NC STATE UNIVERSITY

Tacho Lycos
2018 NASA Student Launch
Preliminary Design Review

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Common Abbreviations & Nomenclature

AGL	=	above ground level
APCP	=	ammonium perchlorate composite propellant
ARRD	=	advanced retention and release device
AV	=	avionics
BBB	=	BeagleBone Black
BP	=	black powder
BRB	=	BigRedBee
CDR	=	Critical Design Review
CG	=	center of gravity
CP	=	center of pressure
EIT	=	electronics and information technology
FAA	=	Federal Aviation Administration
FMECA	=	failure mode, effects, and criticality analysis
FN	=	foreign national
FRR	=	Flight Readiness Review
GPS	=	Global Positioning System
HEO	=	Human Exploration and Operations
HPR	=	High Power Rocketry
HPRC	=	High-Powered Rocketry Club
L3CC	=	Level 3 Certification Committee (NAR)
LCO	=	Launch Control Officer
LRR	=	Launch Readiness Review
MAE	=	Mechanical & Aerospace Engineering Department
MSDS	=	Material Safety Data Sheet
MSFC	=	Marshall Space Flight Center
NAR	=	National Association of Rocketry
NCSU	=	North Carolina State University
NFPA	=	National Fire Protection Association
PDR	=	Preliminary Design Review
PLAR	=	Post-Launch Assessment Review
PPE	=	personal protective equipment
RFP	=	Request for Proposal
RSO	=	Range Safety Officer
SL	=	Student Launch
SLS	=	Space Launch System
SME	=	subject matter expert
SOW	=	statement of work
STEM	=	Science, Technology, Engineering, and Mathematics
TAP	=	Technical Advisory Panel (TRA)
TRA	=	Tripoli Rocketry Association

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1.1 Team Summary

1.1.1 Name and Contact Information

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1.1.2 Mentor Information

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1.2 Launch Vehicle Summary

1.2.1 Size and Mass

The full-scale launch vehicle will be 125.0 in. long with a diameter of 7.5 in. The rocket will have a mass of 46.1 lb on the pad.

1.2.2 Motor Choice

The full-scale launch vehicle will use an AeroTech L2200G motor.

1.2.3 Recovery System

The full-scale launch vehicle will use a dual deploy recovery system with a 3.0 ft drogue parachute deployed at apogee and a 10.0 ft main parachute deployed at 1,000 ft AGL.

1.2.4 Milestone Review Flysheet

These documents were submitted separately.

1.3 Deployable Rover Payload Summary

The team has chosen to include a deployable rover as the payload for the full-scale launch vehicle. The rover will be stowed in an internal tube that will rotate freely using roller bearings (lazy susan bearings) attached at each end of the payload tube. The upper bearing will be attached to a centering ring aft of the main parachute compartment, and the lower bearing will be attached to a bulkhead forward of the drogue parachute compartment and avionics bay. After landing, a door on the upper centering ring will open to allow the rover to exit the vehicle. The rover will then autonomously drive at least 5 ft and deploy a set of solar panels.

2. Changes Made Since Proposal

2.1 Changes Made to Vehicle Criteria

Table 2-1 lists all changes made to the full-scale launch vehicle since Proposal submission. Note that this report uses the terms “launch vehicle” and “rocket” interchangeably.

Table 2-1: List of Changes Made to Full-Scale Launch Vehicle

Description of Change	Reason for Change
The full-scale launch vehicle will use an AeroTech L2200G motor instead of AeroTech L1420R-P listed in the Proposal.	After correcting for payload weight, the original motor produced an apogee below 1 mi. An AeroTech L2200G motor will give the rocket an apogee of approximately 5,500 ft AGL. This decision required a larger motor casing which increased the length to 125 in.
The engine block bulkhead will replace the aft-most centering ring in the Proposal full-scale design.	The team was unaware that motor retainers attached directly to the aft-most centering ring when designing the Proposal launch vehicle. The engine block will be fixed in place of the aft-most centering ring to resist loading from motor during flight.
The payload bay will be fixed in the rocket between the main parachute compartment and avionics bay instead of being ejected as described in the Proposal.	The team was informed that no part of the payload could be ejected from rocket by NASA SL staff. New payload will be fixed in the rocket by an additional centering ring between the main parachute compartment and the avionics bay.
Door added to entrance of rover bay in payload tube.	Door design was updated to make attachment easier and to save weight The door will now be hinged and a latch at the top will keep it closed during flight.
Avionics bay length reduced to 3.25 in. to allow more room for payload components.	The original avionics bay was 11.25 in. long as listed in the Proposal which the team determined was much larger than necessary to house flight electronics. This decision did not affect the length of the rocket.
Nosecone ballast was increased to 45 oz.	After adding all the design changes, an additional 12 oz of ballast was necessary to maintain a stability margin greater than 2.0.
The main parachute was reduced to a diameter of 10 ft.	After performing additional calculations, it was determined that a 12 ft parachute produced too much drag which extended descent time and wind drift.

Description of Change	Reason for Change
The drogue parachute was reduced to a diameter of 3 ft.	After performing additional calculations, it was determined that a 4 ft parachute produced too much drag which extended descent time and wind drift.

2.2 Changes Made to Payload Criteria

Table 2-2 lists all changes made to the deployable rover payload since Proposal submission.

Table 2-2: List of Changes Made to Payload

Description of Change	Reason for Change
Proposal Dimensions: 12”x4.5”x2.35” Current Dimensions: 9.08”x4”x2.35”	The payload bay had to be shortened to allow the installation of an additional bulkhead to separate the parachute and payload bays. Rover length was also decreased as a result of this change.
Rover body was made hollow to allow for easier fabrication and more efficient use of interior volume for components.	The rover was too heavy in the Proposal design, so the overall weight of the payload needed to decrease.
A Section on each side of the rover was created that protrudes from the middle component Section. Small Sections covering the gap between the wheels and main body Section were also added.	The covers between the wheel and main body prevent dirt and other undesirable materials from interfering with rover operation.
Solar panel deployment method was changed from a latch system with flexible panel to rotating arm with folded panels.	A latch system would be required for each panel, and each latch system would require a motor. This would increase weight.
An electric latch on the payload sled and a latch screw on the rover itself will prevent the rover from moving within the payload tube during flight.	The original concept of latches on the bottom of the payload sled and a servo operating them was determined to be too complex after discovering electric latches that can be triggered remotely.

2.3 Changes Made to Project Plan

Table 2-3 lists all changes made to the project plan and timeline since Proposal submission.

Table 2-3: List of Changes Made to Project Plan

Description of Change	Reason for Change
Subscale launch date moved from October 28 to November 18, 2017.	The team was unable to secure funding in time for earlier launch.

3. Vehicle Criteria

3.1 Compliance to Handbook Requirements

Table 3-1, below, contains a list of the full-scale and subscale launch vehicle requirements listed in the 2018 NASA SL handbook as well as their respective compliance actions.

Table 3-1: Full-Scale and Subscale Launch Vehicle Handbook Item Compliance

Handbook Item	Description of Requirement	Compliance	Verification Method
2.1	The vehicle will deliver the payload to an apogee altitude of 5,280 feet above ground level (AGL).	With consideration of the predicted weight of the launch vehicle, a motor was selected based on thrust and specific impulse that will deliver the payload to an apogee altitude of 5,280 ft AGL. The motor selection is defended in Section 3.3.8.	Analysis. Vehicle weight was calculated and a motor was selected based on those calculations. Analysis is detailed in Section 3.3.8.
2.2	The vehicle will carry one commercially available, barometric altimeter for recording the official altitude used in determining the altitude award winner. Teams will receive the maximum number of altitude points (5,280) if the official scoring altimeter reads a value of exactly 5,280 feet AGL. The team will lose one point for every foot above or below the required altitude.	A StratoLoggerCF altimeter will be used to record the official altitude.	Test. The StratoLoggerCF will be tested in a vacuum chamber to ensure correct functionality.
2.3	Each altimeter will be armed by a dedicated arming switch that is accessible from the exterior of the rocket airframe when the rocket is in the launch configuration on the launch pad.	Both altimeters will be armed by their own key switch which will be accessible from the exterior of the airframe.	Demonstration. The team will demonstrate arming each altimeter at the launch pad.

Handbook Item	Description of Requirement	Compliance	Verification Method
2.4	Each altimeter will have a dedicated power supply.	Each altimeter will be powered by its own 9 V Duracell battery.	Demonstration. The team will demonstrate that the two altimeter circuits are entirely independent.
2.5	Each arming switch will be capable of being locked in the ON position for launch (i.e. cannot be disarmed due to flight forces).	The altimeters will be armed by key switches and will be oriented such that they won't be disarmed by flight forces.	Demonstration. The team will demonstrate that the switches are secure when in the on position.
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.	The launch vehicle will be constructed of Blue Tube to increase durability of the rocket. The main parachute will slow the rate of descent of the launch vehicle to avoid damage upon landing.	Analysis and test. Parachute size was determined by analysis and black powder separation will be tested to ensure deployment of parachutes. Described in Section 3.4
2.7	The launch vehicle will have a maximum of four (4) independent Sections. An independent Section is defined as a Section that is either tethered to the main vehicle or is recovered separately from the main vehicle using its own parachute.	The rocket will only have three (3) tethered Sections: nosecone, midsection (with payload), and fin can. See Section 3.2 for more details.	Inspection. The team will inspect the rocket to ensure that it is constructed out of three Sections.
2.8	The launch vehicle will be limited to a single stage.	The launch vehicle is single stage.	Demonstration. The team will demonstrate that the launch vehicle design only incorporates one motor.
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.	The team will practice assembly using detailed checklists prior to launch to ensure that assembly time does not exceed three (3) hours.	Demonstration. The team will demonstrate the capability to prepare the rocket within 3 hours by creating detailed checklists prior to launch.
2.10	The launch vehicle will be capable of remaining in launch-ready configuration at the pad for a minimum of 1 hour without losing	Recovery system electronics will only be armed once on the launch pad to avoid battery drain.	Inspection. The team will inspect recovery system electronics to ensure that they are not engaged prematurely.

Handbook Item	Description of Requirement	Compliance	Verification Method
	the functionality of any critical on-board components.		
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.	The rocket can be launched using only a standard motor igniter.	Inspection. The team will check the motor before purchase and make sure it is able to be launched with a standard 12-volt direct currant firing system.
2.12	The launch vehicle will require no external circuitry or special ground support equipment to initiate launch (other than what is provided by Range Services).	The rocket can be launched using only a standard motor igniter.	Demonstration. The team will demonstrate that the vehicle is capable of launching with no additional ground support.
2.13	The launch vehicle will use a commercially available solid motor propulsion system using ammonium perchlorate composite propellant (APCP) which is approved and certified by the National Association of Rocketry (NAR), Tripoli Rocketry Association (TRA), and/or the Canadian Association of Rocketry (CAR).	The launch vehicle motor selections are AeroTech motors, defended in Section 3.3.8.	Inspection. The team will inspect the chosen motor before purchase to ensure it meets this requirement.
2.13.1	Final motor choices must be made by the Critical Design Review (CDR).	Motor is already selected; the selection process is described in Section 3.3.8. No changes are currently planned.	Demonstration. The team will demonstrate that all design choices impacting motor selection are finalized along with motor choice before CDR is done.
2.13.2	Any motor changes after CDR must be approved by the NASA Range Safety Officer (RSO), and will only be approved if the change is for the sole	Motors for full-scale and subscale are already selected, the selection process is described in Section 3.3.8. No changes are currently planned.	Analysis. The team will determine the impact of any possible high-level design changes on motor performance, and make a request for

Handbook Item	Description of Requirement	Compliance	Verification Method
	purpose of increasing the safety margin.		motor change only when safety can be increased.
2.14	Pressure vessels on the vehicle will be approved by the RSO.	The launch vehicle will not utilize any pressure vessels.	Inspection. The team will inspect the launch vehicle to ensure that no pressure vessels are present.
2.14.1	The minimum factor of safety (Burst or Ultimate pressure versus Max Expected Operating Pressure) will be 4:1 with supporting design documentation included in all milestone reviews.	The launch vehicle will not utilize any pressure vessels.	Inspection. The team will inspect the launch vehicle to ensure that no pressure vessels are present.
2.14.2	Each pressure vessel will include a pressure relief valve that sees the full pressure of the valve that is capable of withstanding the maximum pressure and flow rate of the tank.	The launch vehicle will not utilize any pressure vessels.	Inspection. The team will inspect the launch vehicle to ensure that no pressure vessels are present.
2.14.3	Full pedigree of the tank will be described, including the application for which the tank was designed, and the history of the tank, including the number of pressure cycles put on the tank, by whom, and when.	The launch vehicle will not utilize any pressure vessels.	Inspection. The team will inspect the launch vehicle to ensure that no pressure vessels are present.
2.15	The total impulse provided by a College and/or University launch vehicle will not exceed 5,120 Newton-seconds (L-class).	The motor selection is based on the amount of thrust required to carry the predicted weight of the designed models to the goal apogee of 5,280 ft.; the motor selection is defended in Section 3.3.8 and has a total impulse within the required range.	Demonstration. The team will demonstrate that they did not purchase or install a motor exceeding L-class.
2.16	The launch vehicle will have a minimum static stability margin of	Current static stability margin at point of rail exit is calculated to be	Analysis. The team will calculate expected static stability margin at

Handbook Item	Description of Requirement	Compliance	Verification Method
	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.	2.08 cal. No changes that would appreciably affect this are currently planned.	point of rail exit after each design choice that would change stability margin and ensure it does not go below beneath 2.0. See Section 3.2.4.
2.17	The launch vehicle will accelerate to a minimum velocity of 52 fps at rail exit.	Section 3.6.5 contains the rail exit velocities and rail length calculations; the launch vehicle, both full-scale and subscale, will each have exit velocities greater than 52 ft/s.	Analysis. The team will calculate expected rail exit velocity after each design change that could alter rail exit velocity, and ensure it does not go below 52 ft/s.
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.	A sub scale launch is planned for Nov. 18th, well before CDR is due with enough time for a relaunch if the first launch is not successful.	Demonstration. The team will demonstrate a subscale launch well before CDR with time to carry out more launches if necessary.
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.	The current subscale is designed to function as closely to the full scale as possible at 52% of the size.	Analysis. The team will design the subscale to function as similarly to the full scale as possible at 52% of the size.
2.18.2	The subscale model will carry an altimeter capable of reporting the model's apogee altitude.	The subscale rocket will contain an altimeter during the test flight.	Demonstration. The team will demonstrate the altimeter reporting the model's apogee altitude.
2.19	All teams will successfully launch and recover their full-scale rocket prior to FRR in its final flight configuration. The rocket flown at FRR must be the same rocket to be flown on launch day. The purpose of the full-scale demonstration flight is to demonstrate the launch vehicle's stability, structural integrity, recovery systems, and the team's ability to prepare the launch vehicle	The same rocket will be flown during the full-scale demonstration flight as the full-scale test flight.	Demonstration. The team will fly the same rocket in the full-scale demonstration flight as in the full-scale test flight.

Handbook Item	Description of Requirement	Compliance	Verification Method
	for flight. A successful flight is defined as a launch in which all hardware is functioning properly (i.e. drogue chute at apogee, main chute at a lower altitude, functioning tracking devices, etc.).		
2.19.1	The vehicle and recovery system will have functioned as designed.	Through OpenRocket, hand calculations, and subsystem experiments, the vehicle and subsystem performance will be predicted prior to flight.	Analysis/Test. OpenRocket and mathematical analyses methods will be implemented for performance goals. Experiments will test these goals.
2.19.2	The payload does not have to be flown during the full-scale test flight.	The payload will be flown during full-scale test flight to demonstrate rover mission performance.	Demonstration. The payload will be flown during full-scale test to demonstrate rover mission performance.
2.19.2.1	If the payload is not flown, mass simulators will be used to simulate the payload mass.	The subscale payload electronics will be offset to simulate full-scale payload axis.	Demonstration. The team is using subscale payload to simulate mass displacement similarly to full scale.
2.19.2.1.1	The mass simulators will be located in the same approximate location on the rocket as the missing payload mass.	The subscale payload electronics will be offset to simulate full-scale payload axis.	Demonstration. The team is using subscale payload to simulate mass displacement similarly to full scale.
2.19.3	If the payload changes the external surfaces of the rocket (such as with camera housings or external probes) or manages the total energy of the vehicle, those systems will be active during the full-scale demonstration flight.	The payload does not change the external surface of the rocket.	Demonstration. The team will demonstrate the payload does not change any external surfaces.
2.19.4	The full-scale motor does not have to be flown during the full-scale test flight. However, it is recommended	The full-scale motor will be flown during the full-scale test flight and will demonstrate full flight readiness	Demonstration. The team will demonstrate the use of the full-scale

Handbook Item	Description of Requirement	Compliance	Verification Method
	that the full-scale motor be used to demonstrate full flight readiness and altitude verification. If the full-scale motor is not flown during the full-scale flight, it is desired that the motor simulates, as closely as possible, the predicted maximum velocity and maximum acceleration of the launch day flight.	and verification through inflight avionics readings.	motor during the full-scale test launch.
2.19.5	The vehicle must be flown in its fully ballasted configuration during the full-scale test flight. Fully ballasted refers to the same amount of ballast that will be flown during the launch day flight. Additional ballast may not be added without a re-flight of the full-scale launch vehicle.	The full-scale ballast is 2.25 lb in the nose and will be flown in full-scale test flight to demonstrate maximum stability.	Demonstration. The team will demonstrate the rocket will be fully ballasted during the full-scale test flight.
2.19.6	After successfully completing the full-scale demonstration flight, the launch vehicle or any of its components will not be modified without the concurrence of the NASA Range Safety Officer (RSO).	The experiments completed in the subscale flight are sufficient for determining design changes and the full-scale design will implement any changes needed.	Inspection. The team will inspect each components to ensure there is not the need to modify any components.
2.19.7	Full scale flights must be completed by the start of FRRs (March 6th, 2018). If the Student Launch office determines that a re-flight is necessary, then an extension to March 28th, 2018 will be granted. This extension is only valid for re-flights; not first-time flights.	The full-scale launch day is February 2018. The exact date is not yet known.	Demonstration. The team will demonstrate the full scale flight will completed by the start of FRRs.

Handbook Item	Description of Requirement	Compliance	Verification Method
2.20	Any structural protuberance on the rocket will be located aft of the burnout center of gravity.	The forward-most structural protuberance is the forward rail button, located 96.375 inches from the tip of the nosecone. The burnout center of gravity is located 72.357 inches from the tip of the nosecone.	Inspection. The team will inspect the rocket body to ensure there are no structural protuberances forward the burnout center of gravity.
2.21.1	The launch vehicle will not utilize forward canards.	The launch vehicle has no forward canards.	Demonstration. The team will demonstrate the rocket does not use forward canards.
2.21.2	The launch vehicle will not utilize forward firing motors.	The launch vehicle has no forward firing motors.	Demonstration. The team will demonstrate the rocket does not use forward firing motors.
2.21.3	The launch vehicle will not utilize motors that expel titanium sponges.	The launch vehicle motor does not expel titanium sponges.	Demonstration. The team will demonstrate the rocket does not use motors that expel titanium sponges.
2.21.4	The launch vehicle will not utilize hybrid motors.	The launch vehicle does not utilize hybrid motors.	Demonstration. The team will demonstrate the rocket does not use hybrid motors.
2.21.5	The launch vehicle will not utilize a cluster of motors.	The launch vehicle uses only a single motor.	Demonstration. The team will demonstrate the rocket does not use cluster motors.
2.21.6	The launch vehicle will not utilize friction fitting for motors.	The launch vehicle motor mount is epoxied into the rocket.	Inspection. The team will inspect each component to ensure it is properly secured.
2.21.7	The launch vehicle will not exceed Mach 1 at any point during flight.	The launch vehicle does not exceed Mach 1 during flight.	Analysis. Simulations using OpenRocket to determine the maximum flight speed will not exceed Mach 0.64. As show in Table 3-3.
2.21.8	Vehicle ballast will not exceed 10% of the total weight of the rocket.	Vehicle ballast will be 2.25 lb. The total weight of the rocket will be 42	Inspection. The team will weigh the ballast to determine it is less than 10% of the rocket's weight.

Handbook Item	Description of Requirement	Compliance	Verification Method
		lb. Vehicle ballast is 5.36% of the total weight of the rocket.	
3.1	The launch vehicle will stage the deployment of its recovery devices, where a drogue parachute is deployed at apogee and a main parachute is deployed at a lower altitude. Tumble or streamer recovery from apogee to main parachute deployment is also permissible, provided that kinetic energy during drogue-stage descent is reasonable, as deemed by the RSO.	The recovery system is defended in Section 3.2.	Test. The team will test the deployment of the rocket's recovery devices in the full-scale test launch.
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.	Ground ejection tests will be done prior to both the subscale and full-scale launches to test the size of charge and capability of completing successful separations for the drogue and main separations.	Test. The team will conduct a ground ejection test for both rockets.
3.3	At landing, each independent Sections of the launch vehicle will have a maximum kinetic energy of 75 ft-lb$_f$.	As provided in Section 1.2.4, the kinetic energy at landing of each separate Section is at a maximum 75ft-lb, defended by calculations of impact velocity and ensured by parachute selection to decrease impact velocity.	Test. The team will conduct a test during test launch to determine the kinetic energy does not exceed 75 ft-lb$_f$.
3.4	The recovery system electrical circuits will be completely independent of any payload electrical circuits.	The recovery system electronics will be housed in their own Section, and will not interact with the payload.	Demonstration. The team will demonstrate the recovery electrical circuits are independent.

Handbook Item	Description of Requirement	Compliance	Verification Method
3.5	All recovery electronics will be powered by commercially available batteries.	All recovery electronics will be powered by 9 volt Duracell batteries.	Demonstration. The team will demonstrate Duracell 9 volt batteries will be used.
3.6	The recovery system will contain redundant, commercially available altimeters. The term "altimeters" includes both simple altimeters and more sophisticated flight computers.	The recovery system is redundant in both the drogue and main separations, using Stratologger altimeters for both. Defended in Section 3.4.	Demonstration. The team will demonstrate the altimeters are redundant.
3.7	Motor ejection is not a permissible form of primary or secondary deployment.	The motor will not be ejected, and its secured position is crucial to the mission completion.	Demonstration. The team will demonstrate the motor in secured within the rocket.
3.8	Removable shear pins will be used for both the main parachute compartment and the drogue parachute compartment.	Defended in Section 1.2.8, 4-40 Nylon shear pins will be used in both the main and drogue Sections for separation.	Demonstration. The team will demonstrate there are shear pins for both parachutes.
3.9	Recovery area will be limited to a 2500 ft. radius from the launch pads.	The parachutes were chosen to ensure that the recovery area will be limited to the 2500 ft radius. Wind drift calculations are shown in Section 3.6.4.	Analysis. The team will conduct calculations to ensure the rocket will not exceed a 2500 ft radius.
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.	One BigRedBee BRB 900 GPS unit will be housed in the payload bay. All launch vehicle Sections are connected.	Demonstration. The Team will demonstrate that there is a functioning tracking device in the launch vehicle.
3.10.1	Any rocket Section, or payload component, which lands untethered to the launch vehicle, will also carry an active electronic tracking device.	All the launch vehicle Sections are tethered. The deployable rover will carry its own GPS unit.	Demonstration. The team will demonstrate that all components are tethered together.
3.10.2	The electronic tracking device will be fully functional during the official flight on launch day.	Tracking devices used for the launch vehicle and the rover will be tested	Test. The team will test the tracking devices prior to launches.

Handbook Item	Description of Requirement	Compliance	Verification Method
		both prior to launch and on launch day to ensure functionality.	
3.11	The recovery system electronics will not be adversely affected by any other on-board electronic devices during flight (from launch until landing).	The power systems and transmitting systems are completely independent from any other electronic subsystem within the launch vehicle.	Inspection. The team will inspect all electronics to ensure their safety.
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.	See Section 3.4.	Inspection. The team will inspect all electronics to ensure their safety.
3.11.2	The recovery system electronics will be shielded from all onboard transmitting devices, to avoid inadvertent excitation of the recovery system electronics.	See Section 3.4.	Inspection. The team will inspect all electronics to ensure their safety.
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.	See Section 3.4.	Inspection. The team will inspect all electronics to ensure their safety.
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.	See Section 3.4.	Inspection. The team will inspect all electronics to ensure their safety.
2.1	The vehicle will deliver the payload to an apogee altitude of 5,280 feet above ground level (AGL).	With consideration of the predicted weight of the launch vehicle, a motor was selected based on thrust and	Analysis. Vehicle weight was calculated and a motor was selected

Handbook Item	Description of Requirement	Compliance	Verification Method
		specific impulse that will deliver the payload to an apogee altitude of 5,280 ft AGL. The motor selection is defended in Section 3.2.10.	based on those calculations. Analysis is detailed in Sections Section 3.2.10.
2.2	The vehicle will carry one commercially available, barometric altimeter for recording the official altitude used in determining the altitude award winner. Teams will receive the maximum number of altitude points (5,280) if the official scoring altimeter reads a value of exactly 5,280 feet AGL. The team will lose one point for every foot above or below the required altitude.	A StratoLoggerCF altimeter will be used to record the official altitude.	Test. The StratoLoggerCF will be tested in a vacuum chamber to ensure correct functionality.
2.3	Each altimeter will be armed by a dedicated arming switch that is accessible from the exterior of the rocket airframe when the rocket is in the launch configuration on the launch pad.	Both altimeters will be armed by their own key switch which will be accessible from the exterior of the airframe.	Demonstration. The team will demonstrate arming each altimeter at the launch pad.
2.4	Each altimeter will have a dedicated power supply.	Each altimeter will be powered by its own 9 V Duracell battery.	Demonstration. The team will demonstrate that the two altimeter circuits are entirely independent.
2.5	Each arming switch will be capable of being locked in the ON position for launch (i.e. cannot be disarmed due to flight forces).	The altimeters will be armed by key switches and will be oriented such that they won't be disarmed by flight forces.	Demonstration. The team will demonstrate that the switches are secure when in the on position.

3.2 Selection, Design, and Rationale of Full-Scale Launch Vehicle

This Section includes the mission goals, technical specifications, and design justifications for the full-scale launch vehicle which is shown in Figure 3-1, below.

Figure 3-1: Isometric View of Full-Scale Launch Vehicle

3.2.1 Mission Statement

The team is proud to present the full-scale launch vehicle design for the 2018 NASA SL competition in the pages below. This rocket is an original design that includes efforts from team members with backgrounds in, but certainly not limited to, high-powered rocket design, structural analysis, electrical design, and slender-body aerodynamics. It is the goal of the team to always choose the rocket and payload options that offer the greatest challenge to expand on the experience for even the most veteran team members. The rocket will satisfy all the requirements listed in Section 3.1 based on the success criteria defined in Section 3.2.2 and will serve as learning tool for all members regardless of background or prior experience on the team.

3.2.2 Mission Success Criteria

The success of the full-scale launch vehicle is based on the challenge criteria listed in Section 3.1, as well as the team-derived requirements presented in Section 6.2. The team has defined a successful rocket launch as one where the vehicle apogee is within 100 ft of 5,280 ft AGL (1 mi), the drogue parachute deploys at apogee, the main parachute deploys at 1,000 ft AGL, and the entire rocket is reusable immediately after landing. To accomplish these goals, every component of the rocket must work as designed and redundancies should be in place for each component critical to the flight. The team will rely on simulations, physical experiments, and test flights to confirm that the vehicle will be successful with regards to the above criteria for every flight. Additionally, the success of the deployable rover is dependent on the success of the launch vehicle since the rover mission starts once the rocket mission ends.

3.2.3 Dimensions

The full-scale launch vehicle was designed using OpenRocket, a free software that is utilized by NAR and TRA rocketeers at all certification levels. The rocket will be 124.5 in. long with a constant body diameter of 7.5 in. after the nosecone. A large body tube diameter was chosen to maximize the volume available within the rocket for the payload with deployable rover. The rocket will have three body Sections: nosecone, midsection, and fin can. Figure 3-2, below, shows the OpenRocket 3D schematic with body Sections labelled respectively.

Figure 3-2: OpenRocket Model with Section Labels

The current rocket configuration in OpenRocket has a predicted weight of 46.2 lb when fully assembled, which acts as the maximum allowable weight for the full-scale design. For comparison, the detailed SolidWorks model of the current rocket design has a predicted weight of 43.0 lb when fully assembled. Though these values do include approximations for the payload, avionics, and motor masses, they do not include weight values for body paint, epoxy, black powder charges, or fasteners. Both models will be updated continuously throughout the project timeline to reflect the latest design changes and provide up-to-date values for total mass. The team will also work on applying additional modelling techniques to include as many physical components in the OpenRocket and SolidWorks models as possible. Figure 3-3, below, shows the detailed SolidWorks model which will be compared to the OpenRocket model in Figure 3-2 for component mass confirmation.

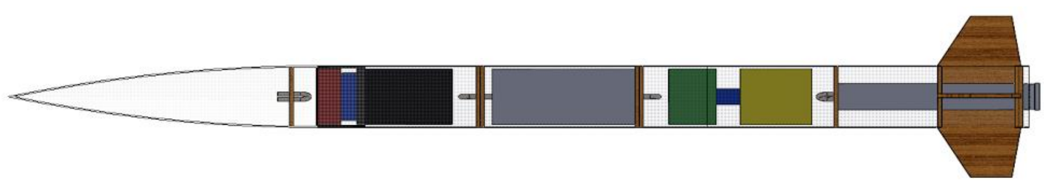


Figure 3-3: Detailed SolidWorks Model

In its current configuration, the predicted CG location of the rocket is at a point 76.9 in. from the nosecone tip as determined using the mass approximations described above in OpenRocket. The CG location was confirmed using the detailed SolidWorks model to be 75.5 in. from the nosecone tip. The CG location between models differs because the OpenRocket model features an ideal weight for the payload of 7.0 lb, while the

SolidWorks model features the actual weight of the payload in its current state as described in Section 5.2. The predicted CG location in each model will become more accurate as additional masses, such as body paint, epoxy, and fasteners, are added to the models throughout the design process.

3.2.4 Flight Stability

According to the OpenRocket model, the CP location of the rocket in its current configuration is at a point 92.2 in. from the nosecone tip. This value was computed automatically by the software at M = 0.3, which corresponds to the approximate average Mach number of the rocket from launch to apogee.

The Barrowman's Method, which defines a simple algebraic method for calculating the CP position on a subsonic rocket, was applied to confirm the OpenRocket prediction for CP position. Barrowman's Method allows the rocket to be split into three parts: nosecone, transition, and fins. Since the rocket does not include a transition Section, only the nosecone and fin equations were considered. The coefficient for nosecones C_N can be defined as a constant equal to 2. The arm length for any ogive nosecone X_N can be defined as:

$$X_N = 0.466L_N \tag{1}$$

where L_N is the length of the nosecone. Using L_N = 37.5 in., X_N was calculated to be 17.475 in. The coefficient for fins C_F can be defined as:

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

where R is the radius of the rocket body, S is the fin semi-span length, N is the number of included fins, d is the diameter of the rocket body, L_F is the fin mid-chord line length, C_R is the fin root chord length, and C_T is the fin tip chord length. Using R = 3.75 in., S = 6.0 in., N = 4, d = 7.5 in., L_F = 6.185 in., C_R = 10.0 in., and C_T = 5.0 in., C_F was calculated to be 7.719. The equation for the arm length of the fins X_F can be defined as:

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

where X_B is the distance from nosecone tip to fin root chord leading edge and X_R is the fin sweep length measured parallel to the rocket body. Using X_B = 112.5 in., X_R = 4.0 in., C_R = 10.0 in., and C_T = 5.0 in., X_F was calculated to be 116.222 in.

The equation for CP position of the entire body is a weighted average of the coefficient for each component, and can be defined as:

$$X_{CP} = \frac{C_N X_N + C_F X_F}{C_N + C_F} \tag{4}$$

The CP position was calculated to be 95.9 in. from the nosecone tip, which is 3.7 in. aft of the CP position from OpenRocket. This is a significant difference, and equates to an increase of 0.5 cal to the stability margin. However, this disparity can be explained by comparing the complexity of the OpenRocket calculation to the simplicity of the Barrowman's Method. When calculating the CP position, OpenRocket considers effects due to flight speed, nosecone shape, nosecone length, body diameter, body length, fin shape, fin location, fin leading edge shape, and surface roughness due to body paint. Compared to Barrowman's Method, which only considers nosecone shape, nosecone length, fin shape, and fin location, the OpenRocket calculation is much more advanced and can be considered more accurate. Though Barrowman's Method was used to approximately confirm the CP position, the OpenRocket prediction for CP position was used when calculating the rocket stability margin and determining the amount of ballast necessary for stable flight.

The equation for the stability margin of a rocket S_M can be defined as:

$$S_M = \frac{X_{CP} - X_{CG}}{d} \tag{5}$$

Where X_{CP} and X_{CG} are the distances from nosecone tip to the rocket CP and CG, respectively, and d is the rocket outside diameter. Stability margin is measured in calibers, where one caliber is equal to the rocket outside diameter. Using the results from OpenRocket, X_{CP} = 92.2 in. and X_{CG} = 76.9 in., the stability margin was calculated to be 2.04 cal. Since this value is the stability margin of the rocket after full assembly and before launch, it exceeds the handbook requirement (see Table 3-1) that the rocket must have a stability margin of at least 2.0 when exiting the launch rail since the stability margin will only increase during flight. Figure 3-4, below, shows the CG and CP locations on the OpenRocket model.

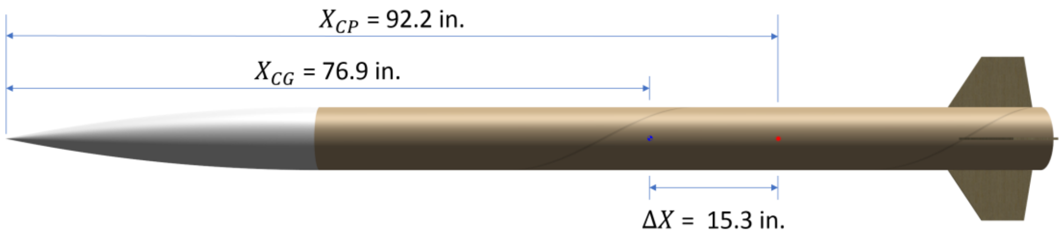


Figure 3-4: OpenRocket Model with CG and CP Locations Shown

As component weights are updated throughout the remainder of the project, the nose ballast will be modified in the OpenRocket model to ensure that the CG is less than 77.2 in. from the nosecone tip, which corresponds to the minimum allowable stability margin of 2.0. Detailed analysis of how the stability margin will change throughout the flight is shown in Section 3.6.1.

3.2.5 Material Selection

The airframe of the launch vehicle houses all internal rocket components such as parachutes, payload, and avionics. Every component of the rocket interfaces with the airframe; therefore, it is critical that the airframe structure be designed to provide support during all phases of flight. High Powered Rockets are typically constructed of filament-wound fiberglass, phenolic, or Blue Tube. Due to strength and durability limitations of phenolic paper tubes, phenolic was not considered for this design. Fiberglass tubes offer excellent strength and rigidity; however, composites can be expensive and difficult to work with. Additionally, features such as drill holes and rail buttons make fiberglass very susceptible to delamination. Blue Tube offers high strength and durability at a fraction of the cost and weight of fiberglass. For these reasons, Blue Tube was chosen as the construction material for NC State's launch vehicle.

The projected weights of the body tubes and coupler were calculated for the full-scale rocket in order to justify the use of Blue Tube on the rocket. The combined weight of the fiberglass forward airframe body tube, fin can body tube, and coupler was calculated to be approximately 7.83 lb. The combined weight of the Blue Tube forward airframe body tube, fin can body tube, and coupler was calculated to be approximately 3.83 lb. Using Blue Tube body tube components results in a weight savings of approximately 4 lb, nearly 10% of the weight of the rocket. Through reducing weight of the rocket, there is also a reduction of loading during flight which provides a higher margin of error for hitting the target altitude of 5,280 ft AGL.

To verify that Blue Tube would be capable of withstanding forces during flight, the compressive strength was analyzed. The principle compressive loads on the airframe are caused by inertia and drag. The peak drag force F_D was calculated using Equation 7 below:

$$F_D = \frac{1}{2}\rho V^2 C_D A \tag{6}$$

where ρ is the air density at sea level, $1.225\frac{kg}{m^3}$; V is the peak velocity of the rocket, $219.56\frac{m}{s}$; C_D is the drag coefficient, 0.451; and A is the frontal area of the rocket, $0.028578\,m^2$. This yields $380.20\,N$. The peak inertial load F_I was calculated using Equation 8 below:

$$F_I = ma \tag{7}$$

where m is the mass of the rocket, $19.090\,kg$, and a is the peak acceleration of the rocket, $142.95\frac{m}{s^2}$. The resulting peak inertial load on the rocket is $2729.1\,N$. The total compressive force F_C on the rocket is given by Equation 9 below:

$$F_C = F_D + F_I = 3109.3\,N = 698.99\,lbf \tag{8}$$

The supplier of Blue Tube, Always Ready Rocketry, has provided axial crush test data for Blue Tube which is shown in Figure 3-5, below.

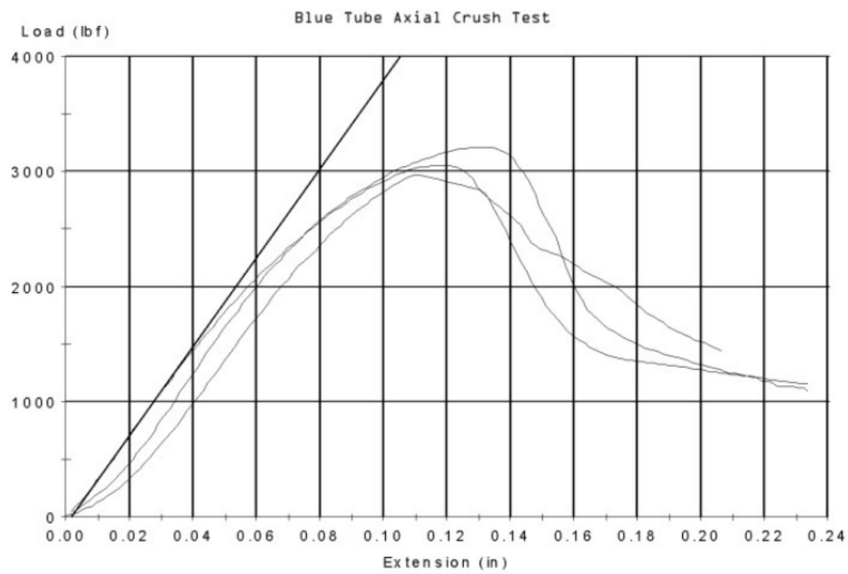


Figure 3-5: Blue Tube Axial Crush Test Data

It can be seen that over 3 trials, the buckling load for Blue Tube was approximately 3,000 lbf.[1] The factor of safety for the Blue Tube airframe on this rocket was calculated to be approximately 4.3, confirming that Blue Tube is suitably strong for this application. This is also a conservative estimate, as the rocket features several bulkheads and centering rings that provide additional support. The team will conduct additional compression testing on cylindrical Blue Tube samples in order to verify the strength of the material. As one of the tests, the samples will be loaded in a hydraulic press until buckling occurs. This experiment will also grant the opportunity to quantify the strengthening effect of bulkheads, as well as the weakening effect of drilled holes and hatches in the body tube.

