allowed range for rover deployment. The vehicle may also face unexpected environmental obstacles like busy roads or water.	2C	The altimeter and ARRD will be prepared by experienced members that are familiar with the setup processed for the components.	Recovery launch procedures as well as TDR 4 require ARRD load testing and inspection of ARRD and altimeters in post flight inspections to ensure proper use and maintenance of components. Full scale flight tests will verify correct timing of recovery events.07.1.1.16.2.2	2E

Table 80: Recovery Equipment Hazard Risk Assessment

7.2.5. VDS Equipment Hazard Risk Assessment

Hazard	Cause	Effect	RBM	Mitigation	Verifications	RWM
7.4 v battery death	Improper charging	If the battery dies prior to launch, the drag blades would not potentially actuate during flight. If the battery dies during ascent the rocket will not reach the intended height	2A	The battery will be charged throughout integration up until the rocket leaves for the launch rail	Before the installation of all batteries they will be checked to ensure they are fully charged. All batteries will go through proper testing to ensure they can last the anticipated time on the pad with a factor of safety of 2. 6.1.60 7.1.1.4	2D
Time variable overflow	Extended run time	VDS drag blazes could potentially actuate on rail, leading to increased rail friction, rail button shear and lower than expected exit velocity	1C	If time on rail is excessive, VDS can be restarted removing the issue of the variable overflow	Software testing will be done to ensure that the VDS runs error free and can operate for extended period of times without issues 6.1.6.1	1D
VDS VN-100 or other sensors are affected by transmitting antenna	Sensor is not properly shielded from transmitting antenna	VDS drag blazes could potentially actuate on rail, leading to increased rail friction, rail button shear and lower than expected exit velocity	1C	Extensive testing will be conducted to determine the level of risk	Testing will be done to ensure the probability of this is mitigated. 6.1.6.1	1D
11.1v battery death	Improper charging	If the battery dies prior to launch, the drag blades would not potentially actuate during flight. If the battery dies during ascent the rocket will not reach the intended height	2A	The battery will be charged throughout integration up until the rocket leaves for the launch rail. VDS was designed to have battery plugs accessible after installation into vehicle	Before the installation of all batteries they will be checked to ensure they are fully charged. All batteries will go through proper testing to ensure they can last the anticipated time on the pad with a factor of safety of 2. 6.1.60 7.1.1.4	2D

Broken gearbox	VDS blades remained actuated during recovery	Permanent damage to VDS assembly and hazard to crowd if recovery is unsuccessful	3B	VDS is programmed to retract blades after apogee. The team is currently investigating recovery force reduction	Through flight testing and opening force calculations the VDS will be testing to ensure the gearbox can handle the induced loads 6.1.1.5	3E
Sensor error due to DC motor feedback	improperly isolated circuits	VDS actuates too early, launch vehicle undershoots altitude resulting in mission failure	3B	New sensors have a built-in sensor filter to eliminate noise and signal line noise from motor encoder reduced	Built in Kalman filter will be tested to ensure that all signal noise is properly filtered 6.1.6.1	3D
Pressure phenomenon from open-ended propulsion bay causes altitude error	Vacuum formed under propulsion bay	VDS actuates too early, launch vehicle undershoots altitude resulting in mission failure	3C	Electronics bay will be airtight from the actuation bay to prevent possible interference	Both research and data analysis have found this to be a pressure anomaly with a single sensor. Proper sensor inspection before and after flight will prevent damaged sensors from providing faulty data 6.1.1.5	3D
slow speed SD card causes delay in data reading	Installed the wrong SD card	VDS fails to respond to accurate real-time data resulting in imprecise system function and higher altitude than anticipated.	2B	This will be mitigated through pre-flight check lists	All SD cards purchased will have a read/write speed greater than 300mb/s to ensure that slow SD card problems don't arise 6.2	2E
Sharp blade edges	Burs may result from blade manufacturing	minor injuries to personnel	3A	Edges will be deburred prior to VDS assembly	All blades will be properly sanded to ensure that burs are removed and the likelihood of injury is mitigated. 3.3.2.2.4	3D

