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Notre Dame Rocket Team
Preliminary Design Review

NASA Student Launch 2020

Lunar Sample Retrieval System and Air Braking System

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365 Fitzpatrick Hall of Engineering
Notre Dame, IN 46556

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Table 1: List of Acronyms

Acronym	Meaning
ABS	Air Braking System
ACCST	Advanced Continuous Channel Shifting Technology
AGL	Above Ground Level
CFD	Computational Fluid Dynamics
CPU	Central Processing Unit
CRAM	Compact Removable Avionics Module
DSM	Digital Spectrum Modulation
ESC	Electronic Speed Controller
FEA	Future Excursion Area
FMEA	Failure Modes and Effects Analysis
FPS	Frames Per Second
FPV	First-Person View
IMU	Inertial Measurement Unit
LED	Light Emitting Diode
LiPo	Lithium Polymer
NDRT	Notre Dame Rocket Team
OpenCV	Open Source Computer Vision Library
OPTO	Optoisolator
PCB	Printed Circuit Board
PDB	Power Distribution Board
PID	Proportional-Integral-Derivative
PLA	Polylactic Acid
PWM	Pulse-Width Modulation
RC	Radio Controlled
RF	Radio Frequency
UAV	Unmanned Aerial Vehicle

1 Summary of Report

1.1 General Information

School Name:	University of Notre Dame
Team Name:	Notre Dame Rocketry Team
Location:	University of Notre Dame 365 Fitzpatrick Hall of Engineering Notre Dame, IN 46556
Faculty Advisor:	Dr. Aleksander Jemcov Research Assistant Professor Department of Aerospace and Mechanical Engineering e: ajemcov@nd.edu p: (574)631-7576
Graduate Student Advisors:	Emma Farnan & Joseph Gonzales Ph.D. Graduate Students Department of Aerospace and Mechanical Engineering e: efarnan@nd.edu & jgonza21@nd.edu p: Emma: (631)572-6091 & Joseph: (703)887-9974
Team Lead:	Collette Gillaspie e: cgillasp@nd.edu p: (402)699-7366
Safety Officer:	Brooke Mumma e: bmumma@nd.edu p: (314)288-8297
Mentor:	Dave Brunsting (NAR/TAR Level 2) e: dacsmema@gmail.com p: (269)838-4275
NAR/TAR Section:	TRA #12340, Michiana Rocketry

1.2 Mission Statement

The Notre Dame Rocketry Team is to build a vehicle to reach a target altitude of 4,444 ft. Success on this criterion will be determined based on readings from an altimeter on board the rocket. The launch vehicle will be designed to be recoverable and reusable without need of repair and have a maximum of four independent sections. The vehicle will be limited to a single stage that is to be launched by a standard 12-volt direct current firing system and a total impulse of 5,120 Newton-seconds and must maintain at least 2 cal stability for the whole duration of flight. Furthermore, two identical subscale vehicles will be built to allow the team to have a test flight and wind tunnel testing.

1.3 Launch Vehicle Summary

After having considered different designs for this year's mission, considerations which are explained in detail under section 3.2, the team has decided on building a variable diameter rocket. The variable diameter allows for extra space for the payload and its deployment mechanism near the nose cone and optimizes the weight by transitioning to a smaller diameter aft of the transition. Material selection include carbon fiber for the aft body tubes and fiber glass for the payload bay area to allow for payload communication which have historically proven strong enough and the team has experience with. The nose cone will be a 3:1 ogive nose cone which will be 3D printed out of acrylonitrile styrene acrylate (ASA) plastic and fins will be made of 0.125 in carbon fiber sheet and have an isosceles trapezoidal shape for stability. The motor selected was an L1395 which allows for a high enough apogee for ABS actuation to bring apogee down from the predicted 4,860 ft to the target 4,444 ft.

1.3.1 Size Statement

The preliminary design of the launch vehicle for this year's Student Launch competition is a variable diameter rocket with a total length of 136 inches and fore and aft diameters of 8 and 6 inches respectively. Size budgets for each subsystem (which includes payload, recovery, ABS, and Telemetry) were determined by the team. The rocket design and construction will strictly adhere to these size budgets. To ensure that these are not exceeded, constant monitoring and open communication with the corresponding squads are in place. The mentioned size budgets and other general dimensions of the rocket can be found below in Table 2. An in-depth size statement is found further in the report in section 3.4.2. center of gravity (CG) and center of pressure (CP) positions are measured from the nose cone and estimated with the use of OpenRocket and RockSim and all diameters are outer diameters.

Table 2: Vehicle Dimensions and Characteristics

Characteristic	Dimension	Units
Length of Rocket	133	in
Fore Diameter	8	in
Aft Diameter	6	in
Transition Length	5	in
Number of Fins	4	in
Fin Root Chort	6	in
Fin Tip Chord	3	in
Fin Height	6.5	in
Sweep Length	1.5	in
CG Position (with motor)	75.67	in
CP Position	96.6	in
Unloaded Weight	663	oz
Loaded Weight	816	oz
Estimated Unloaded Stability Margin	3.91	cal
Estimated Loaded Stability Margin	2.62	cal
Payload Length Budget	20	in
Recovery Length Budget	36	in
ABS Length Budget	12	in

1.3.2 Mass Statement

The weight of general subsystems and components are listed below in Table 3. It is important to understand the distribution of mass across the body of the rocket in order to fully comprehend its aerodynamic performance. Because of this importance, each subsection listed below must adhere to a given weight budget. If any of the components are heavier or lighter than we intended, aspects such as the center of gravity and the stability margin of the launch vehicle could be severely affected. To ensure these weights are achieved, constant monitoring and open communication between subsystem teams will again be in place. While this is a general mass statement, a fully detailed one is found in section 3.4.2.

Table 3: Mass Statement

Characteristic	Weight [oz]
Nose Cone	35
Payload	158
Transition Section	35
Recovery	202
ABS	66
Telemetry	48
Fin Can	120
Motor	152
Total	**816**

1.3.3 Motor Selection

Selecting the motor is a key decision that will create constraints on many of the properties of the vehicle. The three motors considered while designing our rocket were: Cesaroni L1090SS-P, Cesaroni L1395, and Aerotech L1300R-P. Through OpenRocket and RockSim the properties of the motor candidates were compiled. The resulting information for the variable diameter design is shown below in Table 4.

Table 4: Motor Selection Comparison

Motor	Cesaroni L1090SS-P	Cesaroni L1395	Aerotech L1300R-P
Apogee [ft]	4324	4841	4211
Diameter [in]	2.95	2.95	3.86
Length [in]	26.18	24.45	17.44
Cost [USD]	292.99	292.99	304.99

After considering the data in Table 4 the team decided to select the Cesaroni L1395 because of its higher apogee. Due to the difference in simulation data it was determined that a higher apogee than predicted would benefit the team as in most cases it would allow the necessary height to allow ABS to accentuate and decrease the rocket's apogee to our target altitude of 4123 ft.

1.4 Payload Summary

The payload experiment chosen for this year's competition is the Lunar Ice Sample Retrieval. The payload experiment is broken into three subsystems: Rover, UAV, and Deployment. The Rover consists of Rover Mechanical Design, Rover Electrical Design, and the Sample Retrieval System. The UAV consists of UAV Mechanical Design, UAV Electrical Design, and Target Detection. The Deployment system is broken down into the orientation, retention, and deployment systems.

2 Changes Since Proposal

2.1 Changes to Vehicle Criteria

Due to apogee concerns, the nosecone has changed from a fiberglass commercial nosecone to a student fabricated ASA one. A 3D printed nose cone with a high enough infill would decrease our total weight of the rocket and allow us to shoot for a higher apogee with a higher margin for ABS actuation. Because of this weight change the target apogee changed from 4,100 ft to 4,444 ft. Additionally, to make sure the stability margin stayed within the acceptable margin between 2 and 3 calibers, the shape of the fins was changed to isosceles trapezoid. To ensure the strength of the nosecone different physical tests will be preformed.

2.2 Changes to Payload Criteria

Since Proposal, the Lunar Ice Retrieval experimental payload has undergone several design changes. These include the use of a rover and a UAV, a rectangular compartment for retention of the payload, an orientation system that utilizes the spring power of the deployment hatch, and the Archimedes screw sample retrieval system. These changes were due to the conduction of trade studies and changes to other systems inside the vehicle. The combination of a UAV and rover will allow utilize the benefits of each system and optimize the performance of the experimental payload.

2.3 Changes to Project Plan

The primary change to the project plan since Proposal was to set milestones for all test flights. The targeted sub-scale flight has been set for November 10th, 2019 with a backup launch date of November 16th, 2019. Additional funding from a rocket team alum and General Electric have been secured, and a full itemized budget was generated for predicted project expenses.

3 Technical Design: Launch Vehicle

3.1 Mission Success Criteria

In order to achieve ultimate mission success, the team arranged a specific set of criteria apart from NASA's criteria in the hopes that this extra direction will ensure that we utilize the proper technical design to optimize the performance and function of the launch vehicle.

Firstly, the vehicle must have two separation points: two in-flight separation points for a drogue and main parachute deployment that will also serve for access to the CRAM and ABS system. A twist and lock design must be able to hold loads of both parachutes deployment and the weight of the launch vehicle while descending. The recovery body tube cannot exceed 48 inches and the air breaking system cannot exceed 12 inches nor 66 oz; the payload bay must be a fiberglass body tube 20 inches in length and 8 inches in outer diameter. Meanwhile, the vehicle itself is designed with a maximum length of 12 feet and a maximum weight of 70 pounds, as well as with a 4,444 foot goal for its altitude. Also, the weight distribution throughout the launch vehicle must be kept as even as possible to decrease the necessary parachute size. The vehicle's ballast area must hold up to ten percent of total vehicle weight and be fully designed and integrated at the launch vehicle's CG to minimize its effect on vehicular stability, which in turn must have a margin of 2-3 calibers with the motor. This motor must favor overshooting the target apogee in order to ensure the use of the air braking system. Lastly, to record the air breaking system's performance the vehicle must hold a camera angled downward for visibility. A full detailed list of the criteria appointed by the team can be located in Appendix A under Table **??**.

3.2 Vehicle Design Trade Studies

In order to aid in designing the airframe, the team conducted a variety of trade studies for vehicle components. In completing these studies, the team researched different options for the respective airframe component. Trade studies were conducted for the overall airframe design, especially exploring the benefits of a single diameter versus a variable diameter launch vehicle, the nose cone design, transition section design and fabrication, fin shape and quantity, ballasting capabilities, and propulsion selection. These tasks will be further explained in the following sections.

3.2.1 Airframe

The general design of the airframe has cascading implications for systems integration, including weight and volume allowable for various payloads, and therefore several factors must be taken into consideration when outlining decision parameters. These include material, diameter considerations, and length. The trade offs between several options are discussed in more detail below.

3.2.1.1 Materials

In regards to the material of the airframe, the plausible options that are being considered are carbon fiber, fiberglass, and polypropylene. Each of these materials have their advantages and drawbacks in regards to what is needed to achieve the mission objective.

Regarding the nose cone, there are two viable options which are fiberglass and ASA plastic. Carbon fiber is not considered as a material for the nose cone due to the fact that it blocks radio signals to the payload, and therefore hinders communication between the aircraft and ground control. The advantages of using ASA plastic include the design customization due to it being a 3D printing plastic. In the past, fiberglass nose cones resulted in hardships with the sizing. On the other hand, this material has historically proven to be strong enough to endure flight forces. Both allow for transmission of radio signals.

Considering materials for the body tube, similar options surfaced. The most reliable materials for constructing the body tube are carbon fiber, and fiberglass. Both of these materials have advantages and drawbacks when used as the main material for the body tube of the launch vehicle. Compared to fiberglass, carbon fiber presents challenges such as cost, as well as being more difficult to manipulate. On the other hand, carbon fiber is strongest and can absorb shock better. That being said, it also presents itself as the lightest option, making it very appealing to be the body tube of this launch vehicle.

3.2.1.2 Dimensions

In terms of determining the shape of the airframe, the most critical consideration is whether to use a single or variable diameter for the body tubes. A single diameter body tube would render the need for a transition section obsolete, and be comprised of one 8-inch diameter body tube that extends the full length of the launch vehicle. This design would be fabricated entirely out of fiberglass, as it is challenging to find carbon fiber body tubes with an 8-inch diameter, and would be difficult to assign adequate volume to each payload with a 6-inch single diameter airframe. Comparatively, a variable diameter body tube would have a larger diameter of 8- inches behind the nose cone of the launch vehicle, which would then transition (through the use of a transition section) into a 6-inch diameter body tube. The 8 inch fore section, which houses the experimental scoring payload bay, must be fabricated out of fiberglass in order to ensure proper communication between the launch vehicle and remote sensors and controls. The 6 inch aft section would house the recovery and ABS neither of which require as much volume as the experimental payload bay. The fabrication of the transition section is discussed in depth in section 3.2.3 of this document, but it is notable that its inclusion would allow for sensors to be placed on the airframe without impacting the aerodynamic efficiency of the system.