The external airframe has an outer diameter of 7.5 in. and is reinforced with several Baltic birch plywood bulkheads and centering rings. All bulkheads and centering rings will be secured in the launch vehicle using West Systems 2-part epoxy. This epoxy was chosen for its high strength, durability, and working time. The bulkheads are constructed from 0.375 in. Baltic birch plywood. Varying numbers of plies are used for each bulkhead depending on the expected loads at that location in the rocket. The launch vehicle will separate at two points: between the forward airframe and nosecone and between the forward airframe and fin can.

3.2.6 Nosecone Design

The design requirements for the nosecone on the full-scale launch vehicle specified that the nosecone base had to be 7.5 in. in diameter, and that the tip could be removed to allow for the addition of nosecone ballast in the form of a metal tip and/or epoxied lead. Since the team does not have the precise manufacturing capabilities required to produce an original nosecone, only commercially-available nosecones were considered. After searching through catalogs of many different online rocketry retailers, two nosecones were chosen for consideration due to cost and availability: a 5:1 conical and 5:1 ogive.

Note that nosecone ratio corresponds to the relationship between nosecone length and nosecone base diameter, so a 5:1 nosecone is five times longer than it is wide at its base.

3.2.6.1 5:1 Conical Nosecone Justification

Apogee Rockets, an online retailer for rocketry components, advertises a 5:1 conical fiberglass nosecone on their website which would suit the needs of the team. The fiberglass nosecone is 38.4 in. long and 7.5 in. in diameter at its base, which meets the design requirements described in Section 3.2.6. The nosecone mass is also listed as 3.3 lb.

To quantify the effects of a conical nosecone, the OpenRocket model was updated to include a fiberglass conical nosecone with a length of 38.4 in. to match the Apogee Rockets product. Figure 3-6, below, shows the OpenRocket model with a conical nosecone.

Figure 3-6: OpenRocket Model with Conical Nosecone

After adding the conical nosecone, the rocket CP position was predicted to be 94.9 in. from the nosecone tip. After adding 1.75 lb of ballast to the nose to achieve a stability margin of 2.03 cal, the CG of the rocket was predicted to be 79.6 in. from the nosecone tip. Table 3-2, below, shows the results for a flight simulation using an L2200G motor and launching directly vertical from an 8 ft launch rail with no windspeed.

Table 3-2: OpenRocket Simulation Results for Conical Nosecone

Parameter	Value
CP Location (from nosecone tip)	94.9 in.
CG Location (from nosecone tip)	79.6 in.
Stability Margin	2.04 cal
Ballast Required	1.75 lb
Weight at Launch	44.9 lb
Maximum Mach Speed	0.66
Apogee	5,515 ft
Maximum Drag Coefficient	0.51

The results of the flight simulation show that the conical nosecone design will meet or exceed the limits imposed on the rocket by the requirements listed in Section 3.1.

3.2.6.2 5:1 Ogive Nosecone Justification

Mad Cow Rocketry, another online retailer for rocketry components, advertises a 5:1 ogive fiberglass nosecone on their website which would suit the needs of the team. Unfortunately, the online catalog does not list the weight or length of the nosecone, so its length was estimated to be exactly five times its length, at 37.5 in., and its weight was calculated to be 6.4 lb, assuming a wall thickness of 0.06 in. and fiberglass density of 0.07 lb/in^3. Figure 3-7, below, shows the OpenRocket model with an ogive nosecone.

Figure 3-7: OpenRocket Model with Ogive Nosecone

As described in Section 3.2.4, the rocket CP and CG positions were predicted to be 92.2 in. and 76.9 in. from the nosecone tip, respectively. The forward shift of CP as compared to Table 3-2 shows that the ogive nosecone produces greater aerodynamic forces on the rocket than the conical nosecone. A forward shift in CP is generally not desirable for rockets since it will also decrease the stability margin of the rocket, thus requiring more weight on the forward end of the rocket. Fortunately, the fiberglass ogive nosecone is 3.1 lb heavier than the fiberglass conical nosecone, so only 0.5 lb of additional nose ballast will be necessary to maintain a stability margin of 2.04 cal at launch. Table 3-3, below, shows the results for a flight simulation using an L2200G motor and launching directly vertical from an 8 ft launch rail with no windspeed.

Table 3-3: OpenRocket Simulation Results for Ogive Nosecone

Parameter	Value
CP Location (from nosecone tip)	92.2 in.
CG Location (from nosecone tip)	76.9 in.
Stability Margin	2.03 cal
Ballast Required	2.25 lb
Weight at Launch	46.2 lb
Maximum Mach Speed	0.64
Apogee	5,573 ft
Maximum Drag Coefficient	0.45

The results of the flight simulation show that the ogive nosecone outperforms the conical nosecone in apogee even though the overall rocket has a higher weight after full assembly. This can be explained by the fact that the sharp edges on the conical nosecone are designed to create shockwaves in supersonic flight, but do not offer any advantage in subsonic flight. By comparing the maximum simulated drag coefficients, the ogive nosecone has a slightly smaller value than the conical nosecone in subsonic flight. Since the rocket will not exceed M = 1.0, the long curves of the ogive nosecone will produce less aerodynamic drag, which is reflected by the increased altitude performance.

Based on the results of this investigation into nosecone designs, the team chose to use the 5:1 fiberglass ogive nosecone for the full-scale launch vehicle design.

3.2.6.3 Nosecone Material

The nosecone will be constructed out of filament-wound G12 fiberglass. The tip of the nosecone is aluminum to give additional strength and durability. It features a 6.0 in. shoulder that will be secured to the forward airframe using shear pins.

3.2.6.4 Nosecone Bulkhead Design

The nosecone will feature a single bulkhead fixed into the nosecone cavity to be used as a tethering point for the main parachute. The bulkhead will consist of two circular sheets of 0.125 in. aircraft-grade birch plywood sandwiched together using epoxy for a total thickness of 0.75 in. The aft face of the bulkhead will be fixed 3.0 in. from the base of the nosecone. Since the bulkhead will be placed inside the nosecone, OpenRocket was used to determine the outer diameter of the bulkhead to ensure that it will correspond to the inner diameter of the nosecone at its current position. Based on an assumed nosecone wall thickness of 0.08 in., the bulkhead must have a diameter 7.25 in. to fit inside the nosecone cavity.

A U-bolt will be installed through the center of the bulkhead as an attachment point for the main parachute. This will allow the nosecone Section to remain tethered to the rest of the rocket during recovery rather than falling as a separate, self-contained independent Section. The U-bolt shown in Figure 3-8, below, will be permanently fixed to the bulkhead during fabrication using epoxy and Loctite to secure the fastening nuts on the opposite side of the bulkhead.

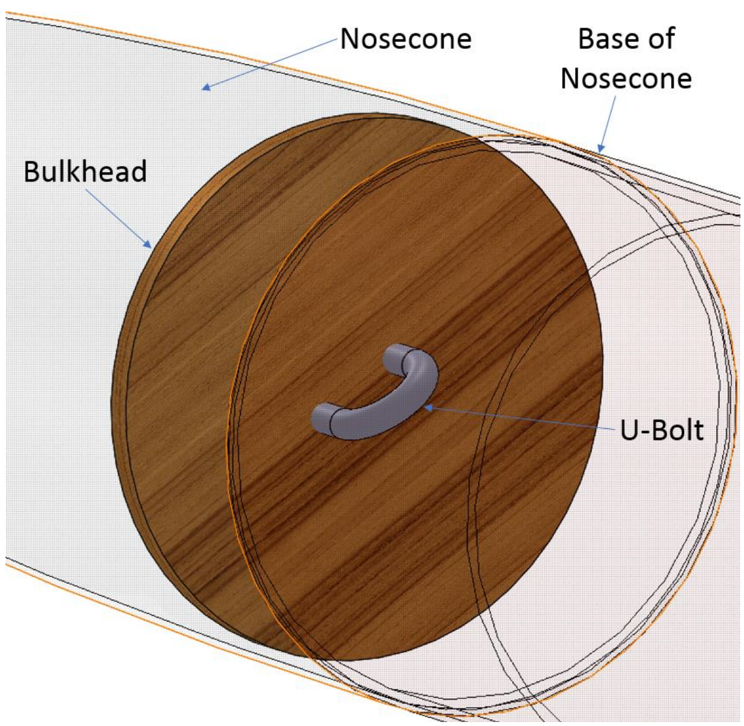


Figure 3-8: Nosecone Bulkhead

Note that once the bulkhead is installed, it will be impossible to access the interior cavity of the nosecone, therefore bulkhead installation will be one of the final steps in the rocket fabrication process.

To attach the nosecone assembly to the rocket midsection, a 6.0 in. shoulder will be included at the base of the nosecone. This shoulder is pre-formed by the nosecone manufacturer and will also use fiberglass material in its design. Shear pins will be used to secure the nosecone to the rocket body for launch.

3.2.7 Midsection Design

The rocket midsection, which is identified in Figure 3-2, will contain the payload bay, avionics bay, and part of each parachute compartment. The midsection body tube will be 45.0 in. long with a constant body diameter of 7.5 in., which will match the base diameter of the nosecone described in Section 3.2.6. Figure 3-9, below, shows the midsection with labels for each internal component and subassembly.

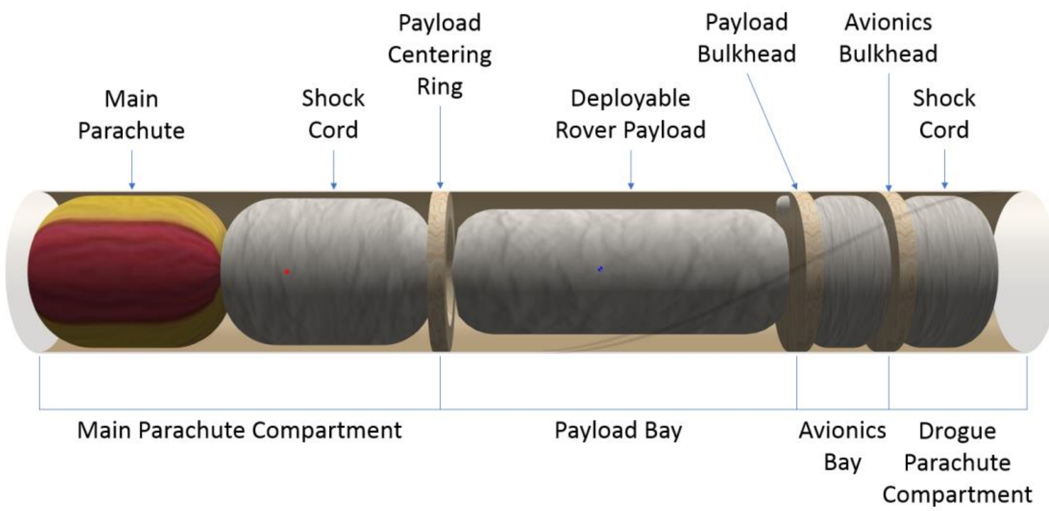


Figure 3-9: Midsection with Component and Subassembly Labels

Note that the 12.0 in. long aft coupler to be installed in the drogue parachute compartment is not shown in Figure 3-9, above. The forward end of the coupler will be installed so that it sits flush against the avionics bulkhead exactly 6.0 in. from the base of the midsection. This will give the midsection a 6.0 in. shoulder that will be used to attach the fin can. The team decided to epoxy the half of the coupler that will sit in the midsection so that the base of the coupler can be used as a straight-edge guide for the avionics nosecone during fabrication.

The main parachute will be housed in the forward parachute compartment, and the drogue parachute will be housed in the aft parachute compartment. The recovery system for the full-scale launch vehicle is described in Section 3.4. The avionics bay will contain the electronics sled with attached altimeters and batteries, which is discussed in Section 3.4.4.

3.2.7.1 Payload Bay Design

Since the payload is expected to account for a significant portion of the total rocket weight, it was determined that placing the payload bay as far forward in the midsection as possible would be most ideal. This would cause the CG position to shift forward in the rocket and would reduce the amount of nose ballast necessary to maintain a stability margin greater than 2.0. The team conducted a study to determine the maximum allowable payload weight by increasing the payload weight in steps of 0.5 lb and comparing the flight simulation results. Table 3-4, below, shows the OpenRocket flight simulation results for various payload weights using an L2200G motor and launching 5° from vertical on an 8 ft launch rail with no windspeed.

Table 3-4: OpenRocket Flight Simulation Results for Varying Payload Weight

Payload Weight (lb)	Rocket Weight (lb)	Nose Ballast (lb)	Apogee (ft AGL)
5.0	44.6	2.62	5,644
5.5	44.9	2.44	5,615
6.0	45.4	2.37	5,573
6.5	45.8	2.30	5,533
7.0	46.2	2.25	5,490
7.5	46.6	2.12	5,453
8.0	47.0	2.00	5,417

The results in Table 3-4 confirm that the amount of nose ballast necessary to maintain a stability margin of at least 2.0 will decrease as the payload weight increases. Since the OpenRocket model does not include the weight of body paint, epoxy, and fasteners, the team decided that a simulated apogee of approximately 5,500 ft AGL was reasonable for this stage in the design. The rocket model with a 7.0 lb payload weight reached 5,490 ft AGL using the simulation settings, which is why it was designated as the maximum allowable weight. As discussed in Section 5, the current payload model has a predicted weight less than 7.0 lb, but the team expects this value to increase as more components are added to the design. The OpenRocket and SolidWorks models will be updated continuously throughout the project to ensure greater accuracy in the flight simulation results.

3.2.7.2 Payload Centering Ring and Bulkhead Design

Since the payload will remain fixed within the midsection for the entire flight, it was necessary to block the rover tube off from the rest of the rocket using a forward centering ring and aft bulkhead which will both be permanently fixed to the inner surface of the body tube. The bulkheads will each consist of two circular sheets of 0.375 in. aircraft-grade birch plywood sandwiched together using epoxy for a total thickness of 0.75 in. The outer diameter of each bulkhead will match the inner diameter of the body tube, approximately 7.3 in., and will be epoxied into place during fabrication. Since the entire payload bay will be closed off after installation of the centering ring and bulkhead, an access hatch will be cut out of the body tube which is described in Section 3.2.7.5.

The forward centering ring will have an inner diameter large enough to allow the rover to exit the payload bay through a door built into the rover tube after landing, which is described in Section 5.2.4.3. Since the door and centering ring combination will close out the main parachute compartment, the blast caps used for nosecone separation and main parachute deployment will be installed on the outer ring. The outer ring is 0.75 in. wide, so only blast caps with diameters of up to 0.75 in. can be used.

The team is confident that the door and centering ring combination will be able to resist the forces applied by the rapid pressure expansion in the main parachute compartment following black powder ignition. However, the team is not confident that the centering ring alone would be able to withstand the significant forces applied by the main parachute upon inflation if the shock cord were to be attached to a U-bolt that is fixed only to the centering ring. Instead, the team has designed the centering ring to include a small notch cut from the outer edge which will allow the shock cord to pass through the gap and attach to the bulkhead located aft of the payload instead. Figure 3-10, below, shows the payload centering ring with blast caps and shock cord cutout.

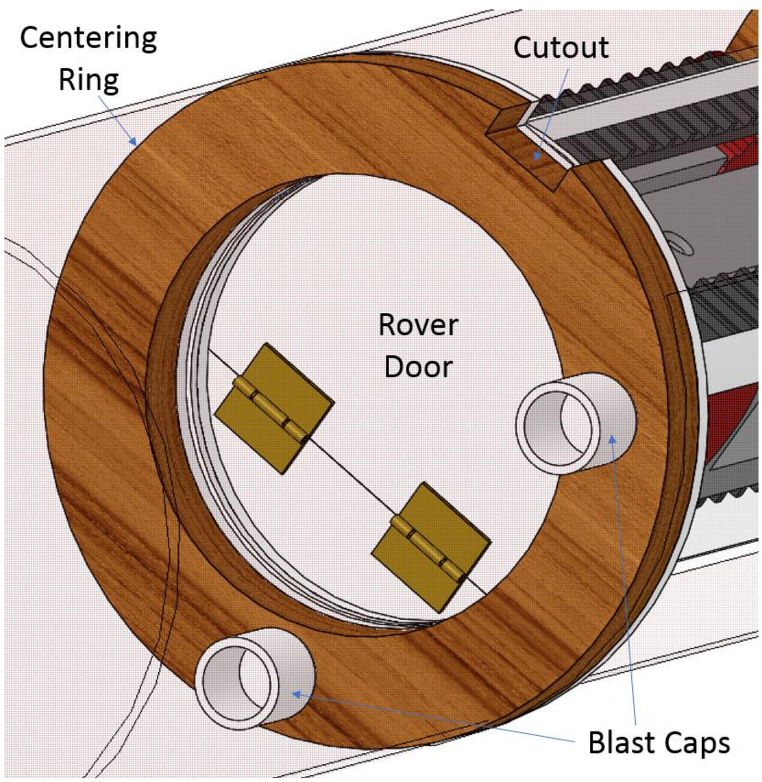


Figure 3-10: Payload Centering Ring

The notch cut from the centering ring to allow the shock cord to be fed through is 1.0 in. across and 0.375 in. deep which will give ample room for the 0.5 in.-wide shock cord described in Section 3.4.10. After the shock cord is installed during final assembly, the cutout will be sealed using a large amount of plumbing sealant to ensure that expansion gases from the black powder ignition do not enter the payload bay volume.

Once installed, the payload centering ring will act as the forward attachment point for the access hatch described in Section 3.2.7.5.

The payload bulkhead, located aft of the rover bay and forward of the avionics bay, will have a U-bolt installed through the outer width surrounding the rover tube lazy susan bearing as described in Section 5.2.2.1(b). The U-bolt shown in Figure 3-11, below, will be permanently fixed to the bulkhead during fabrication using epoxy and Loctite to secure the fastening nuts on the opposite side of the bulkhead.

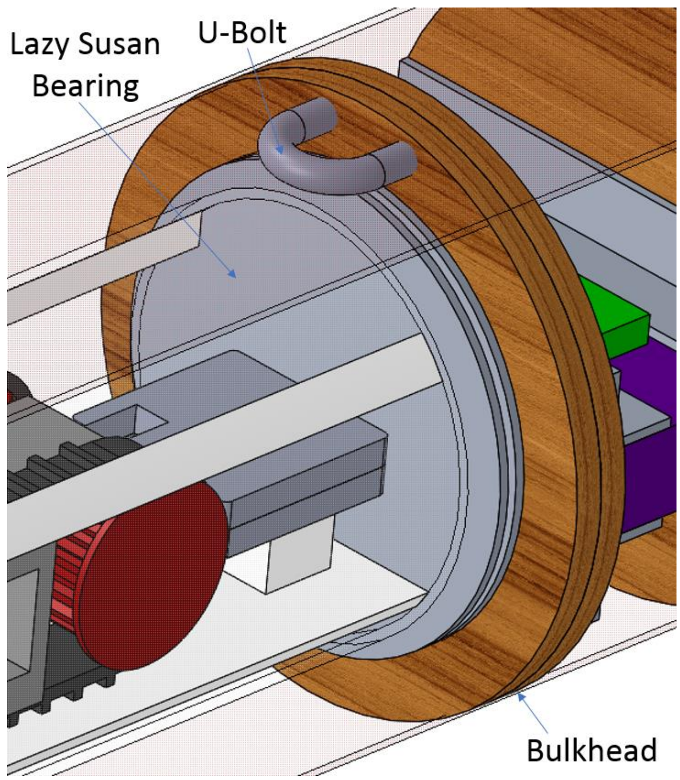


Figure 3-11: Payload Bulkhead

As described in Section 3.2.7.5, an access hatch will be cut out from the body tube over the payload and avionics bay to allow access to the U-bolt shown in Figure 3-11 during final assembly. The main parachute shock cord will be attached to this U-bolt to tether the midsection and nosecone assembly together during rocket recovery.

3.2.7.3 Avionics Bay Design

The avionics bay will contain all the electronics necessary to operate the recovery systems on board the full-scale launch vehicle which are described in Section 3.4. The team utilized the relatively large diameter of the rocket body to design an avionics package ("sled") that could fit horizontally in the avionics bay, rather than the more conventional vertical installation. The avionics bay will be 3.25 in. long which corresponds to an internal volume of 143. 6 in^3. The avionics sled design shown in Section 3.4.4 confirms that this open volume is large enough to contain all the necessary electronics for flight.

Two small holes will be drilled into the side of the rocket to allow for installation of two key switches that will enable the team to power on or off the electronics from the exterior of the rocket. Another small hole will be drilled as a static port for the altimeters. These holes are detailed in Section 3.4.5.

3.2.7.4 Avionics Bay Bulkhead Design

The avionics bay bulkhead will separate the avionics bay, located aft of the payload bay, from the drogue parachute compartment at the aft-most end of the midsection. The bulkhead will consist of two circular sheets of 0.375 in. aircraft-grade birch plywood sandwiched together using epoxy for a total thickness of 0.75 in. The aft face of the bulkhead will be fixed 6.0 in. from the base of the midsection to allow space for the aft coupler described in Section 3.2.7.

A U-bolt will be installed through the center of the bulkhead as an attachment point for the drogue parachute shock cord. This will allow the midsection to remain tethered to the fin can during rocket recovery. The U-bolt shown in Figure 3-12, below, will be permanently fixed to the bulkhead during fabrication using epoxy and Loctite to secure the fastening nuts on the opposite side of the bulkhead.

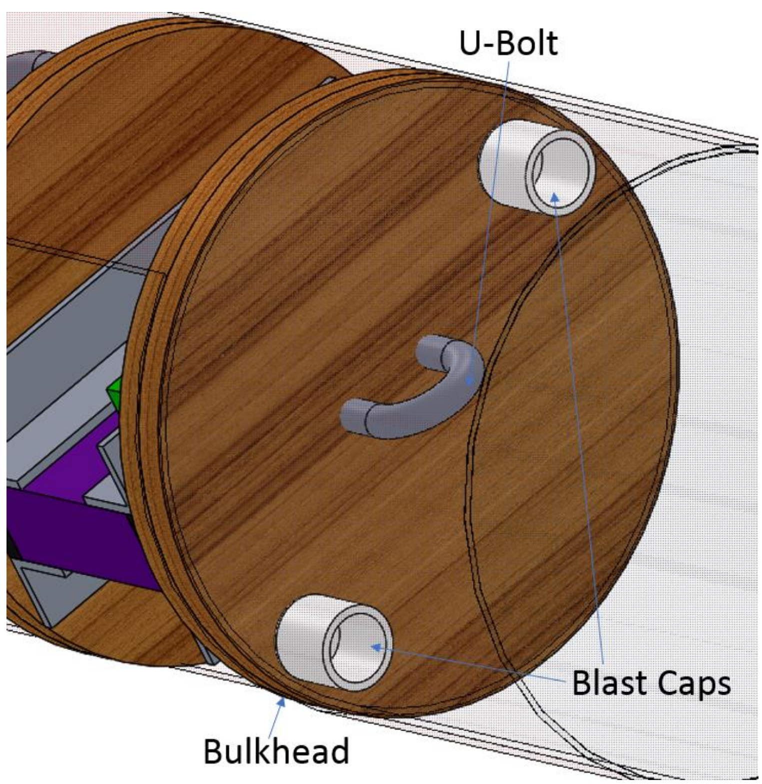


Figure 3-12: Avionics Bay Bulkhead

Once installed, the avionics bay bulkhead will act as the aft attachment point for the access hatch.

3.2.7.5 Access Hatch Design

Since the payload and avionics bays will be inaccessible once installed, it will be necessary to add an access hatch to the rocket body. Figure 3-12 shows the access hatch opening.

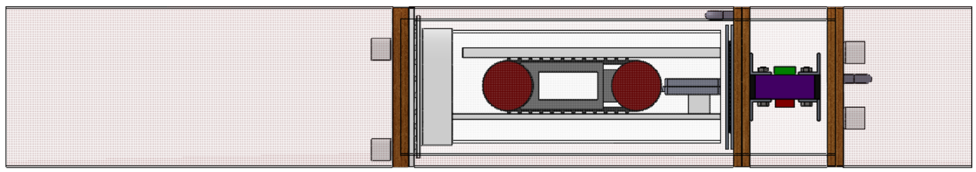


Figure 3-13: Avionics Bay Bulkhead

The hatch will have six mounting locations, with two at each bulkhead: forward payload bay, aft payload bay, and avionics bay. The hatch will also allow access to the main parachute shock cord and U-bolt. The shock cord will be routed from the U-bolt along the inner diameter of the body tube through the slot in the forward payload bulkhead. This location for the shock cord was chosen due to decreased strength in the forward payload bulkhead caused by the hole allowing the rover to deploy; therefore, very little stress will be experienced at the forward payload bulkhead.

3.2.8 Fin Can Design

The fin can Section on the full-scale launch vehicle will be 41.0 in. long and will contain the drogue parachute compartment, motor, and fins. As described in Section 3.2.7.4, the drogue parachute compartment will be split between the midsection and fin can. A 12.0 in. coupler will be attached to the midsection and split between each Section. The 6.0 in. coupler shoulder will fit into the drogue parachute compartment and will be secured using shear pins during final assembly. Figure 3-14, below, shows the fin can Section with all components labelled.

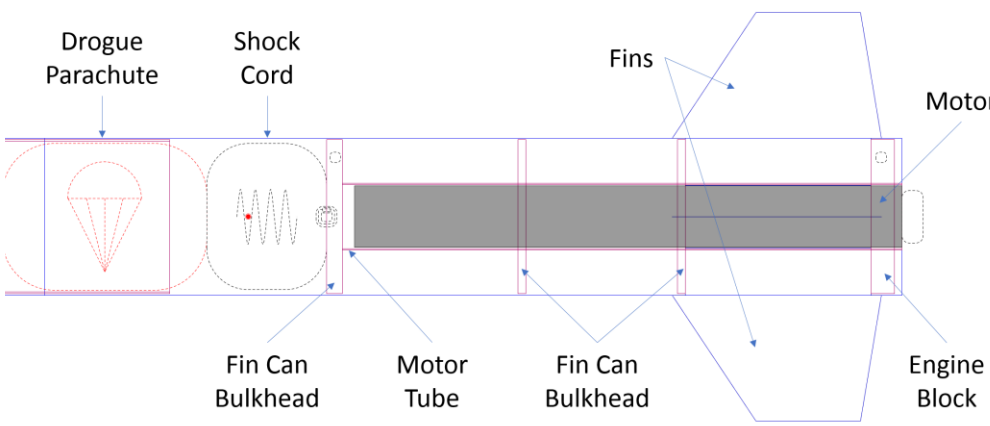


Figure 3-14: Fin Can with Component Labels

As shown in Figure 3-14, above, the fin can will have four fins attached through slots cut in the body tube and secured using epoxy fillets at the interSection with the body tube.

3.2.8.1 Fin Can Bulkhead Design

The fin can bulkhead will separate the drogue parachute compartment and the motor bay, which contains the motor tube, centering rings, and motor. The bulkhead will consist of two circular sheets of 0.375 in. aircraft-grade birch plywood sandwiched together using epoxy for a total thickness of 0.75 in. The forward face of the bulkhead will be fixed 6.0 in. from the base of the midsection to allow space for the drogue parachute and shock cord.

A U-bolt will be installed through the center of the bulkhead as an attachment point for the drogue parachute shock cord. This will allow the fin can to remain tethered to the midsection during rocket recovery. The U-bolt shown in Figure 3-15, below, will be permanently fixed to the bulkhead during fabrication using epoxy and Loctite to secure the fastening nuts on the opposite side of the bulkhead.

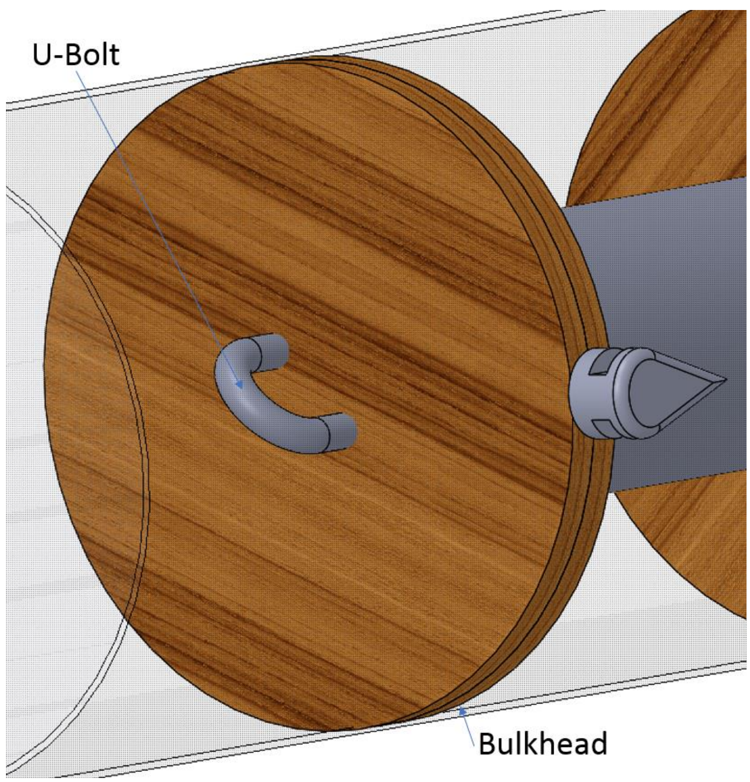


Figure 3-15: Fin Can Bulkhead

Though the motor tube will make contact with the fin can bulkhead, it will not be attached by any means.

3.2.8.2 Motor Tube Design

The motor tube is designed to spread out the loading from the rocket motor retainer to the centering rings. The motor retainer will be fixed to the aft end of the motor tube, and the inner diameter will exactly match the outer diameter of the motor casing. This will allow the motor casing to slide into the tube and be secured using the retainer which will act as the single point of contact between the motor casing and the rest of the rocket. The motor tube is one of the most critical components that will be installed on the rocket, and much caution will be used when handling, installing, and securing the motor tube during fabrication.

The motor tube will have a length of 26.75 in., which is 0.55 in. longer than the motor casing to ensure that the motor casing will be able to fit vertically in the fin can. Since the engine block is recessed into the body tube by 0.125 in., which is explained in Section 3.2.8.4, the bottom of the motor tube will extend out by that amount as well. This extra length was included to allow the motor retainer to fit over the end of the motor tube and rest against the lower face of the engine block.

Since the forces from the motor are transferred directly through the retainer to the motor tube, it is important to consider a material with high compressibility strength. Fiberglass was chosen as the motor tube material due to its high compressibility strength and workability. The fiberglass tube will be purchased as a pre-cut product from an online rocketry component retailer. Fiberglass bonds extremely well when using epoxy, and care will be taken to ensure that the fiberglass surface will be prepared correctly prior to installation.

As described in Section 3.2.8.1, the motor tube will not be attached in any way to the fin can bulkhead. Instead, the motor tube will be attached by epoxy to both centering rings, the engine block, and four fin tabs. The designs for the centering rings, engine block, and fins are detailed in Sections 3.2.8.3, 3.2.8.4, and 3.2.9, respectively. Figure 3-16, below, shows each of the epoxy areas along the length of the motor tube, with motor removed for clarity, as green shapes.

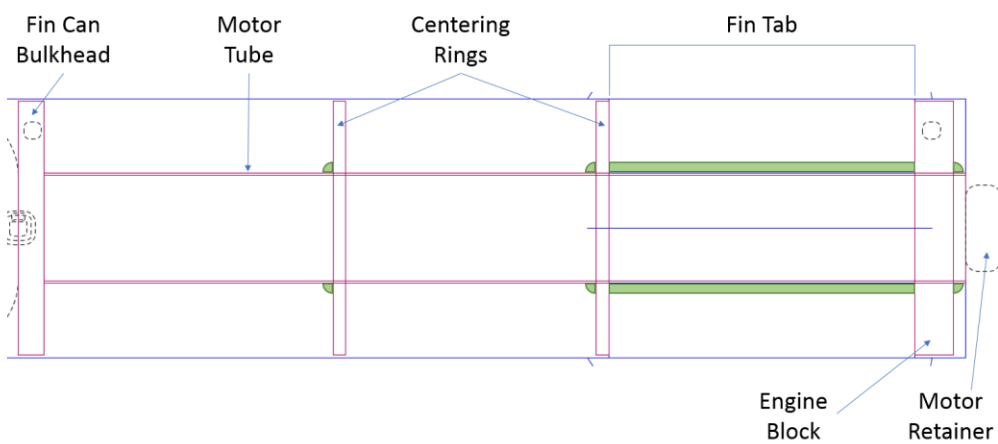


Figure 3-16: Motor Tube with Epoxy Areas Highlighted

As shown in Figure 3-16, the team will not be able to epoxy the aft side of the forward centering ring due to fabrication limitations. During fabrication, the aft centering ring will be installed within the empty body tube first and secured along its outer edge using epoxy. After the epoxy is cured, the motor tube will be slid into the centering

ring hole and secured in place using clamps. The upper corner created between the upper face of the aft centering ring and motor tube will be epoxied and left to cure. The fin tabs will be inserted into their respective slots cut into the body tube and epoxied to the motor tube and lower face of the aft centering ring. The remaining exposed surface of the corner created between the lower face of the aft centering ring and the motor tube will be epoxied. This will allow the aft centering ring to be epoxied on both sides, and the attached fin tabs will aid in transferring the load from the motor to the outer body tube via the motor tube. Once the epoxy used for the fin tabs and aft centering ring is cured, the engine block will be installed and epoxied only on the outer corner created between the motor tube and lower face of the engine block. To ensure that the upper half of the motor tube remains secure, a forward centering ring will be installed around the tube and epoxied to the inner surface of the body tube. Since the lower face of the forward centering ring will be inaccessible after its installation, only the upper face will be epoxied at the corner created by the motor tube and the upper face of the forward centering ring.

The team is confident that this configuration of motor tubes, centering rings, fin tabs, and engine block will spread the load from the motor retainer to the outer body tube in a safe and effective manner. The team will perform load simulations on the SolidWorks model with this configuration to confirm that the rocket will be able to withstand all applied forces of flight.

3.2.8.3 Motor Tube Centering Ring Design

Two identical centering rings will be used to secure the motor tube within the fin can. Each centering ring will consist of two circular sheets of 0.375 in. aircraft-grade birch plywood sandwiched together using epoxy for a total thickness of 0.75 in. The outer diameter of the centering ring will match the inner diameter of the body tube, and the inner diameter of the centering ring will match the outer diameter of the motor tube. Figure 3-17, below, shows the dimensions for each centering ring.

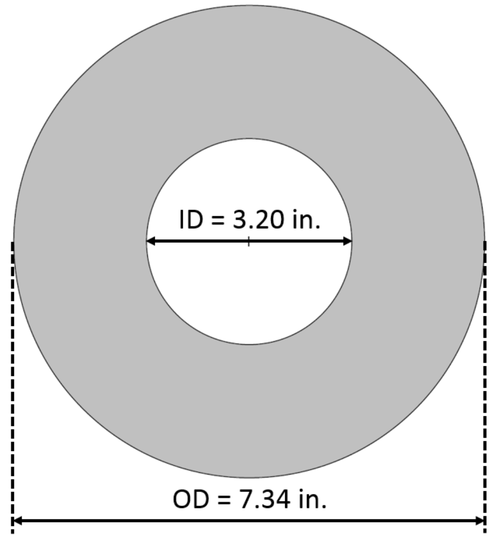


Figure 3-17: Centering Ring Dimensions

The upper faces of the forward centering ring and aft centering ring will be installed 8.375 in. and 16.00 in. from the lower face of the fin can bulkhead, respectively. As shown in Figure 3-18, below, these dimensions also correspond to lengths from the top of the motor tube.

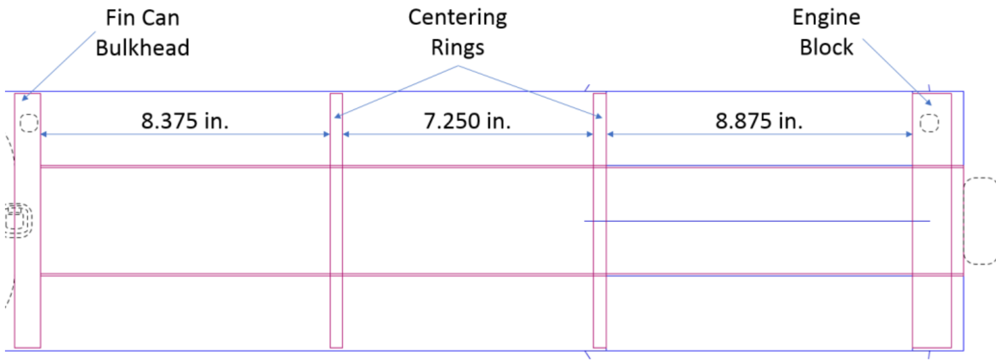


Figure 3-18: Location of Centering Rings Along Motor Tube

Section 3.2.8.2 describes the installation process for each of the centering rings.

3.2.8.4 Engine Block Design

The engine block is designed to withstand the largest forces from the motor during flight, and it will transfer loads directly from the motor retainer to the outer body tube. The engine block will consist of three circular sheets of 0.375 in. aircraft-grade birch plywood sandwiched together using epoxy for a total thickness of 1.125 in. The outer diameter of the centering ring will match the inner diameter of the body tube, and the inner diameter of the centering ring will match the outer diameter of the motor tube. Therefore, the outer diameter of the engine block will be 7.34 in. and the inner diameter will be 3.20 in. These dimensions match those of the centering rings as shown in Figure 3-17.