Table 81: VDS Equipment Hazard Risk Assessment

7.2.6. Personnel Safety Hazard Risk Assessment

Hazard	Cause	Effect	RBM	Mitigation	Verification	RWM
Black powder explosion or premature ignition while handling.	Incorrect charge assembly or storage that allows the charge to be exposed to open flames or sparks.	Mild to severe cuts and burns or extreme trauma if a team member is holding the charge.	1C	Black powder charges will only be made outdoors with two team members present and they will be properly stored in the team's clearly labeled explosive's box.	Vehicle launch procedures require careful preparation and storage of all charges.	1E
Rotating parts and tools that move automatically.	Long hair/jewelry or loose clothing were not tied back or removed prior to working with tools/machines.	Team member could be caught or pulled into the machine, causing serious injury or death.	1B	The Engineering Garage rules require all hair and loose clothing to be tied back and jewelry to be removed prior to operation of machines.	All team members signed the Safety Manual and agreed to follow all rules of the Engineering Garage. They also acknowledged and agreed to the penalty of losing access to the Engineering Garage and team membership. 4.2	1D
Contact with flying debris from machining operations.	Lack of PPE used while machining or incorrect use of machines like closing not closing CNC doors or forgetting to remove the lathe jaw chuck.	Mild to severe cuts, or broken bones, or blunt trauma or death.	1C	Members are not certified on machines until they run machines with careful safety precautions. Two members are required to be in the machining cage at all times to ensure that safety precautions are followed.	All team members signed the Safety Manual and agreed to not use tools or equipment they were not trained on. They also agreed to the penalty of losing access to the Engineering Garage and team membership. 4.2	1D
Cuts from handheld power tools like saws, drills, or Dremel.	Improper training on power tools or lack of attention given to work.	Mild to severe cuts or burns to personnel.	2B	Individuals must be trained on the tool being used. Those not trained should not attempt to learn on their own and should find a trained individual to instruct them.	All team members signed the Safety Manual and agreed to not use tools or equipment they were not trained on. They also agreed to the penalty of losing access to the Engineering Garage and team membership. 4.2	2C

Cuts from heavy/automatic machining equipment like lathes, electric saws, or CNCs.	Lack of formal training or attention to work.	Severe cuts to personnel, damage to vehicle component or equipment.	2B	Individuals must be trained on the tool being used. Those not trained should not attempt to learn on their own and should find a trained individual to instruct them.	All team members signed the Safety Manual and agreed to not use tools or equipment they were not trained on. They also agreed to the penalty of losing access to the Engineering Garage and team membership. 4.2	2D
Electric shock.	Untrained use of welding equipment.	Minor burns to severe nervous system damage or death.	1C	Individuals must be trained on the tool being used. Those not trained should not attempt to learn on their own and should find a trained individual to instruct them.	All team members signed the Safety Manual and agreed to not use tools or equipment they were not trained on. They also agreed to the penalty of losing access to the Engineering Garage and team membership. 4.2	1D
Explosive black powder residue or spill.	Leak in black powder charge or dropped powder bottle during charge making.	Contamination of delicate parts that could be damaged if the residue or spill were to ignite, causing mild to severe burns.	1C	All team members will make black powder charges in accordance with launch procedures. Only non-sparking and non-static producing tools should be used to clean black powder spills or residue and the charges will all be made outside.	Launch testing procedures detail charge preparation and required caution with cleanup.	1E
Inhalation of carbon fibers or particulate matter	Improper use of PPE when sanding or grinding or improper cleanup of work surfaces covered with particulate matter, causing them to become airborne.	Mild to severe pain or asthma from prolonged exposure.	3B	The Engineering Garage requires work tables to be covered with cardboard for protection, making it easy to clean up shavings completely. Safety glasses and respirators will be used by the team when carbon fiber or fiber glass are cut.	All team members received a Safety Briefing prior to PDR and signing the Team Safety Agreement. This agreement was required to participate on the team and detailed the PPE required when cutting and sanding carbon fiber. 4.2	3D
Exposure to chemicals.	Lack of PPE, chemical splash or fumes when from chemical components.	Mild to severe burns on skin or eyes, lung damage or asthma aggravation.	2B	MSDS documents will be readily available at all times and will be thoroughly reviewed prior to working with any chemical. All chemical containers will be marked to identify appropriate precautions that need to be taken, including required safety glasses, nitrile gloves, and working in well-ventilated areas, when working with hazardous materials.	All team members received a Safety Briefing prior to PDR and signing the Team Safety Agreement. This agreement was required to participate on the team and detailed the emergency equipment, like eyewash stations, and permanent location of all team MSDS that list the hazards of each chemical used by the team. 4.2	2D
High decibel levels from machinery.	Use of heavy machinery or power tools for extended periods of time.	Prolonged exposure could lead to permanent ringing in the ear, or deafness.	2C	Ear plugs are readily available throughout the Engineering Garage and should be used when operating tools or machines for longer than 30 minutes.	All team members received an official Safety Briefing that covered the importance of ear protection prior to PDR and signing the signed the team Safety Agreement Form that they understood all included topics. 4.2	2D
Physical contact with hot surfaces.	Team member isn't attentive or a component like a hot glue gun or heat gun is left plugged in.	Mild to severe burns on skin.	3C	All heated elements will be used on a designated table, so members know where to expect heated elements.	A table will be identified in the next Safety Manual revision.	3D
Shavings or particles imbedded in skin or eyes.	Improper use of PPE when sanding or grinding or improper clean-up of work surfaces.	Mild to severe rash and pain.	3B	Long sleeves should be worn at all times when sanding or grinding materials and safety glasses are required when working with any power tools. Proper cleanup of all debris that results from sanding and cutting.	All team members received an official Safety Briefing that covered the importance of PPE requirement documentation included in MSDS prior to PDR. All team members signed	3D