There are benefits and drawbacks to both designs. The main benefit to a single diameter airframe is a decrease in length of the launch vehicle. In order to meet the requirement of an airframe which is 12 ft or less in length, it may be useful to have a larger diameter through the entire body of the rocket, allowing for some payload bays to have a larger volume for a set length. The drawback here is that fiberglass is heavier than carbon fiber, and therefore, the increase in weight overshadows in increase in length efficiency. Additionally, an entirely fiberglass body tube would be less expensive than that of carbon fiber, but the working budget is large enough to accommodate either design specification. If a variable diameter body tube is used, the largest benefit is the weight efficiency, which can be translated into a higher apogee for the same total payload weight. Additionally, the inclusion of sensors onto a transition section is a compelling reason to opt for the variable diameter airframe.

Ultimately, the most aerodynamically efficient option for the airframe is to use a variable diameter with an 8-inch fore section fabricated from fiberglass transitioning to a 6-inch aft section fabricated from carbon fiber, in order to maximize volume while minimizing weight in a feasible manner.

3.2.2 Nose Cone

In the selection of a nose cone, the design criteria considered were material properties, aerodynamic shape, and base diameter.

To ensure that the cone has minimal weight and enough strength to endure in-flight forces, ASA Plastic, fiberglass, carbon fiber, and polypropylene were among the materials considered. In the past, nose cones made of polypropylene have shown signs of warping when the shoulders were cut to integrate into the payload bays, adding to the risk of damage during landing. While carbon fiber was considered due to its highly increased levels of strength, fiberglass proved to be a better option because it offers a higher strength compared to polypropylene while remaining lightweight and inexpensive relative to carbon fiber. On the other hand, ASA plastic could easily be 3D printed and would provide a lightweight and customizeable alternative.

In determining the shape of the nose cone, the dictating factor was the reduction of pressure drag and frictional drag. Because the maximum velocity of the full-scale rocket will be below the transonic region of Mach 0.8, the pressure drag will be negligible. In order to minimize frictional drag, minimizing wetted area and shape discontinuity became the driving factors in choosing a shape. Therefore, either an elliptical or a tangential ogive shape were considered because both have similar surface area and offer a tangential contact point between the nose cone and payload bay. The tangential ogive shape was ultimately chosen due to its proven effectiveness based on historical data from years past.

Because the pressure drag on the nose cone is negligible, the fineness and bluffness ratios were lesser priorities when determining the nose cone size. The driving dimension, then, was the diameter of the cone such that it will fit the desired payload bay diameter of 8 inches. The length was found via the comparison of commercially available ogive-shaped nose cones. This ensured that dimensions where smooth and congruent with the ratio selected. A 3:1 ogive-shaped nose cone was modeled and the dimensions of it can be found in Table 5 below. Depending on the dimensions of the 3D printer that we have available the nose cone might have to be printed in two parts. This will be determined after talking to the facilities on campus.

Table 5: Nose Cone Dimensions

Dimension	Value	Units
Length	24	in
Shoulder Length	6	in
Weight	90	oz
Outer Diameter	8	in
Inner Diameter	7.815	in

3.2.3 Transition Section

In considering whether to implement a single or variable diameter into the airframe of the vehicle, it is necessary to consider the feasibility of incorporating a body tubes of various diameters with the use of a transition section to avoid any flow separation between diameters. The factors that must be taken into consideration with regards to the transition section include: material, source (student fabrication or purchase), the diameter of the upper and lower body tube, and flexibility of design.

Historically, the transition section of the airframe has not been required to be load bearing, which allows for some flexibility in terms of material. Whereas other load bearing sections are required to be fabricated from carbon fiber or fiberglass, the transition section may be constructed out of any reasonably light, cost effective material, such as a plastic. In terms of accessibility, plastic is readily available for use in student fabrication through the 3D printers in the student fabrication laboratory. Additionally, many ready made transition sections are fabricated using plastic. Therefore, should it become evident that the most favorable option is purchase of the transition section, it will not be necessary to custom order due to material constraints. The disadvantages of using plastic include failure rate. Should the section experience an unexpected force or load, plastic is more likely than carbon fiber or fiberglass to fail under that stress. This would result in a necessary replacement of the part; however, plastic is cost effective enough that this outcome is not prohibitive to mission success.

In considering source, one crucial factor is customizability. The two most viable options for source are: purchasing through a rocketry supply store, or fabricating using CAD modeling software such as CREO or SOLIDWORKS, and then 3D printing the part. The main advantage of 3D printing is customizability. Any change in the larger or smaller diameter body tubes would result in the need to change the transition section. Therefore, the ability to customize a part that perfectly fits the current model is vital. Additionally, 3D printing the part would allow for the convenient inclusion of an on-board camera. To avoid additional drag from the inclusion of a camera, a 3D printed transition section will be designed to smoothly integrate the camera into the airframe.

The size of the transition section camera shroud is negligible compared to the size of the entire airframe. The transition section camera shroud is also offset from the fins. Thus, the shroud and camera should not affect the performance of the launch vehicle. The on-board camera will document the behavior of the launch vehicle for flight inspection, social media posts, and corporate sponsorship purposes. To prove that the shroud and camera will not affect the performance of the launch vehicle, CFD analysis of the airframe will be conducted prior to CDR in order to ensure the transition section will not affect the performance of the launch vehicle. Notably, an on-board camera will be able to view the ABS system and this footage can be used to determine whether the ABS is functioning correctly.

Below, Figure 1 shows the tentative design of the transition section with the camera shroud. Additionally, Table 6 has the overall dimensions of the transition section.

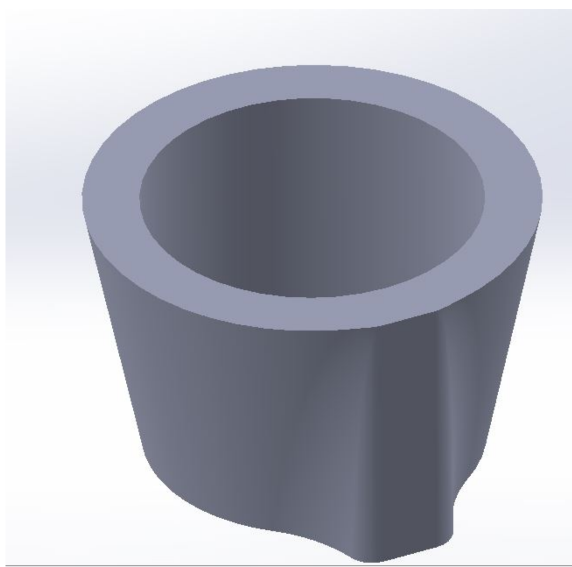

Figure 1: Transition Section and Camera Housing Preliminary Design

Table 6: Transition Section Dimension

Description	Dimension	Unit
Height	5	in
For Diameter	8	in
Aft Diameter	6	in
Camera Housing	2x1	in

3.2.4 Fins

In order to maintain dynamic stability throughout the flight, the fins will be attached at the fin can on the body of the launch vehicle. This will assure that the Center of Pressure remains aft of the Center of Gravity, which will result in a stable flight path and steady angle of attack. In order to reach optimal fin design, we consider different options for material used, airfoil shape, number of fins, and platform shape.

An optimal material for the fins must be strong enough to withstand the forces applied during all stages of the flight. The material must also be lightweight and inexpensive. Considerations for the fin material included carbon fiber and G-10 fiberglass. Available for purchase by Public Missiles Ltd., the G-10 fiberglass fin is cheaper, and can also be purchased pre-made and pre-sanded. Eliminating those steps will remove room for error, and will ensure exact measurements are reached without error. However, carbon fiber is lighter and very strong. While more expensive, carbon fiber has proven to be the most successful material for projects in the past. Therefore, in order to ensure structural integrity, the fins will be constructed using carbon fiber.

The options considered for platform shape included ellipse, trapezoid and parallelogram. Table 7 summarizes the factors considered when choosing the optimal shape.

Table 7: Fin Design Analysis

	Ellipse	Trapezoid	Parallelogram
Effectiveness at low Reynolds numbers	Low drag	Moderate drag	Low drag
Difficulty of construction	Difficult	Moderate	Simple

The best shape for our launch vehicle will be an isosceles trapezoid shape. The model of the fin shape can be seen in Figure 2, as designed on Creo Parametric 4.0.

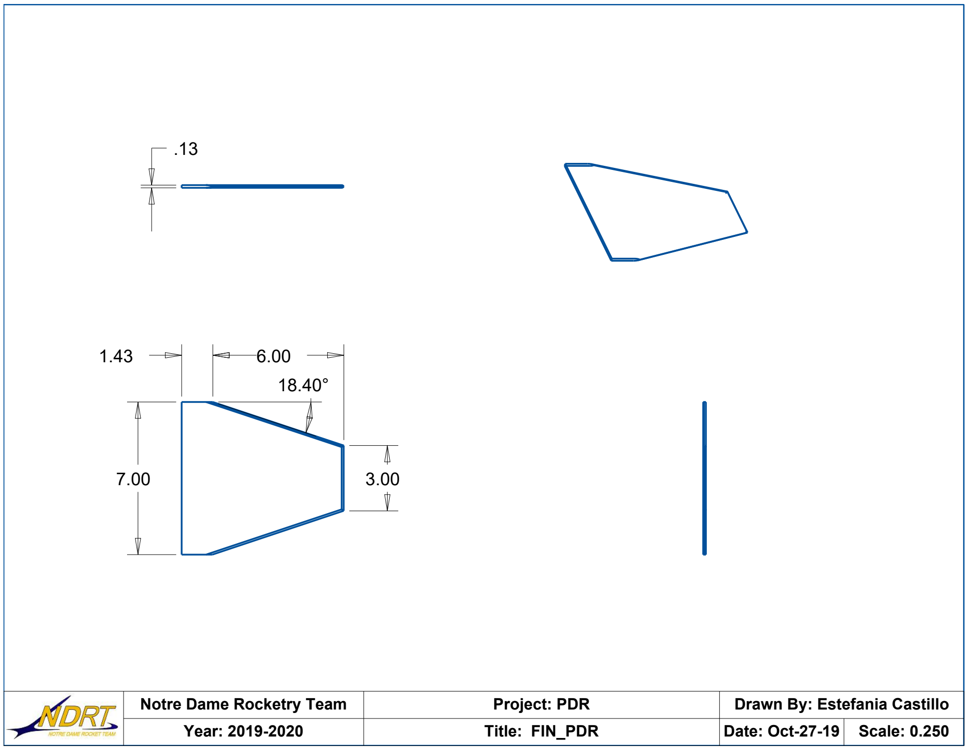

Figure 2: Leading Fin Design

At low Reynolds numbers, the optimum airfoil shape is a rounded leading edge leading to a pointed trailing edge, with a neutral camber to prevent uneven lift forces from acting on the fin surfaces. A symmetric airfoil shape modeled after a NACA 0010 will be sanded onto the fins.

In order to ensure stability and minimize material or weight added, the launch vehicle can have 3-4 fins. If more fins were added, no additional stability would be granted, and instead would be redundant. See Table 8 below for a comparison between 3 and 4 fins.

Table 8: Fin Design Considerations

3 Fins	4 Fins
Less additional interference drag	Higher additional interference drag
Difficult to attach symmetrically around the launch vehicle	Team experience with attaching 4 fins.

This launch vehicle will have 4 fins, which will ensure symmetry, stability, and ease of assembly when attaching to the launch vehicle. See Table 9 for specific dimensions and design choices for the fins.

Table 9: Leading Design Fin Dimensions

Material	Carbon Fiber
Platform shape	Isosceles Trapezoid
Root chord length	6”
Tip chord length	3”
Sweep length	1.5”
Tab length	6.5”
Thickness	0.125”
Number of Fins	4

3.2.5 Ballasting

The launch vehicle will include a ballast chamber, which will be filled with weights if ballast is needed. The ballast section will be located near the launch vehicle’s CG, so that any weight added to the ballast will not affect the launch vehicle’s stability by shifting the center of gravity. If all subsystems of the launch vehicle are operational, the vehicle’s weight is accurate to our simulations, and the vehicle’s predicted apogee is also accurate to simulations, then the ballast housing will remain empty during flight. If, on the other hand, additional weight is needed for optimal launch vehicle performance, then ballast weight will be added.