The engine block will be installed after the fin tabs are secured, as described in Section 3.2.8.2. The motor retainer will be mounted over the portion of the motor tube that extends past by the aft face of the engine block by 0.125 in. This dimension was provided by the motor retainer manufacturer, and ensures that the motor retainer will rest against the engine block.

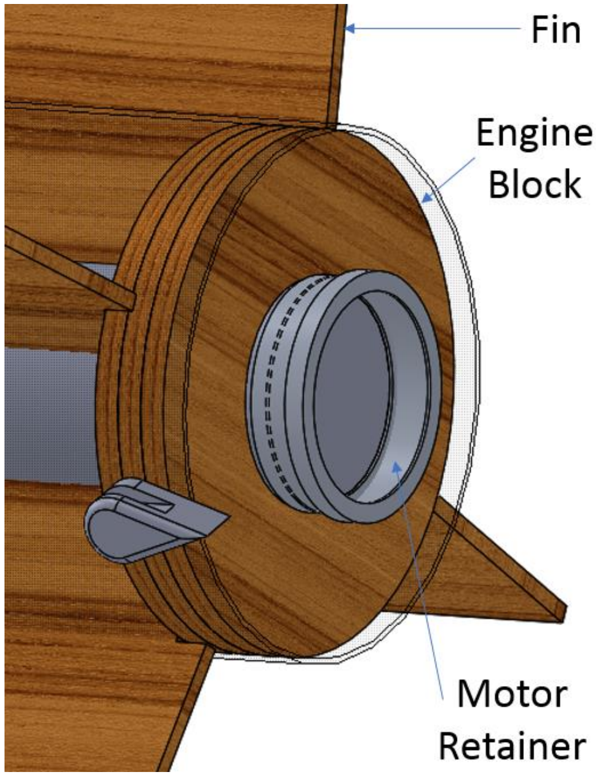


Figure 3-19: Engine Block and Motor Retainer

To further secure the motor retainer to the engine block, the two components will be epoxied and screwed together during fabrication as per the manufacturer's instructions. Once installed, the motor will be able to slide into the motor tube and be secured by the motor retainer ring.

3.2.8.5 Rail Buttons

Two rail buttons will be installed on the exterior surface of the body tube to secure the full-scale launch vehicle on the vertical rail for launch. It is desirable to place rail buttons close to the tail of the rocket so that the rail can guide the vehicle in the vertical direction for as long as possible. It is also important that the rail buttons be installed at strong points in the rocket body as the entire weight of the rocket will be supported by the buttons while on the rail prior to launch.

The forward rail button will be installed at the center of fin can bulkhead, and the aft rail button will be installed at the center of the engine block. To install each rail button, a pilot hole will be drilled through the body tube and bulkhead after all epoxy has cured. The rail buttons will then be screwed in through the pilot holes using the manufacturer-recommended screws.

The team has chosen to use airfoil-shaped rail buttons to decrease the drag penalty from attaching asymmetrical components to the exterior of the rocket. The team intends to perform a future investigation on how the drag from the rail buttons affects the overall flight performance.

3.2.9 Fin Design

The rocket fins are critical to controlling the rocket stability during ascent, but require precise fabrication to limit any errors in manufacturing and installation. The surface area, cross-Sectional shape, and location of the fins control the location of the CP along the length of the rocket, which is described in Section 3.2.4. The greatest effect on CP can be observed when the fins are installed at the aft-most end of the rocket, which also increases the stability margin by pushing the CP closer to the tail if the CG remains unchanged. There are numerous sources of conflicting information regarding fin shapes and sizing available in books and online, so the team relied on advice from mentors and results from OpenRocket flight simulations to design the fins.

Alan Whitmore, one of the team mentors as shown in Section 1.1.2, offered the following advice to the team when asked about fin sizing:

1. Straight lines are easier to design, cut, and install than curved lines.
2. Any part of a fin that extends beyond the bottom of the body tube will be more likely to break off at impact, thus eliminating rocket reusability.
3. The most durable fins have forward swept trailing edges with the rear of the root chord starting some small distance from the bottom of the body tube.
4. Fins with a span that exceeds the root chord length will be more subject to the negative effects of fin flutter during flight.
5. Fins with a span less than half of the root chord length generally do not perform very well during flight.
6. The most common fin design for high-powered rockets is the clipped delta.

Applying the above advice, the team started with a clipped delta fin design where the tip chord length was exactly half of the root chord length. Since the fin span should be within the range of 50-100% the root chord length, a span of 60% the root chord length was chosen. To increase the durability, and thus reusability, of the rocket, the trailing edge was given a slight forward sweep following a slope of 1/6, or 1.0. in. of forward sweep for every 6.0 in. along the span. With the tip chord length unchanged and only shifted forward, the leading edge had a sweep of 2/3, or 2.0 in. of sweep for every 3.0 in. along the span.

After gaining design approval from the team mentors, the team then investigated the requirements for number of fins. Given the choice between three fins or four, the team chose to use four fins for the design citing concerns for installation precision. With four fins installed, any minor installation errors will be overcome by the other three fins during flight, which ensures a greater level of redundancy in the flight. Considering that the fins will be installed by hand, it was important to also consider how they would be installed. The team decided to use an external jig to guide the fins into place and provide a clamping surface to hold them still while the epoxy cures. This method has worked very well in the past and veteran team members have experience designing and installing the jigs. The team is confident that using four fins instead of three will make for a simpler installation process and guarantee a greater chance of mission success through redundancy.

With the number of fins and scalable dimensions known, the team tested different fin designs to investigate their effects on the CP location and stability margin for the rocket. Table 3-5, shows the OpenRocket flight simulation results for four fins with varying root chord spans using an L2200G motor and launching 5° from vertical on an 8 ft launch rail with no windspeed.

Table 3-5: OpenRocket Flight Simulation Results for Various Fin Sizes

Root Chord Length (in.)	Tip Chord Length (in.)	Span Length (in.)	Nose Ballast Required (lb)	Stability Margin (cal)	Apogee (ft AGL)
8.0	4.0	4.8	7.25	2.06	5,126
9.0	4.5	5.4	4.25	2.04	5,386
10.0	5.0	6.0	2.25	2.04	5,528
11.0	5.5	6.6	0.75	2.03	5,606
12.0	6.0	7.2	0.0	2.10	5,605

As described in Section 3.2.7.1, the team used an apogee of 5,500 ft AGL as the reasonable goal altitude at this stage in the design process due to the fact that the OpenRocket model does not contain weight values for body paint, epoxy, or fasteners. The results in Table 3-5, show that a root chord length of 10.0 in. will give an approximate apogee altitude of 5,528 ft AGL which is very close to the goal altitude. Based on these results, the team chose to use a fin design where each dimension relies on a 10.0 in. root chord using the scalability factors detailed above. Figure 3-20, below, shows the dimensions for each fin on the full-scale launch vehicle.

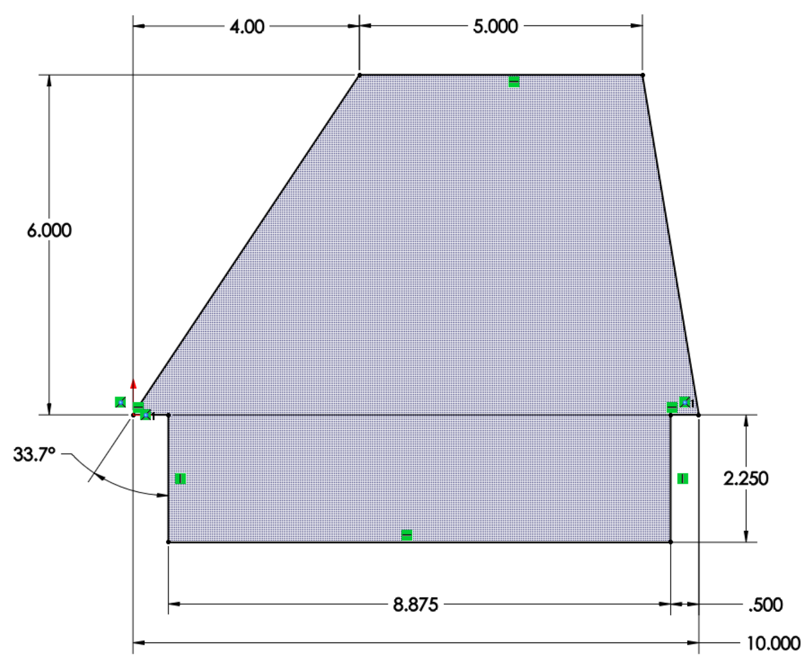


Figure 3-20: Fin Dimensions

Each fin will be installed as described in Section 3.2.8.2, with fin tabs being used to better secure each fin to the rocket body tube as well as to aid in transferring the loads from the motor to the body tube. The length of each fin tab will be equal to the distance between the lower face of the aft centering ring and the upper face of the engine block. The depth of the tabs will be equal to the distance from the outer diameter of the body tube to the outer diameter of the motor tube. These limits correspond to a fin tab length of 8.875 in. and a depth of 2.150 in. Figure 3-21, below, shows the fin tab dimensions.

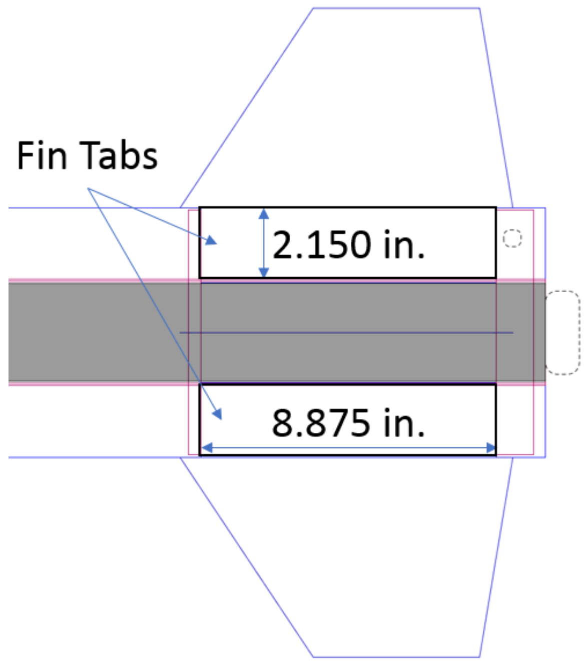


Figure 3-21: Fin Tab Dimensions

Each fin tab will allow forces to be spread across the engine block, motor tube, centering ring, and body tube. To ensure that the fins and fin tabs will be able to withstand the forces of flight, they will be laser cut out of aircraft-grade birch plywood 0.25 in. thick. Though the team is confident from past experience that fins of this thickness will be durable enough to survive multiple flights, stress analysis simulations will be done on the fins, fin tabs, and body tube to ensure that they will be able to withstand the forces of flight.

3.2.10 Motor Selection

The motor selection for the full-scale rocket depends entirely on the weight and dimensions of the rocket and the amount of thrust required for the designed rocket to reach the goal competition altitude of 5,280 ft AGL.

Per Section 2.15 of the 2018 NASA SL Handbook: “The total impulse provided by a College and/or University launch vehicle will not exceed 5120 Newton-seconds (L-class).”

The designed full-scale rocket on-rail weight is predicted to be 47 lb. Modeled in OpenRocket simulating software, the recommended motor to achieve the goal apogee of

5,280 ft is an L-class motor; the selected motor for the full-scale design is an AeroTech L2200G in an AeroTech 75/5120 motor casing. The total impulse of this motor is 5104 N-s and is within the 5120 N-s requirement. The average thrust of the motor is 2243 N, the maximum thrust is 3102 N. The motor thrust curves are included in Section 3.6.2.

According to AeroTech, each motor has a designation which provides important information on that motor. An explanation of the identification of motor sizing is shown below in Figure 3-22.

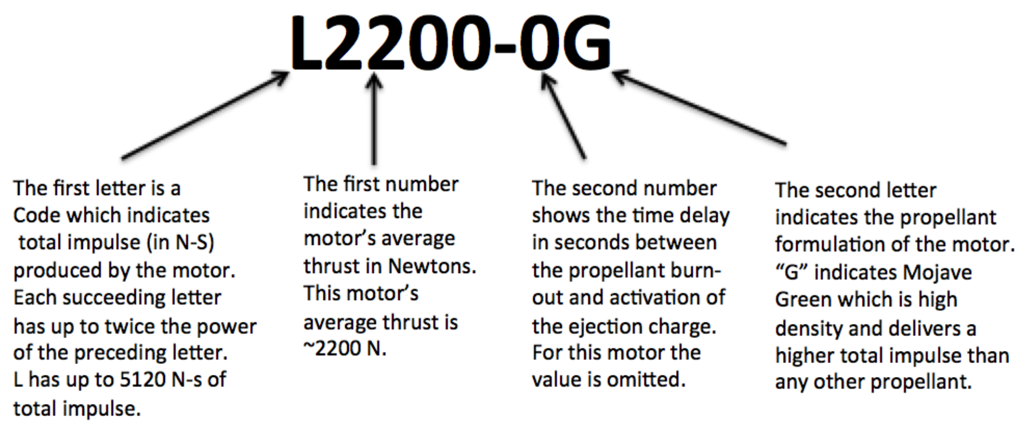


Figure 3-22: AeroTech Rocket Motor Identification

The other option for motor brand is the Cesaroni L1685-SS, which provides a total impulse of 5104 N-s. The motor casing is 3.6 inches longer and the loaded motor and casing combination is 13.3 lb (3 lb heavier) and the empty mass is 4.9 lb. The average thrust throughout flight is 1669 N and the max thrust is 2300 N. This is 800 N less than the max thrust of the AeroTech motor. The positive aspect of the Cesaroni is the consistency in their thrust curves compared to the AeroTech L2200G, but the added weight constrains the design of the rocket. The team has had motor malfunctions in the past using Cesaroni motors and casings, and actively avoids the use of their products.

Initially during design, the motor selected was the AeroTech L1420R-P. This motor did not allow the launch vehicle to achieve the goal apogee of 5,280 ft AGL. During the redesign of the payload following the submission of the design proposal, the AeroTech L2200G was identified using OpenRocket as the motor that would deliver the launch vehicle to the goal competition apogee. The AeroTech L2200G was chosen because of experience with the team's use of the motor. The motor performed reliably and consistently during the 2016-2017 launch year. The team owns an AeroTech 75/5120 motor casing, which saves cost and reduces the budget.

3.3 Selection, Design, and Rationale of Subscale Launch Vehicle

This Section includes the mission goals, technical specifications, and design justifications for the subscale launch vehicle which is shown in #Figure 3-1, below.

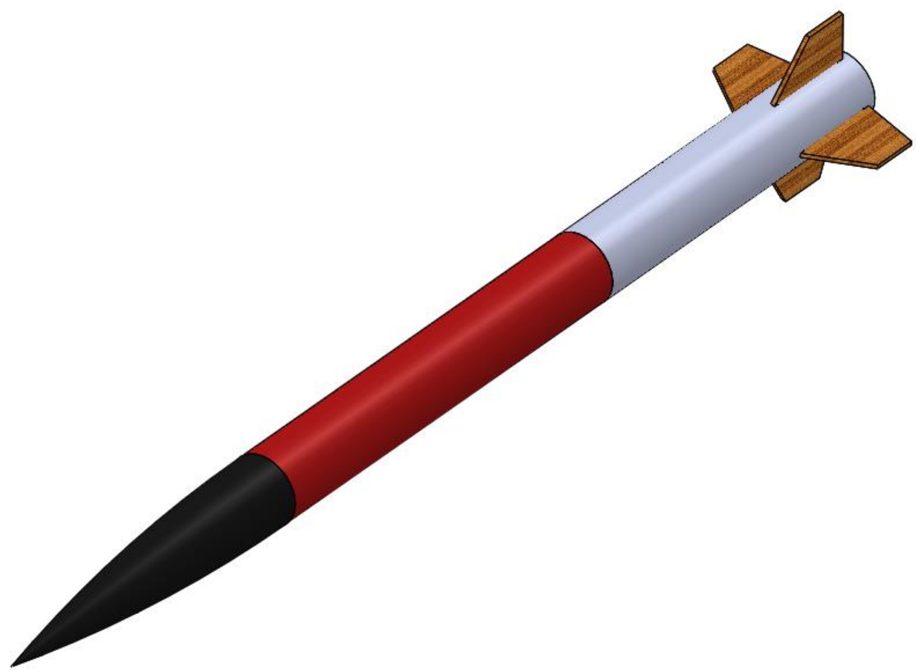


Figure 3-23: Isometric View of Subscale Launch Vehicle

3.3.1 Mission Success Criteria

The success of the subscale launch vehicle is based on challenge criteria presented in Table 3-1, as well as the team-derived requirements presented in Section 6.2. The team has defined a successful launch as one where the vehicle reaches an apogee of at least 1,500 ft AGL, the drogue parachute deploys at apogee, the main parachute deploys at 800 ft AGL, and the entire rocket is reusable immediately after landing. To accomplish these goals, every component of the rocket must work as designed and redundancies should be in place for each component critical to the flight. The team will rely on simulations, physical experiments, and test flights to confirm that the vehicle will be successful with regards to the above criteria for every flight. Additionally, the subscale rocket represents the first chance that the team will be able to design, build, and fly a rocket together, so entire process will be a learning experience for both rookie and veteran members. It is important that the team analyzes any issues that arise during subscale operations to ensure that the same issue will not arise during full-scale operations.

3.3.2 Dimensions

The subscale rocket will be 65.0 in. long with a constant body diameter of 3.9 in. after the nosecone. The diameter was chosen based on phenolic body tube diameters that are

Figure 3-24to full-scale body diameter is 3.9:7.5, or 52%. Since the subscale will be a down-scaled version of the full-scale design, every dimension on the full-scale design should be multiplied by a factor of 0.52 to arrive at the same dimension for the subscale design. Therefore, the subscale will be a 52% scale model of the full-scale design. Identical to the full-scale, the subscale rocket will have three body Sections: nosecone, midsection, and fin can. Figure 3-24, below, shows the OpenRocket 3D schematic with body Sections labelled respectively.

Figure 3-24: OpenRocket Model with Section Labels

The current rocket configuration in OpenRocket has a predicted weight of 9.69 lb when fully assembled. For comparison, the detailed SolidWorks model of the current rocket design has a predicted weight of 8.90 lb when fully assembled. Though these values do include approximations for the payload, avionics, and motor weights, they do not include weight values for body paint, epoxy, black powder charges, or fasteners. Figure 3-25, below, shows the detailed SolidWorks model which will be compared to the OpenRocket model in Figure 3-24 for component mass confirmation.

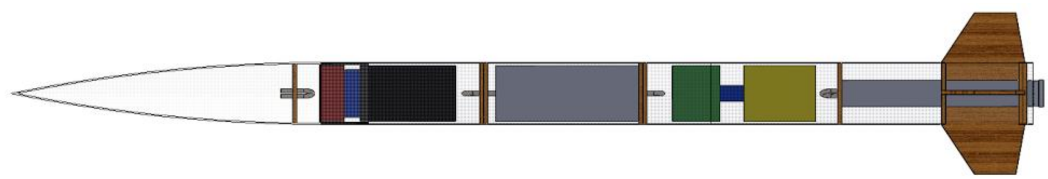


Figure 3-25: Detailed SolidWorks Model

In its current configuration, the predicted CG location of the rocket is at a point 40.4 in. from the nosecone tip as determined using the mass approximations described above in OpenRocket. The CG location was confirmed using the detailed SolidWorks model to also be 40.4 in. from the nosecone tip. The predicted CG location in each model will become more accurate as additional masses, such as body paint, epoxy, and fasteners, are added to the models throughout the design process.

3.3.3 Flight Stability

According to the OpenRocket model, the CP location of the rocket in its current configuration is at a point 48.2 in. from the nosecone tip. Figure 3-26, below, shows the CP and CG locations on the subscale launch vehicle.

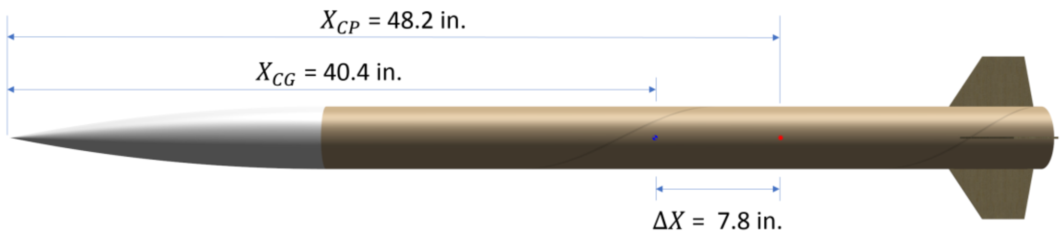


Figure 3-26: OpenRocket Model with CG and CP Locations Shown

Using Equation 6, the stability margin was calculated to be 2.0 cal after final assembly. This indicates that the CG cannot shift beyond 40.4 in. to ensure that the stability margin meets or exceeds the limit of 2.0, as listed in Table 3-1. The team will update the models with more accurate component weights throughout the fabrication process to ensure that the CG value is accurate. After final assembly, the team will use physical methods to identify the CG location on the rocket to determine if any ballast is necessary prior to launch. Section 3.7.1 contains detailed analysis of how the stability margin will change throughout the flight.

3.3.4 Material Selection

The sub-scale body tube components will be constructed from phenolic in order to reduce the cost and weight of the rocket. Due to lesser loads on the subscale rocket, a weaker and cheaper material can be used.

The subscale bulkheads and centering rings will be constructed out of 0.25 in. Baltic birch plywood, versus the 0.375 in. Baltic birch plywood used on the full-scale rocket. Similarly to the full-scale rocket, multiple plies of plywood will be secured together to form 0.5 in. bulkheads using West Systems 2-part epoxy. The use of thinner bulkheads and centering rings helps to further reduce the weight of the sub-scale rocket, where less strength is required.

The subscale nosecone will be constructed out of plastic, versus the filament wound fiberglass nosecone found on the full-scale rocket. This choice was made to reduce cost and weight of the sub-scale rocket. Although plastic is not as strong as fiberglass, the sub-scale rocket will only experience a fraction of the forces the full-scale rocket experiences and does not need to withstand several flights.

3.3.5 Nosecone Design

To match the full-scale 5:1 ogive nosecone shape, a 5:1 ogive nosecone with base diameter of 3.9 in. was selected for use on the subscale launch vehicle. Section #3.2.6.2 contains analysis on why a 5:1 ogive nosecone was chosen for the full-scale launch vehicle. A 5:1 nosecone with base diameter of 3.9 in. will have an ideal length of 19.5 in. from the nosecone base to tip. A plastic nosecone was selected to ensure a higher level of durability and reusability for the subscale launch vehicle. Unlike the full-scale, the subscale nosecone will not have a metal tip, and any necessary nose ballast will be added as lead epoxied into the nosecone inner cavity.

3.3.5.1 Nosecone Bulkhead Design

The nosecone will feature a single bulkhead fixed into the nosecone cavity to be used as a tethering point for the main parachute. The bulkhead will consist of a single circular sheet of 0.25 in. aircraft-grade birch plywood epoxied to the inner diameter of the nosecone. The aft face of the bulkhead will be fixed 1.5 in. from the base of the nosecone. Since the bulkhead will be placed inside the nosecone, OpenRocket was used to determine the outer diameter of the bulkhead to ensure that it will correspond to the inner diameter of the nosecone at its current position. Based on an assumed nosecone wall thickness of 0.08 in., the bulkhead must have a diameter 3.71 in. to fit inside the nosecone cavity.

A U-bolt will be installed through the center of the bulkhead as an attachment point for the main parachute shock cord. This will allow the nosecone Section to remain tethered to the rest of the rocket during recovery rather than falling as a separate, self-contained independent Section. The U-bolt shown in Figure 3-27, below, will be permanently fixed to the bulkhead during fabrication using epoxy and Loctite to secure the fastening nuts on the opposite side of the bulkhead.

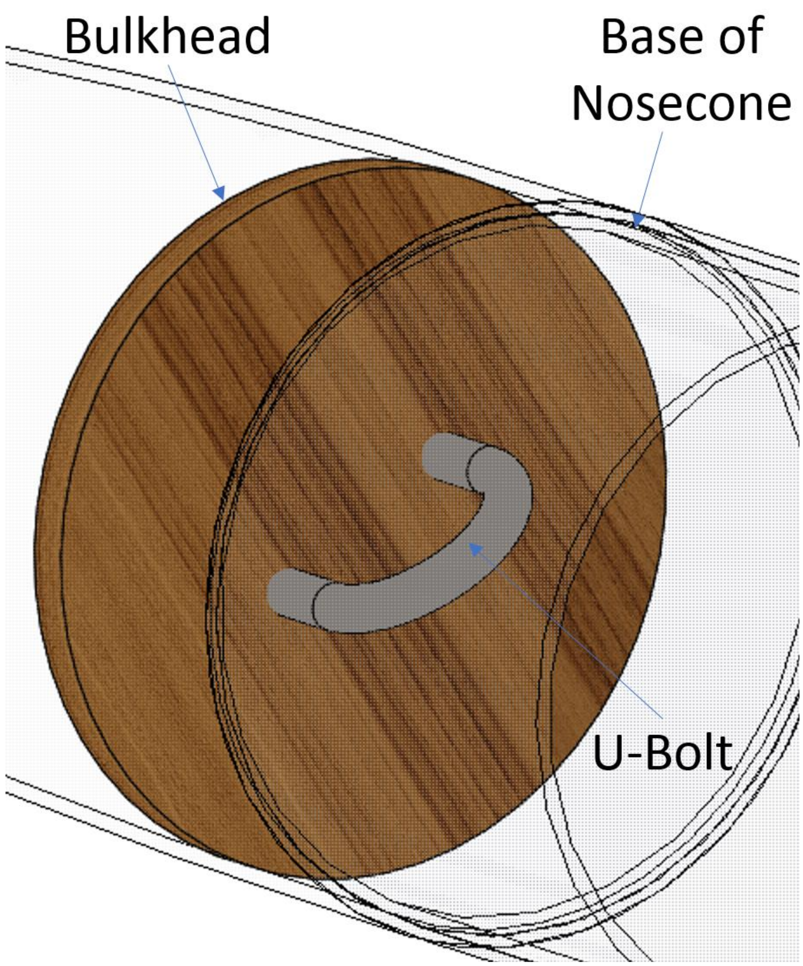


Figure 3-27: Nosecone Bulkhead

Note that once the bulkhead is installed, it will be impossible to access the interior cavity of the nosecone, bulkhead installation will be one of the final steps in the rocket fabrication process.

To attach the nosecone assembly to the rocket midsection, a 3.0 in. shoulder will be included at the base of the nosecone. This shoulder is pre-formed by the nosecone manufacturer as part of the same plastic mold. Shear pins will be used to secure the nosecone to the rocket body for launch.

3.3.6 Midsection Design

The rocket midsection will contain the payload and avionics bay, as well as part of each parachute compartment. The midsection body tube will be 24.5 in. long with a constant body diameter of 3.9 in., which will match the base diameter of the nosecone described in Section 3.3.5. Figure 3-28, below, shows the midsection with labels for each internal component and subassembly.

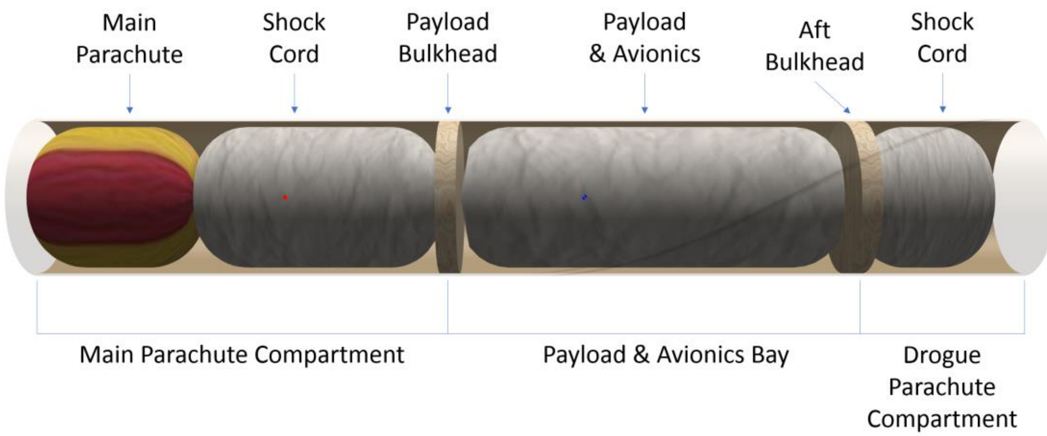


Figure 3-28: Midsection with Component and Subassembly Labels

Note that the 12.0 in. long aft coupler to be installed in the drogue parachute compartment is not shown in Figure 3-28, above. The forward end of the coupler will be installed so that it sits flush against the avionics bulkhead exactly 4.0 in. from the base of the midsection. This will give the midsection an 8.0 in. shoulder that will be used to attach the fin can. The team decided to epoxy the half of the coupler that will sit in the midsection so that the base of the coupler can be used as a straight-edge guide for the avionics nosecone during fabrication.

The main parachute will be housed in the forward parachute compartment, and the drogue parachute will be housed in the aft parachute compartment. The recovery system for the subscale launch vehicle is described in Section 3.5. The payload bay, which is assigned a length of 9.5 in., will contain the experimental subscale payload as well as the electronics sled with attached altimeters and batteries, which is discussed in Section 3.5.2.

3.3.6.1 Payload Bulkhead Design

The payload bay bulkhead will separate the main parachute compartment from the payload bay. The bulkhead will consist of two circular sheets of 0.25 in. aircraft-grade birch plywood sandwiched together using epoxy for a total thickness of 0.5 in. The forward face of the bulkhead will be fixed 10.0 in. from the top of the midsection to allow space for the main parachute. A U-bolt will be installed through the center of the bulkhead as an attachment point for the main parachute shock cord.

3.3.6.2 Aft Bulkhead Design

The midsection aft bulkhead will separate the drogue parachute compartment from the payload bay. The bulkhead will consist of two circular sheets of 0.25 in. aircraft-grade birch plywood sandwiched together using epoxy for a total thickness of 0.5 in. The aft face of the bulkhead will be fixed 4.0 in. from the bottom of the midsection to allow space for the drogue parachute. A U-bolt will be installed through the center of the bulkhead as an attachment point for the drogue parachute shock cord.

3.3.7 Fin Can Design

The fin can Section on the subscale launch vehicle will be 20.25 in. long and will contain the drogue parachute compartment, motor, and fins. The drogue parachute will be split between the midsection and fin can. A 12.0 in. coupler will be attached to the midsection leaving an 8.0 in. shoulder that will fit into the fin can drogue parachute compartment to be secured using shear pins during final assembly. Figure 3-29, below, shows the fin can Section with all components labelled.

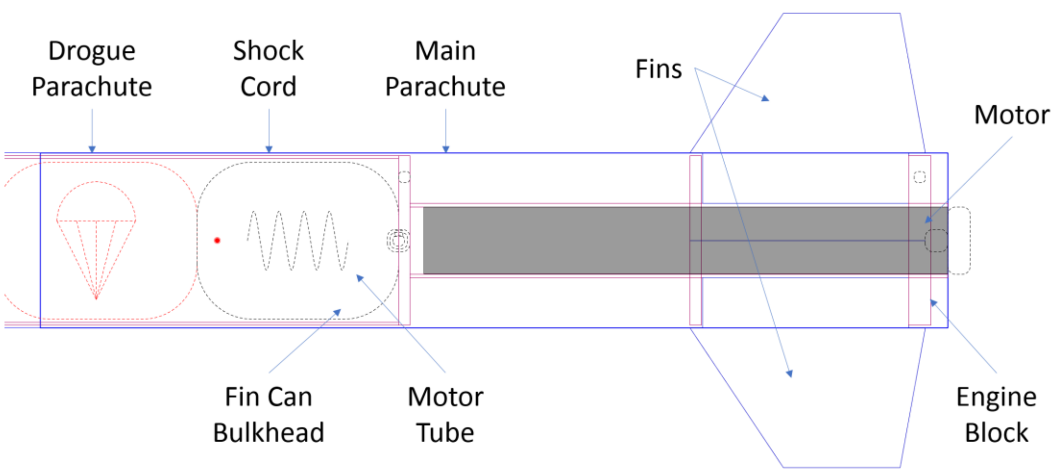


Figure 3-29: Fin Can with Component Labels

As shown in Figure 3-29, above, the fin can will have four fins attached through slots cut in the body tube and secured using epoxy fillets at the interSections with the body tube.

3.3.7.1 Fin Can Bulkhead Design

The midsection aft bulkhead will separate the drogue parachute compartment from the payload bay. The bulkhead will consist of two circular sheets of 0.25 in. aircraft-grade birch plywood sandwiched together using epoxy for a total thickness of 0.5 in. The aft face of the bulkhead will be fixed 4.0 in. from the bottom of the midsection to allow space for the drogue parachute. A U-bolt will be installed through the center of the bulkhead as an attachment point for the drogue parachute shock cord.

3.3.7.2 Motor Tube Design

The motor tube is designed to spread out the loading from the rocket motor retainer to the centering rings. The motor retainer will be fixed to the aft end of the motor tube, and the inner diameter will exactly match the outer diameter of the motor casing. This will allow the motor casing to slide into the tube and be secured using the retainer which will act as the single point of contact between the motor casing and the rest of the rocket. The motor tube is one of the most critical components that will be installed on the rocket, and much caution will be used when handling, installing, and securing the motor tube during fabrication.

The motor tube will have a length of 20.25 in., which is 0.30 in. longer than the motor casing to ensure that the motor casing will be able to fit vertically in the fin can. Since the engine block is recessed into the body tube by 0.125 in., which is explained in Section 3.2.8.4, the bottom of the motor tube will extend out by that amount as well. This extra length was included to allow the motor retainer to fit over the end of the motor tube and rest against the lower face of the engine block. Fiberglass was chosen for the motor tube, which is explained in Section 3.2.8.2. The motor tube will not be attached in any way to the fin can bulkhead. Instead, the motor tube will be attached by epoxy to the centering ring, engine block, and four fin tabs.

3.3.8 Motor Selection

Per Section 2.15 of the 2018 NASA SL Handbook: “The total impulse provided by a College and/or University launch vehicle will not exceed 5120 Newton-seconds (L-class).”

The subscale design is a 52% scale down of the full-scale rocket design and the subscale weight and dimensions are based around this scale. As there is no specified apogee goal for the subscale rocket in the 2018 NASA SL Handbook, OpenRocket simulations were used to determine a motor that would achieve approximately half the altitude of the full-scale goal altitude.

On-rail, the weight of the subscale rocket is 9.69 lb. An AeroTech I435T motor was selected with an AeroTech 38/600 Motor casing, which will allow the launch vehicle to achieve an apogee of 2006 ft AGL. The total impulse of the subscale motor is 568.9 N-s and is less than the upper limit of 5,120 N-s. The motor thrust curve is included in Section 3.7.2.

The motor selection is based on the full-scale design. A similar product from the same manufacturer was selected for performance comparison.

3.4 Selection and Rationale for Full-Scale Launch Vehicle Recovery System

This Section contains details and analysis of dual-deploy recovery system that will be employed on the full-scale launch vehicle.

3.4.1 Description of Recovery Events

Figure 3-30, below, displays an overview of the recovery system for the subscale and full-scale rockets. The main components of the system are integrated to contribute to 2 successful separation events, each releasing a parachute (first the drogue, and then the main).

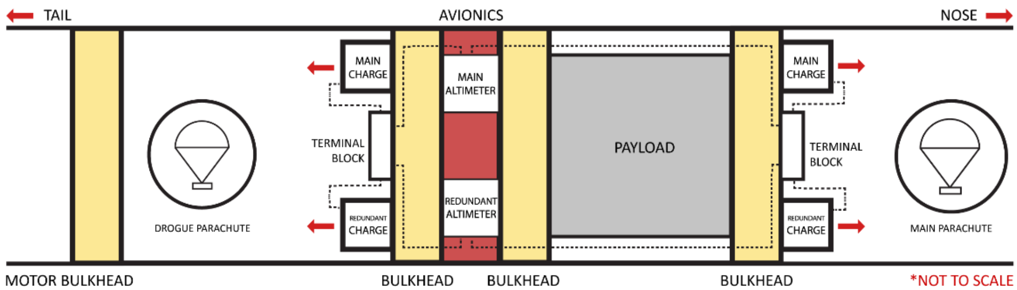


Figure 3-30: Recovery System Diagram

Successful recovery system performance begins in the avionics bay. For the recovery system, the avionics bay houses two altimeters, the main altimeter and the redundant altimeter. Ideally the system would function with one altimeter, but redundancy lies in including a second.

For the full-scale, at an apogee of 5,280 ft AGL, a signal is sent from the main altimeter to the terminal block in the drogue compartment, aft of the avionics bay and forward of the fin can/motor Section. The terminal block relays the signal through an E-match to a small PVC cap housing enough black powder to complete the first separation, covered by 3M blue painters tape to secure the E-match within the cap. The calculations for the exact black powder charge sizes are in Section 3.4.11 and will be defended in ground ejection tests prior to launch. The transmitted signal will cause the first separation to occur, and the drogue parachute will release. For redundancy, 1-second later, a second, redundant altimeter will send a signal through the terminal block in the drogue compartment to an E-match inserted into a second, same-sized black-powder charge in an identical PVC set-up, releasing the drogue parachute should there be an interruption or failure in the first, main system charge. At this point, the first separation of the recovery system is complete.

The second step of the recovery system is a successful, second separation and release of the main parachute. For the full-scale launch vehicle, the second separation will occur at 1000 ft AGL. At 1000 ft AGL, from the avionics bay, the main altimeter will send an electrical signal up through the payload bay, to the terminal block in the compartment housing the main parachute. The signal will transmit from the terminal block through an E-match connected to a PVC cap filled with an appropriately sized black powder charge, secured with 3M blue painters tape. This charge will pressurize the main cavity and forcefully separate the nosecone from the midsection of the rocket, releasing the main parachute deployment bag, which will allow for more time during the unfurling of shroud lines and parachute opening, decreasing the force from the parachute opening on the separated body Sections. For redundancy, 1 second later, the redundant altimeter will send a signal through the terminal block to a second E-match connected to a second, separate PVC cap containing the same-sized black powder charge, also sealed by painters' tape to contain the charge. The calculations for charge sizes are included in Section 3.4.11 and will also be defended with ground ejection tests prior to launch. At this point, the second, main separation of the recovery Section is complete.