					the Safety Agreement Form confirming that they understood all included topics. 4.2	
Dangerous fumes produced while soldering.	The use of leaded solder or resting of soldering iron on plastic.	Team members become sick due to inhalation of toxic fumes, with prolonged exposure possibly leading to asthma.	2C	The team will use well ventilated areas while soldering and automatic ceiling fans will remain on while soldering. Team members must be trained and certified to use the Engineering Garage soldering equipment that includes iron coils/stands.	The required certification teaches members about the fumes generated by soldering and how to properly clean and store the soldering iron. 4.2	2D
Potential burns while soldering.	Lack of attention or untrained use of soldering iron.	Minor burns on hands or fingers.	2C	Team members must be trained and certified to use the Engineering Garage soldering equipment that includes iron coils/stands.	The required certification teaches members about the fumes generated by soldering and how to properly clean and store the soldering iron. 4.2	2D
Failed black powder test.	Too much black powder was used, causing the structure to fail or damage to the tube was not noticed prior to the charge test.	The tube being testing could be ruptured or thrown by the charge, causing debris to possibly fly toward team members, causing damage to the vehicle or onboard systems and causing the rocket to follow a ballistic path, endangering all team members and launch spectators.	1C	All team members must wear safety glasses and stand 15 feet away from the charge during all testing procedures that require black powder. All charges will be made outside with two members present immediately prior to black powder testing. PPE will be required for all preparations incolving black poweder.	The Safety Officer or sub-team lead will be present at each black powder space to make sure that all safety precautions are followed.4.2	1D
Intense light from welding.	Improper use of PPE while welding.	Injury to eyesight may occur. May result in loss of eyesight at an early age if welding without proper PPE over long periods of time.	2A	A welding helmet, fitted with a filter shade must be worn at all times while welding.	Safety cards are placed at all equipment that indicate the PPE required for safe operation. 4.2	2E
Radiation and burns from welding.	Improper use of PPE while welding.	Mild to severe burns to skin.	2C	A welding helmet, heat resistant jacket and gloves, and close toed shoes must be worn at all times while welding.	Safety cards are placed at all equipment that indicate the PPE required for safe operation. 4.2	2E
Carbon fiber tow splinters.	Carbon fiber strands splinter when cut or stretched, becoming loose.	Splinters can imbed in the skin or eyes.	3B	Team members are required to wear cut resistant gloves, long sleeves, and safety glasses when handling carbon fiber.	All team members received an official Safety Briefing that covered the importance of ear protection prior to PDR and signing the signed the team Safety Agreement Form that they understood all included topics. 4.2	3D
Overcurrent from power source while testing.	Failure to correctly regulate power to circuits during testing or failure to identify a short.	Team members could suffer electrical shocks which could cause burns to heart arrhythmia.	2D	The circuits will be analyzed before they are powered to ensure they don't pull too much power. The circuits will be checked for shorts prior to being powered. Power supplies will also be set to the correct levels.	All circuits will be checked for continuity to identify the presence of shorts.7.1.1.4	2E
Cutting fluid contacts skin or eyes.	Use cutting fluid when machining metals.	The fluid contains known carcinogens that could lead to serious health hazards later in life.	2C	Face shields and long sleeves must be worn when using cutting fluid to prevent the fluid from splashing onto skin.	All team members received an official Safety Briefing that covered the importance of chemical risks and PPE requirement documentation included in MSDS prior to PDR. All team members signed the Safety Agreement Form confirming that they understood all included topics. 4.2	2E