One construction option for the chamber would be to utilize CAD modeling such as Fusion 360, Creo Parametric 4.0, or SOLIDWORKS and fabricating it with a 3D printer. This allows for an increased amount of customizability. Given that the location of the ballast chamber has yet to be determined, the exact size is still unknown. Using CAD modeling would allow for any changes in diameter or height to be reflected in the design immediately without having to contact suppliers. Because the ballast chamber itself is not a load bearing component of the vehicle, any lightweight, reasonably strong material such as plastic will work. Plastic is inexpensive and readily available in the fabrication lab, meaning that in case of any unexpected damage to the ballast chamber in a test flight, it would be easy to replace. CAD modeling also allows for more versatility in the types of weights to be housed in the chamber.

The design of the ballast chamber needs to allow for a large amount of versatility in the amount of weight it can hold. The weights that will be utilized will most likely be Newton weights. The chamber itself will most likely have threaded bolts attached vertically to the bottom of the chamber. These will allow the weights to be secured by a nut. This will keep the weight stationary in the ballast chamber while in flight. Figure 3 below shows a tentative design of the ballasting section.

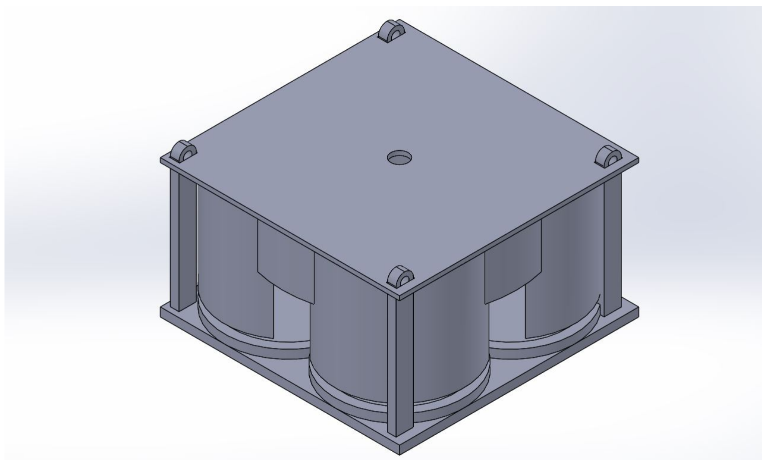

Figure 3: Leading Ballasting Model

3.2.6 Propulsion

To make a preliminary motor selection, a number of motor configurations were simulated on a model of the full-scale launch vehicle created in Open Rocket and RockSim simulation software. The major design criteria in the selection process of a motor were estimated apogee altitude and weight.

To estimate the altitude at apogee, the simulation software takes into account many parameters, including the vehicle shape, material finish, weight, and component density which have been updated from the proposal based on continued design from payloads. With the chosen target apogee of 4,444 ft, it was necessary to choose a motor that would result in an apogee comfortably above the target apogee in order to allow the Air Braking System (ABS) to precisely control the actual final altitude at apogee.

After many simulations with a number of Cesaroni, Loki Research, and Aerotech motors, the three L-class motors considered for the current configuration were the Cesaroni L1090SS-P, the Cesaroni L1395-BS-0, and the Aerotech L1300R-P.

The effects of the total impulse, maximum acceleration, and burn time were important features to consider in determining how the thrust applied by the motor will affect internal components through burnout. These quantities and other relevant motor information is listed below in Table 10. Based on historical team data, these have proven to be reasonable values that will not damage internal launch vehicle components and payloads.

Table 10: Relevant Motor Information

Manufacturer	AeroTech	Cesaroni Technology Inc	Cesaroni Technology Inc
Classification	L1300R-P	L1090SS-P	L1395-BS-0
Predicted Apogee for Variable Diameter Airframe [ft]	4211	4324	4841
Predicted Apogee for Single Diameter Airframe [ft]	3521	3624	4054
Diameter [in]	3.86	2.95	2.95
Length [in]	17.4	26.2	24.4
Propelant Weight [oz]	83.8	69.5	65.2
Loaded Weight [oz]	172	193	152
Average Thrust [lbf]	297	247	329
Maximum Thrust [lbf]	349	368	405
Total Impulse [lbf s]	1024	1054	1100
Burn Time [sec]	3.44	4.27	3.34

The Cesaroni L1395-BS-0 was finally chosen because it met the criteria listed above while giving the highest project altitude about 340 feet above the target altitude. Due to differences between simulation and physical performance, this additional potential apogee will allow ABS to have a marked effect in precisely controlling final altitude. Additionally, this motor takes up less volume inside the launch vehicle while also weighing the least of the three motors compared.

3.3 Leading Vehicle Design

After analyzing the benefits and drawbacks of various design elements in the above trade studies, an optimal design for the airframe was selected. Many criteria were considered, including cost, durability, feasibility, and access, and each decision was made to maximize benefit while minimizing drawbacks. Some overarching desirable characteristics of each aspect include low weight for high durability, which is necessary in order to provide adequate weight budgets to each payload without compromising the integrity of the part, feasibility for reasonable expense, and aerodynamic efficiency. Specifics of the final design are enumerated below.

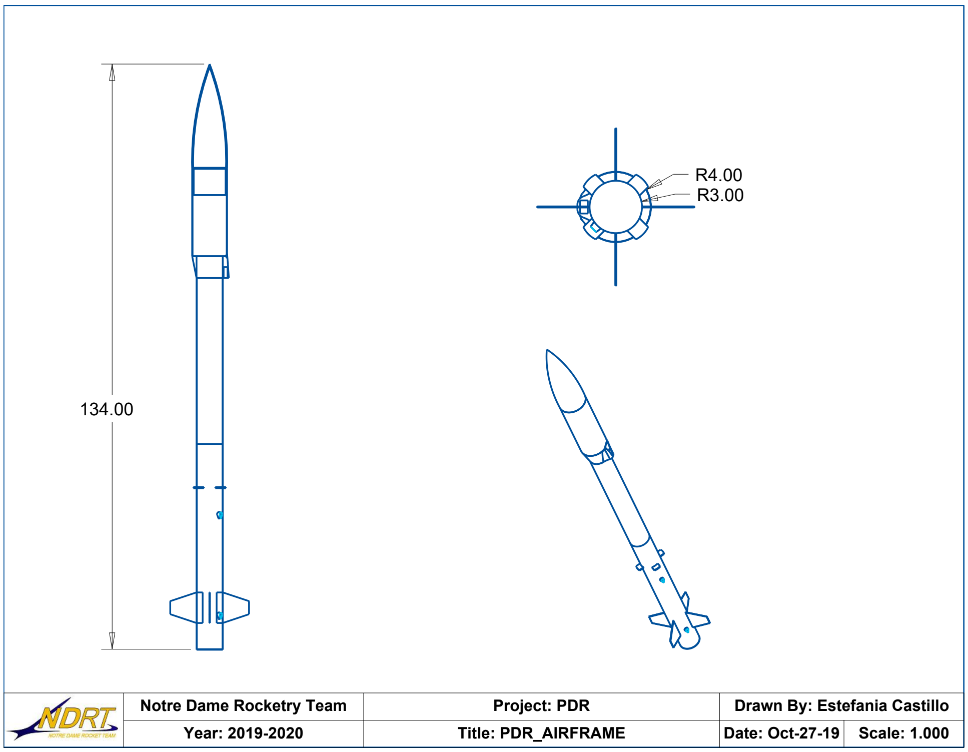

Figure 4: Leading Vehicle Design Drawing

3.3.1 Materials Selection

Fiberglass will be used for the nose cone due to its strength and relatively low cost. The materials selected for construction of the body tube are fiberglass and carbon fiber. These materials were chosen because of their strength and their previous success in similar scenarios. The reason that fiberglass is used in conjunction with carbon fiber is that the carbon fiber would block radio signals to the UAV payload, which requires such a signal.

The material selected for the fins is carbon fiber. As with the rest of the launch vehicle, carbon fiber was selected because it is strong, responds well to shock, and durable. The material selected for the centering rings and bulkheads will be determined after solids testing is done on plywood, fiberglass, and carbon fiber. For the couplers and motor mount, the team will be using carbon fiber. Carbon fiber provides more strength and reliability than other materials the team has used in the past, namely phenolic tubing. The team will also use various adhesives when constructing both the subscale and and full scale launch vehicle. Great Planes 30 minute epoxy will be used for the attachment of the phenolic portions for subscale production and Glenmare RocketPoxy will be used for attaching the carbon fiber and fiberglass pieces to the full scale launch vehicle. The motor will be attached with JB weld because of its high heat tolerance, an important factor when choosing motor adhesive.

3.3.2 Vehicle Layout

Found below, Table 11 outlines the proposed compositions and purposes of the subsections of the launch vehicle. Figure 5 below displays the layout of the sections and subsections.

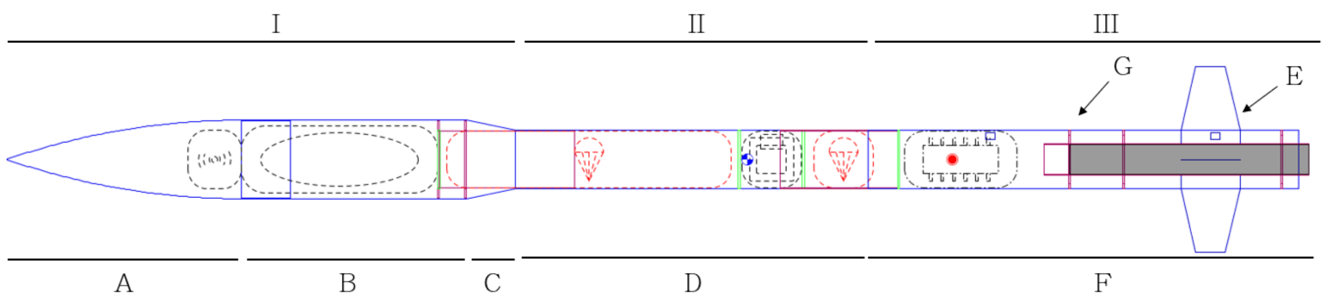

Figure 5: Leading Vehicle Design Breakdown

Table 11: Vehicle Breakdown Description

Section	Sub-Section	Label	Composition	Description
I	Nose Cone	A	Hollow ASA Plastic nose cone with metal tip. Length of 24" and a diameter of 8".	Foremost component. Connected to the payload bay.
	Payload Bay	B	Fiberglass payload bay with a length of 20", an outer diameter of 8.005", an inner diameter of 7.815", and a wall thickness of .095 inches.	Contains Lunar Vehicle payload and connects to the transition section.
	Transition Section	C	The 3D printed section is 5" in length with a fore diameter of 8" and an aft diameter of 6 inches.	Transition section between payload bay and recovery tube.
II	Recovery Tube	D	Made of carbon fiber, it is 44" in length with a cram length of 6", diameter of 6", and a thickness of .056".	Holds main parachute and CRAM (Compact Removable Avionics Module)
III	Fins	E	Made of G-10 fiberglass	Provides aerodynamic stability. Connected to fin can.
	Fin Can	F	Carbon fiber fin can, 42" in length with a diameter of 6" and a thickness of .056 inches.	Secures fins and contains motor mount and ABS.
	Motor Mount	G	24" in length with a diameter of 3.2" and a thickness of 3 inches.	Secures the motor inside the vehicle.

3.3.3 Detailed Mass Statement

A detailed mass statement of parts and subsystems is listed below in Table 12.

Table 12: Detailed Section and Part Masses

Part Name	Mass [oz]	Material
Nose cone	35	ASA Plastic
GPS Solution	48	Various
Payload bay	58	Fiberglass
Payload	100	Various
Transition Section	12	ASA Plastic
Recovery Tube	39	Carbon Fiber
Parachutes	76	Ripstop Nylon
CRAM	53	Various
ABS	70	Various
Fin Can	47	Carbon Fiber
Fin (each)	4.5	Carbon Fiber
Motor mount	41	Carbon Fiber
Motor	152	Vaiour
Tube coupler (each)	13	Carbon Fiber
Ballast (max)	80	Newton Weights

Material Properties can be found below in Table 13.

Table 13: Material Densities

Material	Density (oz/in^3)
G10 Fiberglass	1.51
Carbon fiber	0.91
6061 Aluminum	1.56
ASA Plastic	0.255

3.3.4 Propulsion

After having considered three motors the selected motor was the Ceseroni L1395-BS. It was selected due to its fast burn time which increases stability and because of its high apogee prediction which allows for ABS enough overshoot to actuate. The expected thrust curve is found below in Figure 6.