3.4.2 Avionics

The launch vehicle will contain a single avionics bay which will house all recovery system electronics. In order to keep the launch vehicle at the appropriate length, the length of the avionics bay will be 3.25 in. The avionics (AV) bay will contain a sled that is slid into the bay through the removable hatch in the body tube. The AV sled will hold two 9 V Duracell batteries, one Entacore AIM USB 3.0 altimeter, and one StratoLoggerCF altimeter. Two different brands of altimeters will be used to avoid the possibility of both altimeters failing due to a common manufacturers defect. The StratoLoggerCF and Entacore AIM USB 3.0 altimeters were chosen because the team has used those altimeters in the past and has not had any issues with them. The altimeters will be independently powered by their own 9 V batteries. The Duracell brand was chosen because an experiment was conducted by the team in a previous year which found that Duracell brand batteries tended to deviate from 9 V less than other brands. The StratoLogger will be designated as the official competition altimeter because it has successfully been used as the main altimeter before. The StratoLogger will be wired to the two main ejection charges: one for the separation of the fin can for drogue deployment and one for the separation of the nose cone for main deployment. The altimeter will be programmed to activate drogue deployment at apogee and main deployment at an altitude of 1000 ft. The Entacore altimeter will be wired to two backup charges, one for each separation event, and will be programmed to activate at a 1 s. delay compared to the main charge. The one second delay is added to the redundant charges to avoid over pressurizing the tube and damaging bulkheads or the body tube. A 1 s. delay has been used by the team before and has proven to be a sufficient delay. Each altimeter will be armed with a dedicated key switch accessible from the exterior of the rocket airframe when the rocket is in the launch configuration on the launch pad. The switches will not be armed until the vehicle is erected on the launch rail to avoid battery drain, and will be capable of being locked in the ON position for launch. Since each altimeter has a dedicated battery and switch, the two systems will operate on separate circuits and will be independently redundant.

3.4.2.1 GPS Tracking

GPS tracking of the full-scale rocket will be done by a BigRedBee BRB 900 transmitter which will be installed in the payload. A BRB 900 transmitter will be used because the team has successfully used them in the past. The payload bay and the AV bay are separated by a bulkhead which will provide sufficient shielding from stray EM waves, the recovery system will not experience interference from the GPS.

3.4.3 Electrical Schematic for Recovery System

Figure 3-31, below, shows a block diagram of the recovery system responsible for the deployment of the drogue and main parachutes.

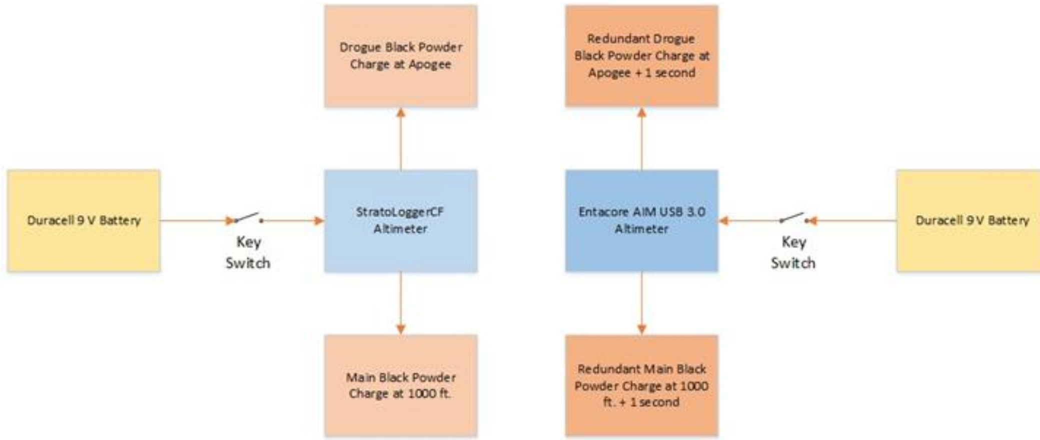


Figure 3-31: Recovery System Electronics Diagram

3.4.4 Avionics Sled

Figure 3-32, below, shows the current design for the avionics sled when fully assembled with altimeters and batteries installed.

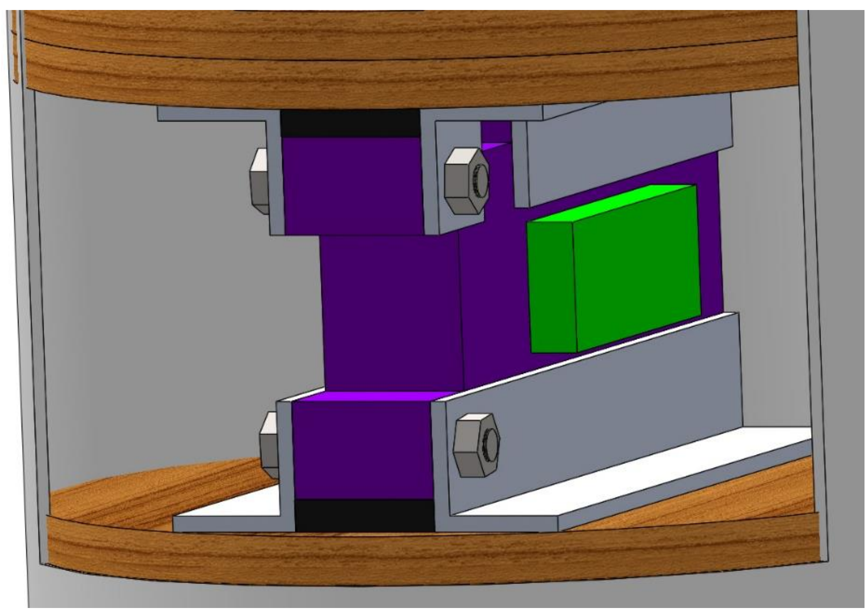


Figure 3-32: AV Sled Final Assembly

An exploded view of the AV sled assembly is shown in Figure 3-33 below.

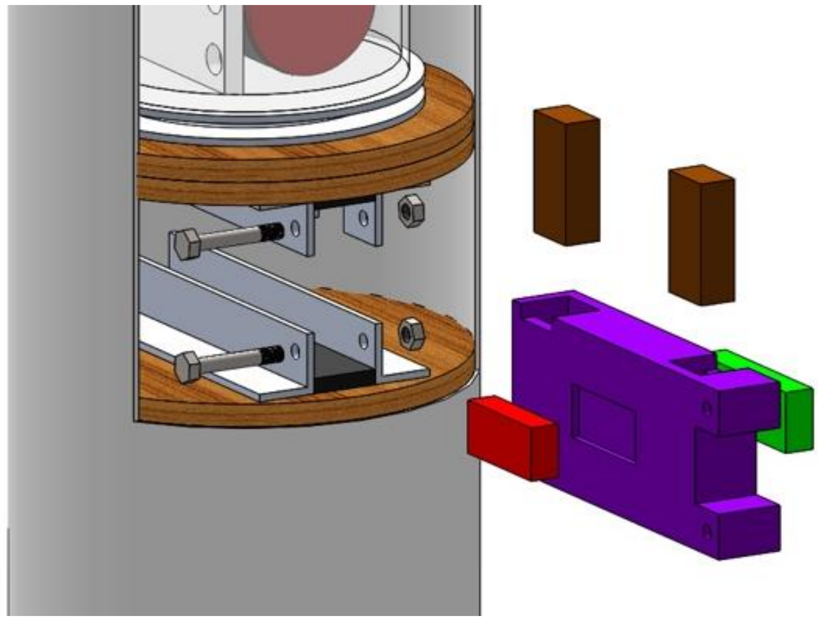


Figure 3-33: Exploded View of AV Sled

In Figure 3-32 and Figure 3-33, the StratoLoggerCF altimeter is shown in red and the Entacore AIM USB 3.0 altimeter is shown in green. Two 9 V Duracell batteries are shown in brown and the 3D printed sled is shown in purple. Figure #, below, shows a dimensional drawing of the sled which has two 0.125 in. indents on opposite faces to hold the altimeters in place during flight.

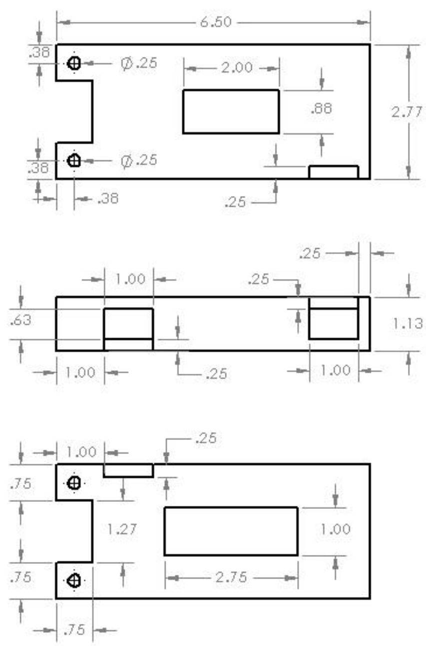


Figure 3-34: Exploded View of AV Sled

The indents provide some support to the board of the altimeter, but are kept shallow to allow for wiring. The altimeters will be screwed to the sled at their built-in attachment points, and the sled is kept thick between the altimeter slots in order to have material to be screwed into. The top of the sled has two slots 2.25 in. deep that the 9 V batteries will slide into. There is enough depth to fit the battery and the battery cap, and there are slits for the wires of the battery cap to exit the enclosure. Before they are inserted, batteries will be zip tied to their battery caps to ensure that the battery caps remain connected during flight. The walls of the slots are 0.25 in. thick, which should provide more than enough strength to keep the batteries contained. Vertical slots for holding the batteries where chosen because this configuration will keep the batteries enclosed on all sides once the sled is slid inside the AV bay. This will prevent the batteries from moving, which will reduce the risk of batteries getting disconnected from altimeters. When the sled is inside the AV bay, 4 aluminum 1 in. x 1 in. L brackets will keep the sled immobile. Slots will be made in the brackets where battery cap wires need to exit the battery enclosure. Two 0.25 in. thick foam inserts will be between the sled and the bulkheads. The foam will provide some cushioning and will keep the fit snug, reducing the risk of heavy vibration of the sled. The foam inserts and the brackets will both be permanently epoxied to the bulkheads. Once the sled is in place, two bolts will secure the sled to the L brackets to prevent the sled from sliding. The sled will be 3D printed out of AVC plastic using a Rostock Max V2 SeeMeCNC printer. 3D printing was chosen because of the complex shape of the sled. AVC plastic will be used because the team has this type of plastic on hand, and has successfully used it before. Due to the shape of the sled, it will be printed in several parts which will then be glued together.

3.4.4.1 AV Sled Weight

The goal weight of the AV bay is 1.5 lb. SolidWorks estimates the current design for the full avionics bay assembly to weigh 1.54 lb. This estimate assumes aluminum 6061 density for the brackets, AVC plastic density for the sled, and forged steel for the bolts, using standard SolidWorks libraries. The altimeters and batteries were weighed with a scale and the weights for those components were added to solid works. The weight estimate assumes solid AVC plastic for the sled, while in reality the sled won’t contain 100% infill, so the sled will weigh less than is estimated. The estimate also doesn’t include the foam inserts, wires, battery caps, terminal blocks, key switches or epoxy, which will add to the weight. Once all materials are acquired and a model of the sled is printed, a more accurate AV bay weight will be estimated. At that point, the team will focus on bringing the weight to 1.5 lb. Should the weight be over 1.5 lb., several modifications can be made to reduce weight. The walls of the battery enclosure can be reduced in thickness without sacrificing structural integrity. The central part of the sled where the altimeters are attached can be reduced in thickness by changing the overall shape of the sled from a box to an I shape. Several short segments of the L bracket can be used instead of the full-length L bracket.

3.4.5 Avionics Bay Pressure Sampling Holes

The airframe of the avionics bay must contain static pressure sampling holes to allow for the altimeters to sample outside air pressure. The holes will need to be large enough to accurately sample outside air pressure, but small enough to avoid pressure variation due to wind currents. Since the StratoLogger will be used as the competition altimeter, the StratoLoggerCF manual will be used for porthole sizing to ensure optimal performance. The StratoLogger manual recommends four portholes placed 90 degrees apart from each other around the body tube surface. The diameter of these portholes can be defined as:

$$Port\ hole\ diameter = D^2 * L * 0.0008 \tag{9}$$

where D is the AV bay diameter and L is the AV bay length. Given a body tube diameter of 7.5 in. and an AV bay length of 3 in., each of the four pressure sampling holes will be approximately 0.135 in. After drilling, the area around the portholes will be sanded to ensure there are no raised edges which might prevent smooth airflow through the holes.

3.4.6 Recovery System Alternatives

The following list contains the focal points of the recovery system and parachute selection for the team:

- the integration and protection of an electronics-heavy payload
- the forces transferred to the payload, hinges/doors, and integration technology
- the influence of wind drift on landing position

During the design of the recovery system and layout of rocket compartments, several critical brainstorming events among the team identified these focal points. The logical approach to inducing greater drag and executing a soft landing is to employ a set of larger parachutes, though this introduces packing constraints, weight limits and deployment altitude limits.

One concern is that in the current design, the main parachute will eject adjacent to the payload compartment housing the rover and any ejection charge forces, or forces from the extension and tightening of the shock cords, would transfer directly through the payload compartment. Technologies that reduce deployment force would decrease risk involved with damaging the payload during flight due to sudden force exertion on components with strength limits.

Working with the additional challenges of budget and giving priority to available hardware and resources, the concepts of reefing and variable drag technologies would allow for larger parachutes to be used by delaying the full deployment and inflation of the main parachute.

Reefing can be accomplished with several techniques, but the options currently considered employ the use of deployment bags, pre-reefed center shroud lines, and/or a reefing ring. Reefing is the process of partially or completely restricting the shroud lines of a parachute to delay or prohibit the full inflation of the parachute canopy. The affects of implementing reefing processes are outlined in Figure 3-35, below.[2]

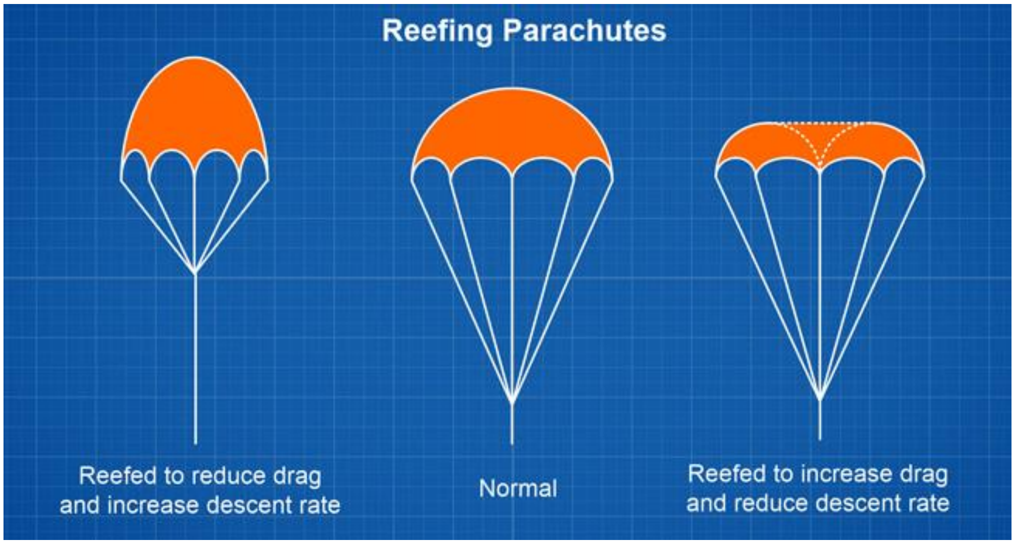


Figure 3-35: Methods of Reefing Parachutes

Due to the intricate folding and line interference of using reefing rings, the selected method is pre-reefed parachutes. The Fruity Chutes Iris Ultra Compact parachutes employ reefing of the center diameter shroud lines, which line the inner diameter of the parachute. Reefing these lines decreases the amount of surface available for airflow to resist during descent. In changing the shape of the parachute in this way, drag is increased and the coefficient of drag, C_d, is increased. The coefficient of drag for the un-reefed, elliptical parachute is 1.5, where the coefficient of drag of a reefed parachute with the same diameter is 2.2.

For mathematical modeling purposes, two 60 in. parachutes are compared, one without reefing and one with, and both with a mass of 20 lb falling beneath. The first parachute is an elliptical parachute with a C_d of 1.5. With the absence of reefing, the descent velocity is 24.38 ft/s. The second parachute is an Iris Ultra Compact parachute with centered reefing has a C_d of 2.2 and a descent velocity of 20.04 ft/s. Table 3-6, below, includes the descent rates for several sizes of both types of parachutes under the same theoretical 20 lb mass.

Table 3-6: Comparison of Reefed and Non-Reefed Parachute Descent Rates

Diameter of Parachute (in.)	Iris Ultra Compact Reefed Descent Rate (ft/s)	Standard Elliptical Non-reefed Descent Rate (ft/s)
18	N/A	81.27
24	N/A	60.67
30	40.08	48.76
36	33.40	40.63
48	25.05	30.33
60	20.04	24.47
72	16.70	N/A
96	12.52	N/A

From the table above, it can be concluded that a smaller, lighter parachute with reefing would act similar to a larger elliptical parachute without reefing. For the inclusion of reefing into the recovery system, the Iris Ultra Compact parachutes will be used as the main parachutes of the subscale and full-scale launch vehicles. The known main C_d is 2.2.

For the drogue parachute, the focus is to find a small parachute with minimal opportunity for influence by wind drift, so an elliptical parachute shape with C_d of 1.5 will be chosen. This selection is expanded on in Section 3.4.8

3.4.7 Kinetic Energy at Landing

To address the kinetic energy requirement 3.3 within the 2018 NASA SLI Handbook, "At landing, each independent Sections of the launch vehicle will have a maximum kinetic energy of 75 ft-lbf", maximum impact velocities for each individual Section are determined from the masses of each of the Sections in order to set a limit for terminal velocities. Limiting the terminal velocities establishes an upper limit for the impact velocity/descent rate of the falling rocket and with parachute sizing will land with a kinetic energy less than the established maximum requirement.

Using MATLAB for analysis, the maximum impact velocities of each Section of the falling rocket body, with parachutes, can be defined as:

$$KE = \frac{1}{2}mV_i^2 \rightarrow V_i = \sqrt{\frac{2KE}{m}} \tag{10}$$

where KE is kinetic energy, m is mass, and V_i is the impact velocity. Using the assigned kinetic energy of 75 ft-lbf and the known empty rocket mass of 1.288 slugs, the maximum allowable descent velocity was calculated as 10.79 ft/s.

Table 3-7 below contains the masses of each of the three separate body Sections, as required by Section 3.3 in the 2018 NASA SL Handbook limits the individual Sections' kinetic energies at landing.

Table 3-7: Comparison of Reefed and Non-Reefed Parachute Descent Rates

Body Section	Mass (slugs)	Maximum Descent Velocity (ft/s)
Nose Cone	0.24	25.00
Midsection	0.47	17.86
Fin Can	0.53	16.82

The MATLAB code for kinetic energy calculations is included in Section 8. Using the minimum of the body Section's descent velocity requirements, 16.82 ft/s, parachute sizes are determined.

3.4.8 Drogue Parachute

For the full-scale, considering a sensitive, electronics-heavy, payload and the influence of wind drift on various parachute sizes and deployment altitudes, parachute sizing for both the drogue and main parachutes is mission critical.

The purpose of the drogue parachute is to take advantage of the introduction of drag at the point where vertical velocity is 0 ft/s, apogee. At apogee, after all of the motor propellant has burnt on ascent, all of the forces on the rocket are momentarily balanced. Though a large parachute deployed at apogee would promote a soft, safe landing and a recoverable, reusable rocket as listed in requirement 2.18 of the 2018 NASA SLI Handbook, the effect of deploying a large parachute at approximately 5,280 ft AGL could result in a landing position outside of the recovery range posted in requirement 3.9 of the 2018 NASA SLI Handbook, which limits landing distance from the launch pad to a 2500 ft radius.

For drogue sizing, the C_{d_d} will be 1.5 for an elliptical parachute. The following equation can be used to determine the equilibrium descent velocity for parachutes of specific surface areas:

$$C_{d_d} = \frac{2mg}{V_{ed}^2 S_0 \rho} \tag{11}$$

where m is the mass of the body falling beneath the parachute, which is the empty rocket mass of 41.35 lb or 1.28 slugs, g is the coefficient of gravity 32.174 ft/s^2, V_{ed} is the terminal velocity of the falling mass with the drogue, S_0 is the surface area of the employed drogue, and ρ is the air density, .002377 slugs/ft^3. In order to determine the size of parachute needed, a maximum desired drogue terminal velocity, V_{ed} was set to 60 ft/s and different diameter parachutes were tested numerically. The results can be seen below in Figure 3-36.

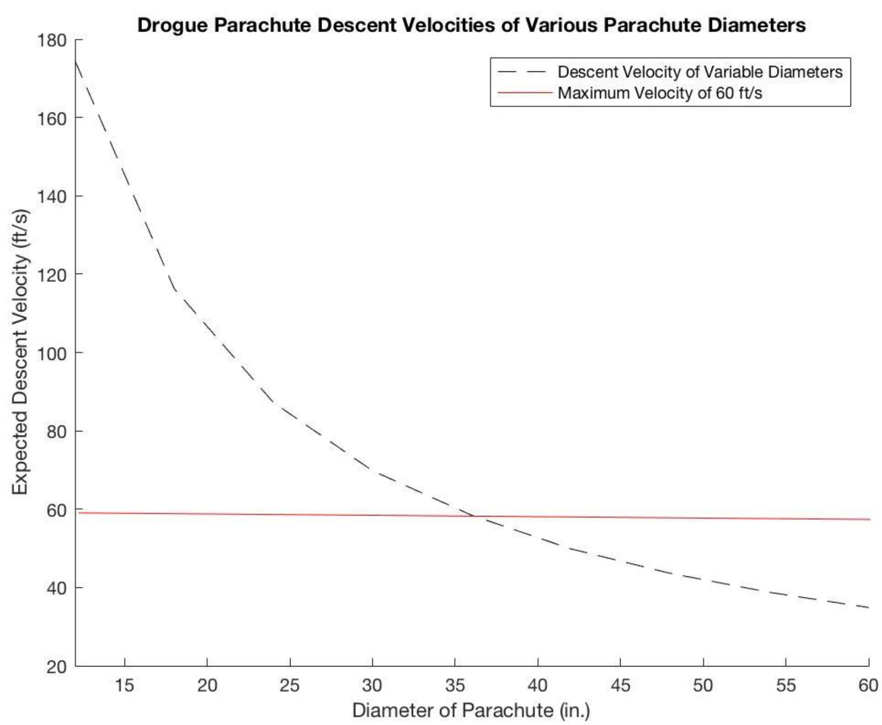


Figure 3-36: Full-scale Drogue Sizing vs. Descent Velocity Prediction

From this figure, a drogue parachute of 36 in. was selected. The terminal velocity of the rocket beneath the drogue will be 58.43 ft/s. The Fruity Chutes Standard Elliptical 36 in. parachute was selected as the full-scale drogue parachute.

3.4.9 Main Parachute

At 1000 ft AGL, the main parachute will eject and deploy. A similar equation to the drogue calculation will be used:

$$C_{d_m} = \frac{2mg}{V_{em}^2 S_{0_m} \rho} \tag{12}$$

but the terminal velocity of the mass beneath the main parachute, V_{em} will be set as a maximum of 16.82 ft/s, the maximum impact velocity for a kinetic energy of 75ft-lbf, calculated in Section 3.4.7. The chosen style of parachute C_{d_m} is 2.2 as shown above. Figure 3-37, below, shows the predicted descent velocities of Iris Ultra Compact parachutes of various diameters.

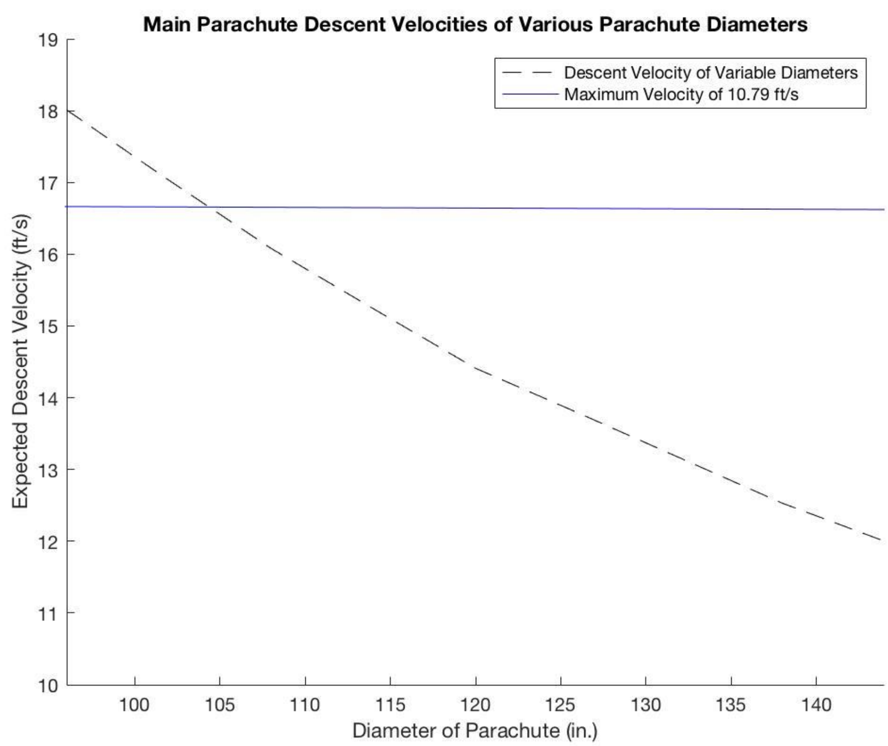


Figure 3-37: Full-scale Main Sizing vs. Descent Velocity Prediction

From Figure 3-37, for the maximum impact velocity of 16.82 ft/s, a parachute diameter of at least 108 in. is required. For a safe landing, and to account for any weight variances in construction, the Iris Ultra Compact 120 in. parachute will be used for the main parachute and will allow for a descent velocity of 14.41 ft/s, which is below the maximum impact velocity of 16.82 ft/s. The predicted kinetic energies of each Section at landing are included in the Table 3-8 below.

Table 3-8: Full-Scale Predicted Kinetic Energy Values at Impact

Body Section	Mass (slugs)	Maximum Descent Velocity (ft/s)	Kinetic Energy at Landing (ft-lbf)
Nose Cone	0.24	14.41	24.92
Midsection	0.47	14.41	48.79
Fin Can	0.53	14.41	55.03

3.4.10 Shock Cord Sizing

For the full-scale launch vehicle, both separations will require 480-in of ½-in tubular Kevlar shock cord in each compartment. The chosen shock cord is 2200 lb strength from Giant Leap Rocketry and will withstand the opposing forces from ejection and black powder forces explained in Section 3.4.11. Figure 3-38 below is a diagram for shock cord arrangement during separation.

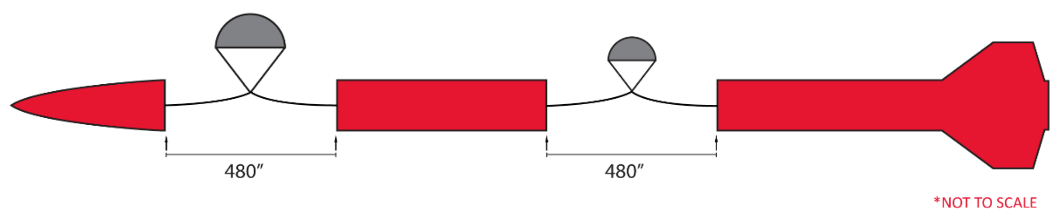


Figure 3-38: Full-Scale Shock Cord Sizing Diagram

3.4.11 Black Powder Sizing

Black powder ejection methods will be used for the full-scale Section separations. Goex 4F black powder will be used in both ground testing, and ejection separations in flight. In order to determine the mass of black powder that is required to complete each separation, the following equation is used:

$$C_{d_m} = \frac{2mg}{V_{em}^2 S_{0_m} \rho} \tag{13}$$

where m is the mass in grams, 0.006 is a constant for converting in^3 to grams and is also related to the amount of pressure in the filled cavity, in this case 15 psi, L_{sect} is the length of the body Section in inches that must move in order to separate and D_{sect} is the diameter of the Section in inches.

Figure 3-39 below is a diagram of the lengths of Sections for the full-scale, and the separation lengths.

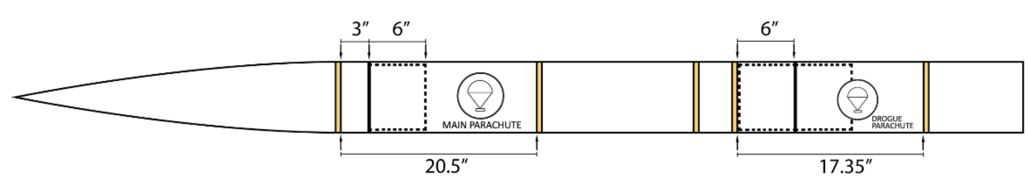


Figure 3-39: Full-Scale Separation Sections and Charge Compartments

For the forward nosecone Section,

$$m_{main} = 0.006 * 20.5\ in.* (7.34\ in.)^2 = 6.6\ g$$

$$m_{drogue} = 0.006 * 17.35\ in.* (7.34\ in.)^2 = 5.6\ g$$

both of these numbers seem high, but in order for the black powder to occupy the full cavity volumes and eject the parachutes to complete the separations and recovery stages, two main charges of 6.6 g and two drogue charges of 5.6 g will be used. Using the assumed pressure of 15psi, the ejection force can be calculated as:

$$F = P * A \tag{14}$$

where F is the ejection force, P is the pressure that occupies the Section volume and A is the area of the forward bulkhead, opposite of the charge.

$$F = P * A \tag{15}$$

where F is the ejection force, P is the pressure that occupies the Section volume and A is the area of the forward bulkhead, opposite of the charge.

$$F = 15\frac{lb}{in.^2} * \left(\pi\left(\frac{7.34\ in}{2}\right)^2\right) = \ 635\ lbf \tag{16}$$

During the drogue separation the ejection force will maximize at 635 lbf. With this data, the number of sheer pins needed is determined. The familiar shear pins used are 4-40 Nylon, typically 0.5 in. length. One 4-40 Nylon sheer pin can handle a maximum force of 76 lb. As the number of shear pins is increased, the force is additive, the limit for the number of shear pins that can be used is that cumulatively, the force required to break the shear pins must not exceed 635 lb. In the drogue Section, (4) 4-40 Nylon shear pins ½ in. long will require a force of 304 lb to separate and release the drogue chute.

Similarly, the 6.6 g charge meant to complete the main separation will produce a larger force of 730 lb, but the same number and size of shear pins will be acceptable, as the force will still overcome the 304 lb of shear force required to remove the pins and complete the main separation, ejecting the main parachute.

3.5 Selection and Rationale for Subscale Launch Vehicle Recovery System

This Section contains details and analysis of dual-deploy recovery system that will be employed on the full-scale launch vehicle.

3.5.1 Description of Recovery Events

For a visual diagram of recovery events, see Figure 3-30.

For the subscale launch vehicle, at an apogee of 2,006 ft AGL, a signal is sent from the main altimeter in the avionics bay to the terminal block in the drogue compartment, aft of the avionics bay and forward of the fin can/motor Section. The terminal block relays the signal through an E-match to a small PVC cap, housing enough black powder to complete the first separation. The PVC cap is covered by 3M blue painters tape to secure the match and powder within the cap. The calculations for the exact charge sizes are completed in Section 3.5.9 and will be defended with ground ejection tests prior to launch. The transmitted signal will cause the first separation to occur, and the drogue parachute will release. For redundancy, 1-second later, a second, redundant altimeter will send a signal through the terminal block of the drogue Section to an E-match inserted into a second, same-sized black-powder charge in an identical PVC set-up, releasing the drogue parachute should there be an interruption or failure in the first ejection charge. At this point, the first separation of the recovery system is complete.

The second step of the recovery system is a successful second separation and release of the main parachute. For the subscale launch vehicle, the second separation will occur at 800 ft AGL. At 800 ft AGL, from the avionics bay, the main altimeter will send a signal up through the payload bay, to the terminal block in the compartment housing the main parachute. The signal will transmit from the terminal block through an E-match connected to a PVC cap filled with an appropriately sized black powder charge, secured with 3M blue painters tape. This charge will pressurize the main compartment and forcefully separate the nosecone from the midsection of the rocket, releasing the main parachute deployment bag. The deployment bag will allow for more time during the unfurling of shroud lines and parachute opening, decreasing the opening force transferred to the separated body tubes. For redundancy, 1 second later, the redundant altimeter will send a second signal through the terminal block to a second E-match connected to a second, separate PVC cap containing the same-sized black powder charge, also sealed by painters tape to contain the charge. The calculations for charge sizes are included in Section 3.5.9 and will be defended with ground ejection tests prior to launch. At this point, the second, main separation of the recovery Section is complete.

3.5.2 Avionics

The Subscale rocket will contain a combined avionics and payload bay which will house the payload in addition to all recovery system electronics. A removable bulkhead will separate the payload from the avionics. The length of the avionics portion of the bay will be 2.75 in. Two removable threaded rods will be attached to the forward and aft bulkheads, and an avionics sled will be slid onto the rods during assembly. The AV sled will hold two 9 V Duracell batteries, one Entacore AIM USB 3.0 altimeter, and one StratoLoggerCF altimeter. Two different brands of altimeters will be used to avoid the possibility of both altimeters failing due to a common manufacturers defect. The StratoLoggerCF and Entacore AIM USB 3.0 altimeters were chosen because the team has used those altimeters in the past and has not had any issues with them. The altimeters will be independently powered by their own 9 V batteries. The Duracell brand was chosen because an experiment was conducted by the team in a previous year which found that Duracell brand batteries tended to deviate from 9 V less than other brands. The StratoLogger will be wired to the two main ejection charges: one for the separation of the fin can for drogue deployment and one for the separation of the nose cone for main deployment. The altimeter will be programmed to activate drogue deployment at apogee and main deployment at an altitude of 800 ft. The Entacore altimeter will be wired to two backup charges, one for each separation event, and will be programmed to activate at a 1 s. delay compared to the main charge. The 1 s. delay is added to the redundant charges to avoid over pressurizing the tube and damaging bulkheads or the body tube. A 1 s. delay has been used by the team before and has proven to be a sufficient delay. Each altimeter will be armed with a dedicated key switch accessible from the exterior of the rocket airframe when the rocket is in the launch configuration on the launch pad. The switches will not be armed until the vehicle is erected on the launch rail to avoid battery drain, and will be capable of being locked in the ON position for launch. Since each altimeter has a dedicated battery and switch, the two systems will operate on separate circuits and will be independently redundant.

Figure 3-40, below, shows the block diagram of the recovery system electronics responsible for the deployment of the drogue and main parachutes.

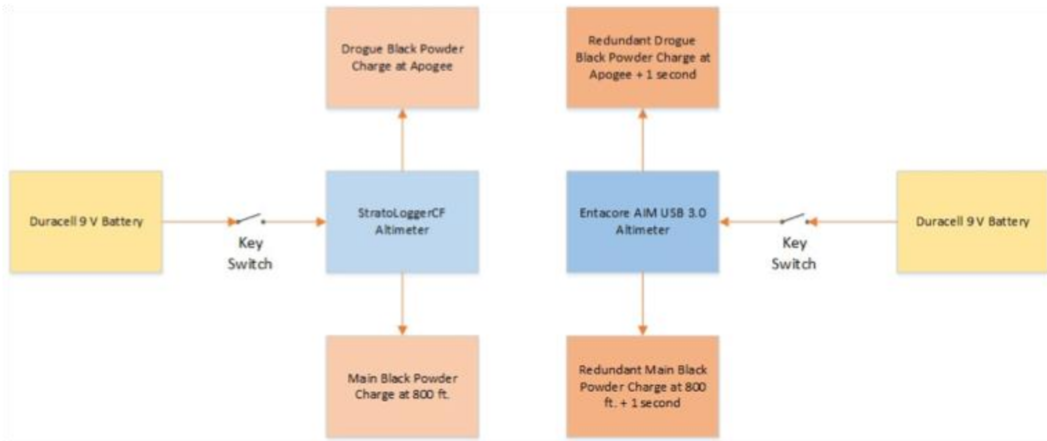


Figure 3-40: Subscale Recovery Electrical System

3.5.2.1 GPS Tracking

GPS tracking of the sub scale rocket will be done by a BRB 900 transmitter, which will be screwed into the bulkhead aft of the AV bay. A BRB 900 transmitter will be used because the team has successfully used them in the past. The altimeters that the team will be using are not sensitive to RF radiation, so the recovery system will not experience interference from the GPS

3.5.3 Subscale Avionics Sled

Figure 3-41, below, is a model of the avionics sled which will hold the recovery system electrical components in the subscale rocket.

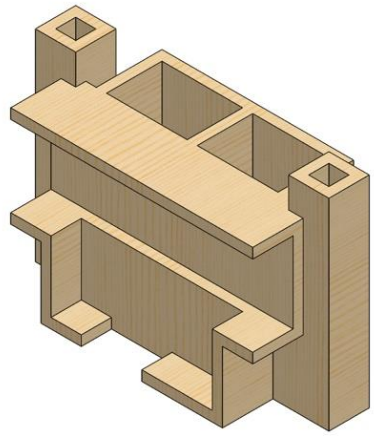


Figure 3-41: Subscale Avionics Sled

The subscale avionics sled will be made out of laser cut 0.125 in. aircraft plywood birch and epoxied together. Plywood was chosen because it is relatively lightweight and has high strength. There are two slots that hold 9 V batteries vertically on one face of the sled, and two horizontal slots for altimeters on the other face. This configuration was chosen to minimize sled size. The raised edges of the altimeter slots provide support to the altimeters while allowing for wiring, and several plywood layers below the altimeters allow for material for the altimeters to be screwed into. Once the batteries and altimeters are wired, two zip ties will be pulled taught around the sled, over a battery hole opening, and over both altimeters. The payload will be slid onto two threaded rods during assembly, and will be kept in place with locked nuts forward and aft of the sled on each rod.