Use of white lithium grease.	Use in installing motor into casing on threads and O-ring seals.	Irritation to skin, eyes, and lungs from contact or specific inhalation of fumes.	3C	Nitrile gloves and safety glasses are to be worn when applying grease. When applying grease, it should be done in a well-ventilated area to avoid inhaling fumes.	All team members received an official Safety Briefing that covered the importance of chemical risks and PPE requirement documentation included in MSDS prior to PDR. All team members signed the Safety Agreement Form confirming that they understood all included topics. 4.2	3D

Table 82: Personnel Safety Hazard Risk Assessment

7.2.7. Environmental Hazards to Rocket Risk Assessment

Hazard	Cause	Effect	RBM	Mitigation	Verifications	RWM
Unlevel launch pad.	Failing to properly level the launch pad or sinking of the launch due to excessively soft ground.	Unanticipated vehicle trajectory.	2B	Launch pad will be leveled directly prior to the launch vehicle being installed on it	Both the vehicle leads and either the co-captain or safety officer will sign off on the leveling of the launch pad section of launch procedures prior to the ignition of the launch vehicle motor 7.1.1.5	2D
Difficulty assembling vehicle components in field.	Excessive changes in humidity and or temperature causing unequal swelling or shrinking of components.	Hole misalignments and new stresses are induced due to preloading of materials resulting in increased separation friction and possibility of failed separations.	1C	All fits will be verified prior to leaving for the field, separations will be tested in similar weather conditions and sand paper will be brought to the field to make finite adjustments to ensure proper fits.	These steps will require a signature of both the vehicle lead and either a co-captain or the safety officer on the launch procedures to ensure all steps are properly done and risks have been mitigated. 6.1.2.11	2E
Rover mission halted due to large unanticipated debris.	Large debris in front of the rover's path.	The rover's path is blocked halting the mission and resulting in a failure in the payload mission.	1C	An onboard lidar sensor scans for debris in front of the rover and will scan for a clear path that the rover can take.	Lidar testing will ensure that the onboard system properly identifies obstacles and avoids them.	1D
Rover becomes stuck due to loose dirt mounds or mud.	Recent rain or tilling of the field results in pockets of mud or mounts of dirt that are difficult to gain traction on.	Difficult terrain will result in the wheels failing to gain traction and the rover getting stuck and ultimately failing to complete its mission of driving 5 ft.	1B	Substantial rover drive system testing will be done to simulate different terrains that the rover may encounter to ensure the rover can complete its mission on all terrains the rover may encounter.	Rover drive system testing will ensure that the rover can surmount terrains you intend to encounter.	1D
VDA, Recovery, or rover electronics damaged by environmental conditions..	High winds, rain and cold and hot temperatures	Electronics fail to function properly resulting in mission critical electronics failing..	1A	All electronics will be shielded from light precipitation and will be verified to operate under all anticipated operating temperatures. It will be ensured the rocket and payload do not operate under high winds and under amounts of heavy precipitation	All mission critical electronics will be encased in 3d printed sleds protecting it from light precipitation and all flight weather conditions will be verified prior to launch to prevent testing during high winds or heavy precipitation. 5.1.40.1	1E
Rocket Structure failure or launch pad fire..	Direct sunlight and high outside temperatures can result in high temperatures inside the rocket	Increased temperatures due to extended sunlight exposure and high temperatures results in overheating of batteries and leads to battery fire.	1B	Ensuring that the rocket is covered under a tent during assembly and that all batteries are stored inside insulated bags until needed.	Tents will be listed on the travel check list to ensure they are taken with and set up and all batteries will travel in insulated battery bags and remain there until needed in the assembly process. 7.1.1	1D