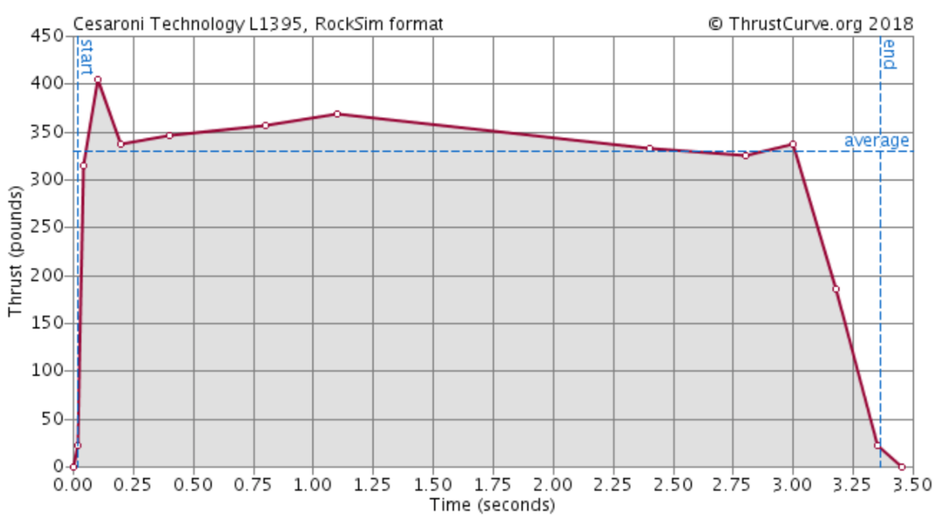

Figure 6: Thrust Curve for Cesaroni L1395

3.4 Air Braking Subsystem

3.4.1 Overview of ABS

The purpose of the Air Braking System (ABS) is to implement a control system for inducing a variable drag force in order to meet the target apogee of 4,444 feet. To achieve this, a set of drag surfaces, hereby referred to as drag tabs, will be extended from the body of the launch vehicle to increase the acting drag force, therefore decreasing the projected apogee, until the target has been achieved. The system will use a microcontroller to keep track of altitude and velocity sensor data and run a closed loop PID control algorithm to adjust the extension of the drag tabs until the predicted apogee matches the target apogee. The microcontroller will adjust the extension of the drag tabs by controlling a servo motor, which will drive a mechanism to actuate the drag tabs. The success of the ABS will be evaluated based on the following criteria:

- The launch vehicle shall achieve an apogee within ± 25 feet of the target apogee.
- The drag tabs shall extend at a location no more than ± 1 inch from the center of pressure.
- The drag tabs shall not actuate until burnout has occurred.
- The drag tabs shall retract completely while the projected apogee is at or below the target apogee, and after actual apogee is detected.

3.4.2 ABS Aerodynamic Design

The location of the ABS within the launch vehicle will have significant implications for stability. It is ideal for the drag tabs to actuate at the exact location of the CP, because the added pressure from the induced drag force would otherwise alter the location of the CP, therefore changing the stability margin. The ABS will be located within the fin can because this is where Openlaunch vehicle models show the CP to be, and the mechanism will be situated to ensure that the drag tabs will extend at a location no more than ±1 inch from the CP of the launch vehicle. Additionally, the drag tabs must not generate unsteady flow over the fins, as this would also alter the stability of the launch vehicle, so the tabs will be offset relative to the fins such that the flow that passes over the fins is unaffected by the drag tabs.

In order to make apogee calculations more accurate, the air braking system will not deploy its tabs until burnout has been detected. This ensures that gravity, drag due to the launch vehicle, and drag due to the drag tabs are the only forces that the ABS will need to take into account when using kinematics to predict apogee. In order to model the drag force that the tabs will induce during flight, the team will use the drag equation

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

Where D is drag force, ρ is the density of air, v is airspeed, A is the surface area of the drag tabs relative to the direction of airflow, and C_d is the drag coefficient. In order for the ABS to induce as much drag as possible, the drag tabs will be designed to have a large drag coefficient. For preliminary modeling, the tabs have been assumed to be flat plates perpendicular to the direction of flow, yielding a constant theoretical drag coefficient of 1.28. This value will be adjusted based on measured results obtained from wind tunnel testing and subscale launches.

CAD models show that the system will be capable of producing a total drag tab area of 9.20 in^2 at full extension. In order to verify that this will be sufficient to reduce the projected apogee to the target, a MATLAB model of the flight profile from burnout to apogee was generated with initial conditions based on Openlaunch vehicle simulations at burnout. For the selected motor, the Cesaroni L1090SS-P, which is projected to being the launch vehicle to an altitude of 4,832 feet without the ABS, the MATLAB model shows that the launch vehicle can be brought to 4,461 feet with drag tabs at half extension, and 4,261 feet with drag tabs fully extended for the entire duration of flight. The generated flight profiles for these three scenarios are shown in Figure 7.

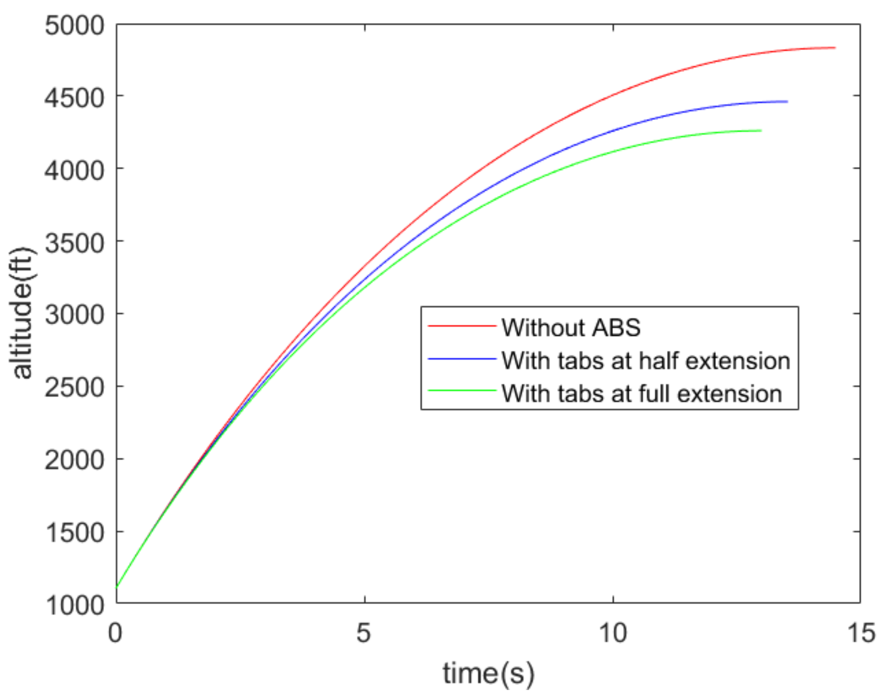

Figure 7: Effect of ABS drag tabs on flight profile

3.4.3 ABS Mechanism

The purpose of the ABS mechanical system is to control the distance the drag tabs extend from the launch vehicle body. Precise, active control is required to provide sufficient induced drag upon the launch vehicle in order to reach a set apogee. These drag tabs must be deployed in a manner that does not produce any moments or destabilizing forces for the launch vehicle. Only a drag force parallel and opposite to the velocity vector of the launch vehicle should be induced by the drag tabs, which will be achieved by designing a mechanism that deploys the drag tabs at equal lengths symmetrically around the body of the launch vehicle. The tabs must also have a large surface area perpendicular to the airflow in order to provide a large induced drag force. Additionally, larger tabs will minimize the effects of systematic error from the code and the mechanical system, as larger tabs will have to deployed for less time than smaller tabs.

3.4.3.1 Mechanical Design Options

The goal of the ABS mechanical design is to achieve precise, accurate, and active control for extending and retracting drag tabs from the launch vehicle body during flight. For this year's ABS mechanism, three systems were examined and compared. The first mechanism under consideration uses tabs that sit parallel and flush to the outside of the launch vehicle body and are deployed radially outwards. This method will be referred to as the Radial Displacement Mechanism (RDM). The second mechanism under consideration linearly displaces the tabs from within the body using a central rotating hub, with linkages that translate the rotation to linear extension. This mechanism will be referred to as the Linear Displacement Linkage Mechanism (LDLM). The third mechanism also linearly displaces the tabs, but uses grooves cut in the tabs that interface with a central hub. This mechanism will be referred to as the Linear Displacement Groove Mechanism (LDGM).

The first option is the Radial Displacement Mechanism (RDM). This mechanism uses a stepper motor to turn a central lead screw that runs parallel to the length of the launch vehicle. Along this lead screw rides a hub that can move up and down depending upon the rotation of the lead screw. Connected to this hub are arms that connect to tabs that sit flush with the outside of the launch vehicle body. As the hub moves upwards, the arms push the tabs radially outward. The design and movement can be seen in Figure 8.

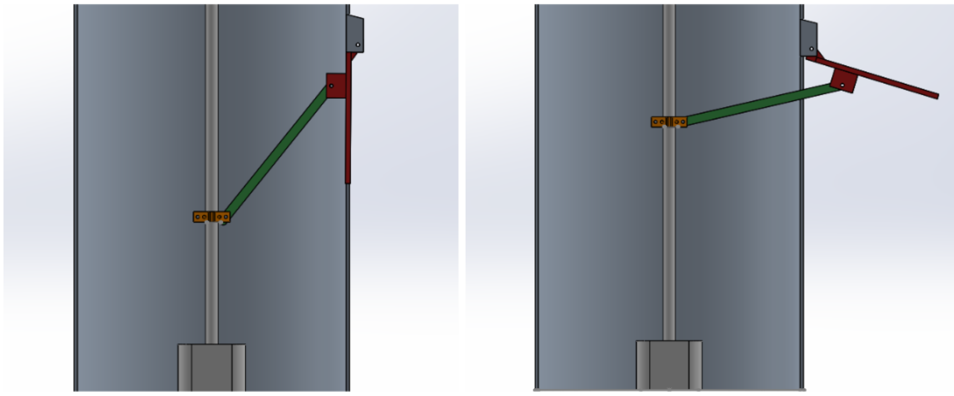

Figure 8: CAD Model of the Radial Displacement Mechanism

The second option under consideration is the Linear Displacement Linkage Mechanism (LDLM). This mechanism uses a central servo motor to turn a central hub. Attached to this central hub are four linkages that attach to the four drag tabs. These tabs sit within a plastic disk with four slots cut into it. When the servo turns the hub, the linkages force the tabs linearly outwards within the grooves. The design and movement can be seen in Figure 9.

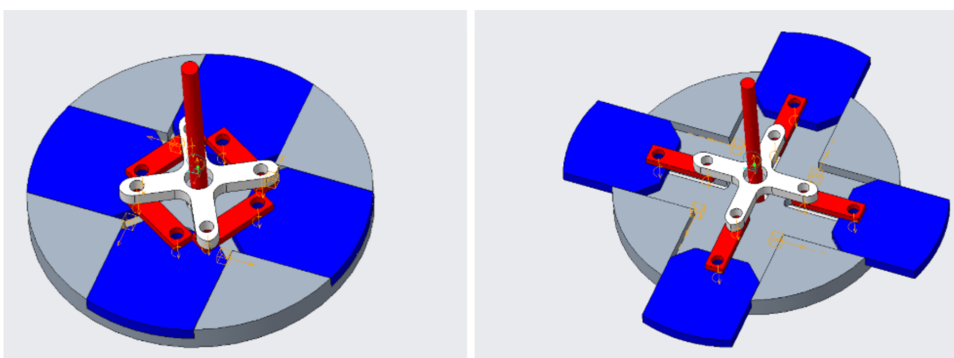

Figure 9: CAD Model of the Linear Displacement Linkage Mechanism

The third option under consideration is the Linear Displacement Groove Mechanism (LDGM). This mechanism uses a servo and hub similar to the LDLM. However, in this system, grooves are cut in the tabs that interface with a central hub. The grooves are cut at an angle which allows the tabs to move linearly outwards when the servo rotates the hub. The design and movement can be seen in Figure 10.