3.5.4 Avionics Bay Pressure Sampling Holes

The airframe of the avionics bay must contain static pressure sampling holes to allow for the altimeters to sample outside air pressure. The holes will need to be large enough to accurately sample outside air pressure, but small enough to avoid pressure variation due to wind currents. Since the StratoLogger will be used as the competition altimeter, the StratoLoggerCF manual will be used for porthole sizing to ensure optimal performance. The StratoLogger manual recommends four portholes placed 90 degrees apart from each other around the body tube surface. The diameter of these portholes can be defined as:

$$Port\ hole\ diameter = D^2 * L * 0.0008 \qquad (17)$$

where D is the AV bay diameter and L is the AV bay length. Given a body tube diameter of 3.9 in. and an AV bay length of 2.75 in., each of the four pressure sampling holes will have a diameter of around 0.03 in. After drilling, the area around the portholes will be sanded to ensure there are no raised edges, which might prevent smooth airflow through the holes.

3.5.5 Kinetic Energy at Landing (Team-Derived Requirement)

For the recovery of the subscale, a 52% scale for calculation of kinetic energy at landing was used because the subscale model is 52% of the full-scale model. 52% of the 75ft-lbf requirement makes a maximum kinetic energy at landing of 39 ft-lbf. This is a team-derived requirement.

Using MATLAB for analysis, the maximum impact velocities of each Section of the falling rocket body, with parachutes, can be defined as:

$$KE = \frac{1}{2}mV_i^2 \rightarrow V_i = \sqrt{\frac{2KE}{m}} \tag{18}$$

where KE is kinetic energy, m is mass, and V_i is the impact velocity. Using the assigned kinetic energy of 75 ft-lbf and the known empty rocket mass of 0.282 slugs, the maximum allowable descent velocity was calculated as 16.63 ft/s.

Table 3-9, below, contains the masses of each of the three separate body Sections, as requirement 3.3 limits the individual Sections' kinetic energies at landing.

Table 3-9: Body Section Max Descent Velocity

Body Section	Mass (slugs)	Maximum Descent Velocity (ft/s)
Nose Cone	0.03	50.99
Midsection	0.14	23.60
Fin Can	0.09	29.43

The MATLAB code for kinetic energy calculations is included in Appendix A. Using the minimum of the body Section's descent velocity requirements, 23.60 ft/s, parachute sizes were determined.

3.5.6 Drogue Parachute

For drogue sizing of the subscale, the C_{d_d} will be 1.5 for an elliptical parachute. The following equation can be used to determine the equilibrium descent velocity for parachutes of specific surface areas:

$$C_{d_d} = \frac{2mg}{V_{ed}^2 S_0 \rho} \tag{19}$$

where m is the mass of the body falling beneath the parachute which is the empty rocket mass of 9.1 lb or .282 slugs, g is the coefficient of gravity 32.174 ft/s^2, V_{ed} is the terminal velocity of the falling mass with the drogue, S_0 is the surface area of the employed drogue, and ρ is the air density, .002377 slugs/ft^3. In order to determine the size of parachute needed, a maximum desired drogue terminal velocity, V_{ed} was set to In order to determine the size of parachute needed, a maximum desired drogue descent velocity V_{ed} was set to 60 ft/s and different diameter parachutes were tested. The results can be seen below in Figure 3-42.

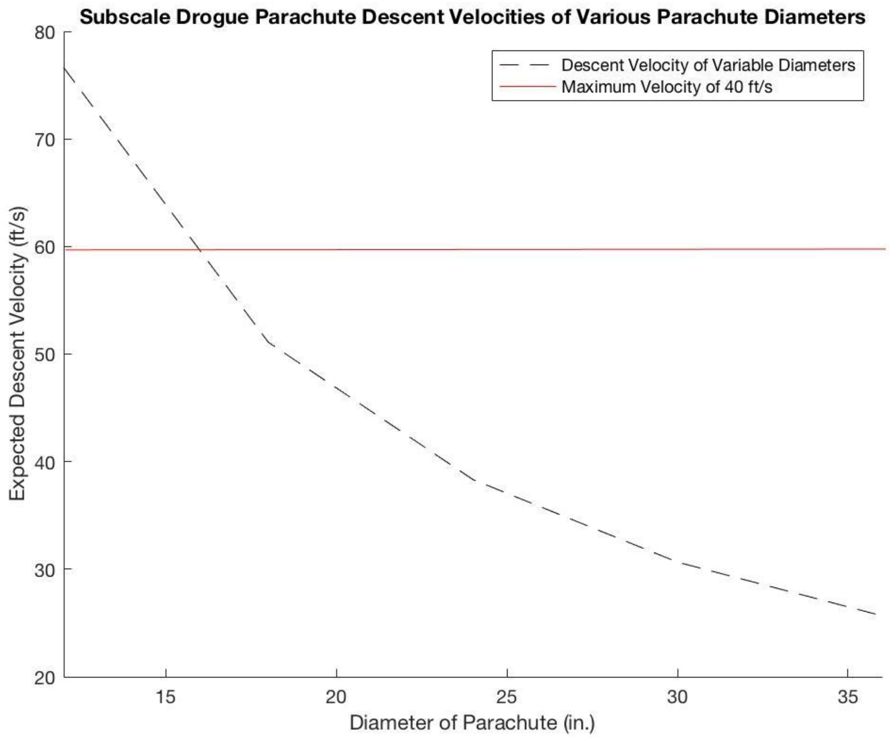


Figure 3-42: Subscale Drogue Sizing vs. Descent Velocity Prediction

From Figure 3-42, it can be seen that with a maximum velocity of 60 ft/s following drogue release an 18 in. elliptical parachute is needed, this achieves a terminal velocity of 51.10 ft/s. The 18 in. Fruity Chutes Elliptical parachute is the selected drogue parachute.

3.5.7 Main Parachute

At 800 ft AGL, the main parachute will eject and deployed soon after. A similar equation to the drogue calculation for coefficient of drag is used:

$$C_{d_m} = \frac{2mg}{V_{mi}^2 S_{0_m} \rho} \tag{20}$$

but the velocity of the mass beneath the main parachute, V_{em} will be set as a maximum of 15 ft/s, less than the maximum required impact/terminal velocity calculated in Section 3.5.5, 23.60 ft/s. The C_{d_m} of the chosen parachute style, the Iris Ultra Compact, is 2.0. Figure 3-43, below, shows predicted descent velocities of Iris Ultra Compact parachutes of various diameters.

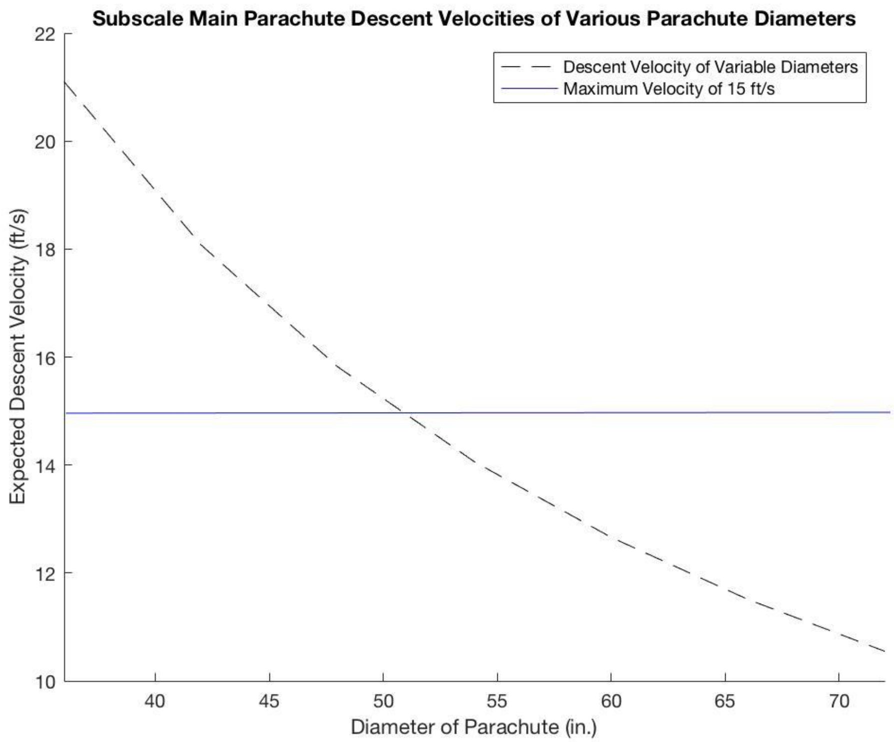


Figure 3-43: Subscale Main Sizing vs. Descent Velocity Prediction

As seen in **Error! Reference source not found.** above, in order to reach a descent velocity of 15 ft/s the minimum parachute diameter required is 54 in.

A 60-in Iris Ultra Compact parachute is selected for the subscale main parachute. The predicted descent velocity/impact velocity will be 12.66 ft/s. This velocity value is less than the predicted full-scale value, but the parachute is chosen because there is no deployable rover and because the kinetic energy requirement is halved for the subscale design. The predicted kinetic energies of each Section at landing are included in the Table 3-10 below.

Table 3-10: Subscale Predicted Kinetic Energy Values at Impact

Body Section	Mass (slugs)	Maximum Descent Velocity (ft/s)	Kinetic Energy at Landing (ft-lbf)
Nose Cone	0.03	12.66	2.404
Midsection	0.14	12.66	11.219
Fin Can	0.09	12.66	7.212

3.5.8 Shock Cord Sizing

For the subscale, both compartments require 360 in. of shock chord. The selected shock cord is ¼ in. tubular Kevlar from Giant Leap Rocketry will be used. The selected shock cord is 2200 lb strength and will withstand the opposing forces from the black powder ejection forces explained in Section 3.5.9 below. Figure 3-44 below shows a diagram of shock cord for both separations.

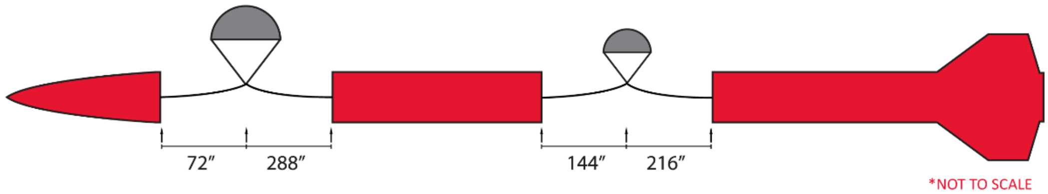


Figure 3-44: Subscale Shock Cord Diagram

3.5.9 Black Powder Sizing

Black powder ejection methods will be used for the subscale Section separations. In order to determine the mass of black powder that is required for each separation, the following equation is used:

$$m = 0.006 L_{sect} D_{sect}^2 \tag{21}$$

where m is the mass in grams, 0.006 is a constant for converting in^3 to grams and is also related to the amount of pressure in the filled cavity, in this case 15 psi, L_{sect} is the length of the body Section in inches that must move in order to separate and D_{sect} is the diameter of the Section in inches.

Figure 3-45, below, shows a diagram of the lengths of Sections for the subscale, and the separation lengths.

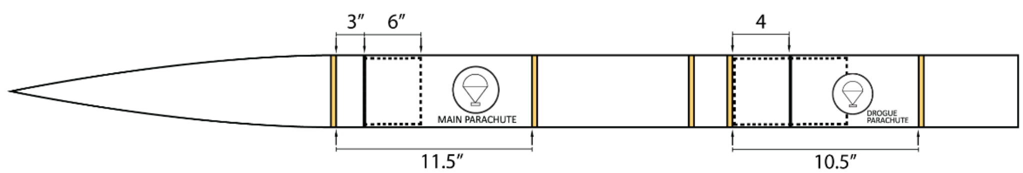


Figure 3-45: Subscale Separation Sections and Charge Compartments

For the forward nosecone Section,

$$m_{main} = 0.006 * 11.5\ in. * (3.76\ in.)^2 = .97\ grams = rounded\ to\ 1.00\ gram$$

$$m_{drogue} = 0.006 * 10.5\ in * (3.76\ in)^2 = .89\ grams = rounded\ to\ 1.00\ gram$$

and both of these values seem appropriate for the ejection of the drogue and main parachutes with respect to the volume of their respective cavities. The charge sizes will be tested and finalized through ground ejection charge testing. Using the assumed pressure of 15psi the ejection force can be calculated:

$$F = P * A \tag{22}$$

where F is the ejection force, P is the pressure that occupies the Section volume and A is the area of the forward bulkhead, opposite of the charge.

$$F = 15\frac{lb}{in.^2} * \left(\pi\left(\frac{3.776\ in}{2}\right)^2\right) = 168\ lbf \tag{23}$$

During the drogue and main separations, the ejection force will maximize at 168 lbf. With this data, the number of shear pins needed is determined. The familiar shear pins used are 4-40 Nylon, typically ½ in. length. One 4-40 Nylon sheer pin can handle a maximum force of 76 lbf. As the number of shear pins is increased, the force is additive, the limit for the number of shear pins that can be used is that cumulatively, the force required to break the shear pins must not exceed 168 lb. In the drogue and main Sections, (2) 4-40 Nylon shear pins ½ in. long will require a force of 152 lb to separate and release the drogue chute and then the main chute.

Because the calculation for black powder mass above is the minimum mass required, the charge will be increased to 1.5 g to force ejection and ensure the separation of Sections while also relying on the security of having two shear pins in place for each Section.

3.6 Mission Performance Predictions for Full-Scale Launch Vehicle

This Section contains results and analysis for full-scale launch simulations.

3.6.1 Stability Margin

As the motor burns and ejects mass, the CG will translate forward in the rocket, thereby increasing the stability margin throughout the entire powered phase of flight. However, a large stability margin is not required as it may cause the rocket to pitch off-course due to an external perturbation such as a gust of wind. Once perturbed, the rocket flight will become unpredictable which is not ideal in any scenario. To increase spectator safety and confirm that the rocket will perform as expected, the team investigated how the stability margin changed during flight. Table 3-11 below, shows the results of this investigation, which shows the rocket stability margin as a function of rocket mass and CG location for the ascent profile.

Table 3-11: Stability Margin at Various Events during Flight

Wind Condition (mph)	Event	Stability Margin (cal)
0	Rail Clearance	2.0754
	Burnout	2.753
	Apogee	5.4843
5	Rail Clearance	1.4579
	Burnout	2.7335
	Apogee	2.7712
10	Rail Clearance	1.0538
	Burnout	2.716
	Apogee	1.7485
15	Rail Clearance	0.7783
	Burnout	2.7019
	Apogee	0.9239

Wind Condition (mph)	Event	Stability Margin (cal)
20	Rail Clearance	0.5853
	Burnout	2.694
	Apogee	1.3343

Figure 3-47, below, shows the stability margin as a function of flight time and altitude for the ascent profile.

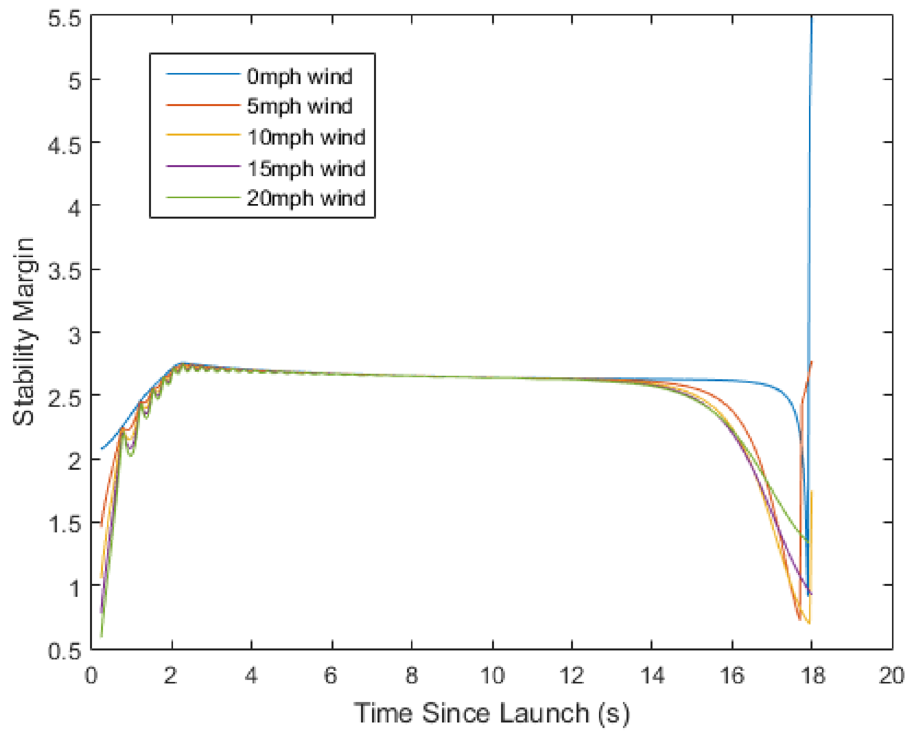


Figure 3-46: Time vs. Stability Margin

Stability margin, the distance between the center of gravity and the center of pressure, increases in an oscillatory motion for one quarter of the time to apogee. After engine burnout, the margin remains relatively constant until before apogee where it drops dramatically then rises at apogee.

3.6.2 Simulated Motor Thrust Curve

To support the motor selection, the motor thrust curves for both the Cesaroni L1685-SS and the AeroTech L2200GP were analyzed. Below, Figure 3-47: Cesaroni L1685-SS Motor Thrust Curve shows the thrust curve for the Cesaroni L1685-SS motor.

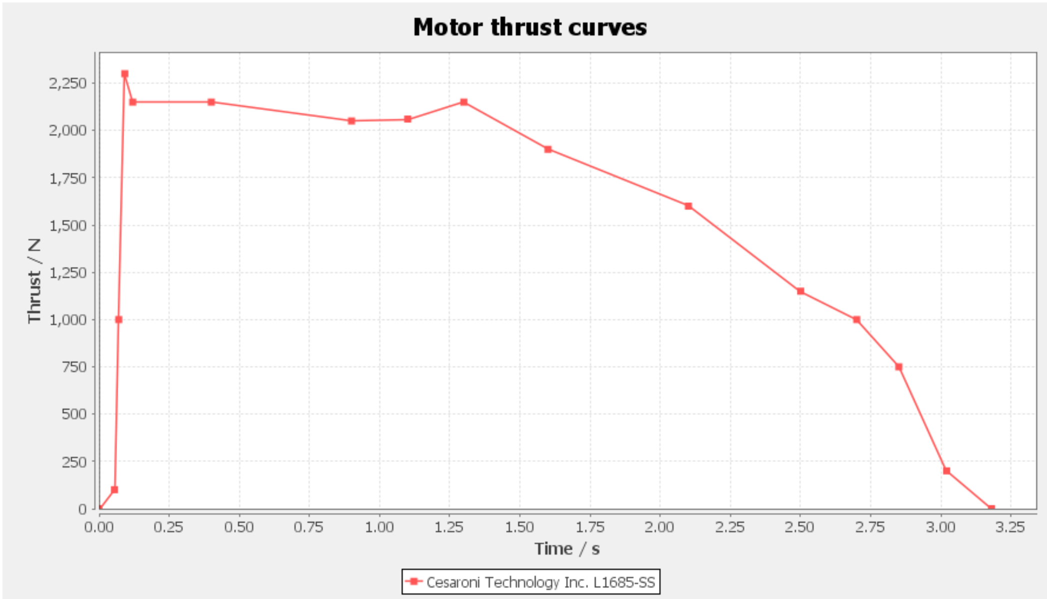


Figure 3-47: Cesaroni L1685-SS Motor Thrust Curve

This motor shows a smooth thrust trend and takes approximately 3.25 seconds to burn. The max thrust is 2300 N. The second, chosen contender for the full-scale motor was the AeroTech L2200GP, with the motor thrust curve shown below in Figure 3-48.

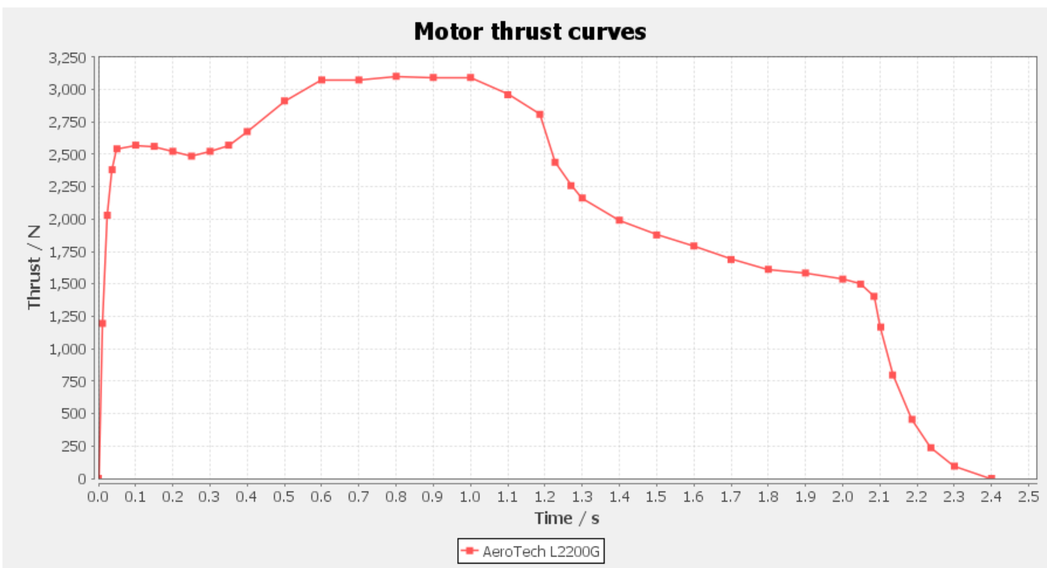


Figure 3-48: AeroTech L2200GP Motor Thrust Curve

From the thrust curve in Figure 3-48, the performance is not as smooth as that seen in Figure 3-47, but the AeroTech motor provides higher thrust and a higher total impulse in less burn time (2.4 s), stabilizing the rocket sooner in flight. The AeroTech has a max thrust of 3100 N, approximately 800 N greater than that of the Cesaroni.

3.6.3 Simulated Flight Profiles

The team has relied on flight simulation results to determine the effectiveness of the full-scale launch vehicle design. Figure 3-49, below, shows the results from one of these simulated launches, where location was set to Huntsville, AL and windspeed was a constant 10 mph, launching from an 8 ft launch rail angled 5° from vertical.

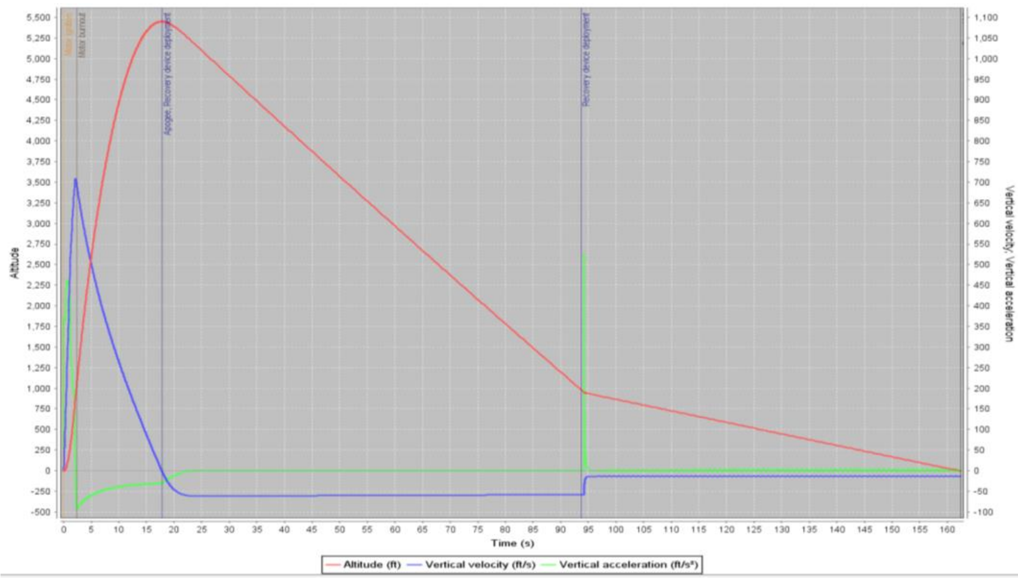


Figure 3-49: OpenRocket Simulation with 10 mph Winds at Huntsville, AL

Figure 3-49 shows that the rocket motor will burn out at approximately 3 seconds to let the rocket coast for another 12 seconds until apogee at 5,449 ft AGL. The total flight lasts 162 seconds and the ground impact velocity is 14.7 ft/s, well below the limit assigned in Section 3.4.7. The team is confident that the numerous flight simulations run using OpenRocket are reasonable to use when iterating the rocket design.

Table #, below, shows the results for various flight simulations using the same environmental parameters described above.

Table 3-12: OpenRocket Simulation Results for Varying Winds

Wind Speed (mph)	Apogee (ft AGL)	Flight Time (s)	Impact Velocity (ft/s)
0	5,575	165	14.7
5	5,569	164	13.0
10	5,449	162	14.7
15	5,527	163	14.7
20	5,496	164	13.0

The simulation results show that the rocket has a very predictable and steady flight pattern. Regardless of wind speed, the apogee never deviates more than 100 ft, and the impact velocity is never greater than 14.7 ft/s. The team is confident that the rocket will perform as expected, and will continue to use OpenRocket simulations to observe the effects that any small change may have on the overall design.

3.6.4 Simulated Wind Drift

To defend requirement 3.9 of the 2018 NASA SLI Handbook, where recovery distance must remain within a 2500 ft radius of the launch pad, wind drift calculations are completed to predict performance in various wind conditions. To simulate wind drift for 0 mph, 5 mph, 10 mph, 15mph and 20 mph winds, two methods of calculations are used; OpenRocket simulations provided drift distances and descent times and then to compare, hand calculations were completed with the equation below:

$$d_{drift}\,(ft) = t_{descent}\,(s) * V_{wind}\left(\frac{ft}{s}\right) \tag{24}$$

where d_{drift} is the lateral distance at landing from the launch pad, $t_{descent}$ is the time the rocket Sections take to reach the ground from apogee and V_{wind} is the wind speed converted to ft/s from mph. The mathematical results from method 1, hand calculations are included in Table 3-13, below.

Table 3-13: Predicted Wind Effect on Altitude and Drift

Wind Speed (mph)	Predicted Altitude (ft AGL)	Descent Time (s)	Wind Speed (ft/s)	Lateral Drift Distance from OpenRocket (ft)	Lateral Drift Distance Hand Calculation (ft/s)
0	5572.9	145.70	0.00	7.77	0.00
5	5566.6	145.79	7.33	821.82	1068.58
10	5549.8	146.89	14.67	1686.40	2154.87
15	5524.9	146.86	22.00	2571.20	3230.92
20	5493.7	146.13	29.33	3464.40	4286.11

It is seen from the values above that with winds 15 mph and above, the potential for wind drift could carry the rocket outside of the required 2500 ft radius limit. Typically, rockets are not launched at 15 mph, because that would break the drift limit.

It can be seen from the table that the OpenRocket simulations predict lower drift distances than the hand calculations; this is due to a margin of error set manually within OpenRocket.

The predicted altitudes during various wind speeds are all above the goal apogee of 5,280 ft AGL, but the rocket design being simulated is before paint, ballast, and any variable weight changes during construction of the full scale.

Figure 3-50, below, shows a plot of the drift distances it feet.

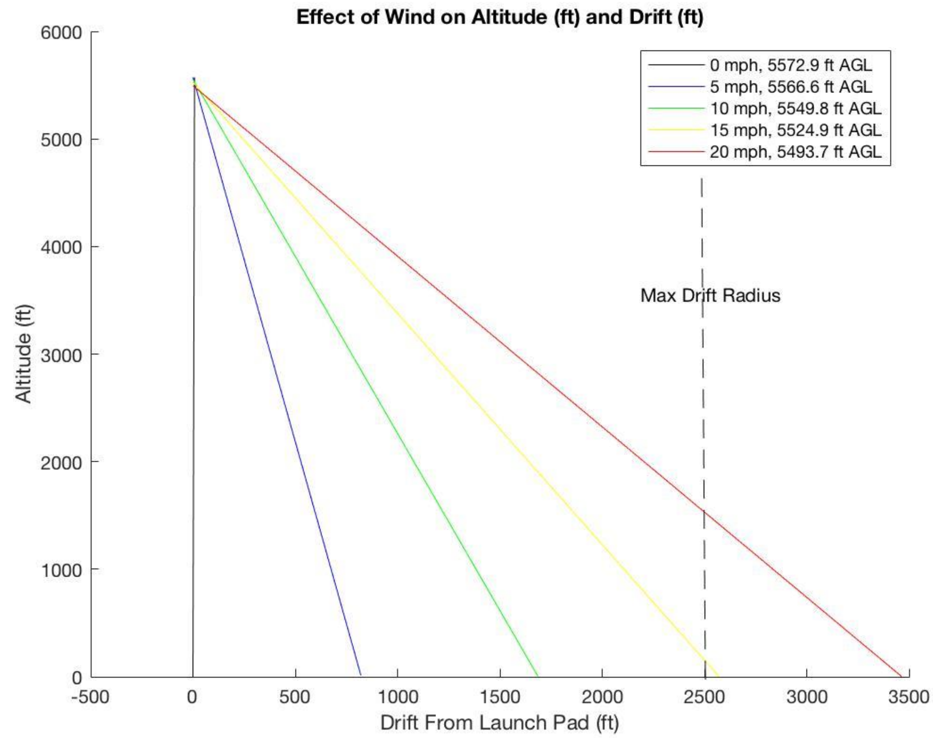


Figure 3-50: Full-scale Effect of Wind on Altitude and Drift from Launchpad

These values correspond to the predicted OpenRocket altitude and drift values. Figure 3-51, below, shows a plot of the surrounding areas for lateral drift distances.

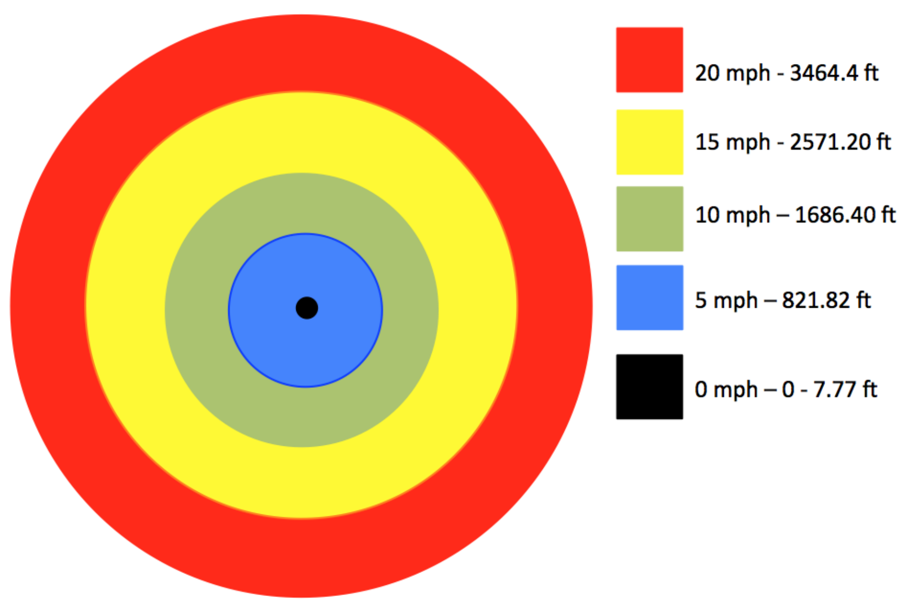


Figure 3-51: Radial View of Wind Drift Lateral Distances from Launch Pad

3.6.5 Launch Rail Exit Velocity

The rail exit velocity requirement 2.17 of the 2018 NASA SLI Handbook sets a lower limit of 52 ft/s for the exit speed of the rocket from the launch rail. In order to determine the launch speed of the full-scale design, the locations of rail buttons and the size of the launch rail were determined.

Equation 8, shown below, is the equation for launch rail exit velocity:

$$V_{exit} = \sqrt{\frac{2L(T - W\sin\theta)}{m}} \tag{25}$$

where L is the distance the top rail button travels before leaving the rail (ft) and is equivalent to:

$$L(in.) = L_{rail}(in.) - d_{rb_{aft}}(in.) \tag{26}$$

which is equivalent to the difference in the length of the rail and the distance between the top most rail button and the aft most point of the rocket. For the full-scale:

$$L = 8\,ft - 2.35\,ft = 5.65\,ft$$

and T is the average thrust (lbf) over the first instant of flight, taken from Open Rocket simulation. The predicted values for both subscale and full-scale motors are provided below:

$$T = 2243\,N = 504.25\,lbf$$

W and M are the weight and mass of the launch vehicle, listed below**:**

$$W = 47lbf$$

$$m = 1.46\,slugs$$

θ is the angle of the launch rail from horizontal, which is typically 85 degrees from horizontal for launch purposes for both full-scale and subscale.

$$\theta = 85°$$

With this information, Matlab is used to analyze the rail exit velocity, and is plugged in below for clarification. The MATLAB code is included in Section 8.

$$V_{exit} = \sqrt{\frac{2 * (5.65\,ft) * (504.25\,lbf - (47\,lbf)\sin(85°))}{1.46\,slugs}}$$

$$V_{exit} = 59.46\,\frac{ft}{s}\ and\ 59.46\,\frac{ft}{s} \geq 52\frac{ft}{s}$$

3.7 Mission Performance Predictions for Subscale Launch Vehicle

3.7.1 Stability Margin

As the motor burns and ejects mass, the CG will translate forward in the rocket, thereby increasing the stability margin throughout the entire powered phase of flight. However, a large stability margin is not required as it may cause the rocket to pitch off-course due to an external perturbation such as a gust of wind. Once perturbed, the rocket flight will become unpredictable which is not ideal in any scenario. To increase spectator safety and confirm that the rocket will perform as expected, the team investigated how the stability margin changed during flight. Table 3-14 below, shows the results of this investigation, which shows the rocket stability margin as a function of rocket mass and CG location for the ascent profile.

Table 3-14: Stability Margin at Various Events During Flight

Wind Condition (mph)	Event	Stability Margin (calibers)
0	Rail Clearance	2.0487
	Burnout	2.3254
	Apogee	5.3429
5	Rail Clearance	1.4322
	Burnout	2.3164
	Apogee	2.6721
10	Rail Clearance	1.0538
	Burnout	2.2996
	Apogee	1.8057
15	Rail Clearance	0.75035
	Burnout	2.2145
	Apogee	0.47859
20	Rail Clearance	0.55567
	Burnout	2.1537
	Apogee	0.79493

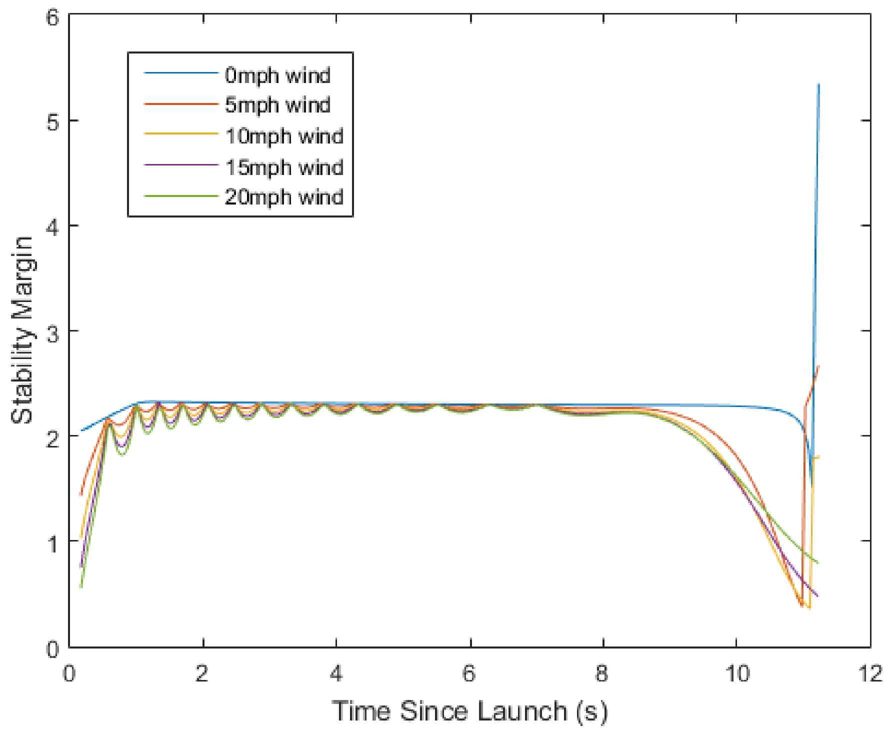


Figure 3-52: Time vs. Stability Margin

Similar to the full-scale relation, stability margin increases in an oscillatory motion for one quarter of the time to apogee. After engine burnout, the margin remains relatively constant until before apogee where it drops dramatically then rises at apogee.

3.7.2 Simulated Motor Thrust Curve

The subscale motor thrust curve for the AeroTech I435T is shown below in Figure 3-53.

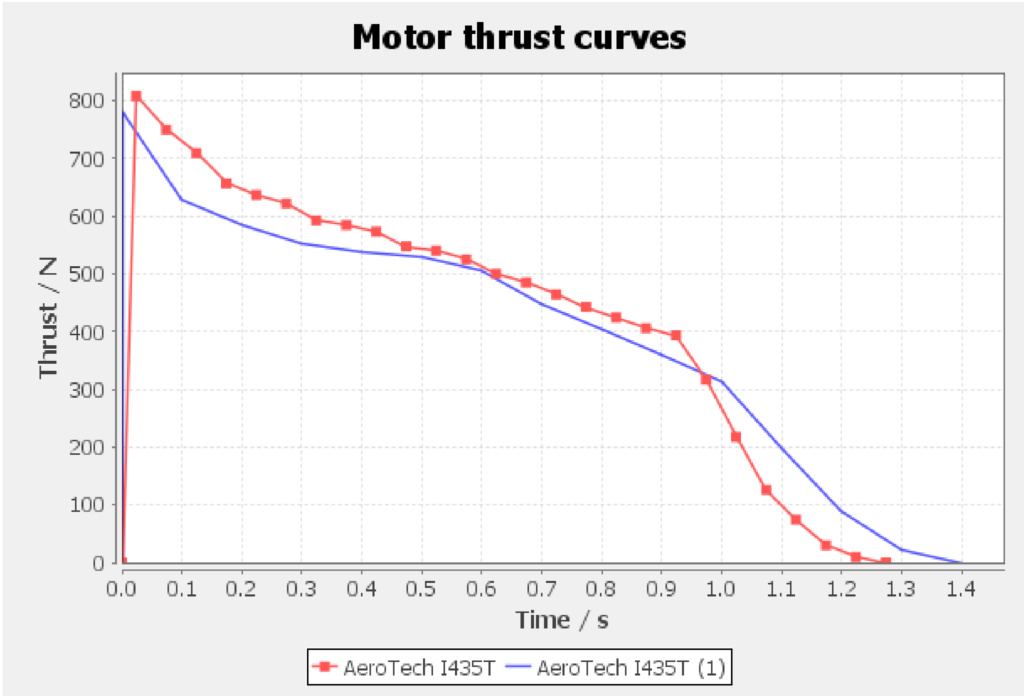


Figure 3-53: AeroTech I435T Motor Thrust Curve

The performance of the thrust curve throughout the flight resembles the flight curve of the AeroTech L2200G, the selected motor for the full-scale, in the rough pattern. This is due to the White Lightning propellant used that initially exerts max thrust of about 800 N and the most energy right after takeoff, as the propellant burns the hottest. The burn time for the motor is 1.4 seconds.