Parachute and rocket body damage.	Excessive winds and nearby trees and obstacles.	Recovery equipment being damaged and the rover unable to deploy due to not having ground to deploy onto.	3B	To mitigate these issues the team will not launch with winds exceeding 15mph and will ensure that each launch field adheres to proper launching distances stipulated in the NAR handbook.	A wind speed data logger will be brought to every launch and both the wind speed data logger speed and the local weather station's report of wind speeds will be recorded on the flight procedures by both captains before launch to ensure wind speeds are at an appropriate level along with a discussion with the range safety officer to ensure the rocket pad is at the appropriate distance away 7.1.1	3D
Ice buildup on launch vehicle resulting in sealing of vent holes.	Ice buildup results in pressure sensor holes being fully or partially closed off.	The change in the vent hole size results in failure to proper read altitudes resulting recovery failing to deploy at the proper altitude and the vehicle lands at higher than nominal speed.	1D	To mitigate a visual check of the vehicle will be done to ensure no ice buildup has occurred and that all vent holes are open and free of obstruction.	A visual inspection of the vehicle will be done by both the vehicle and the recovery lead to ensure ice has not accumulated and all vent holes are unobstructed. 7.1.1.2 7.1.1.1	1E

Table 83: Environmental Hazards to Rocket Risk Assessment

7.2.8. Rocket Hazards to Environment Risk Assessment

Hazard	Cause	Effect	RBM	Mitigation	Verifications	RWM
Unsuccessful deployment of recovery systems	Deployment charges failing to ignite, insufficient deployment charges or improper pressure readings	Launch vehicle plummets to the earth at higher than nominal speed resulting in the launch vehicle getting damaged and debris is scattered around launch area	1B	Both separation charge calculations and separation tests will be done to ensure all sections separate properly	All black powder charges will be calculated then tested according to. 6.1.2.8	1D
Launch Pad Fire	Launch pad fire caused by brush and dry grass catch fire following motor ignition.	Brush and dry grass around the launch pad igniting results in a wild fire in the local area surrounding the launch area, destroying local wildlife and habitats and creating extra pollution.	1B	to mitigate this no launch will happen within 100ft of any dry grass or brush and a fire extinguisher will be brought to every launch as a precautionary measure	Listed in launch procedures and in compliance with NAR regulations we will not launch within 100ft of brush or dry grass and according to launch procedures a fire extinguisher will be brought to every launch. 4.2	1E
Motor CATO	Improper packing of a motor or motor defect	Launch motor fires through launch vehicle, destroying it and scattering parts throughout launch field	1C	All motors will be packed by two certified team members including the vehicle lead	Following the packing of motors both packing members must sign the launch procedures under motor packing to verify the motor was packed properly.7.1.1.2	1D
Rocket part debris	Rocket parts loosely secured or free floating inside the rocket body	Failing to properly secure all rocket parts results in debris scattered throughout the launch field resulting in a potential hazard for local animals	2B	All rocket parts must be fully tethered to a recovered part of the vehicle while all tethering must be able to withstand opening forces	FEA and testing will be done to ensure all mounts are capable of withstanding launch and opening forces. 3, 5	2E
chemical contamination of local water sources	Leaking batteries and other hazardous materials leaks into body of water the rocket lands in.	Leaking batteries contaminating local drinking water and making it hazardous to local animals	1B	All batteries will be inspected prior to installation into launch vehicle	Launch procedures requires visual inspection of all batteries and a sign off box to confirm this step has been complete 7.1.1.4	1D

Table 84: Rocket Hazards to Environment Risk Assessment