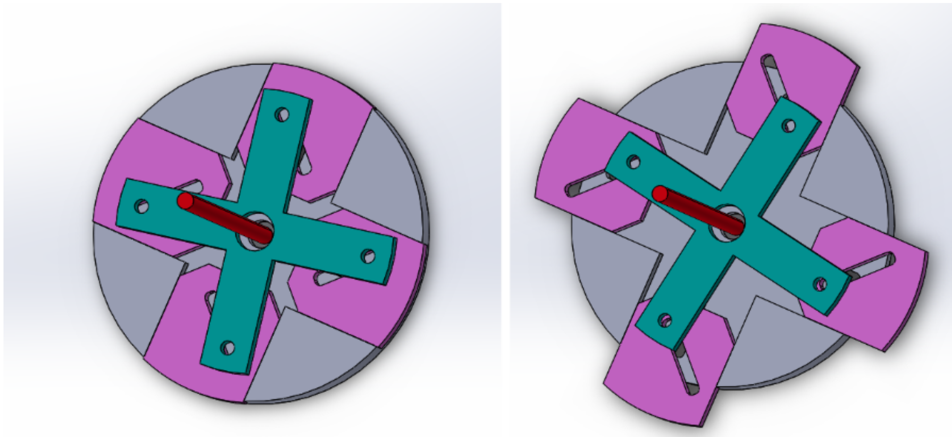

Figure 10: CAD Model of the Linear Displacement Groove Mechanism

3.4.3.1.1 Mechanical Design Trade Study

The three mechanical design options above were compared in a trade study, comparing manufacturability, cost, precision, ability to produce large drag tab areas, speed of deployment, complexity of computations, and power consumption. The RDM design provides a size advantage, as a larger tab area can be extended to generate more drag. The RDM converts large rotational movements of the stepper motor to small changes in the angle, meaning a high amount of precision can be achieved. However, this design is also heavier, more complex, and takes up more space compared to other designs. Lead screws are quite heavy and the arms connecting the hub to the tabs require large amounts of space to move. Additionally, the drag coefficient is not constant during the radial displacement, as the angle of the drag tabs with respect to the airflow changes. This greatly increases the complexity of the coding and calculations, and would require extensive wind tunnel testing in order to determine a useful relation between drag coefficient and angular displacement. Furthermore, the flaps do not fully retract into the system, as they must sit flush with the outside of the launch vehicle body, which prevents the system's ability to be easily inserted into and removed from the launch vehicle, which is required in order to make adjustments and charge the batteries.

The LDLM offers a more compact ABS mechanical system. Another benefit of this design is that the extension of the drag tabs is linear, therefore the drag coefficient can be assumed to be constant, which simplifies calculations of the drag force. This assumption will be verified through wind tunnel testing. The LDLM can also be easily inserted into and removed from the body tube because it does not include any components that will interface with the launch vehicle body. However, this design limits the amount of induced drag upon the launch vehicle, as the tabs must be small enough to retract completely into the body of the launch vehicle.

The LDGM requires fewer moving pieces compared to the RDM and the LDLM. However, the system has numerous disadvantages. The interfacing of a pin and groove provides less mechanical advantage compared to the linkages in the LDLM, meaning that a larger torque would be required from the motor. Additionally, due to the groove cut along the length of the tabs, the LDGM cannot extend the drag tabs as far as either of the other two designs, limiting the drag force it is able to induce. The three mechanisms under consideration were compared in a trade study, shown in Table 14.

Table 14: Mechanism Design Trade Study

Criteria	Weight	RDM	LDAM	LDGM
Manufacturability	15%	4	5	3
Cost	10%	4	6	6
Precision/Strength	15%	7	4	3
Effective Surface Area	10%	9	4	2
Speed of Deployment	10%	3	5	5
Testability	10%	3	8	8
Weight/Size	10%	3	6	6
Complexity	15%	4	5	7
Power Consumption	5%	9	3	3
Total		4.9	5.15	4.8

Based on the trade study results, the LDLM system was chosen as the mechanism to deploy the tabs from the launch vehicle body. The RDM added too much complexity, took much longer to deploy fully compared to the other systems, and is not removable from the body tube. Comparing the LDLM to the LDGM, the LDGM fell short due to its smaller effective surface area and the difficulty of precise manufacturing required. Overall, the LDLM proved to be the best solution. Previous NDRT launch vehicles employed similar systems with no major issues with the mechanism arising. Therefore, the tabs will be deployed using the Linear Displacement Linkages Mechanism.

3.4.4 ABS Material Selection

The materials selected for both the drag tabs and the mechanical system must be able to endure in-flight forces, be easily machinable using the resources available to the team, have a low coefficient of friction so that the friction they create is not too much for the torque of the motor to handle, have a low density in order to minimize weight, and be relatively inexpensive. The team decided to machine the parts in house for greater control over the tolerances of the parts and to reduce the cost of production. The two machining methods most readily available to the team are 3D printing using a Makerbot Replicator+ and CNC milling using a Techno Router, which were compared in a trade study shown below in Table 15. The items considered during the trade study were manufacturing cost, compatibility of machining method and materials, manufacturing time, and machine tolerance.

Table 15: Machining method

Criteria	Weight	3D Printing with Makerbot Replicator+	Grade	CNC Milling with Techno Router	Grade
Material Compatibility	50%	Only PLA, ABS	2	Many options	**9**
Tolerance	30%	Tend to deform when cooling	5	0.01% of the total dimension	**7**
Time	10%	A few hours	5	About an hour	**8**
Manufacturing Cost	10%	$0 for printing and materials	**10**	$0 for cutting and cost of materials	7
Weighted Score	—	—	4	—	**8.1**

Using the results of the trade study, the team decided to use the Techno Router (weighted score of 8.1) because it is compatible with many more plastics, takes less time to complete, and will allow for greater precision and accuracy in machining. The price of materials is the only difference between the manufacturing cost of 3D printing and milling the part, which the team considered a fair trade-off.

3.4.4.1 Drag Tab Material Trade Study

A wide range of materials, including metals, woods, and plastics, are compatible with the Techno Router, but the team decided to limit the search to plastics because of the easy machinability, low cost, and low density of the material. The other major factor in considering the material of the drag tabs is the coefficient of friction, because it is important that the torque provided by the servo motor is sufficient to overcome the friction generated by the actuation of the drag tabs. In order to determine material strength criteria, a Finite Element Analysis of one drag tab was performed using ANSYS to estimate the maximum von-Mises stress under the worst-case scenario loading condition, in which the drag force is approximately 7 lbf, and the drag tab is supported as a cantilever. This simulation calculated the maximum von-Mises stress the drag tab will undergo to be approximately 1231 psi. An image of the drag tab and the stress distribution generated by the ANSYS software is shown below in Figure 11.

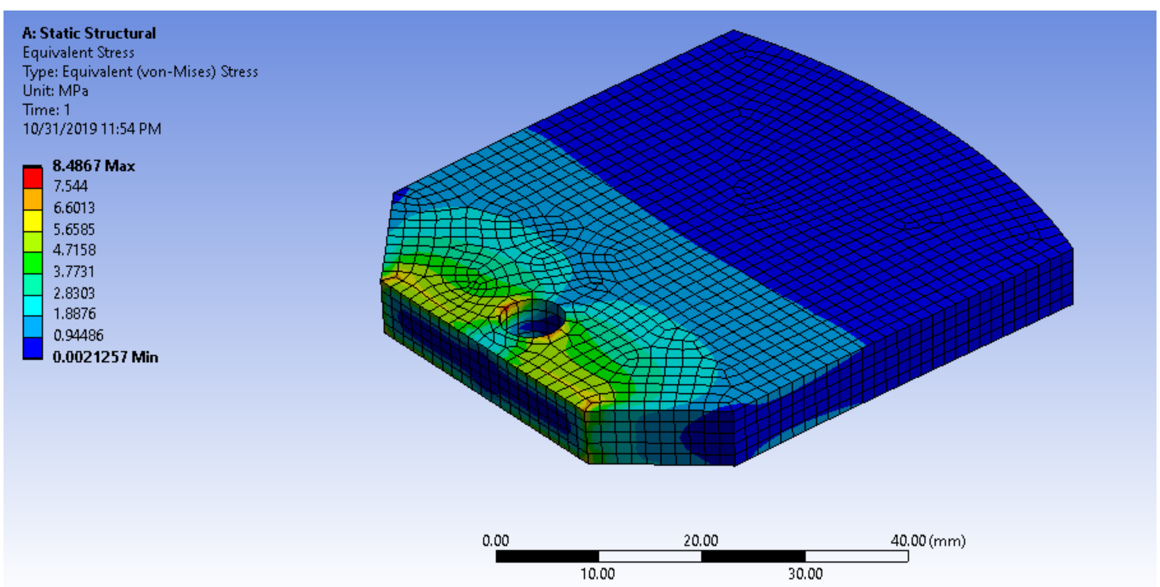

Figure 11: von-Mises stress for worst-case loading of a drag tab

Several materials that are easily machinable using the Techno Router were compared based on cost, factor of safety calculated from the predicted maximum stress of 1231 psi, density, and coefficient of friction, with scoring weights shown in Table 17. These criteria were the basis for the trade study shown in Tables 16 and 18 to determine the material of the drag tabs. According to a NASA report on structural design requirements for spaceflight hardware, the factor of safety for nonmetallic flight structures of a launch vehicle must be at least 2.0 at discontinuity areas. Nearly all materials considered exceeded this factor of safety except for PTFE, which also had a high cost and density. The material selected last year was Delrin, which has a high yield stress and low coefficient of friction; however, the high cost of the material outweighed the benefits of its marginally lower coefficient of friction. Instead, the preliminary choice for the drag tab materials is Nylon 6/6. Nylon 6/6 has the optimal values for each of the criteria set by the team, and lubrication will be used to further lower the coefficient of friction between the moving components.

Table 16: Trade study of drag tab materials

Material	Cost (6"x6" sheet)	Grade	Yield Stress [psi]	Factor of Safety	Grade	Density [g/cm^3]	Grade	Coefficient of Friction	Grade
Delrin 150	74.45	3	10500	8.5324	10	1.412	6	0.25	8
HDPE	7.57	9	4060	3.2992	10	0.958	9	0.31	7
ABS	9.67	9	4650	3.7786	10	0.969	9	0.35	7
Nylon 6/6	14.97	9	11750	9.5482	10	1.135	8	0.26	8
Acrylic	8.97	9	10625	8.634	10	1.19	8	0.8	3
Polycarbonate	10.56	9	9200	7.476	10	1.246	7	0.5	5
TECAFORM AH	58.35	4	9300	7.5573	10	1.41	6	0.21	8
PTFE	103.42	2	2250	1.8284	5	2.15	2	0.1	9

Table 17: Scoring weights for criteria

Criteria	Scoring Weight
Cost (6” x 6” sheet)	40%
Yield Stress & Factor of Safety	10%
Density	20%
Coefficient of Friction	30%

Table 18: Final material scores

Material	Final Score
Delrin 150	5.8
HDPE	8.57
ABS	8.5
Nylon 6/6	8.6
Acrylic	7.1
Polycarbonate	7.5
TECAFORM AH	6.2
PTFE	4.4

The components of the mechanism will also be machined from Nylon 6/6. This will ensure that all of the contacting surfaces in the mechanism will have low friction when the tabs actuate.

3.4.5 ABS Electrical Design

In order to design a successful Air Braking System, it is necessary to select sensors, actuators, and microcontrollers that can be interfaced with one another. It is important that the sensors the team selects provide accurate data at a high rate, and that they provide the data in a format that can interface with a microcontroller. Actuators, most notably servo motors, need to be able to provide sufficient torque for the actuation of the drag tabs, and need to respond to signals in a quick and consistent way. The microcontroller will be used to process the data collected from the sensors, and then use that processed data to control the servo within the system. The microcontroller must have a high processing speed, and needs to be able to handle several data inputs while controlling the servo motor using a PWM signal. In order to select all of the electrical components within this system, the team conducted several trade studies in order to create an electronic system that will allow the ABS to succeed.

The team is also considering whether to utilize a printed circuit board (PCB). A printed circuit board would allow the sensors to be connected to the microcontroller using soldered pins, which would be much more durable than loose wires. The main factors that are affecting our decision include the cost of manufacturing the PCB and the restriction of movement that comes with attaching sensors to the PCB. The issue of This decision will be made once the sensors arrive and we have a finalized concept of how the system is going to be structured.

3.4.5.1 Accelerometer Selection

One of the sensors being used in this system is an accelerometer. The accelerometer will provide real-time acceleration data with respect to 3 axes, which allows the ABS to track how the motion of the launch vehicle is changing over time. Ideally, this sensor will also provide information about the orientation of the launch vehicle, which can be used in the calculation of the projected flight apogee. The team is choosing the BNO055 for the accelerometer for a variety of reasons. This sensor provides data at 100 Hz, which will provide enough samples for the system to alter the drag if necessary, and it provides more than just 3-axis acceleration. This sensor provides useful information such as orientation, linear acceleration (without gravity), and acceleration with gravity. These values are going to be critical in the algorithms that are going to be used in ABS, and the BNO055 is the only reasonably priced sensor available that made this information accessible. It is possible to calculate some of these values by utilizing a 3-axis accelerometer and calibrating it with respect to gravitational acceleration, but the team decided that it would be much more efficient and reliable to implement a sensor that presents this information as raw data. Orientation is critical in case the launch vehicle is travelling at an angle rather than directly upward, because the apogee can be significantly altered by this difference. This accelerometer is also easily programmable, and different sensors can be activated/deactivated if necessary. These different metrics can be used to improve the system's accuracy, and all of these data points are provided at 100 Hz. This accelerometer is expensive compared to other options, but the team has determined that the benefits justify the cost. The chosen BNO055 accelerometer is shown in Figure 12.