3.7.3 Launch Rail Exit Velocity (Team-Derived Requirement)

While there were no specified rail exit velocity requirements for the subscale design, the same standard minimum limit for rail exit velocity of 52 ft/s was used as a team-derived requirement. The requirement was set to familiarize the team with the limitations of the rail exit, rail sizing and to become familiar with the calculations. The same steps from full-scale calculations are repeated below for the subscale design.

$$V_{exit} = \sqrt{\frac{2L(T - W\sin\theta)}{m}} \tag{27}$$

where L is the distance the top rail button travels before leaving the rail (ft):

$$L = L_{rail}\,(in.) - d_{rb_{aft}}(in.) \tag{28}$$

and is equivalent to the difference in the length of the rail and the distance between the top most rail button and the aft most point of the rocket. For the subscale,

$$L = 6\ ft - 1.08\ ft = 4.92\ ft$$

and T is the average thrust (lbf) over the first instant of flight, taken from Open Rocket simulation. The predicted values for both subscale and full-scale motors are provided below.

$$T = 482\ N = 108.36\ lbf$$

W and M are the weight and mass of the launch vehicle, listed below.

$$W = 9.69\ lbf$$

$$m = 0.30\ slugs$$

θ is the angle of the launch rail from horizontal, which is typically 85 degrees for launch purposes for both full-scale and subscale.

$$\theta = 85°$$

With this information, Matlab was used to analyze the rail exit velocity, and is plugged in below for clarification. The MATLAB code is included in Section 8.

$$V_{exit} = \sqrt{\frac{2 * (4.92\ ft) * (108.36\ lbf - (9.69\ lbf)\sin(85°))}{0.30\ slugs}}$$

$$V_{exit} = 56.78\frac{ft}{s}\ and\ 56.78\frac{ft}{s} \geq 52\frac{ft}{s}$$

4. Safety

4.1 Compliance to Handbook Requirements

Handbook Item	Description of Requirement	Compliance Action
5.1	Each team will use a launch and safety checklist. The final checklists will be included in the FRR report and used during the Launch Readiness Review (LRR) and any launch day operations.	The team will follow to provided TRA Pre-Flight Checklist Outline
5.2	Each team must identify a student safety officer who will be responsible for all items in Section 5.3.	Safety Officer: Erik Benson
5.3	The role and responsibilities of each safety officer will include, but are not limited to, handbook items 5.3.1-5.3.4, below.	See Below
5.3.1	Monitor team activities with an emphasis on safety during: Design of vehicle and payload Construction of vehicle and payload Assembly of vehicle and payload Ground testing of vehicle and payload Sub-scale launch test(s) Full-scale launch test(s) Launch day activities Recovery activities Educational engagement activities	All members have been briefed on proper lab conduct and all relevant safety equipment is made available to include storage of energetics.
5.3.2	Implement procedures developed by the team for construction, assembly, launch, and recovery activities.	The safety officer and the team will bear the responsibility of being up to date on all risk associated with materials.
5.3.3	Manage and maintain current revisions of the team's hazard analyses, failure modes analyses, procedures, and MSDS/chemical inventory data.	The safety officer will be the first point of contact to record and distribute changes to hazard, risk and failure documentation.
5.3.4	Assist in the writing and development of the team's hazard analyses, failure modes analyses, and procedures.	For finalization of documentation the safety officer will be responsible for the accuracy and quality of relevant information.

Handbook Item	Description of Requirement	Compliance Action
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.	The safety officer will direct the club to fully cooperate with the directions and wishes of the the NAR Staff and RSO's at all junctures of the competition.
5.5	Teams will abide by all rules set forth by the FAA.	Through all other outlined safety measures the team will abide by the FAA rules.

4.2 Personnel Hazard Analysis

Safety Concern	Mitigation	Confidence
Assembly and Handling of Motor	Minimal individuals will be allowed contact with the motor. The primary handlers of the motor would be the safety officer and club advisors. In unforeseen circumstances another club officer may be necessary but unlikely. The directions of the manufacturer will be followed strictly	The motor will be in optimal conditions to perform to specifications due to the proper care given by the adherence to the instructions of the manufacturer in addition to guidance given by HPRC club advisors.
Transport and Handling of the Launch Vehicle	The launch will consistently supported by multiple points of contact in effort to reduce the risk of damage to the body or internal components. This will be paid special attention to when being transported to and from road vehicles.	With these concerns and wishes expressed to all members, an effective pool of members will be available at any moment to be called upon or take initiative at any time that a need may arise.

Safety Concern	Mitigation	Confidence
Launch Vicinity	Careful management of all personnel and bystanders present at the launch site and enforcing the rule, specifically the safe distance, will be necessary at all launches.	These actions will promote a safe launch and allow officials to maintain focus on other launch related tasks.
Location	The team will only launch at NAR/TRA approved launch sites. The team plans to launch its subscale and full-scale rockets at the Bayboro, NC launch site which meets the minimum range requirements described. The final, competition launch will occur at another NAR/TRA approved launch site near Huntsville, AL.	This will ensure the launches are able to be safely conducted with the expectation that all other safety protocols are followed.

4.3 Personnel Reference Material

Dewalt Power Drill DC 759 Manual
Dewalt Power Sander D26450 Manual
Delta Belt Sander 90183 Manual
Craftsman 137.216020 Scroll Saw
Delta Band Saw 28-180
GOEX Black Powder
Klean-Strip Acetone
West System 105 Epoxy Resin
West System 206 Slow Hardener
Fiberglass Fabric
Batteries
Cotton Flock
Baby Wipes
Igniters
Liquid Nails
Glass Microspheres
WD-40
Blue Tube

4.4 Environmental Hazard Analysis

Safety Concern	Hazard	Mitigation
Cloud Cover	NAR Safety Code does not allow for launches into cloudy conditions	Use weather forecasts to find a launch date with favorable conditions.
Precipitation	Precipitation usually accompanied with cloudy conditions (see above). Precipitation puts the electrical components at higher risk of being damaged.	Use weather forecasts to find a launch date with favorable conditions. Waterproof materials will be brought to shelter the rocket and its components.
High Wind	NAR Safety Code prohibits launching in wind conditions exceeding 20 miles per hour and if launched the rocket may become unrecoverable	Use weather forecasts to find a launch date with favorable conditions.
Water Features	The rocket when launched may land in a water feature and would become unrecoverable or damage the rocket.	Choose a launch site that has no or minimal water features in the feasible landing area.
Severe Cold	Cold temperatures can cause power sources to discharge more rapidly and can weaken the structural integrity of the rocket itself.	Use weather forecasts to find a launch date with favorable conditions. If the outdoor temperatures are determined to be detrimental, thermally shelter the rocket and components as long as possible.
Severe Humidity	The moisture in the air can compromise the integrity of the black powder charges as well as affecting the calibration of altimeters.	Keep components and charges stored in optimal conditions prior to the launch.
Severe UV Exposure	UV exposure can be detrimental to the integrity of adhesives used on the rocket as well as degrading the material of the parachute.	Limit the amount of direct sunlight that are experienced and periodically inspect the rocket.
Hard Landing Surface	Hard surfaces can damage the rocket upon landing	Choose an appropriate launch site that has soft surfaces for the rocket to land on

Safety Concern	Hazard	Mitigation
Trees	The rocket and parachute may become lodged into the trees making the recovery difficult or even impossible.	Read the conditions with the intention to reduce if not eliminate the risk of the rocket landing in a wooded area.
Bystanders	Bystanders on the ground could be injured by parts of the rocket or not following the directions of the RSO.	All members must abide by the safety regulations as well as imposing them upon all bystanders.

4.5 Failure Mode, Effects, and Criticality Analysis (FMECA)

See Section 9 for FMECA tables relevant to this project.

4.6 Project Risk Analysis

Risk	Likelihood	Impact	Mitigation	Mitigation Cost
Overspending	Low	Medium	Sticking to a strict budget	Fundraising may become more necessary
Delays	Medium	High	Rigid and realistic time management	Loss in flexibility for unforeseen challenge
Lack of Resources	Low	High	Clearly communicated needs for materials	Unexpected costs and subsequent delays
Improper Design	Medium	High	Cooperative review of the rockets systems on a regular basis	Need for an expensive and time consuming redesign late in the competition
Motor Failure	Low	High	Buy reputable and quality motors	An increase in cost of motors
Rover Deployment Failure	Low	High	Rigorous testing and troubleshooting	An accelerated development cycle
Power Supply Failure	Low	High	Using a large power supply	Rocket has a higher weight

5. Payload Criteria

5.1 Compliance to Handbook Requirements

Handbook Item	Description of Requirement	Compliance Action
4.1	Each team will choose one design experiment option from the following list: target detection, deployable rover, or landing coordinates via triangulation.	The team chose option 2: the deployable rover.
4.2	Additional experiments (limit of 1) are allowed, and may be flown, but they will not contribute to scoring.	The team has chosen not to perform additional experiments.
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.	The team has chosen not to perform additional experiments.
4.5	Deployable rover challenge requirements.	See below, Section 5.2.
4.5.1	Teams will design a custom rover that will deploy from the internal structure of the launch vehicle.	The rover will be custom designed and 3D printed in order to meet all the requirements.
4.5.2	At landing, the team will remotely activate a trigger to deploy the rover from the rocket.	A transceiver will receive signals from the team which will trigger mechanisms in the payload Section and on the rover itself that will deploy the rover.
4.5.3	After deployment, the rover will autonomously move at least 5 ft. (in any direction) from the launch vehicle.	The motor that is responsible for the rover's motion will remain active long enough for the rover to move 5 ft.
4.5.4	Once the rover has reached its final destination, it will deploy a set of foldable solar cell panels.	A sail system, outlined below, will be deployed by an arm controlled by a motor on top of the rover. This will be the rover's solar panel deployment method.

5.2 Selection, Design, and Rationale for Full-Scale Payload System

This Section contains the design decisions, descriptions, and justifications for the full-scale payload system.

5.2.1 Mission Statement

The payload is a custom designed rover, which is to deploy from the internal structure of the launch vehicle upon remote triggering. During flight, it is housed in the payload tube, which is rotating about a Lazy Susan bearing system. Upon landing, the payload tube self-rights, and the electric latch keeping the rover in the tube is unlocked. At the same time, the rover exits the payload tube is to autonomously drive 5 ft in any direction and deploy foldable solar cell panels upon reaching the final resting point. The solar panels will be deployed using a rotating arm with folded panels attached. As the arm rotates, the panels unfurl. The experiment is considered to be successful when the rover deploys from the launch vehicle, travels the required distance, and deploys the solar cells with minimal damage.

5.2.2 Subsystem Alternative Design

This Section discusses the differed alternative designs for each subsystem of the payload.

5.2.2.1 Structure

This subSection will discuss the payload housing, the rolling system, and payload security of the payload.

5.2.2.1(a) Housing Tube

Initially, the payload design had two concentric body tubes where the inner tube would freely roll within the outer tube. The outer tube had an outer diameter of 6 inches and 1/8 inch walls while the inner tube had an outer diameter of 5.25 inches with 1/8 inch walls. The objective of this was to allow the payload to eject from the launch vehicle during descent in order to ensure the rover would not have issues deploying. The center of mass location would keep the rover in the upright position when landing. Having two tubes would be an effective method of keeping the rover in a position to deploy during the whole flight. It also would protect the electronics from the black powder charges during separation. Having two tubes, however, adds a significant amount of weight to the payload.

Three tube materials were discussed in the planning of the payload: fiberglass, blue tube, and acrylic tube. Fiberglass is strong but heavy and expensive. It has a shear strength of 8 ksi and a density of 0.072 lb/in^3, according to SolidWorks Material Database. Blue tube is significantly cheaper than fiberglass, strong, and light. It has a density of 0.047 lb/in^3, according to SolidWorks Material Database. Blue tube, however, would require a fiberglass wrapping to protect the tubing. Acrylic plastic is strong, easy to work with, and light. It has a shear modulus of 129 ksi and a density of 0.043 lb/in^3, according to SolidWorks Material Database.

Also, it is cheaper than fiberglass but is more expensive than blue tube. Acrylic is the best option for the payload housing.

The next design for the housing tubes is to have a single tube that freely rolls within the launch vehicle. This tube used is the same as the inner tube in the first design. Removing the outer tube from the first design opens up 1.6 lb of weight that can be used towards other aspects of the payload which is better than using two tubes.

5.2.2.1(b) Rolling System

There have been three designs for the rolling system. The first one had free moving bearings in between the two tubes of the initial payload design, the second was a 3D printed track that housed the bearings, and the third was a Lazy Susan bearing system. All three were designed for the two tube system.

The free-moving bearing system is simple and only requires bearings. However, it does require a large number of bearings to fill the entire payload. It also requires a cap to be created to contain the bearings in between the two tubes. Since the two tubes are designed to have a .25 inch gap between the walls, this would require having ball bearings that size. The large number of large ball bearings would produce a significant amount of weight.

The team would be able to create the 3D printed tracks with a 3D printer. It would require smaller and significantly less ball bearings. Two sets of tracks would be needed to be constructed for the forward and aft ends of the payload. This design, however, will be difficult to install in between the tubes and would not be removable after the installation. Also, there would not be a lot of material used in this design which is potentially weak and could fracture.

The Lazy Susan system could be purchased from a manufacturer, which means that it will be strong and durable. This system can be easily attached to allow the tube to spin freely. The require number of ball bearings purchased for the payload is reduced because the bearings will be included in the Lazy Susan. This option will be heavier than 3D printed bearing tracks. The Lazy Susan bearing is the best option for ensuring the payload tube spins.

5.2.2.2 Rover Security

5.2.2.2(a) Rover Platform

The initial design for the rover platform was a continuous extruded face with curved edges that are concentric with the payload tube. The top edge of the platform is 1.25 inches from the horizontal axis. This design allows the platform to sit easily in the tube. The thickness of the tube is

.25 inches which allows it to be strong enough to support the while not requiring a large amount of material that would increase the payload weight.

The next design idea for the platform was to make the front face an inch tall and an inch thick. The rest of the platform is .25 inches thick and continues for the entire length of the payload. This design allows for a door to be mounted to the platform. The added material to the from face moves the center of gravity of the payload down while not adding a significant amount of weight to the entire payload.

The final design is similar to the initial design but instead of the front face only being an inch thick, it fills the entire profile of the tube between 1.25 inches below the horizontal to the inner wall edge. This design allows the door to mount to the platform and reduces the size of the door. This design is the optimal option because it minimizes the weight contributed to the payload and protects the electronics the platform houses on its bottom side.

5.2.2.2(b) Rover Latch

Initially, the rover was going to be secured by hooks attached to the underside of the rover. The hooks were to be controlled by the servo in a push-pull fashion. This design would be sufficient to prevent the rover from moving vertically off the platform and horizontally towards the door but there is no simple design that would prevent it from moving horizontally in the opposite direction and functioning incorrectly.

The next design for the latch is securing the rover with a Southco R4-EM 4&6 Series Electronic Rotary Latch. Commonly used in screen doors, this electronic latch hooks onto a screw on the back of the rover (see Section 0). This latch will lock the rover in place with a hook and will prevent it from moving in the lateral direction. However, this latch will not prevent the rover from moving vertically. When the launch vehicle lands, this latch will be remotely triggered to release the rover to allow the rover to travel the require distance. The latch will be connected to a servo stored in the payload. Despite this, it is the best rover security design as well as rover deployment.

5.2.2.2(c) Braces

Braces were initially designed to prevent the platform from being displaced from the tube's inner edge. This design consisted of two runners that had a profile of a curved edge that matched the tube's inner edge's curve and a half inch horizontal edge that created a 90° angle with a vertical edge. The horizontal edge was in contact with the platform. These runners were extruded the length of the platform. An epoxy resin is used to attach the braces to the tube. This design is sufficient to prevent the platform from moving because the platform was wedged in between the tube and the braces.

The other brace design is identical to the platform braces but instead of mounting them on the platform, they would be in contact with the rover's treads. This design prevents the rover and the platform from moving in the vertical direction during the flight and landing. This is the best design because it keeps the rover and platform wedged in place.

5.2.2.3 Door System

A door system was required to be designed in order to protect the payload and electronics from harm from the black powder charges or from the ground when landing. The initial design for the door was a two piece door that matched the profile of the tube with the platform. The purpose of the two pieces was to allow the door to open with hinges. The top piece of the door was the same shape as the tube above the platform's top edge, while the bottom piece was the shape of the remaining area of the tube. The bottom piece was to be secured to the rover platform with screws to ensure the door would spin along with the rest of the payload system.

An alternative version to this design does not involve the initial design's lower piece. Instead, the door fits inside the tube and rests on the platform. It is attached to the platform with brass hinges allowing the door to swing open when released. The door is secured closed during the flight by a .25 inch diameter x 1 inch long stainless steel pin. This latch system attaches the door to the tube by a brace with a hole secured to the top of the interior of the tube. A wire attached to the pin is removed by a push-pull servo attached underneath the platform to open the door. The wire will be guided around the interior of the tube to minimize difficulty in pulling the pin. This pin should be sufficient in keeping the door secure because stainless steel has a shear modulus of 12500 ksi, according to SolidWorks Material Database, making it very strong and rigid.

The rover will push open the door as it exits, turning the door into a ramp. The door is 3.75 inches tall and the distance from the bottom of the door to the ground is 2.46 inches. The Pythagorean Theorem finds the distance the door extends from the platform 2.83 inches.

The door does come in contact with the inside edge of the bulkhead before touching the interior of the body tube, however. This produces a gap of 0.078 inches between the edge of the door and the body tube interior. The angle of the door opens to is approximately 50.5°. Even though the angle is steep and there is a gap when the door is fully open this will not be an issue. Since the rover is on treads, it will have no trouble exiting over the gap. The treads will always remain in contact with the payload, whether on the platform or the door, until it makes contact with the body tube.

In the case this design does not work as intended, the team will continue to study different door designs that will allow the rover to accomplish its mission.

5.2.3 Current Payload Design

This subSection describes the current design for the payload system.

5.2.3.1 Structure

The chosen payload housing system is a single tube that freely spins inside the rocket body with the use of both purchased and created Lazy Susan bearing systems. These designs were chosen in order to minimize the weight of the payload and to ensure the payload is able to spin freely during the flight. Figure 5-1 and Figure 5-2are 3D models showing the tube and bearing mountings in an isometric view and full Sectioned side view, respectively.

5.2.4 Current Payload Design

This subSection describes the current design for the payload system.

5.2.4.1 Structure

The chosen payload housing system is a single tube that freely spins inside the rocket body with the use of both purchased and created Lazy Susan bearing systems. These designs were chosen in order to minimize the weight of the payload and to ensure the payload is able to spin freely during the flight. Figure 5-1 and Figure 5-2 are 3D models showing the tube and bearing mountings in an isometric view and full Sectioned side view, respectively.

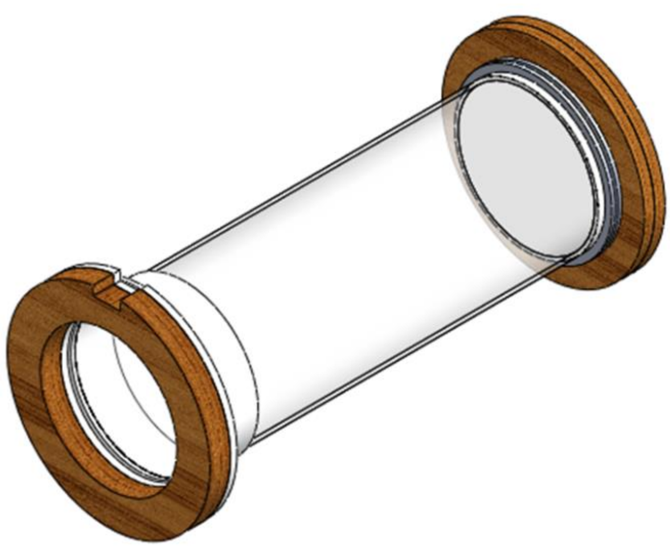


Figure 5-1: Isometric view of the payload housing and bearing system

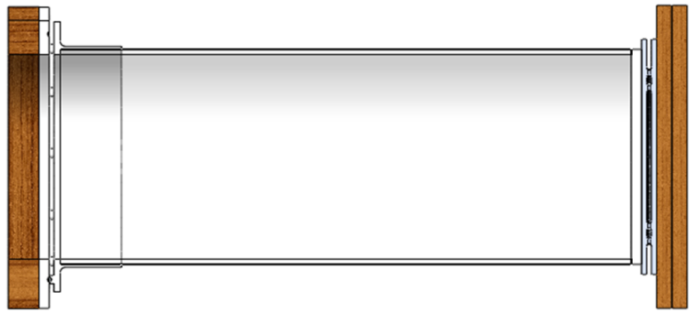


Figure 5-2: Full Sectioned side view of the payload housing and bearing system

5.2.4.2 Rover Security

The selected designs for the rover security are the most recent designs. The platform with the largest frontal face allows the door to be secured properly and allows the door to be attached and protect the electronics. The Southco R4-EM 4&6 Series Electronic Rotary Latch is the most secure locking mechanism for the rover in the lateral direction. The chosen position for the braces was to secure them above the treads. This was chosen because it restrains the rover and the platform and prevents them from moving vertically. Figure 5-3 and Figure 5-4 are 3D models showing the rover security subsystem in an isometric view and full Section side view, respectively.

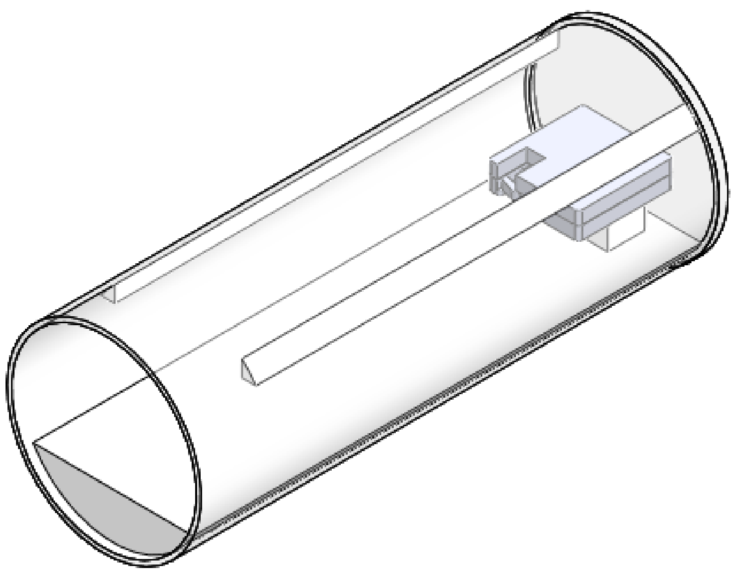


Figure 5-3: Isometric View Showing the Rover Security System

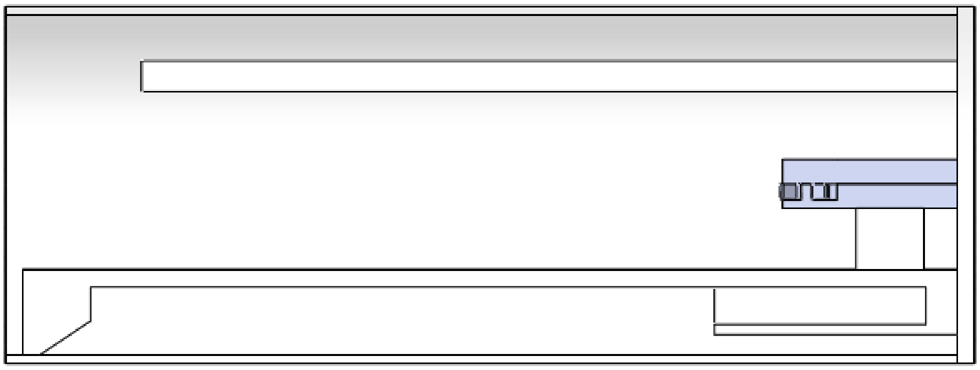


Figure 5-4: Full Sectioned Side View of The Rover Security System

5.2.4.3 Door System

The door system that is utilized in this rocket is the latest design that rests on top of the platform and is hinged to the platform. This design will protect the rover and electronics during the flight and will act as ramp for the rover to exit on. Despite the steep angle that is predicted, the rover will not have any difficulty exiting because of its treads. The metal pin and the 3D printed latch should be adequate to keep the door closed during all flight stages. Figure 5-5, Figure 5-6, and Figure 5-7 are 3D models showing an isometric view of the door system on the payload, a full Sectioned side view of the design, and a full Sectioned side view with the door open.

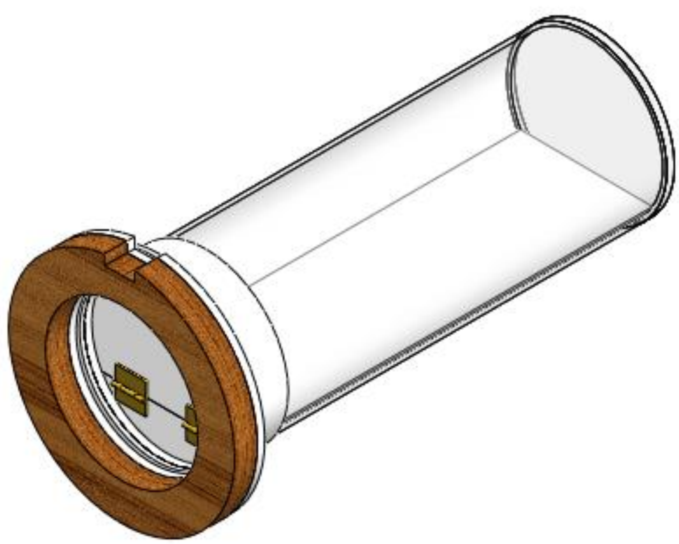


Figure 5-5: Isometric View of the Payload Showing the Door System

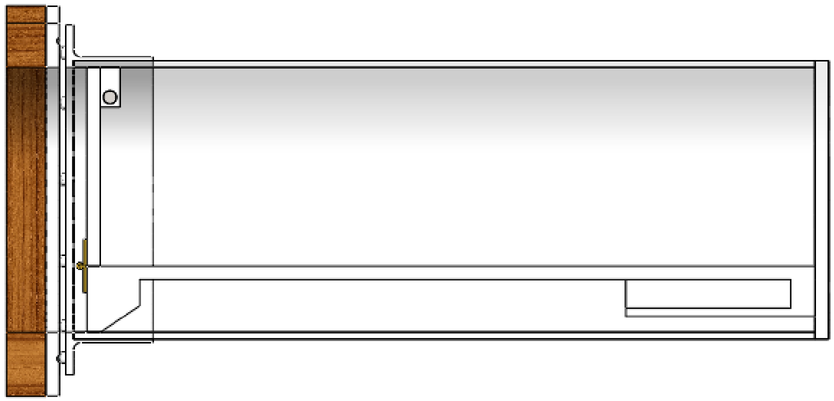


Figure 5-6: Full Sectioned Side View of the Payload Showing the Door System

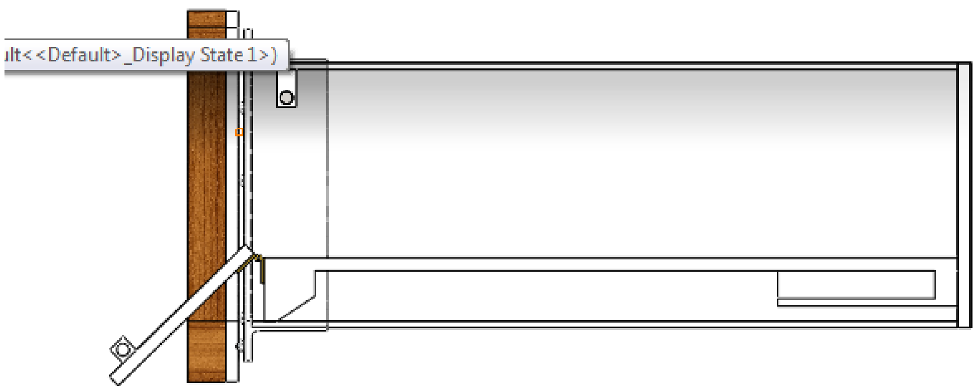


Figure 5-7: Full Sectioned Side View of the Payload Showing the Designed Opening of the Door

5.2.4.4 Full Design

The weight of the payload without the bulkheads and centering ring is 5.585 lb and is 6.730 lb. Table 5-1 below, shows the mass breakdown of each component.

Table 5-1: Weight of Each Payload Component

Part	Weight (lb)
Tube	1.222
Forward Lazy Susan	0.360
Aft Lazy Susan	0.696
Rover Platform	0.768
Tube Mounting Piece	0.199
Door	0.148
Door Pin	0.014
Door Latch	0.003
Door Hinges x2	0.046
Rover Braces x2	0.080
Rover Latch	0.152
Rover Latch Mount	0.067
Battery	0.158
Servo	0.017
Arduino	0.002
Rover	1.667
Total	**5.585**

The desired weight to match the simulation results is 7 lb. This means that there is approximately .25 lb that can be used for the epoxy resin to attach the required parts. Figure 5-8 and Figure 5-9are 3D models showing the entire designed payload, including the rover, in an isometric view and full Sectioned side view, respectively, and Figure 5-10 illustrates the designed rover's fit in the tube in an offset front view.

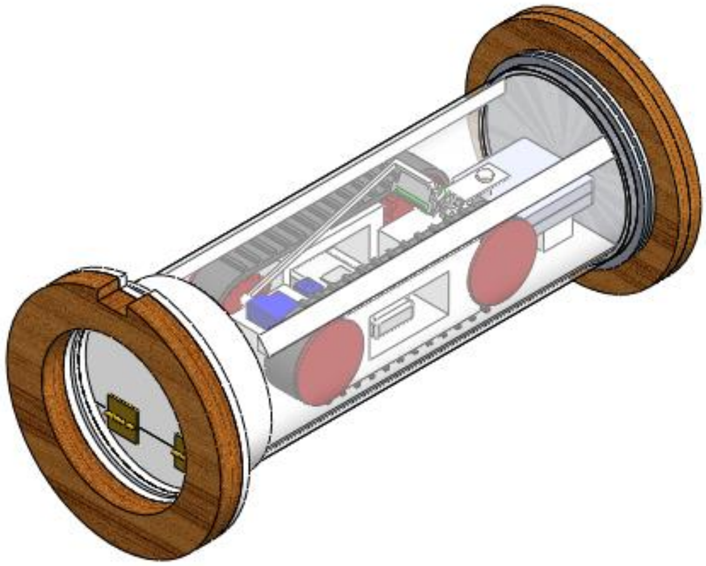


Figure 5-8: Isometric View of the Entire Payload

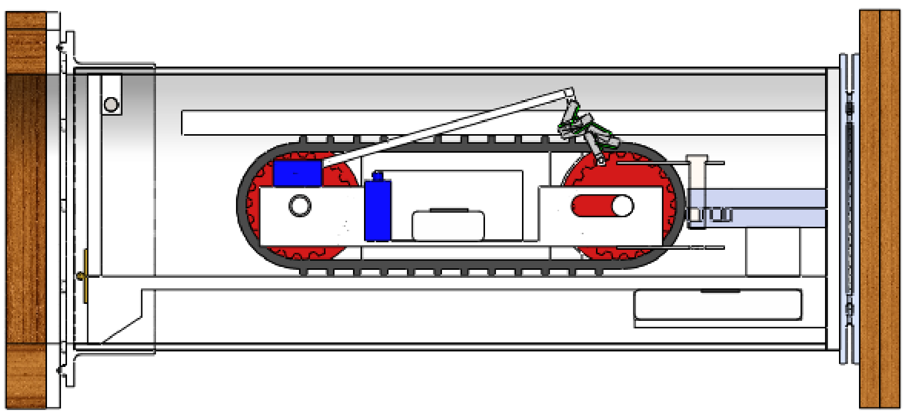


Figure 5-9: Full Sectioned Side View of the Payload

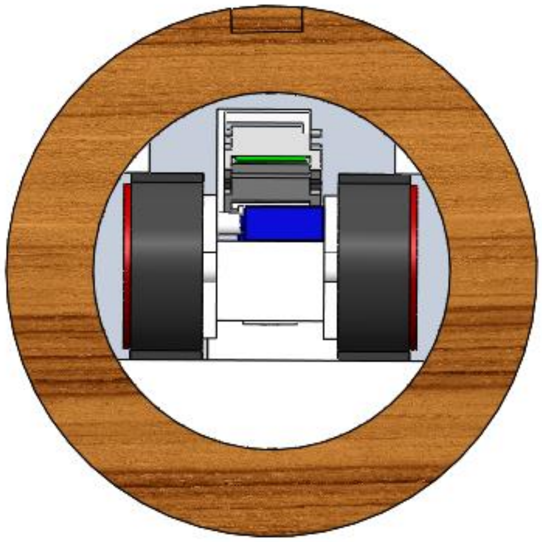


Figure 5-10: Front View of Showing the Rover's Fit in the Payload

The center of gravity in of the payload rests below the lateral axis. This will cause the rover to spin and settle in the optimal exiting position regardless of the rocket's landing position. Figure 5-11illustrates the center of gravity's position in a full Sectioned side view.

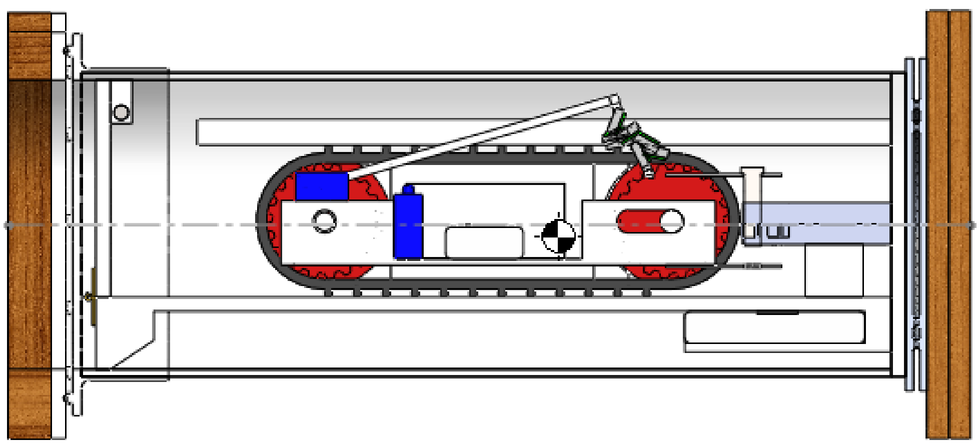


Figure 5-11: Position of the Center of Gravity With Respect to the Lateral Axis

It can be seen in Figure 1.11 that the center of gravity is just below the lateral axis. This means in the case the payload is spinning during the flight, flight stability will not be compromised.

5.2.4.5 Interfaces between Payload and Launch Vehicle

The payload housing is attached to the rocket body by the centering ring on the forward end and the bulkhead on the aft end of the payload.

5.3 Selection, Design, and Rationale for Subscale Payload System

This Section describes the selection process, the design, and the rationale for the payload system for the subscale launch vehicle.

5.3.1 Mission Statement and Success Criteria

The objective of the subscale payload is to replicate the full-scale payload's design at a scaled weight for data collection during the subscale rocket launch. This payload will collect data on the angular momentum and the forces acting upon the payload during its flight. Any data collected during the launch that is usable in the full-scale payload design will deem this launch a success.

5.3.2 Subscale Payload Design

This subsection describes the design of the subscale payload.

5.3.2.1 Structures

The subscale payload is modelled similarly to the full scale payload. The payload tube is free rolling during the flight. The electronics it houses will record the angular acceleration the payload experiences during flight. The rolling is caused by a 3D printed Lazy Susan-style system on the aft end and a cap that houses 3mm ball bearing attached to the forward end. These systems are 3D printed because they can be produced cheaply and ABS plastic is light and strong with a shear modulus of 46 ksi and a density of 0.037 lb/in^3, according to the SolidWorks Material Database. The aft end rolling system is secured to the bulkhead which is attached to the launch vehicle.

The tube is going to be acrylic plastic because it is strong and light. It has a shear modulus 129 ksi and a density of 0.043 lb/in^3, according to the SolidWorks Material Database.

5.3.2.2 Security

The electronics in the payload will be contained to a 3D printed sled that is concentric to the payload tube. This will allow the payload to spin freely during the launch to collect the necessary data. This sled is secured to the tube by two braces that will be mounted to the tube. The sled will be able to slide into the tube and will be secured once the end cap is mounted.

To add extra support, two steel threaded rods will extend the entire length of the payload from the forward end bearing cap, through the avionics sled in the aft end. To prevent any components from sliding, nuts will be fastened on each side of the bearing cap, the aft end bearing cap and bulkhead, and the avionics sled.

5.3.2.3 Full Design

To fit the scaling of the subscale, the payload must weigh 3 lb. This is in order to keep the stability margin the same as the full scale. The designed payload in its current state is 1.945 lb. Table 1.1, below, shows the mass breakdown of each component.

Table 5-2: Weight of Each Payload Component

Part	Weight (lb)
Tube	0.291
Forward Cap	0.117
Aft Lazy Susan	0.086
Payload Sled w/ Electronics	0.772
Avionics Sled w/ Electronics	0.309
Bulkhead	0.073
Threaded Rods	0.297
Total	**1.945**

This means the team must come up with 1 lb of weight to be added in order to get accurate results from the experiments conducted with the subscale rocket.

Figures 1.12 and 1.13 are 3D models showing the entire designed subscale payload in an isometric view and full-Sectioned side view, respectively.