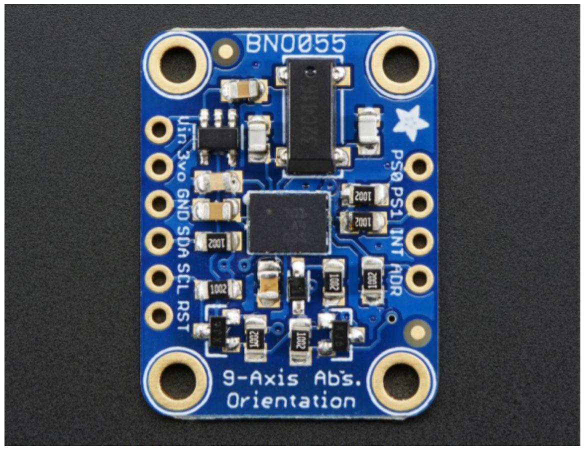

Figure 12: BNO055 accelerometer

3.4.5.2 Barometer Selection

Another sensor that is going to be utilized within the system is a barometer, which will allow for real-time sensing of air pressure. This air pressure reading can then be used to calculate the altitude and velocity of the launch vehicle, which are important metrics in determining how much to extend the drag tabs. When considering different barometers, accuracy, ease of implementation, and sampling rate are very important. These factors were compared for two candidate barometric pressure sensors in the trade study shown in Figure 19.

Table 19: Barometer Trade Study

Criteria	Weight	MPL3115A2	Grade	BMP388	Grade
Ease of Implementation	20%	Python Interface	9	Python Interface	9
Sampling Rate (Hz)	30%	100	5	200	8
Accuracy (meters)	40%	0.3	9	0.5	6
Cost	10%	$10	7	$10	7
Score		7.6		7.3	

From this trade study, the MPL3115A2 edges out the BMP388, even though they are both good choices. The MPL3115A2 has a slightly slower sampling rate, but it is more accurate. This difference in accuracy can result in a significant difference in the system's projected apogee calculation, so accuracy is key. An image of the MPL3115A2 is shown in Figure 13.

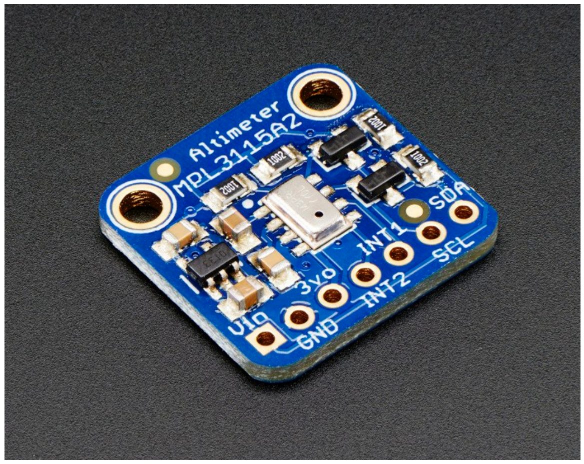

Figure 13: MPL3115A2 barometer

3.4.5.3 Servo Motor Selection

The selection of a servo motor is crucial to this system's success, as it is the actuator that controls the movement of the tabs. In order for the ABS to work properly, the servo needs to be able to provide enough torque to overcome the reactionary friction force that the drag tabs will generate during extension, and it needs to provide accurate rotations to ensure that the actual drag tab extension does not deviate from the extension expected by the system. In the trade study below in Table 20, various servos are analyzed.

Table 20: Servo Motor Technical Specifications

Criteria	D845WP	D980TW	D950TW
Weight (oz)	8	2.76	2.4
Speed at 7.4 V (sec/60°)	0.17	0.17	0.14
Torque at 7.4 V (oz-in)	694	611	486
Cost	$100	$170	$130
Max Travel (degrees)	202.5	120.5	118.5
Operating Current Draw (A)	1.6 A	0.5 A	0.5 A
Durability	Water/Dust Resistant	Splash Proof	Splash Proof

The three servos under consideration were the D845WP, the D980TW, and the D950TW. The main factors that differentiated these servos were: weight, cost, operating current drive, and torque. When looking at all of these factors, the team decided that the D845WP best suits the current design constraints. It provides a very high torque, which is ideal, but this high torque comes at the cost of high weight and current draw. The weight of 8 oz was determined to be acceptable within the ABS weight budget, and the high current draw can be compensated for by activating the servo on the launchpad to preserve battery life. Additionally, the D845WP provides its own internal feedback via a 5 kΩ potentiometer, which will be crucial for ensuring that it executes the expected rotations.

3.4.5.4 Microcontroller Selection

The microcontroller used in the ABS will take inputs from the accelerometer and barometer, filter the data, run a PID control algorithm to determine the desired extension of the drag tabs, and output a PWM signal to the servo motor to actuate the drag tabs. The microcontroller is one of the most important aspects of the ABS, and so the team is taking care in selecting which microcontroller to use. Some things that are necessary in a microcontroller are high sampling and output rate, ability to interface with sensors, and ability to run the code that is written.

There are two microcontrollers being considered currently: the Arduino MKR Zero and the Raspberry Pi. The team has used the MKR Zero in previous years, and it has proven to be a solid option, but this year the team is considering shifting to a Raspberry Pi. A Raspberry Pi has higher processing power, can run code written in Python, and has libraries that will allow the team to develop the system comparatively easily (PyKalman, CircuitPython, etc.). These aspects are very appealing to the team, but a Pi cannot simply be powered by a standard battery; in order for the Pi to function properly, it needs to have a source that provides a constant voltage and current. This involves the use of a battery pack, which adds weight to the system.

The team has not yet decided which microcontroller is going to be implemented. This decision will be made after gathering data from the sensors in subscale, and then running this data through some tests to see which algorithm best suits the team's needs. If a high powered algorithm such as the Kalman filter is chosen, then the Pi will be necessary in order to properly implement it. However, if it is determined that a simpler algorithm such as an averaging filter is sufficient, then the Arduino MKR Zero will be easier to implement and test. In conclusion, more testing is required before reaching a final decision regarding the selected microcontroller.

3.4.5.5 Battery Selection

In the Air Braking System this year, two batteries will be necessary in order to properly power the entire system. One battery will be used to power the servo motor, and the other will power the microcontroller and sensors. To match the required specifications for the chosen D845WP servo motor, the selected battery must supply a voltage of 7.4 V, and must be rechargeable for use in multiple tests and launches. Primary factors under consideration for the battery are size, weight, cost, and capacity. Three batteries under consideration for powering the servo motor are shown in Table 21, comparing the specifications for each one.

Table 21: Battery considerations for the Air Braking System

Battery Name	Capacity [mAh]	Voltage [V]	Discharge Rating [C]	Mass [g]	Dimensions [mm]	Price [$]
Tenergy LiPo Battery, for Syma X8C X8W X8G	2200	7.4	30	120	90.5mm x 36.5mm x 19.8mm	14.99
Tenergy Li-ion 18650 Rechargeable Battery	2200	7.4	30	97	71mm x 37mm x 19mm	14.99
Turnigy nano-tech 2S2P Hardcase Lipo Pack	6000	7.4	65	313	138mm x 46mm x 25mm	44.83

Based on the specifications shown, the Tenergy Li-ion 18650 Rechargeable Battery is currently the leading choice to power the servo motor, due to the advantage of its low price, comparatively low weight, and its ability to power the servo motor for a sufficient run time given a capacity of 2200mAh.

If the Raspberry Pi is selected as the team's microcontroller, a separate battery pack is needed to ensure that there is a constant current supply. The operating voltage needed is 5V. To serve this purpose, the team has selected the MakerFocus 3800mAh Lithium Battery for Raspberry Pi 3 due to its affordable cost and reasonably small dimensions and weight. Its specifications are shown below in Table 22.

Table 22: MakerFocus 3800mAh Technical Specifications

MakerFocus 3800mAh Lithium Battery	
Cost	$23.99
Weight	50 g
Dimensions	86 x 56 x 18mm

If an Arduino MKRZero is used rather than the Raspberry Pi, then the team will need to purchase a 3.7 V LiPo battery with a minimum capacity of 700 mAh. In this case, the team would purchase a battery that has a capacity of either 2000 mAh or 2500 mAh, which would cost $10-15 from Adafruit Industries. The final decision on batteries will depend heavily on which microcontroller is chosen.

3.4.6 Integration of System Components

The ABS will be separated into two main sections: a mechanical compartment and an electrical compartment. The mechanical subsystem will consist of the mechanism, the servo motor, and the battery powering the servo motor, while the electrical compartment will house the microcontroller, the battery powering the microcontroller, and the sensors. The two components will be separated by a bulkhead and a thin sheet of copper. The copper sheet will prevent the magnetic field due to the current supplying the servo motor from interfering with the sensors in the electrical subsystem, which was an issue in last years' system. This system layout is shown in Figure 14, where the electrical components are housed in the upper section, while the mechanism, servo motor, and its battery are housed in the lower sections.

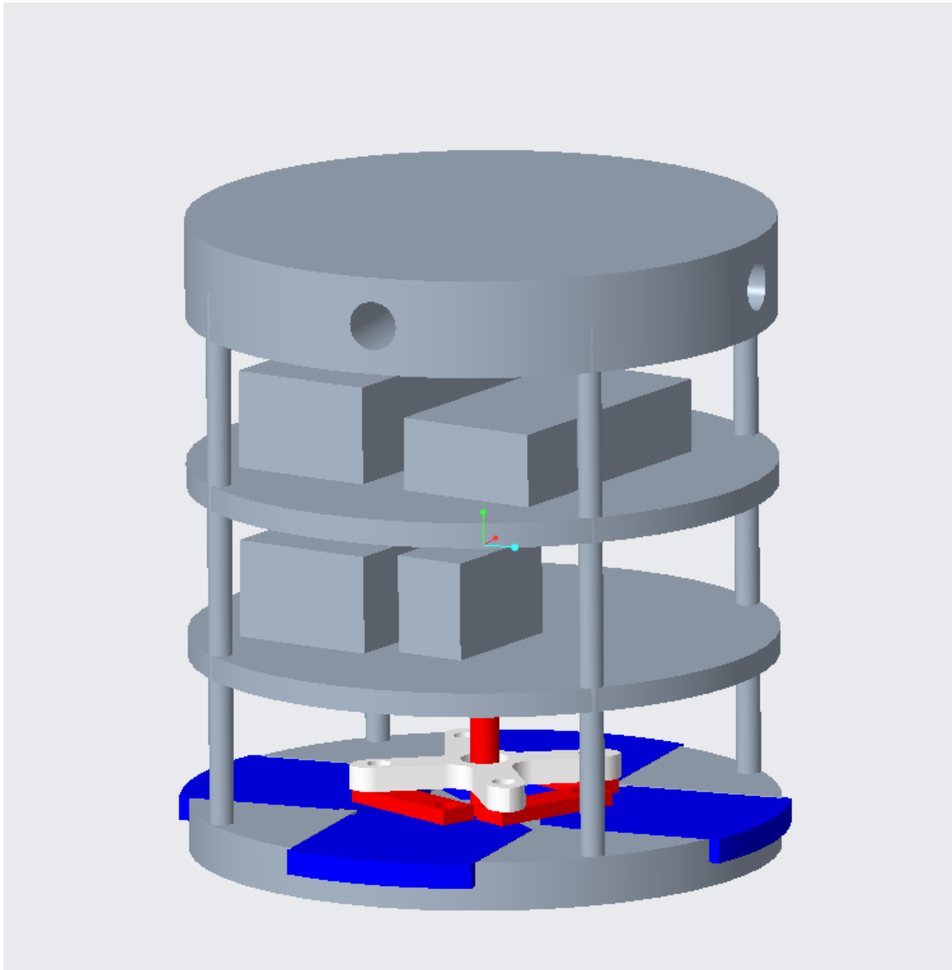

Figure 14: ABS Integration of Components

3.4.7 ABS Control Structure

The ABS control code first activates on the launchpad, giving visual confirmation through LED status lights that it is acquiring data from the accelerometer and barometer. The system will be able to write to an SD card in order to provide detailed logs of the sensor data and filter outputs. This connection to an SD card will also be indicated by an LED. Upon activation of the arming switch, a third LED will indicate that the system is armed. Sensor data will then be collected continuously and analyzed by a data filter. The system will first detect when liftoff has occurred. Once liftoff has occurred, the system will again use data fed into the filter to determine when burnout has occurred.

After burnout, the filtered data will be read into a proportional-integral-derivative (PID) controller to estimate the optimal drag tab extension. The launch vehicle's velocity at its current altitude will be compared to the velocity at that altitude of a pre-calculated ideal flight; the difference between these two values produces an error value which the algorithm then attempts to compensate by sending a signal to the servo to change drag tab extension.

The system will act as a closed-loop controller, continuously recalculating a new drag tab extension based on the error and communicating that extension to the servo motor controlling the drag tabs. This process terminates when sensor data indicates that the launch vehicle has reached apogee, at which point the drag tabs will retract for the remainder of the flight. A flow chart of the ABS control structure is shown in Figure 15.

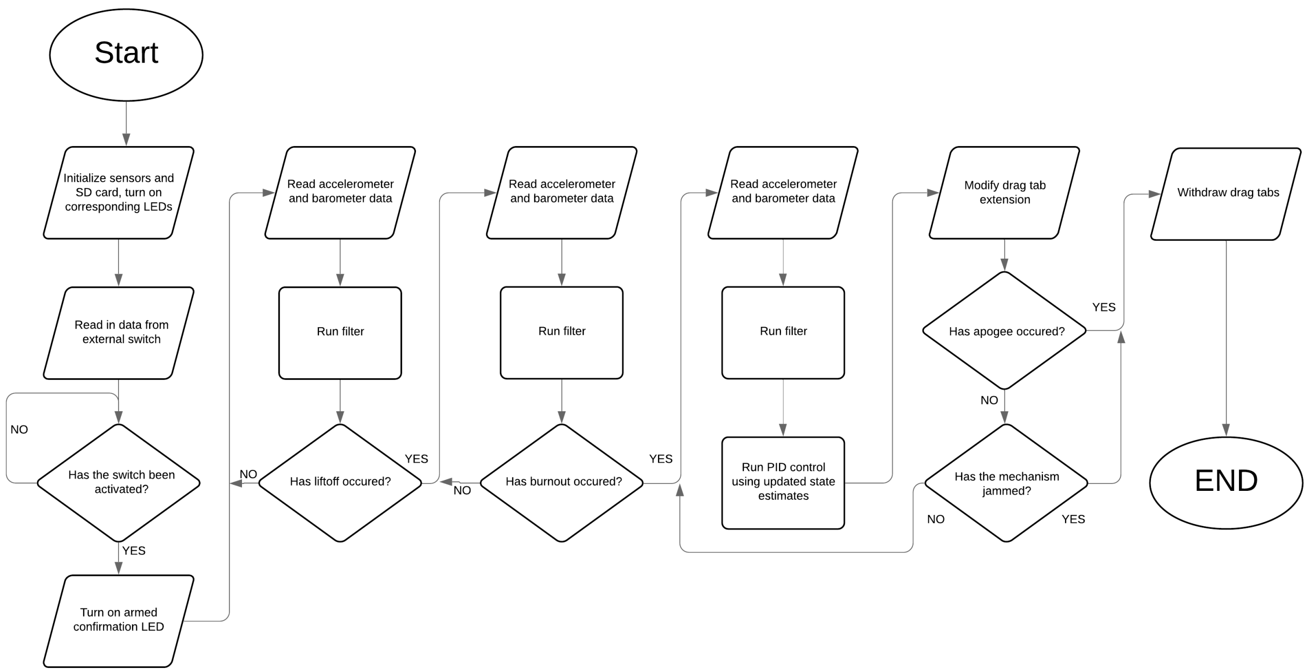

Figure 15: ABS Control Code Flow Chart

3.4.7.1 Data Filter Trade Study

In past years, one of the largest problems preventing the success of the ABS has been insufficient data filtering. The team has identified several potential approaches to the implementation of a data filter, for which the important factors considered include speed, memory efficiency, accuracy, ease of implementation, and ease of testing. The goal of the trade study was to find a data filtering algorithm that is relatively fast, yet provides accurate information about the current state of the launch vehicle.

3.4.7.1.1 Noise-Reduction Filters

The goal of noise-reduction filters is to read in the sensor data and filter out any noise which may disrupt accurate measurements. Noise can come from several sources, but if too much noise is present in the signal, tab adjustments will not be optimal. Filters in this category solely seek to reduce levels of noise, providing a smoother and more accurate signal to our PID control algorithm. Filters in this category include low-pass filters and moving average filters.

Low-pass filters work by filtering out higher frequencies in an input signal. Since acceleration and height should be changing fairly smoothly, they are unlikely to produce a high frequency signal. Because of this, any higher frequencies in the signal are likely to be noise. Low-pass filters can be implemented in hardware with resistors and capacitors or inductors, or in software through an FIR (finite impulse response) filter.

Moving average filters work by averaging the past n points of an input signal to be treated as an output signal. This filter also works to reduce noise by averaging any sudden spikes over a long period of time, which mitigates the effect of any given spike. This technique is great for noise reduction, and would give us more accurate data for a relatively small computational cost.

3.4.7.1.2 Bayesian Filters

Bayesian filters work over a statistical distribution of possible states, combining information from the sensors to determine a likely current state for the launch vehicle, while also attaching a certain uncertainty to that state. A popular example of this type of filter is the Kalman filter. Kalman filters combine an internal state, knowledge of physical laws, external influence and uncertainty, and sensor data to create a robust and accurate model of the physical state of the system. Compared to noise-reduction filters, Kalman filters have the advantage of integrating the data from all of the sensors onboard the ABS into a single internal state which takes full advantage of the information from each source. The disadvantage of this approach is that it is more expensive and time-consuming to implement, requiring the tuning of several parameters.

3.4.7.1.3 Adaptive Filters

Adaptive filters can improve themselves over time in response to data. This is usually done using the Least-Mean Squares algorithm, which works to minimize the error of the output over time. The advantage of this type of filter is that it can become increasingly accurate over time, and with more flight data could give increasingly accurate outputs which would in turn allow the PID control algorithm to adjust drag tabs to the optimal level more easily. The main problem with this algorithm would likely be the definition of error. In order to improve, adaptive filters need a good error signal, which may be difficult to create without some sort of additional filtering, which would make the adaptive filter somewhat redundant.

3.4.7.1.4 Trade Study Results

Several factors were examined when constructing the trade study of filtering techniques. We define speed as the time required for a single pass of the filter. This is important because the higher the output frequency of the filter, the more frequently the PID control algorithm can adjust tab extension, and the more optimally the system can operate. Memory efficiency is a measure of how much memory the filter requires to operate. If an algorithm requires the storage of large amounts of data in memory to operate effectively, it could compromise the operation of the rest of the controller. Accuracy is a measure of how well an algorithm translates sensor data into the true height and speed of the launch vehicle. This is very important and the whole point of data filtering in the first place, because if noisy, inaccurate data is fed to the PID control algorithm, it will result in drag tab extensions that will not put the launch vehicle on the desired trajectory for the target apogee. Ease of implementation is a measure of how complicated an algorithm will be to implement. If an algorithm is simpler to implement, it will be easier to test and optimize, which will increase the stability of the system. Finally, ease of testing is a measure of how well each of these algorithms can be verified. This testing is in terms of software bugs, as well as problems inherent to the algorithm. Each category is rated on a scale of 1 to 5 for each algorithm, based on the relative merits of each algorithm in each category.

Table 23: Data filter trade study

Criteria	Weight	Averaging Filter	Low-Pass Filter	Kalman Filter	Adaptive Filter
Speed	0.3	5	5	3	4
Memory Efficiency	0.05	4	4	3	4
Accuracy	0.3	2	2	5	4
Ease of Implementation	0.25	5	4	4	3
Ease of Testing	0.1	5	5	5	3
Score		4.05	3.8	4.05	3.65

The results of our trade study suggest that it is worth examining a couple methods going forward. A Kalman filter, if optimized correctly, will likely provide the most accurate results of the algorithms examined. If it is possible to implement this algorithm at a fast enough speed, this algorithm will likely prove to be the most useful. However, if the Kalman filter becomes too difficult to optimize (as it has been for the team in previous years), it will likely be worth implementing a noise-reduction filter. These filters have the advantages of speed and simplicity. It is worth testing both low-pass and averaging filters to see which smooths out flight data more accurately. An adaptive filter could prove useful if a good error metric can be determined and once flight data becomes available from test flights, but for now, simpler methods will likely be more effective.

3.4.7.1.5 Unit Testing

Before implementing the functionalities of the filtering algorithm and PID control algorithm, unit testing cases for each important functionality should be constructed with specified inputs and designated outputs. For the filtering algorithm, the following functionalities in Table 24 should be tested with unit testing cases.

Table 24: Functionalities with unit testing cases

Functionality	Description	Testing Input	Designated Output
Armed Module	When the external switch for sensors are activated, turn on the armed confirmation LED , start to read data from sensors, and run the filtering module.	A Boolean value that shows whether the switch is on or off.	If the input Boolean is TRUE, set the state of LED to be “light” and change the state of the system to “armed.” If the input is False, there is no output.

Launched Module	When the launch vehicle starts to lift off, the module changes to launched module.	Two numerical inputs that show the acceleration of the launch vehicle from the accelerometer and the air pressure from the barometer.	If either the input acceleration is greater than the threshold acceleration for a lift-off or the input pressure is greater than the threshold pressure, change the state of the system to "launched." If neither cases occur, there is no output.
Burnout Module	When the launch vehicle reaches burnout, the module changes to burnout module and run PID control module.	A numerical input that shows the acceleration of the launch vehicle from the accelerometer.	If the input acceleration is smaller than the threshold acceleration for a burnout, change the state of the system to "burnout" and run PID control module. If not, there is no output.
Apogee Module	When the launch vehicle reaches the apogee height, the module changes to apogee module and PID control module is off.	A numerical input that shows the air pressure of the launch vehicle from the barometer	If the input air pressure is smaller than the threshold air pressure for the designated height of apogee, change the state of the system to "apogee" and stops PID control module.

Filtering Module	After the launch vehicle is launched, this module filters noises from data read by sensors.	Two numerical inputs that show the acceleration of the launch vehicle from the accelerometer and the air pressure from the barometer.	A description of the current state of the launch vehicle, which includes a position, velocity, and acceleration.
PID Control Module	After a burnoff occurs, this module reads data from the filter, update the flight state, and adjust the extension length of the tab.	Two numeric inputs corresponding to the position and velocity calculated by the data filter	A numerical value that shows the designated extension length of the tab in order to reach the apogee.

Because the outputs of the filtering module and PID control module in the unit testing cannot be designated by basic calculations, simulation and subscale flight data are needed to further test these modules.

3.4.7.1.6 Simulation and Ground Testing

The purpose of simulation is to generate dummy data to test the PID controller and the filter. To this end, the data files of flights from previous years can be used to generate dummy data files. Additionally, simulated data from a mathematical model of the flight path can be used, as well as sensor data from the subscale flight.

For the data filter, raw data that are read from sensors and filtered data that are input to the PID controller in previous testing flights and future subscale flights can be used to test the filter. After raw data are put into the filter, the output will be compared to the filtered data in previous flights.

For the PID controller, filtered data that are input to the PID controller and the ideal position after the extension of the tab can be used to simulate a flight. After filtered data is inputted, the output from the PID controller will be used to calculate the position after extending the tab, and this position will be compared to the ideal position from simulated flights.

Additionally, a logistic regression algorithm or flight simulator software will be used to simulate new data sets from the previous data, or to examine how the system would respond to unusual launch conditions. Logistic regression will be used to generate new dummy data after training by data from previous years. These tools are valuable because of their ability to provide insight into a wider variety of possible flight profiles. When combined with flight logs from previous years, they will allow for a comprehensive testing program.

When the hardware of the ABS system is assembled, numerous ground tests for the ABS will be conducted. In ground testing, a set of generated sensor data will be fed into the system. The team will then be able to test how the data filter and PID control algorithm are working to calculate an optimal drag tab extension, and will be able to visually confirm that the drag tabs are extending the desired amount. This will give confirmation that the data pipeline is functioning correctly, and that the servo and mechanism are producing the expected drag tab extensions.

3.5 Recovery Subsystem

3.5.1 System Overview

The recovery system consists of a pyrotechnic parachute deployment system, a main parachute, a drogue parachute, and the recovery harness. The drogue parachute will be deployed from the vehicle at the vehicle's apogee, and the main parachute at approximately 600 feet. The parachutes will be deployed by black powder charges contained in PVC charge wells and will be ignited by altimeters powered by batteries. Three independently powered altimeters, contained within the Compact Removable Avionics Module, or CRAM, each control drogue deployment charge and a main deployment charge. Thus, each of the three altimeters are independently capable of deploying both parachutes.

3.5.2 Deployment System Selection

Several different parachute deployment systems were considered. The most common method of parachute deployment in rockets of this size is via black powder separation charges. In this system, the controlling altimeters send current through electronic matches, which ignite black powder charges contained in PVC charge wells. The gas produced by the black powder pressurizes the parachute compartment, breaking the shear pins holding the sections of the rocket together and allowing the parachute to exit the vehicle and slow the vehicle's descent. This system is the lightest weight and simplest of all the options; however, it requires the use of potentially dangerous energetics, and the deployment charges can burn the parachute or the recovery harness if not properly protected.