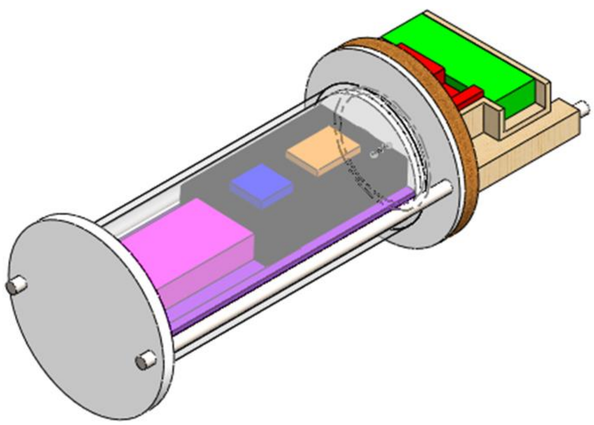


Figure 5-12: Isometric View of the Designed Subscale Payload

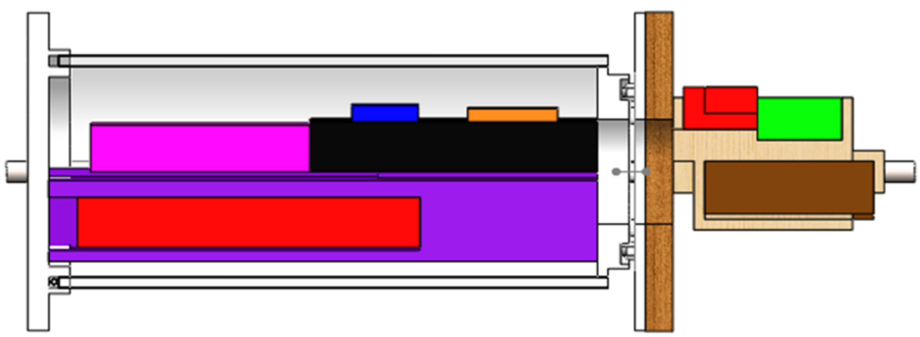


Figure 5-13: Full Sectioned Side View of the Designed Subscale Payload

The center of gravity in the subscale payload is below the longitudinal axis, similarly to the designed full scale payload. After the subscale rocket lands, the payload is expected to come to rest at the bottom of the tube. This will mimic the full scale payload after landing. Figure 1.14 illustrates the center of gravity's position in a full Sectioned side view.

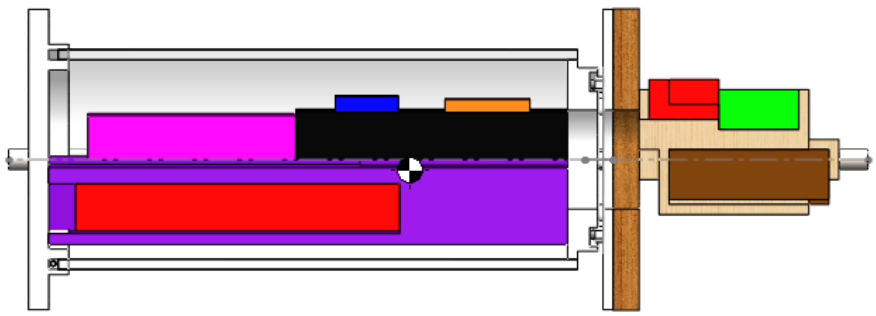


Figure 5-14: Subscale Payload's Center of Gravity's Position in Full Sectioned Side View

5.3.3 Deployable Rover Design

To meet the requirements outlined in Section 4.5 of the NASA Student Launch handbook, the team chose to design a rover with tracks. This was done in order to maximize traction, power efficiency, and volume efficiency. When compared to wheels, treaded tracks allow for greater grip when driving over rough terrain and are less susceptible to getting stuck. Tracks also require less power to perform at similar levels to wheels, and the space inside the tracks and between the powered gears can be utilized for additional storage. During flight, the rover is kept in place by an electric latch at the rear and by slots above. The latch prevents lateral motion within the payload tube, and the slots keep the rover from moving vertically. Upon landing, the self-righting payload tube will rotate into the correct orientation due to the weight imbalance. A radio signal will be simultaneously sent to the payload and rover after landing is confirmed. Transceivers on the rover and payload will trigger a chain of events that includes unlatching the rover, unlocking the door, and driving out of the compartment. The rover will utilize a single motor located at the front to rotate the front gear wheels, thus pulling the tracks around their housings to drive the rover forward. The motor will remain active long enough for the rover to travel at least 5 ft laterally from the payload landing site, and then it will stop in place. To meet the requirement of deployable solar panels, an arm will be attached to a servo on top of the rover. This method of solar panel deployment is discussed below.

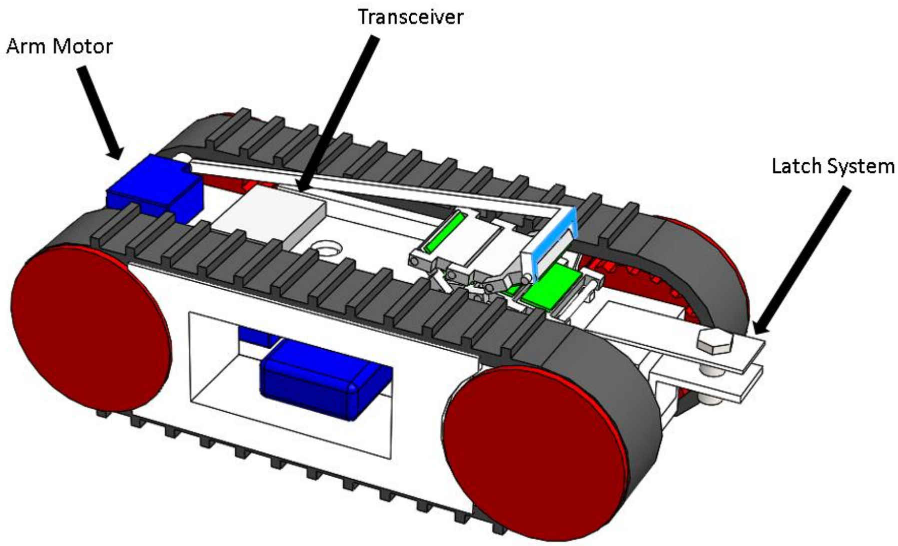


Figure 5-15: Current Rover Model with Solar Panels Stowed

The updated model was shortened due to constraints set forth by the payload bay sizing. Pictured on the right is the latch system, which will be used to secure the rover inside the payload bay. The electric latch, attached to the sled, will wrap around the screw and will unlatch when it receives a signal from the team upon landing. The stowed solar sail is also pictured here.

The rover body will consist of two 3D printed shells and two track housing pieces designed to hold all electronics. These include an Arduino Pro Mini, transceiver, battery, and two servos. The main body will be printed in two pieces to simplify the process of inserting components and minimize the likelihood of errors during printing. The geometry of the body, specifically the axle slots, makes it difficult to print in one piece, so splitting it in half reduces the possibility of printing malfunctions taking place in high risk areas. The team decided to 3D print rather than use other methods because 3D printing allows for rapid prototyping and is much simpler than traditional manufacturing methods. Changes can be made quickly and at a low cost, and they can be tested in a timely manner. The shell design will allow for quick and easy rover assembly, and the track housing pieces will protect the electrical components from dirt and debris after exiting the payload.

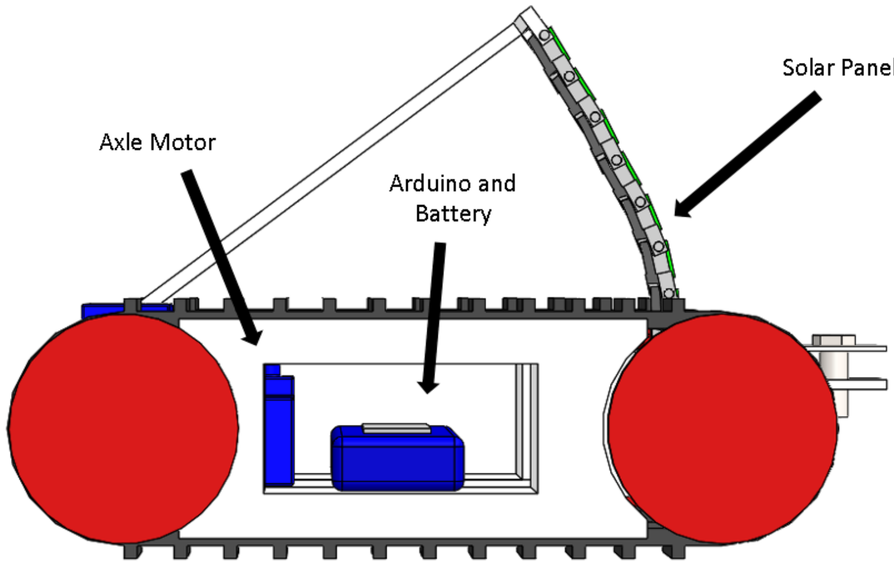


Figure 5-16: Side View of Current Rover Model – Main Body Interior

A more space efficient design was created after running into sizing issues with the initial proposal design. The main body interior allows for sensitive components such as the Arduino board to be stored away from potential hazards.

An Arduino board was chosen for its simplicity, cost, and reliability. Extensive literature is available for team members not experienced in programming to become familiar with its operation, and reference material can be used in the event that any issues arise. The rover will use two FS90 servos: one for motion and one for the solar panel arm. These were chosen because of their reliability, size, and output power. An XBee Pro 900 XSC S3B radio transceiver will be used because of its high-performance capabilities, low power consumption, and long range. A small 1000 mAh LiPo battery will be used to power all electronics onboard and will have the power capacity necessary to keep the system alive while waiting on the launch pad.

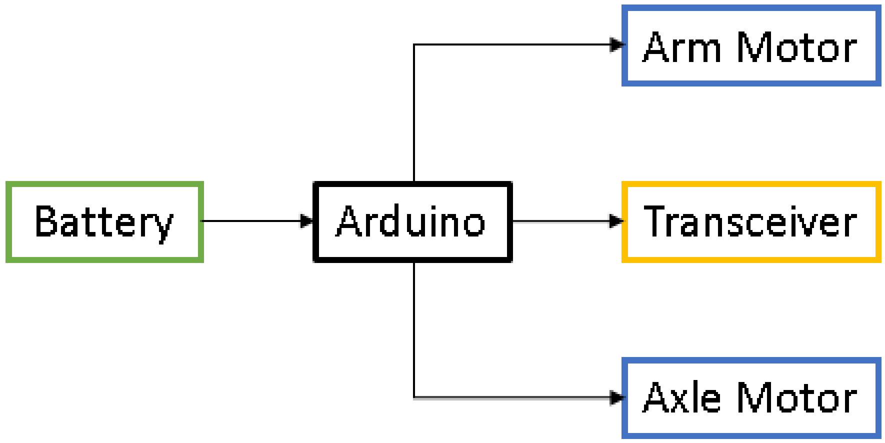


Figure 5-17: Side View of Current Rover Model – Main Body Interior

The battery provides power to the Arduino which provides power to and controls both motors and the transceiver.

Below is the estimated weight of the rover taking into account all parts that are necessary for the current rover model.

Table 5-3: Rover Parts List

Part	Material	Weight (lb)	Quantity	Total Weight (lb)
Body	ABS	0.360	1	0.360
Opening	ABS	0.160	2	0.320
Wheel	ABS	0.120	4	0.480
Tread	Rubber	0.170	2	0.340
Arm	ABS	0.010	1	0.010
Front Axle	ABS	0.020	1	0.020
Rear Axle	ABS	0.020	1	0.020
Latch Screw	Steel	0.020	1	0.020
Screw Guide Bottom	ABS	0.010	1	0.010
Screw Guide Top	ABS	0.010	1	0.010
Panasonic - BSG AM-1456CA Solar Panels	n/a	0.002	7	0.011
Accordion	Paper/Plastic	0.010	1	0.010
FEETECH FS90R	n/a	0.020	2	0.040

Part	Material	Weight (lb)	Quantity	Total Weight (lb)
Turnigy 1000mAh 2S 20C LiPoly Pack	n/a	0.181	1	0.181
Arduino Pro Mini 328	n/a	0.004	1	0.004
Misc. Wires (estimate)	n/a	0.020	1	0.020
			TOTAL (lb)	**1.856**

The mass of the accordion for solar panel deployment as well as the mass of the miscellaneous wires are dependent on decisions made once construction begins, so an estimate of their masses are given. The estimated masses of all parts with identified materials were found using the Mass Properties function in SolidWorks, and those without an identified mass were found via the manufacturers' documentation.

5.3.3.1 3D Printer Optimization Experiments

3D printing was chosen as the primary means of production because it allows for the team to rapidly construct prototypes and create intricate parts much more easily than traditional manufacturing methods. During initial printing, empty space created by the low fill percentage caused cavities to form. Because of this, the print was deformed and would have been unable to function properly if used to construct the rover. In this experiment, the team used multiple printing configurations to determine the optimal fill percentage and temperature for 3D printing rover parts.

Procedure:

Step 1. 3D print 0.5x0.5x0.5 in. cubes with triangular supports at fill percentages of 30%, 40%, and 50%. At each fill percentage, print at the following temperatures (in Celsius): 205, 210, 215, 220, and 225.

Step 2. Let prints cool down for at least one (1) hour to allow them to settle and reach their final size.

Step 3. Visually inspect the cube, looking for any deformities such as warping, cracks, bumps, or holes.

Step 4. Using a set of digital calipers, measure the length, width, and height of the cube at multiple locations to determine the average dimensions.

Step 5. The optimal fill percentage and temperature settings will be determined by selecting the print with dimensions closest to the theoretical dimensions. If two prints are equally close to the theoretical dimensions, the visual inspection will determine the optimal print settings.

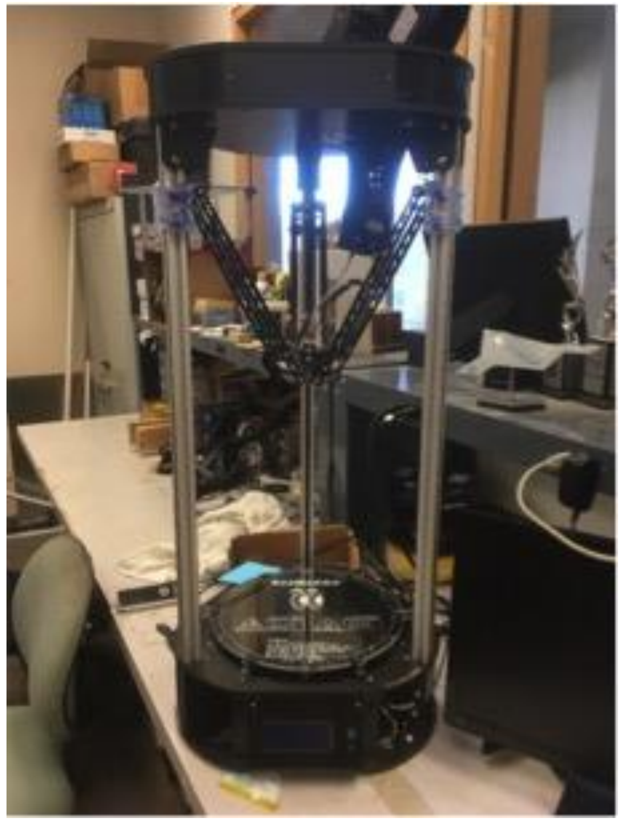


Figure 5-18: The SeeMeCNC Rostock Max V2 3-D Printer

The printer pictured above is in the High-Powered Rocketry Club lab and is the printer that will be used for testing as well as rover body printing.

At the time of writing this document, the experiment was still underway. The following table shows the current data gathered.

Table 5-4: Weight Estimate for Rover

Fill %	Extruder Temp (C)	Quality	Average Dimensions (in)
30	225	Good, warping on edges and rough patch on one side	0.496x0.499x0.497
40	225	Good, less warping than on 30%, still rough patch on one side	0.497x0.506x0.496
50	225	Okay, large warping on edge of side that has rough patch	0.497x0.500x0.502
30	220	Great, minimal warping, small rough patch on one side	0.499x0.497x0.501

Fill %	Extruder Temp (C)	Quality	Average Dimensions (in)
40	220	Great, minimal warping, small rough patch on one side	0.493x0.496x0.499

Current results indicate an increase in print quality following a decrease in temperature. There will most likely be a temperature above which the quality maximized. This is the temperature at which the team will print all the rover components.

5.3.4 Solar Panel Deployment

In order to meet the requirements of deployable solar panels, the rover must contain a folded set of solar panels that deploy after reaching its destination. The final panel deployment system is outlined below.

5.3.4.1 Sail System

In this design, the solar panels are folded in an "accordion" style, which allows for an efficient use of space. To create the accordion, the panels are mounted either on a foldable surface such as paper or plastic or attached to links that can be stowed in a folded state. One end of the accordion is attached to a rotating arm spanning the length of the rover that is controlled by a motor at the front. The other end of the accordion is linked to the body of the rover at the rear. In its stowed position, the arm is parallel with the rover, and the panels are folded and not receiving sunlight. In its deployed position, the arm is upright with the solar panels extended, resembling the sail of a ship.

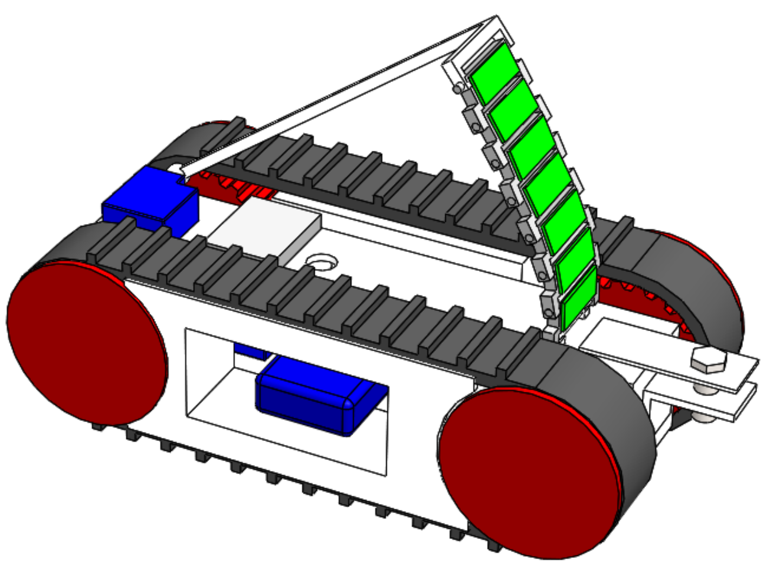


Figure 5-19: Current Rover Model with Solar Panels Deployed

When the motor activates, it will rotate the arm, thus deploying the solar panels. Although no power will be generated by the solar panels, this serves as a simulation of a real solar panel deployment.

The Panasonic - BSG AM-1456CA solar cells were selected because of their small size, light weight, and low cost. The solar cell can ship quickly and in large quantities. Its size and weight are ideal because they will not require large amounts of torque to be lifted out of the folded position, so a small servo can still be used. This keeps the weight of the entire rover low. Other options were researched such as Sundance Solar Products Small Solar Panel and the AMX3d 5V 30mA Micro Mini Power Solar Cell. However, upon narrowing the rover body, these were too wide and failed to fit in a way that the team determined was acceptable.

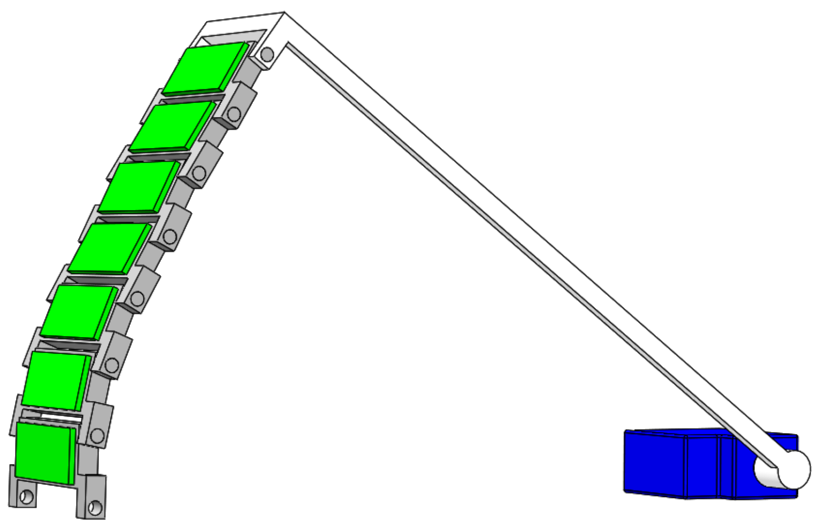


Figure 5-20: Current Rover Model with Solar Panels Deployed

The solar panels above will deploy after the rover reaches its final destination. The linkages modeled are currently a placeholder for the accordion method, which could not be modeled due to limitations in SolidWorks.

All parts from the previous mass table that are relevant to the function of the solar panel deployment are outlined below.

Table 5-5: Weight Estimate for Sail System

Part	Material	Weight (lb)	Quantity	Total Weight (lb)
Arm	ABS	0.010	1	0.010
Panasonic - BSG AM-1456CA Solar Panels	n/a	0.002	7	0.011
Accordion	Paper/Plastic	0.010	1	0.010
FEETECH FS90R	n/a	0.020	1	0.020
			TOTAL	0.051

The Sail System adds very little weight to the rover. This is due to the extremely lightweight solar panels that will be used.

In addition to the system discussed above, two other methods were considered during the design process:

5.3.4.2 Flexible Solar Panels

This was the initial design that was briefly mentioned in the proposal. The solar panels chosen were flexible panels capable of being rolled up into the tread housing to create a space efficient design. After reaching the final destination of 5 ft, a small latch controlled by a motor would activate and allow the panels to unfurl.

The team chose not to pursue this design because it presented several issues, the first of which being its adherence to the criterion in Section 4.5.4 of the Handbook, which states “foldable solar cell panels.” The team could not come to a conclusion regarding the technical definition of rolling in terms of a folding method. Secondly, positioning rolled solar panels in each tread housing would require a latch for each panel, and each latch would require a motor. More motors require more power and space, and with the rover design that was under consideration at the time, this was not the best solution, so the team decided to move on.

5.3.4.3 Rail System

A pair of rails would sit atop the rover body between the two tracks. Instead of the flexible solar panels outlined in the latch method, a set of rigid solar panels would be attached the rail system and linked to each other in an “accordion” style similar to the Sail System. A wire would be connected to the leading panel and be pulled by a motor. This action would extend the “accordion” and allow the solar panels to unfold.

The team chose not to pursue this design because of physical limitations. The added height of the rail system and the folded solar panels atop those rails would exceed the inner diameter of the payload tube. Multiple locking mechanisms would be necessary for the “accordion” to not unfold or detach during flight or rover operation. It was determined by the team that this design was too complex and that a simpler method could be developed.

5.3.5 Alternative Rover Designs

Before deciding to pursue the self-righting payload with a small, treaded rover, the team discussed other possible payload options as well as the challenges associated with each.

5.3.5.1 Quad Recovery System

The quad recovery option would have used a self-powered, autonomous quadrotor vehicle to have a powered descent to landing. The quad would have its arms folded up to stow in the payload compartment during launch. At apogee, the quad would deploy and follow a sequence of commands to unfold, check power supply, power individual rotors, and finally apply power across all rotors. This would allow for a quick system diagnostics test before the rotors are given full power. A backup parachute would be installed and ready to deploy in case the flight computer ever detected an anomaly or if the RSO triggered the event. After landing, the rover would either fold out from the side of the quad or drive out from a platform suspended below.

The team chose not to pursue this recovery option due to its complexity, high cost, and the team’s lack of experience with quadrotor design and autonomous flight. The team approached several other students, clubs, and instructors that had built similar vehicles in the past, and they were very helpful in building a parts list and understanding the full technical requirements.

5.3.5.2 Omni Wheel Rover

The omni-wheel rover design is an alternate solution to using a self-righting floor and treads. An omni-wheel is a commercial product that features 6-8 small discs mounted on the outside edges of a single larger wheel. If installed on the rover, it would allow the rover to roll freely within the payload tube and drive out after landing at any angle. The wheels are relatively cheap and would allow for a slightly larger rover since the floor underneath would be unnecessary.

The team chose not to pursue this option because it presented several issues with regards to stowing the rover during flight as well as battery and radio receiver placement. Since the rover itself is free-spinning, it would be unable to “disconnect” from a payload base like the current payload design, so all payload electronics would have to be housed in the rover itself. This would require larger onboard batteries which would increase the weight of the rover itself.

5.3.5.3 “Hamster Wheel” Rover

A Section of the body tube that could roll around on its own using internal wheels and motors, hence the “hamster wheel” name, would allow the rover to have a large internal volume to store wheels, motors, parachutes, electronics, and batteries. The internal portion of the rover would be set within the body tube so that the wheels could run against the inner wall of the body tube. As the powered, internal wheels spin against the inside of the body tube, the entire Section would roll around.

The team chose not to pursue this option because it would require a Section of the rocket body to separate completely which would require additional parachutes and GPS trackers. Since the rocket is already at its maximum weight to reach an apogee of approximately 5,280 ft AGL, it would not be possible to add all the additional components as well as a larger rover.

5.4 Selection, Design, and Rationale for Subscale Payload System

The subscale launch vehicle will contain an experimental payload that will simulate the payload flight profile for the full-scale rocket. Similar to the full-scale, the subscale payload will have a mass which is off-center and allowed to rotate freely. Onboard electronics will measure and record important parameters such as revolutions per minute, maximum loading, and maximum acceleration, among other variables. The team intends to fly the subscale rocket twice to double the amount of useful data from the subscale payload.

5.4.1 Experimental Payload Design

5.4.1.1 Subscale Payload Sled

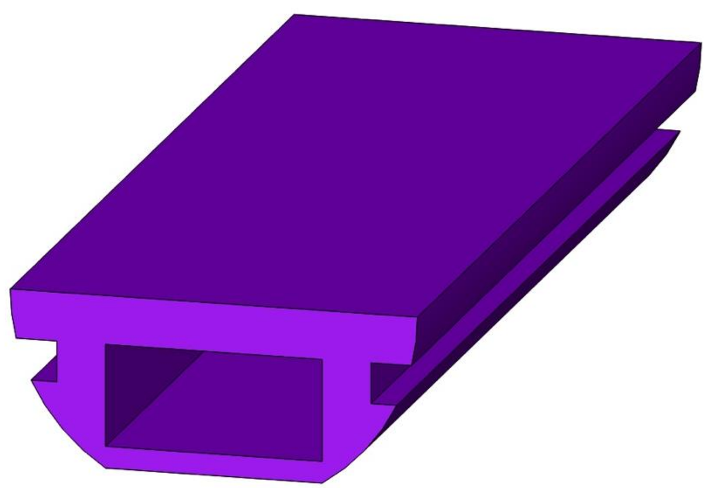


Figure 5-21: Subscale Payload Sled

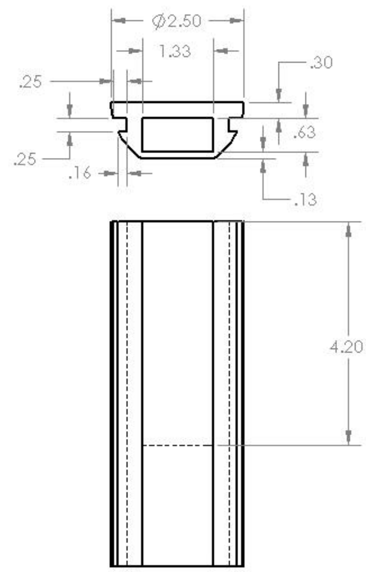


Figure 5-22: Subscale Payload Sled Dimensions

A sled will hold experiment components inside the rotating payload tube. The sled will be 3D printed out of AVC plastic using a Rostock Max V2 SeeMeCNC printer. 3D printing was chosen because it allows the team to create a curved surface that follows the contour of the tube. AVC plastic was chosen because the team has it on hand. The design of the payload sled is roughly semi cylindrical, with experiment electronics screwed into the top, and a slot for the battery in the bottom. As the heaviest component, the battery was placed as close to the edge of the rotating cylinder because the subscale experiment requires the center of gravity to be off center. Two wooden tracks will be cut and sanded to size, and epoxied to the interior of the payload tube. During assembly, the sled will slide onto the tracks, which only allow for one degree of motion. Once the forward cap of the payload is attached, the sled will be immobile inside the tube, while the tube will be able to spin freely.

5.4.2 Subscale Payload Electronics

The payload sled will hold a HobbyTiger 7.4V 2700mAh 30C Lipo Battery to power the electronics. The electronics will include a BeagleBone Black, a BNO 055 Absolute Orientation Sensor, a BMP 180 Barometric Altimeter, and a DROK Voltage Regulator. All of these components where chosen because the team has used them before, and has them on hand. The battery will power the BBB through the voltage regulator, and the altimeter and orientation sensor will be soldered to a BBB hood, which will be connected to the BBB.

Figure 5-23below shows a block diagram of the subscale payload electronics.

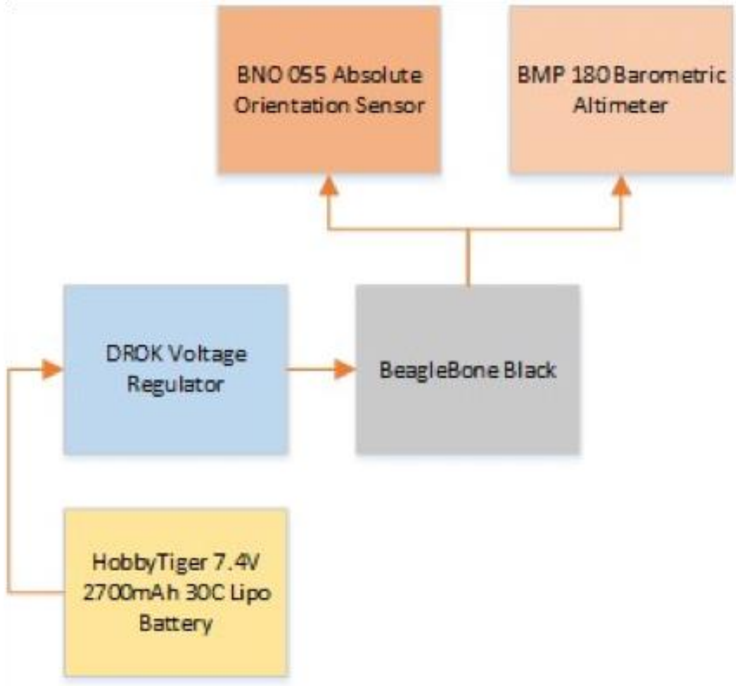


Figure 5-23: Subscale Payload Electrical System

6. Project Plan

6.1 Project Verification Plan

The team has placed the verification methods and descriptions into Table 3-1.

6.2 Team-Derived Requirements

Subteam	Description of Requirement	Compliance Action
Recovery	Subscale KE at landing will not exceed 39 ft-lbf (52% of the Full-scale Requirement).	Hand Calculations, OpenRocket Simulations and parachute sizing will allow the kinetic energy of the subscale launch vehicle to scale the expected result of the full-scale launch vehicle.
Recovery	Launch Rail Exit Velocity of Subscale will be a minimum of 52 ft/s to match that of the full-scale design.	Through hand calculations and OpenRocket simulations, the rail size and rail button locations on the subscale will match those of the full-scale to familiarize the team with rail sizing and predicted performance.
Recovery	Prioritize Hardware available to decrease budget.	In scaling the systems new technologies including reefing and deployment bags are being used to adapt available hardware to complete required tasks.
Recovery	Utilize deployment bag.	Since this is the first time utilizing deployment bags in our recovery system, several separation tests and parachute drops will be completed to demonstrate the function and accurate deployment of the main parachute.
Structures	Minimize cost of materials by using leftover or currently available materials when possible.	Buckling experiment will be conducted using leftover Blue Tube samples from previous projects.
Rover	Create a completely custom body with no premanufactured parts.	The body, axles, and wheels will be 3D printed to ensure the rover can perform as intended.

Rover	Minimize moving parts in order to reduce the number of failure points.	Only two motors will be used: one for motion and one for the solar sail system arm.

6.3 Outreach

Each outreach event is planned different to correspond with the STEM topics requested by the outreach coordinator or contact. Most outreach events include a presentation on the club, NASA SL details, and a topic in rocket design, STEM, general aerodynamics, or all three. After the presentation, there is a hands-on demo that varies according to the specific event. Most hands-on demos teach participants to build and fly their own water bottle rockets. After each flight, participants are encouraged to make changes to their design to compare the effects of adding fins or nosecones, and how the level of water “propellant” affects the flight. This exposes students (and often parents) to the engineering design cycle that is integral to the field. For the current school year, upcoming demos will include more small experiments to teach participants about other topics including Newton’s Laws of Physics, thermodynamics, and dynamics.

6.3.1 Previous Outreach Events

Figure 6-1 Tacho Lycos at NASA Langley Research Center Open House

6.3.1.1 NASA Langley Research Center Open House

Members of NC State University's (Tacho Lycos) team went to NASA Langley Research Center to represent North Carolina Space Grant as part of the NASA Langley Research Center's Open House celebrating their 100-year anniversary. Members of the club set up a table with rockets from previous years to answer questions that participants may have. In addition, the club had a small hands-on activity in which participants could make a straw rocket and learn about Newton's Laws through launching the straw rocket.

6.3.2 Planned Outreach

6.3.2.1 2017 JY Joyner Science Go Round

The team plans to attend Science Go Round, a STEM event hosted by JY Joyner Elementary. This event will be set up like classes with groups of students rotating through different classrooms to see presentations. Other science and engineering organizations will be present at the Science Go Round talking about their involvement with STEM.

6.3.2.2 2018 Astronomy Days

The team will continue its support of Tripoli Rocketry Association to help at their booth for Astronomy Days hosted by the North Carolina Museum of Natural Science. Team members will talk to the general public about high-powered rockets and our participation in NASA SL using past rockets as examples of completed projects.

6.3.2.3 2018 Weatherstone STEM Expo

The team will be manning a booth for the 2018 Weatherstone Elementary School STEM Expo where members of the club will help participants learn about Newton's Laws of Motion and build water bottle rockets. In addition, the team will be discussing our involvement in NASA Student Launch and this year's design.

6.4 Project Budget

Table 6-1, below, lists all the projected expenses for the team to compete in the 2017-18 NASA SL challenge.

Table 6-1: HPRC Projected Expenses for 2017-18 School Year

	Item	Quantity	Price per Unit	Item Total
Subscale Rocket	4” Phenolic Airframe Tube	2	$20.99	$41.98
	4" Phenolic Tube Coupler	1	$4.99	$4.99
	4" Plastic Nose Cone Standard	1	$23.95	$23.95
	Aircraft Spruce Domestic Birch Plywood 1/4"x4x4	2	$56.38	$112.76
	¼” Threaded Rods	2	$4.54	$70.00
	Rail Buttons	1	$7.35	$7.35
	U-Bolts	4	$2.96	$11.84
	I435T Motor	2	$55.99	$111.98
	Motor Casing (subscale)	1	$240.00	$240.00
	Motor Retainer	1	$33.50	$33.50
	12” Motor Mounting Tube MMT-1.525 (38mm)	1	$5.49	$5.49
	StratoLogger Altimeters	2	$55.00	$110.00
	Entacore Altimeters	2	$115.00	$230.00
	Key Switches	2	$12.00	$24.00
	1 in PVC Pipe Caps	4	$0.83	$3.32
	Acrylic Tubing (OD 2.75”, 1/8” thick)	1	$43.27	$43.27
	Terminal Blocks	3	$7.00	$21.00
	3mm Ball Bearings (1000 count)	2	$8.22	$16.44
	Adafruit BNO055 10DoF sensor	1	$28.05	$28.05
			Subtotal:	**$1,139.92**
Full-Scale	Fiberglass 7.5" Filament Wound Metal Tip	1	$170.00	$170.00
	7.5"x 0.08 wall x 12" Blue Tube ARR Standard Coupler	2	$30.00	$60.00
	7.5" x 0.08 wall x 48" ARR Airframe Blue Tube	2	$90.00	$180.00

	Item	Quantity	Price per Unit	Item Total
	Aircraft Spruce Domestic Birch Plywood 1/8"x4x4	3	$50.00	$150.00
	Aircraft Spruce Domestic Birch Plywood 3/8"x2x4	2	$70.00	$140.00
	Rail Buttons	4	$2.50	$10.00
	U-Bolts	3	$1.00	$3.00
	Aerotech L1420R-P	2	$250.00	$500.00
	Motor Casing	1	$390.00	$390.00
	3" G12 Fiberglass Filament Wound Tube 48" Long	1	$91.00	$91.00
	Motor Retainer	1	$25.00	$25.00
	Wires	1	$30.00	$30.00
	Connectors	1	$20.00	$20.00
	StratoLogger Altimeters	2	$55.00	$110.00
	Entacore Altimeters	2	$115.00	$230.00
	BRB 900 Transmitter	3	$200.00	$600.00
	Key Switches	2	$12.00	$24.00
Full-Scale Rocket	Blast Caps	4	$2.50	$10.00
	Terminal Blocks	3	$7.00	$21.00
	E-matches	-	$20.00	$20.00
	Black Powder (lb)	1	$30.00	$30.00
	Sanding Sealer (qt)	1	$17.00	$17.00
	Epoxy and Hardener	1	$50.00	$50.00
	Paint	1	$30.00	$30.00
	Nuts (box)	1	$5.50	$5.50
	Screws (box)	1	$5.00	$5.00
			Subtotal:	**$2,921.50**
Payload	5.25" x .125" Wall Acrylic Tube	1	$145.00	$145.00
	LiPo Batteries	2	$15.00	$30.00
	Servo Motor	1	$20.00	$20.00
	Arduino Nano	1	$22.00	$22.00
	120mm Lazy Susan Bearing	2	$9.37	$18.74
	Southco R4-EM 4&6 Series Electronic Rotary Latch	1	$45.00	$45.00
	1” Brass Hinges (4 pack)	1	$6.28	$6.28
			Subtotal:	**$287.02**
Rover	Arduino Pro Mini 238	1	$9.95	$9.95
	XBee Pro 900 XSC S3B Wire Rx	1	$66.95	$66.95
	FS90 Servo	2	$4.95	$9.90
	Turnigy 1000mAh 25 20C LiPoly Pack	1	$4.73	$4.73
	1" wide Molded track Set	1	$82.00	$82.00
	Panasonic-BSG AM-1456CA	7	$1.55	$10.85
			Subtotal:	**$184.38**
Re	Jolly Logic Chute Release	1	$130.00	$130.00

	Item	Quantity	Price per Unit	Item Total
	36 in. Standard Elliptical Parachute	1	$89.00	$89.00
	120 in. Iris Ultra Compact Parachute	1	$685.00	$685.00
	7.5” deployment bag	1	$69.00	$69.00
	Kevlar Shock Cord (yd) (full-scale)	28	$4.34	$121.52
	¼” Kevlar Tube Shock Cord (yd) (subscale)	20	$2.84	$56.80
	Quick Links	5	$1.25	$6.25
			Subtotal:	**$1,157.57**
Travel	Hotel (7 rooms, 6 days)	7	-	$4,000.00
Travel	Van Rental (2 vans, 1,200 miles each)	2,400	$0.69	$1,656.00
Travel	Gas	-	-	$344.00
			Subtotal:	**$6,000.00**
Promotional	T-Shirts	25	$15.00	$375.00
Promotional	Polos	25	$30.00	$750.00
Promotional	Pens	500	$0.24	$120.00
Promotional	Stickers	500	$0.26	$130.00
			Subtotal:	**$1,375.00**
Other	Incidentals (replacement tools, hardware, safety items)	-	-	$1,000.00
Other	Shipping Costs	-	-	$750.00
			Subtotal:	**$1,750.00**
			Total Expenses:	**$14,815.39**

6.5 Project Funding Plan

The Tacho Lycos team and HPRC gets funding from multiple NC State University organizations as well as the North Carolina Space Grant (NCSG).

The Engineering Technology Fee (ETF) fund from the MAE department at NC State provides monetary support to any student organizations that satisfy the Senior Design project curriculum requirements. Since the NASA SL project will complete the requirements for the Aerospace Engineering Senior Design course, the team will receive approximately $2,000 from the ETF fund for the entire school year.

The Engineers’ Council (E-Council) at NC State University is a student-led organization that oversees events hosted by the College of Engineering. E-Council also allocates funds to different engineering organizations through a proposal, presentation, and appeals process that occurs twice per school year. This academic year, HPRC has received $850 from E-Council for the fall session. A similar request of $900 will be placed for the Spring session, given that the E-Council budget remains the same.

The NC State University Student Government Association’s Appropriations Committee is responsible for distributing university funds to campus organizations. The application process is similar to E-Council with a proposal, presentation, and an in-person interview. This academic year, HPRC has received $865 for the fall session. A similar request of $950 will be made for the Spring and should be received given that Student Government's budget remains the same.

The Department of Mechanical and Aerospace Engineering at NC State has committed to paying up to $6,000 dollars of HPRC's travel costs for the year.

In addition to funding through NC State organizations, the North Carolina Space Grant will provide a large amount of monetary support to the club. NCSG accepts funding proposals during the fall semester and teams can request up to $5,000 for participation in NASA competitions, and another $3,000 for Senior Design projects related to aerospace research. NCSG will review the proposal and inform the club on the amount awarded by mid-November, 2017.

These totals are listed in Table 6-2, below, which compares the projected costs and incoming grants for the 2017-18 school year.

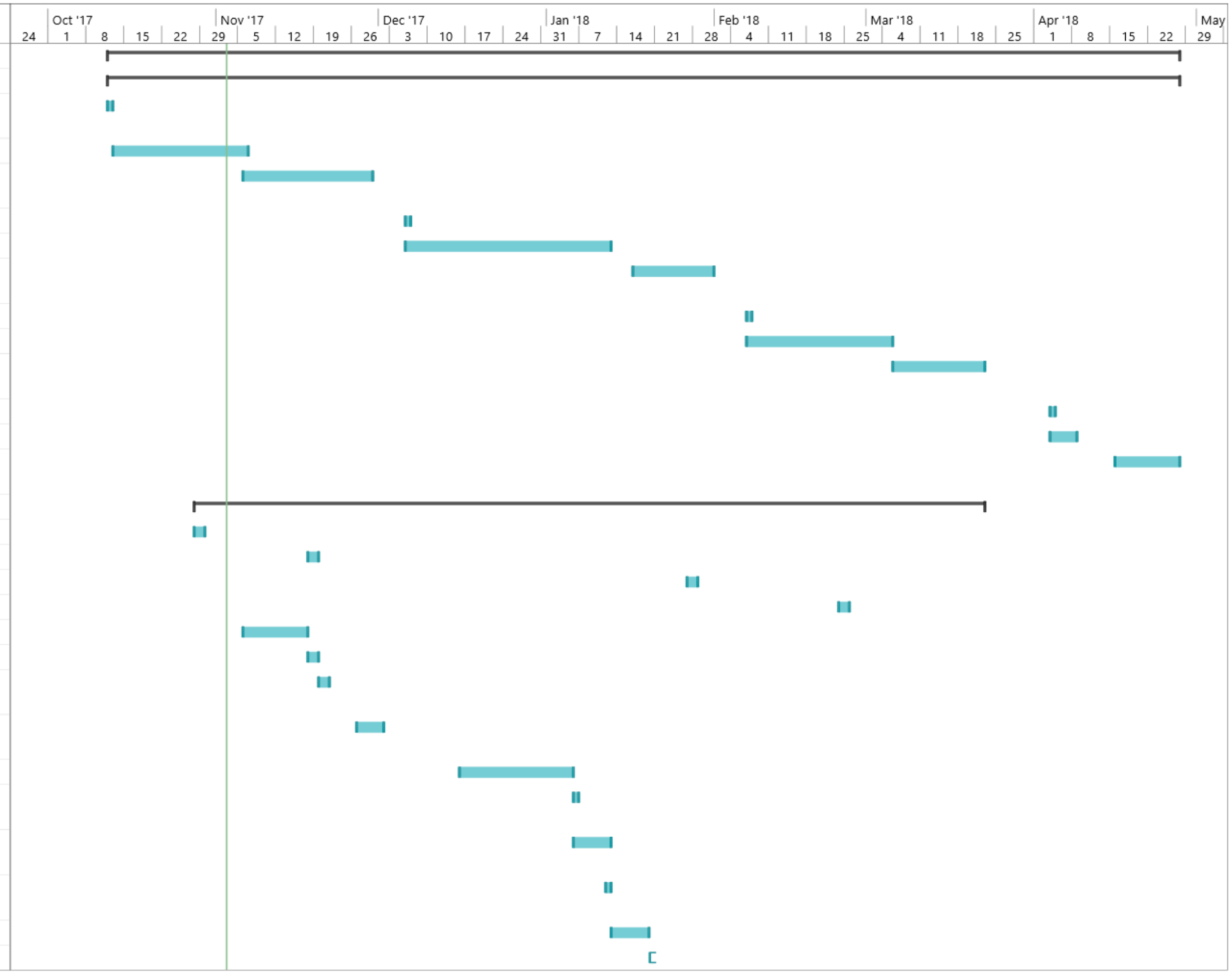
Table 6-2: HPRC Projected Funding for 2017-18 School Year

Organization	Fall Semester Amount	Spring Semester Amount	School Year Total
E-Council	$850.00	$900.00	$1,750.00
SGA Appropriations	$865.00	$950.00	$1,815.00
NC Space Grant	-	-	$8,000.00
ETF Fund	-	-	$2,000.00
MAE Department	-	-	$6,000.00
		Total Funding:	**$19,565.00**
		Total Expenses:	**$14,815.39**
		Difference:	**+$4,749.61**

6.6 Project Timeline

A project plan and timeline were established to enable the team to effectively plan events and tasks through the course of the year. The project plan is presented below is Figure 6.3 and Figure 6.4

ID	Task Name	Duration	Start	Finish
1	**2017-2018_Tacho_Lycos_HPRC_Sched**	**142 days**	**Thu 10/12/17**	**Fri 4/27/18**
2	**NASA Student Launch Schedule**	**142 days**	**Thu 10/12/17**	**Fri 4/27/18**
3	Preliminary Design Review (PDR) Q&A	1 day	Thu 10/12/17	Thu 10/12/17
4	PDR	17 days	Fri 10/13/17	Mon 11/6/17
5	PDR Team Teleconference Window	18 days	Mon 11/6/17	Wed 11/29/17
6	Critical Design Review (CDR) Q&A	1 day	Wed 12/6/17	Wed 12/6/17
7	CDR	28 days	Wed 12/6/17	Fri 1/12/18
8	CDR Team Teleconference Window	11 days	Wed 1/17/18	Wed 1/31/18
9	Flight Readiness Review (FRR) Q&	1 day	Wed 2/7/18	Wed 2/7/18
10	FRR	19 days	Wed 2/7/18	Mon 3/5/18
11	FRR Team Teleconference Window	13 days	Tue 3/6/18	Thu 3/22/18
12	Launch Readiness Review (LRR)	1 day	Wed 4/4/18	Wed 4/4/18
13	NASA Launch Week Events	4 days	Wed 4/4/18	Sun 4/8/18
14	Post-Launc Assessment Review (PLAR)	10 days	Mon 4/16/18	Fri 4/27/18
15	**Tacho Lycos Schedule**	**104 days**	**Sat 10/28/17**	**Thu 3/22/18**
16	Subscale Flight Test Window 1	2 days	Sat 10/28/17	Sun 10/29/17
17	Subscale Flight Test Window 2	2 days	Sat 11/18/17	Sun 11/19/17
18	Full-Scale Flight Test Window 1	2 days	Sat 1/27/18	Sun 1/28/18
19	Full-Scale Flight Test Window 2	2 days	Sat 2/24/18	Sun 2/25/18
20	Subscale construction and testing	10 days	Mon 11/6/17	Fri 11/17/17
21	Subscale Launch	2 days	Sat 11/18/17	Sun 11/19/17
22	Review subscale data & and analyze results	2 days	Mon 11/20/17	Tue 11/21/17
23	Incorporate design changes to full scale, finalize motor selection	5 days	Mon 11/27/17	Fri 12/1/17
24	Write first draft of CDR	16 days	Sat 12/16/17	Fri 1/5/18
25	Submit CDR draft to advisors/mentors for review	1 day	Sat 1/6/18	Sat 1/6/18
26	Fine-tune CDR and incorporate changes	6 days	Sat 1/6/18	Fri 1/12/18
27	Post CDR to website and email NASA SL office	1 day	Fri 1/12/18	Fri 1/12/18
28	Purchase orders for full-scale	6 days	Sat 1/13/18	Fri 1/19/18
29	Full-scale construction and testing		Sat 1/20/18	TBD



Project: Tacho_Lycos_2017-201
Date: Fri 11/3/17

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

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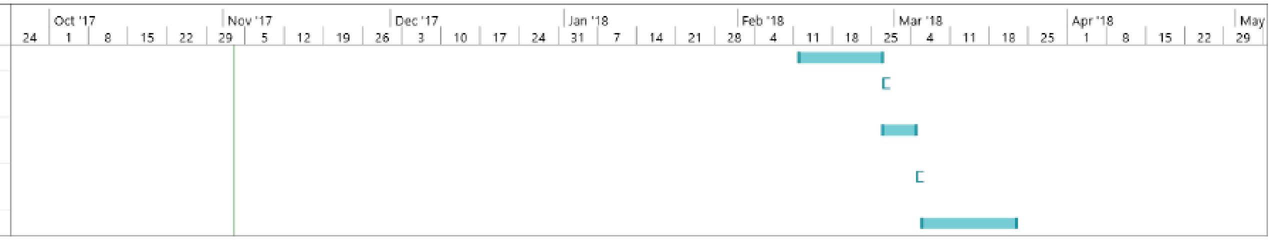
Figure 6-2: Project Plan (pg 1)

ID	Task Name	Duration	Start	Finish
30	Write first draft of FRR	11 days	Mon 2/12/18	Mon 2/26/18
31	Submit FRR draft to advisors/mentors for review		Tue 2/27/18	
32	Fine-tune FRR and incorporate changes	5 days	Tue 2/27/18	Sun 3/4/18
33	Post FRR to website and email NASA SL office		Mon 3/5/18	
34	FRR video conference TBD	13 days	Tue 3/6/18	Thu 3/22/18



Project: Tacho_Lycos_2017-201
Date: Fri 11/3/17

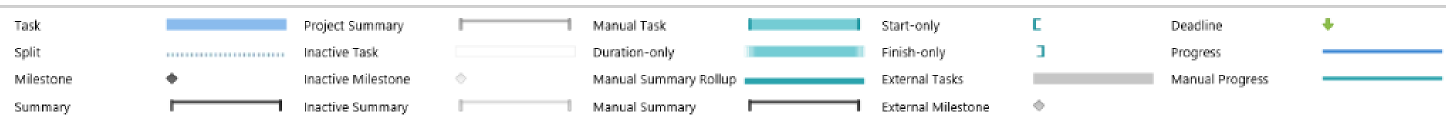


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Table 6-3: Project Timeline (pg 2)

7. References

7.1 Structures References

[1] Ebersole, D. (n.d.). What Is Blue Tube 2.0. Retrieved October 29, 2017, from https://www.alwaysreadyrocketry.com/blue-tube-2-0/

7.2 Recovery References

[2] K, J., & K, P. (2016, June 25). Reefing Parachutes [Three methods of reefing parachutes for variable drag]. Retrieved October 18, 2017, from http://www.aircommandrockets.com/day177.htm

8. Appendix A – MATLAB Code

8.1 Recovery System MATLAB Code

```
clear all
close all
clc

% NCSU Tacho Lycos PDR
%Propulsion and Recovery

%% Exit Velocity

% Subscale
d_inches = 12.9;
R_inches = 72;
T_Newton = 482;
W_lbf = 9.69;
theta_degrees = 85;

% calculations
d_feet = d_inches / 12
R_feet = R_inches / 12
Lin_feet = R_feet - d_feet
T_lbf = T_Newton * 0.224809
m_slugs = W_lbf / 32.174
vExit_feet = sqrt((2 * Lin_feet * (T_lbf-W_lbf*sind(theta_degrees)))/m_slugs)
vExit_inches = vExit_feet * 12

% Full-scale
% initial conditions
d_inches2 = 28.25;
R_inches2 = 96;
T_Newton2 = 2243;
W_lbf2 = 47;
theta_degrees2 = 85;

% calculations
d_feet2 = d_inches2 / 12
R_feet2 = R_inches2 / 12
Lin_feet2 = R_feet2 - d_feet2
T_lbf2 = T_Newton2 * 0.224809
m_slugs2 = W_lbf2 / 32.174
vExit_feet2 = sqrt((2 * Lin_feet2 * (T_lbf2-W_lbf2*sind(theta_degrees2)))/m_slugs2)
vExit_inches2 = vExit_feet2 * 12



%% Kinetic Energy

% full-scale KE
KE_f_lbf = 75 %75 ft-lbf (NASA SL Handbook Requirement)
W = 41.45;
```

```
m = W/32.174
syms V1
eqn = KE_f_lbf == 0.5 * m * V1^2;
answer = solve(eqn,V1) %I don't why this keeps happening :(

% subscale KE
KE_f_lbf2 = 39 % 52% of 75 ft-lbf (NASA SL Handbook Requirement)
W2 = 9.1;
m2 = W2/32.174
syms V2
eqn2 = KE_f_lbf2 == 0.5 * m2 * V2^2;
answer2 = solve(eqn2,V2)
%% Drogue and main

% Subscale Parachute sizing equations

Vs = 51.33; %feet per second
ms = .248; %slugs
g = 32.174; %feetpersecond^2
ps = .002378; %slugs/ft^3
C_dds = 1.5; %average sphereical chute
Ss = (2*ms*g)/((Vs^2)*C_dds*ps);
Ds = (C_dds*Ss*ps*Vs^2)/2;
C_dms = 2.2; %average value for iris ultra
V_ms = 17.73; %ft/s
S_ms = (2*ms*g)/((V_ms^2)*C_dms*ps);

%subscale drogue
dia = [12 18 24 30 36 ];
desc_vel = [76.64 51.10 38.34 30.66 25.67];
ma = 60 ; % ft/s

figure
hold on
xlim([12 36])
title('Subscale Drogue Parachute Descent Velocities of Various Parachute Diameters')
xlabel('Diameter of Parachute (in.)')
ylabel('Expected Descent Velocity (ft/s)')
plot(dia,desc_vel,'--k');
plot(dia,ma,'-r')
legend('Descent Velocity of Variable Diameters','Maximum Velocity of 50 ft/s')
hold off

% Subscale main
di = [36 42 48 54 60 66 72 ];
desc_velo = [21.1 18.08 15.82 14.06 12.66 11.51 10.55];
maxx = 15 ; % ft/s

figure
hold on
xlim([36 72])
title('Subscale Main Parachute Descent Velocities of Various Parachute Diameters')
xlabel('Diameter of Parachute (in.)')
ylabel('Expected Descent Velocity (ft/s)')
```

```
plot(di,desc_velo,'--k');
plot(di,maxx,'-b')
legend('Descent Velocity of Variable Diameters','Maximum Velocity of 15 ft/s')
hold off

% full-scale parachute
Vf = 180; %feet per second **** need this 5 sec after apogee from raven
mf = 1.28; %slugs
g = 32.174; %feetpersecond^2
pf = .002378; %slugs/ft^3
C_ddf = 1.5; %average elliptical chute
Sf = (2*mf*g)/((Vf^2)*C_ddf*pf)
Df = (C_ddf*Sf*pf*Vf^2)/2
C_dmf = 2.2 ;%average value for iris ultra
V_mf = 17.73; %ft/s ** comes from KE eqn
S_m = (2*mf*g)/((V_mf^2)*C_ddf*pf) % area of main in ft^2

% full-scale drogue
diam = [12 18 24 30 36 42 48 54 60];
desc_veloc = [174.46 116.31 87.23 69.79 58.43 49.85 43.62 38.77 34.89];
max = 60 ; % ft/s

figure
hold on
xlim([12 60])
title('Drogue Parachute Descent Velocities of Various Parachute Diameters')
xlabel('Diameter of Parachute (in.)')
ylabel('Expected Descent Velocity (ft/s)')
plot(diam,desc_veloc,'--k');
plot(diam,max,'-r')
legend('Descent Velocity of Variable Diameters','Maximum Velocity of 60 ft/s')
hold off
% full-scale main
diam2 = [96 102 108 114 120 126 132 138 144];
desc_veloc2 = [18.01 17.03 16.08 15.24 14.41 13.79 13.16 12.53 12];
max2 = 10.79 ; % ft/s

figure
hold on
xlim([96 144])
title('Main Parachute Descent Velocities of Various Parachute Diameters')
xlabel('Diameter of Parachute (in.)')
ylabel('Expected Descent Velocity (ft/s)')
plot(diam2,desc_veloc2,'--k');
plot(diam2,max2,'-b')
legend('Descent Velocity of Variable Diameters','Maximum Velocity of 10.79 ft/s')
hold off

%% black powder
%L_d = length drogue Section (in)
%D_d = diam drogue sect (in^2)
% 0.006 is the constant that converts in^3 to grams

%SUBSCALE drogue
L_ds = 14.5;
```

```
D_ds = 3.776;
m_ds = 0.006*L_ds*(D_ds^2);

% SUBSCALE main
L_ms = 14.5;
D_ms = 3.776;
m_ms = 0.006*L_ms*(D_ms^2);

%full-scale drogue
L_df = 17.35;
D_df = 7.34;
m_df = 0.006*L_df*(D_df^2);

%full-scale main
L_mf = 20.5;
D_mf = 7.34;
m_mf = 0.006*L_mf*(D_mf^2);

%% wind drift

%0
%alt
y0= 5572.9; %ft
y00 = 0:y0;
x0= 7.77;%ft
x_0 = 0:x0;
%drift line eqn
m0 = y0/(-x0);
b0 = y0 - (y0/(-x0));
y_0 = m0*(x_0)+b0;
zero = 0;
%5
%alt
y5=5566.6;
y55= 0:5566.6;% ft
x5= 821.82; %ft
x_5=0:x5;
%drift
b5 = y5 - (y5/(-x5));
m5 = y5/(-x5);
y_5 = m5*(x_5)+b5 ;

%10
%alt
y10=5549.8;
y1010= 0:5549.8;%ft
x10= 1686.4; %ft
x_10 = 0:x10;
%drift
b10 = y10 - (y10/(-x10));
m10 = y10/(-x10);
y_10= m10*(x_10)+b10;

%15
```

```
%alt
y15= 5524.9;
y1515= 0:5524.9;%ft
x15= 2571.2;%ft
x_15=0:x15;
%drift
b15 = y15 - (y15/(-x15));
m15 = y15/(-x15);
y_15 = m15*(x_15) + b15 ;

%20
%alt
y20 = 5493.7;
y2020= 0:5493.7;%ft
x20= 3464.4;%ft
x_20 = 0:x20;

%drift
b20 = y20 - (y20/(-x20));
m20 = y20/(-x20);
y_20 = m20*(x_20) + b20 ;

% plots
figure
hold on
title('Effect of Wind on Altitude (ft) and Drift (ft)')
xlabel('Drift From Launch Pad (ft)')
ylabel('Altitude (ft)')

%plot(zero,y1515)
%plot(zero, y2020)
xlim([-500 3500])
plot([0,7.7],[0,5572.9],'-k')
plot(x_5,y_5,'-b')
plot(x_10,y_10,'-g')
plot(x_15, y_15,'-y')
plot(x_20, y_20,'-r')
legend('0 mph, 5572.9 ft AGL','5 mph, 5566.6 ft AGL','10 mph, 5549.8 ft AGL', '15 mph, 5524.9 ft AGL', '20 mph, 5493.7 ft AGL')
```

9. Appendix B - FMECAs

The FMECA table below uses the following hazard level classification:

1) Rocket mission failure; rocket is not recoverable; payload is not recoverable; rover mission failure
2) Rocket mission failure; rocket is recoverable; payload may be recoverable; rover mission may be complete
3) Rocket mission complete; rocket is recoverable; payload is not recoverable; rover mission failure
4) Rocket mission complete; rocket is recoverable; payload is recoverable; rover mission failure

System	Subsystem/ Component	Failure Mode	Casual Factors	Failure Effects		Hazard	Recommendations
				Subsystem	System		
Launch Vehicle	Nosecone	Cracks or breaks	Object in flight path	Loss of nose cone recoverability	Loss of controlled and stabilized flight	2	Ensure that skies are clear of any foreign objects per NAR operations
			Damaged during handling or assembly			2	Inspect for cracks, chips, or other damage during assembly
			Nosecone collides with other rocket component during recovery		N/A	2	Ensure shock cord is long enough to separate nosecone from other components
			Ground impact			2	Metal nose tip will mitigate damage caused by surface impact
		Premature separation from midsection	Damaged during handling or assembly	Potential for permanent structural damage	Loss of controlled and stabilized flight	2	Inspect for cracks, chips, or other damage during assembly

			Shear pins not installed correctly			2	Follow design specifications for sizing, inspecting, and installing shear pins during assembly
			Epoxy not cured properly			2	Follow proper procedures for mixing, applying, and curing epoxy
	Blue Tube Airframe	Cracks or breaks	Manufacturing defect	Loss of structural integrity or usability of Blue Tube body Sections or components	Premature separation of launch vehicle Sections during flight	1	Visual inspection after shipping and before assembly
			Loads experienced beyond design specifications			1	Ensure body tube components can hold flight forces in accordance with design specifications Ensure that all components can maintain a factor of safety of at least 1.5 during all regimes of flight

			Damaged during handling or assembly			1	Inspect each body tube component for cracks, bends, warping, or other damage during assembly Replace any damaged components if possible
			Improper storage			1	Store Blue Tube in a dry space and avoid any liquid spills Do not store items on top of Blue Tube
	Fins	Severe weather-cocking	Fin dimensions are not cut according to design	N/A	Decreased flight stability, unpredictable flightpath, and possible damage to other components	2	Laser cut fins to ensure manufacturing precision Ensure that excessive material is not removed during sanding
			Fins not installed at even increments around fin can (90 degrees to each other)			2	Use laser-cut jig with slots for fins exactly 90 degrees from each other

			Assembled rocket CG is too far forward (Stability Margin >> 2.0)			2	Ensure components and masses are installed according to design specifications
		Fin separation	Loads experienced beyond design specifications	N/A	Any one failure of a fin could lead to additional fin failures which will decrease flight stability and will likely cause a catastrophic failure	1	Analyze flight data and simulations to confirm that factor of safety is sufficient
			Damaged during handling or assembly			1	Inspect for cracks, chips, or other damage during assembly Replace any damaged components
			Fin flutter			2	Rocket will not exceed velocity necessary to induce significant flutter
			Ground impact			2	Implement a recover system design that ensures a low-speed surface impact

	Motor	Motor fails to ignite	Igniter not installed correctly	Failure of vehicle to start launch	Team member and RSO must insert new igniter and restart launch sequence	4	Follow launch checklist and use mentor/RSO supervision to install igniter correctly
			Faulty igniter used			4	Test batch of igniters prior to launch day to ensure quality
			Motor assembled incorrectly			4	Follow launch checklist and use mentor/RSO supervision to install motor correctly
		Catastrophic motor failure	Damaged during handling or assembly	Possible destruction of launch vehicle	Complete mission failure and additional hazard to ground crew and spectators	1	Carefully inspect for cracks, chips, or other damage during assembly
			Motor assembled incorrectly			1	Follow launch checklist and use mentor/RSO supervision to install motor correctly

			Motor casing dislodged during motor burn			1	Ensure all connection points between motor tube, centering rings, and fins are joined properly using epoxy Perform careful inspection of joints prior to launch
		Damage to motor casing	Superficial damage	Motor casing cannot be used	Rocket is not safe to launch if damage is major	4	Carefully inspect for cracks, chips, or other damage during assembly
		Propellant contamination	Rocket fails to launch	Reduced performance of rocket motor	Rocket does not launch or perform as expected	2	Store and maintain motor fuel properly and in isolation/ order from reputable source
			Over-oxidized reaction			2	
			Reduced fuel efficiency			3	
	Bulkheads	Bulkhead separation from airframe during motor burn	Manufacturing defect			1	Visual inspection after shipping and before assembly

			Loads experienced beyond design specifications			1	Ensure body tube components can hold flight forces in accordance with design specifications Ensure that all components can maintain a factor of safety of at least 1.5 during all regimes of flight
			Damaged during handling or assembly			1	Inspect each bulkhead for cracks, warping, chips, or other damage during assembly Replace any damaged bulkheads
			Epoxy not cured properly			1	Follow proper procedures for mixing, applying, and curing epoxy

		U-bolt separation from bulkhead during recovery	Loads experienced beyond design specifications	Launch vehicle components not tethered to a parachute will continue accelerating during descent	Loss of safe and effective recovery system	1	Ensure body tube components can hold flight forces in accordance with design specifications Ensure U-bolt fasteners can handle near-instantaneous loading from parachute deployment
			Epoxy not cured properly			1	Follow proper procedures for mixing, applying, and curing epoxy
	Rail Buttons/ Launch Rail	Vehicle does not leave launch rail as intended	Rail button(s) separate from launch vehicle	Vehicle leaves rail at unpredictable orientation and velocity	Possible mission failure and additional hazard to ground crew and spectators	1	Epoxy rail buttons into body tube to mitigate risk of separation
			Damaged during handling or assembly			1	Inspect each pin for cracks, bends, chips, or other damage during assembly Replace any damaged components
			Launch rail breaks			2	Ensure that the rail is assembled correctly prior to launch

		Vehicle does not leave launch rail at all	Rail button(s) becomes stuck in launch rail	N/A	Mission failure as flight does not take place	2	Ensure that rail buttons match size of launch rail slot Lubricate the launch rail and rail buttons prior to launch Ensure that the vehicle moves smoothly on the launch rail during assembly and launch rail erection
	Shear Pins	Pins break before charge detonation	Manufacturing defect	Loose assembly of compartment	Premature rocket separation and recovery system deployment	2	Visual inspection after shipping and before assembly
			Damaged during handling or assembly			2	Inspect each pin for cracks, bends, chips, or other damage during assembly Replace any damaged components
			Pins fall out of respective holes			2	Ensure size of holes drilled in body tube match diameter of shear pins

			Loads beyond design specifications			2	Ensure pins can hold flight forces in accordance with design specifications
		Pins don't break at charge detonation	Manufacturing defect	Failure to separate compartment	Loss of safe and effective recovery system	1	Inspect each pin for cracks, bends, chips, or other damage during assembly Replace any damaged components
			Pins too tight in body tube holes			1	Ensure size of holes drilled in body tube match diameter of shear pins
			Poor design			1	Use calculations to ensure that pins will break from forces of detonation

	Shock Cord	Incorrect or partial deployment of shock cord	Snags, tears, or rips during ejection	Parachute no longer tethered to entirety of launch vehicle airframe	Loss of safe and effective recovery system	1	Inspect shock cord for damage prior to launch Ensure high-strength shock cord is used with a maximum loading greater than 1,500 lb Ensure shock cord is folded and stowed properly in launch vehicle Reduce/eliminate sharp edges in design to mitigate risk of snagging shock cord and parachutes
			Shock cord disconnects from airframe or parachutes			1	Ensure that connections between the shock cord, airframe, and parachutes are tight and secure

			Shock cord stuck within launch vehicle airframe	Parachute not entirely deployed		1	Ensure that the shock cord and parachutes are folded and stowed properly in launch vehicle Reduce/eliminate sharp edges in design to mitigate risk of blocking shock cord and parachutes
	Parachute Deployment	Drogue parachute fails to deploy correctly	Drogue shock cord tangling	Parachute does not deploy correctly	Rocket is recoverable	2	Ensure that shock cords and parachutes are folded correctly
			Shock cord connections come loose			2	Test shock cord connections before flight, make sure secure
			Parachute bag does not fully open			2	Fold bags correctly and make sure nothing can snag bags
		Parachute does not perform as expected	Tears/holes	Parachute deploys but does not perform as expected	Rocket is recoverable	2	Inspect parachute before folding and packing
		Main parachute fails to deploy correctly	Charge is inadequate	Parachute does not deploy correctly	Separation 2 is not successful, rocket is not recovered safely	1	Test charge measurements before flight

			Payload blocks parachute		Separation 2 is not successful, rocket is not recovered safely	1	Make sure payload ejection hardware does not impact parachute release path
			Shock cords tangled		Rocket is recoverable	2	Ensure shock cords and parachutes are folded correctly
			Shock cord connections loose		Rocket is not recovered safely	1	Test shock cord connections before flight, make sure secure
	Black Powder Charges	Single detonation failure	E-match doesn't light	Failure of one or more black powder charges	Will result in loss of safe and effective recovery system if redundant black powder charge(s) do not detonate	2	Conduct ground tests to ensure that enough black powder will be used for proper separation Thoroughly check redundant systems prior to launch Confirm that wires are attached per design specifications
			Altimeter Malfunction			2	Ensure altimeters are functional prior to launch Test altimeters regularly to ensure component integrity

		Redundant detonation failure	E-match doesn't light	Failure of both ejection charges	Rocket fails to separate and deploy parachutes	1	Conduct ground tests to ensure that enough black powder will be used for proper separation Thoroughly check redundant systems prior to launch Confirm that wires are attached per design specifications
			Altimeter Malfunction			1	Ensure altimeters are functional prior to launch Test altimeters regularly to ensure component integrity

		Charge causes damage to any component other than shear pins	Charge is too big	Causes violent separation and/or damage to surrounding area	Potential to cause permanent damage to bulkheads or shock cord, resulting in a possible failure of parachute deployment	2	Verify that charges are sealed properly and the correct amount of black powder is used with pre-flight checklist Conduct ground tests to ensure that enough black powder will be used for proper separation without damage to other components
		Charge ignites but fails to cause separation	Charge is too small	No ejection	Failure of parachute deployment	1	Conduct ground tests to ensure that enough black powder will be used for proper separation
	Altimeters	No power to altimeters	Uncharged or insufficiently charged batteries	Loss of real-time altitude data, failure to ignite e-match	Failure of parachute deployment	1	Install new/unopened batteries at each launch Confirm that all batteries have the correct voltage before flight using a multimeter

			Battery becomes disconnected from altimeter			1	Ensure that altimeters are properly wired and that wires are secure prior to launch Listen for appropriate chirps when powering on altimeters
			Wiring short			1	Ensure that all wire is properly insulated and that all wires are securely contained in their respective terminals
		No launch detected	Manufacturing defect	Lack of flight data	Failure of parachute deployment	1	Test altimeters in vacuum chamber prior to launch Listen for fault codes at launch site
		False apogee detected	Manufacturing defect	Premature/late ejection of drogue parachutes	Increased load on drogue recovery hardware and bulkheads	2	Test altimeters in vacuum chamber prior to launch Listen for fault codes at launch site

			Incorrect altimeter readings			2	Ensure that pressure ports are sized correctly and listen for fault codes at launch site
		Main parachute deploys at wrong altitude	Incorrect pressure readings or improper programming	Main deployment between apogee and 1,200 ft	Excessive drift, but surface impact will remain below required maximums	2	Verify each altimeter chirps the appropriate program at the launch site Test altimeters in vacuum chamber prior to launch date Ensure pressure ports are sized correctly
				Main deployment lower than 800 ft	Kinetic energy at surface impact will likely exceed 75 ft-lb parachute	1	
	GPS	Ground system failure	Loss of power to ground receiver or the laptop	Inability to receive data from the GPS	Inability to track and recover the rocket in less than an hour	3	Ensure that the receiver and laptop are fully charged at least 6 hours prior to flight
		Loss of signal	Environment or rocket materials blocking signal			3	Perform range tests to ensure reliability of the system at simulated altitudes and ground distances

		Radio interference	Multiple radio devices on the same local frequency and channel			3	Ensure that all transmitting devices are on separate channels and confirm with other teams and launch officials that no frequency conflict exists
		Loss of power	Flight forces cause GPS to disconnect from power supply			3	Ensure that all GPS units are fully charged and use simulated load tests to determine the necessary procedures to secure the units
	Avionics Sled	Detaches from secure position	Loads beyond design specifications	Damage to/loose wiring of avionics components	Loss of recovery system initiation	1	Use simulated load tests and add a sufficient factor of safety when designing sled
			Damage during handling			1	Team members will be taught proper handling and installation procedures for the avionics sled
			Improper maintenance			1	Pre- and post-launch thorough inspections of the avionics sled
Payload	Payload Exterior	Acrylic Tube Door Fails to Open	Manufacturing defects	Rover Systems at Risk	Electronics and rover can be damaged	3	Visual inspection prior to use

		Door Pin System Breaks Door Hinges Come Off	Servo fails	Prevents Rover Deployment	Rover will not complete its task	3	Test servo connection prior to use
			Blocked by dirt after landing	Prevents Rover Deployment	Rover will not complete its task	3	Shape the end of the payload to ensure it lands laterally in order for rover to deploy
			Blocked by centering ring	Prevents Rover Deployment	Possible structural damage to the payload	3	Make the door a smaller size than the centering ring
			Fracture from Force	Unable to transfer loads	Increased loads on other structural members	3	Ensure materials used are strong enough to withstand the forces felt during the flight
			Improper Attachment	Door may Detach	Rover will not complete its task	3	Ensure the screws are properly secured before every flight
	Payload Bulkhead	Bulkhead Separation from payload	Poor Design	Load Transfer Failure	Unintended Structures Bear the Load	2	FEA of bulkhead fixed support to ensure bulkheads can support the loading
			Manufacturing Defect			2	QC of manufacturing process to ensure there is no damage present
			Loads Greater than Designed			2	Maintain vehicle within planned design

			Damaged During Handling			2	Ensure analysis includes handling loads/adhere to proper handling procedure
			Improper Attachment			2	Inspect each bulkhead to ensure proper attachment before and after each launch
	Rover Bay	Separation of Rover from platform	Improper attachment	Rocket weight imbalance during flight	Rocket flight disrupted	3	Test the attachment of the weight and ensure it is properly secure
				Rover fails to rest at bottom of the tube	Prevents proper deployment	3	
		Electronics Fails	Circuitry becomes Damaged	Payload hardware experiences catastrophic failure	Rover fails to deploy	3	Electronics tested thoroughly
	Hardware Rover Tracks	Malfunction Rail button sheared off upon payload jettison Cracks or Breaks	Unable to gain traction	Rover cannot move forward Rover cannot move forward Rover performance hindered	Prevents rover from deploying at the correct orientation Possible damage to other components N/A N/A	4	Test treads on various services
			Cannot traverse obstruction			4	Test treads on various terrain Run structural analysis
			Damage during flight			4	
		Premature detonation	Manufacturing defect	Rover performance hindered	Payload drifts farther than planned	4	Follow proper additive manufacturing technique Test rover on various terrains
			Damage during rover operation	Rover performance hindered		4	

	Hardware	Break	Damage during flight	Rover cannot complete mission	Prevents rover from deploying at the correct orientation	4	Test the structural integrity of the parts and run structural analysis
		Rail button sheared off upon payload jettison	Manufacturing defect	Rover cannot complete mission	Possible damage to other components	4	Follow proper additive manufacturing technique
	Rover Body Rover Gears/ Motor System	Cracks or Breaks Jammed Do not operate	Damage during rover operation	Rover cannot complete mission	N/A N/A N/A N/A	4	Run tests to determine rover capabilities
			Foreign objects get stuck in the gears/motor	Rover performance hindered		4	Run tests in similar conditions to landing site
			Dead battery	Rover cannot complete mission		4	Ensure proper battery charging and handling techniques are followed
		Break Do not deploy	Signal is not sent properly	Rover cannot complete mission		4	Extensively test transceiver in all conditions
			Programming bug	Rover cannot complete mission		4	Run tests on Arduino to ensure high performance
	Rover Body	Cracks or Breaks Jammed Do not operate Jammed Do not operate Low Charge	Signal is not sent properly	Design challenge is not completed	N/A	4	Extensively test transceiver in all conditions
			Foreign objects get stuck in solar panel housing	Design challenge is not completed		4	Test solar panels in conditions similar to landing site

			Dead battery	Design challenge is not completed		4	Ensure proper battery charging and handling techniques are followed
		Break Do not deploy Fire	Improper charging techniques	Rover performance hindered		4	Adhere to proper charging technique
			Improper storage	Rover performance hindered		4	Adhere to proper storage technique
	Rover Gears/ Motor System Rover Solar Panels Rover Battery		Not following proper safety protocol	Rover cannot complete mission	N/A N/A N/A Damage to payload	3	Maintain a high level of safety
		Jammed					
		Do not operate Low Charge					
	Rover Solar Panels						