CO_2 parachute deployment was also considered. In this system, the altimeters would send current through a commercial ejection device, such as a Tinder Rocketry Peregrine or FruityChutes Hawk, which would release the gas contained in a single-use CO_2 cartridge, pressurizing the parachute compartment, breaking the shear pins connecting the rocket sections together, and deploying the parachute. This system does not damage the parachute on ejection, uses less dangerous energetics, and is fairly lightweight (though not as lightweight as the black powder system). However, it is also much more expensive than the other systems considered, more complex, and potentially less reliable than the black powder system.

The last deployment system to consider was an entirely mechanical spring system. A series of springs would be held under compression by cords attached to a latch, with the parachute compartment above the springs. The altimeters would send a signal to a servo motor which would open the latch, releasing the springs. The springs would then push on connecting rods, that would push on the opposite bulkhead of the parachute compartment, breaking the shear pins connecting the rocket sections and deploying the parachute. This system is the least dangerous, using only compressed springs instead of chemical explosives or high-pressure gas. It also does not damage the parachute on ejection. It is, however, the heaviest system under consideration, and it is much more complex than either the CO_2 or black powder systems. Table 25 displays a deployment system trade study, taking into consideration system weight, cost, reliability, ease of assembly, and ease of manufacturing. Higher grades in the trade study indicate more favorable values for the relevant criteria. For example, a higher grade for the "Weight" criteria indicates a lower system weight, as low system weight is preferred over high system weight.

Table 25: Deployment System Design Trade Study

Criteria	Weight	Black Powder	CO_2	Spring
Weight	20%	9	7	4
Cost	15%	8	3	6
Reliability	35%	8	7	3
Ease of Assembly	15%	7	5	3
Manufactuability	15%	7	7	3
Total		7.90	6.40	3.65

3.5.3 CRAM Design

The CRAM is a system for securing the recovery altimeters, batteries, and ejection charges, while maintaining ease of removal for rapid repair and data retrieval. The CRAM consists of several components: the body, the core, and the bulkheads.

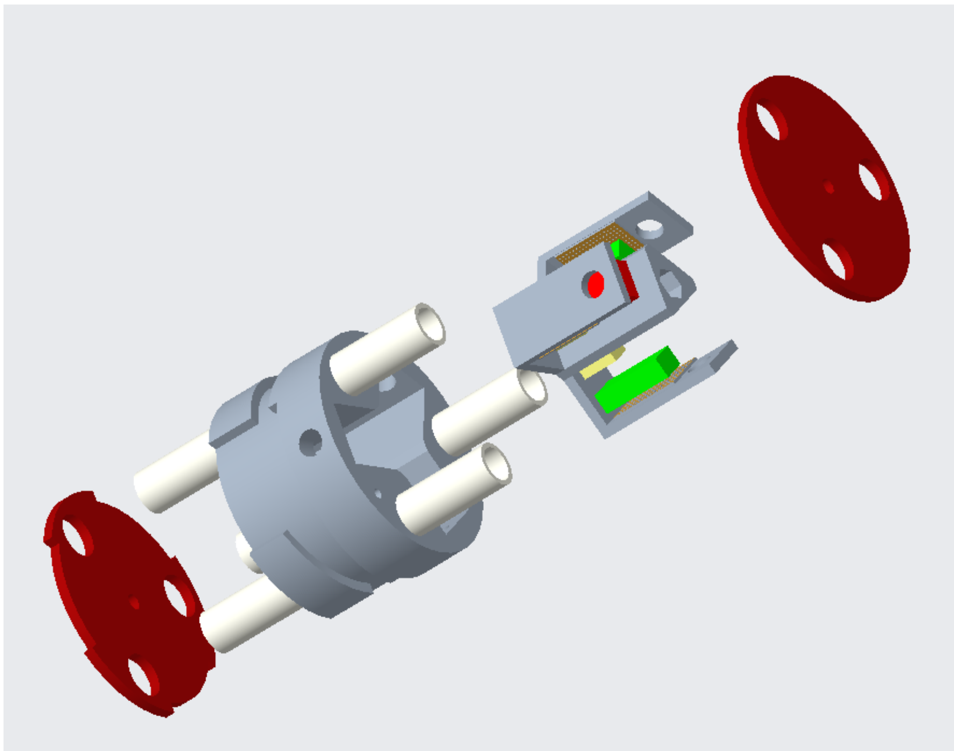

Figure 16: Exploded view of CRAM CAD model

The CRAM body serves as the casing for the recovery electronics, and secures the CRAM in the rocket body tube. It consists of a cylinder with a three-winged shape cut through its entire length, where the CRAM core is placed. On the exterior of the CRAM body, tapered screw cutouts mate with matching protrusions on the CRAM adapter, epoxied into the body of the rocket. This allows the CRAM to be secured in the rocket by simply inserting it into the adapter and twisting it 30^o. Also on the exterior of the CRAM are holes for access to switches to turn on the altimeters, and air holes for accurate air pressure measurement by the altimeters.The top and bottom of the CRAM body has PVC charge wells epoxied in place to hold the black powder ejection charges in flight, and holes all the way through the body to allow for bolts to hold the CRAM together. Figure 17 is a preliminary CAD drawing of the CRAM body, as currently designed.

The material chosen for the CRAM body is essential for ensuring the integrity of the recovery system during the flight of the rocket, as it is one of the main structural components

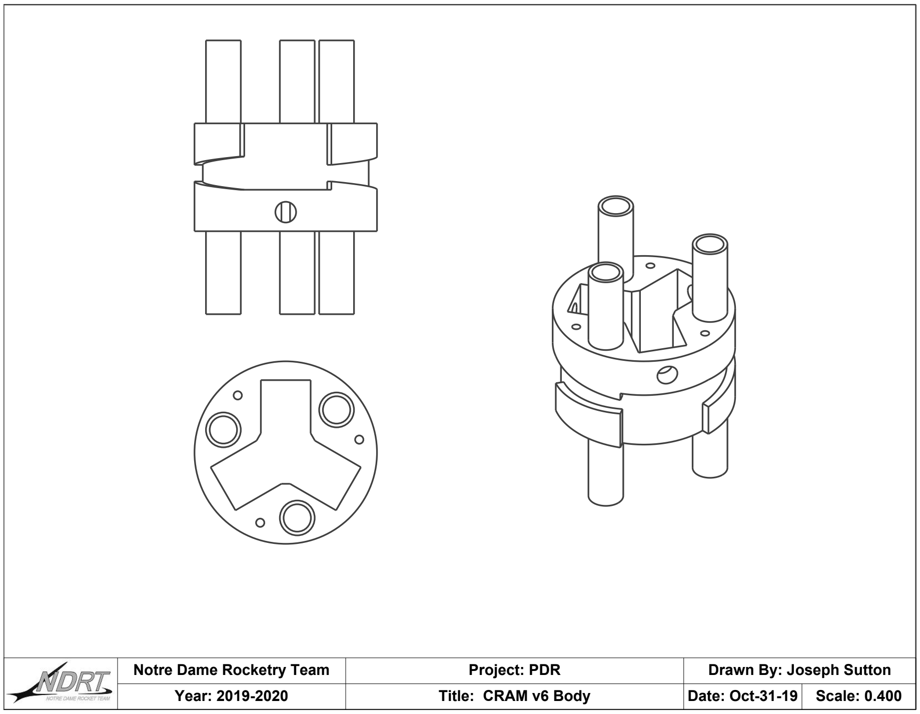

Figure 17: CAD Drawing of CRAM Body

of the recovery subsystem. Pressure-treated yellow pine has been selected as the material for the CRAM body due to its high strength-to-weight ratio, low cost, wide availability, and machinability, all required to produce an effective screw-in locking mechanism. Other materials considered included aluminum, which was ruled out due to cost and weight; 3D printed ABS plastic, which was ruled out based due to sourcing difficulties and low strength; and HDPE, which was ruled out due to high cost. Initial prototypes of the CRAM body will be 3D printed in PLA plastic in order to iterate on the design quickly and verify the CRAM geometry before committing the time and effort to fully machine a piece of wood. Table 26 describes the results of a trade study performed to select the CRAM material, taking into consideration cost, strength, ease of production, weight, ease of modification (ability to be sanded after production), and availability.

Table 26: CRAM Body Material Trade Study

Criteria	Weight	Pressure-Treated Wood	Printed ABS	HDPE
Cost	10%	10	8	6
Strength	25%	8	6	10
Ease of Production	25%	6	10	8
Weight	20%	8	10	10
Ease of Modification	15%	10	6	6
Availability	5%	10	6	4
Total		8.1	8	7.85

The CRAM core serves as the component to which the altimeters, batteries, and other electrical components of the recovery system are mounted. The core itself can be separated into two pieces for ease of CRAM assembly. The core has a central hexagonal component which contains a coupling nut for attachment recovery eyebolts and a mounting location for the recovery batteries, and the wings, to which the altimeters and recovery switches are mounted. The CRAM core will be 3D printed from PLA, due to availability, as the core has no need for high strength. Figure 18 shows some preliminary models of the CRAM core assembly, with the electronics mounted.

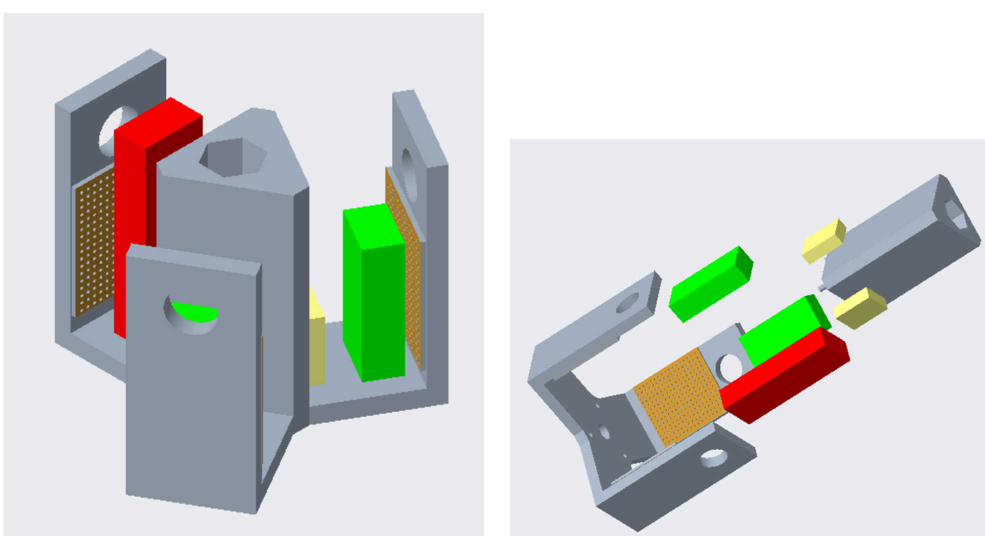

Figure 18: Preliminary CAD models of CRAM Core

On either side of the CRAM body are bulkheads, which serve to retain the core in the CRAM body and protect the electronics from the black powder ejection charges. The bulkheads have cutouts to allow for the PVC charge wells, the central eyebolts, bolt holes, and wire holes for electrical connection between the altimeters and the ejection charges. The bulkheads are machined from 1/4 inch G-10 Garolite fiberglass, chosen due to its high tensile strength and extremely good impact strength. Also considered were birch plywood, acrylic, and HDPE. Table 27 displays the results of a trade study performed to select the bulkhead materials.

Table 27: CRAM Bulkhead Material Trade Study

Criteria	Weight	Garolite G10	Birch Plywood	HDPE	Acrylic
Cost	10%	7	9	7	7
Density	30%	7	9	8	7
Tensile Strength	20%	10	8	7	7
Impact Strength	40%	10	8	6	4
Total		8.8	8.4	7.6	5.8

3.5.4 Electrical Design

In order to deploy the parachutes at the appropriate locations in flight, barometric altimeters control the ignition of the deployment charge. At apogee, the altimeters send a signal to the electronic matches (e-matches), which ignite the black powder ejection charges. Two of these altimeters are Featherweight Raven 3s, while a third is a StratoLogger SL100. Two different models of altimeter are used to in the case that one model of altimeter fails due to design error. Each altimeter will be powered by a 170 mAh 1S battery pack, and each altimeter is connected to a main deployment charge and a drogue deployment charge.

Figure 19 shows how the Raven 3 altimeters and deployment charges are connected. Two switches, a Featherweight magnetic switch and a rotary switch, control the flow of electricity to the altimeter and the e-matches, respectively.

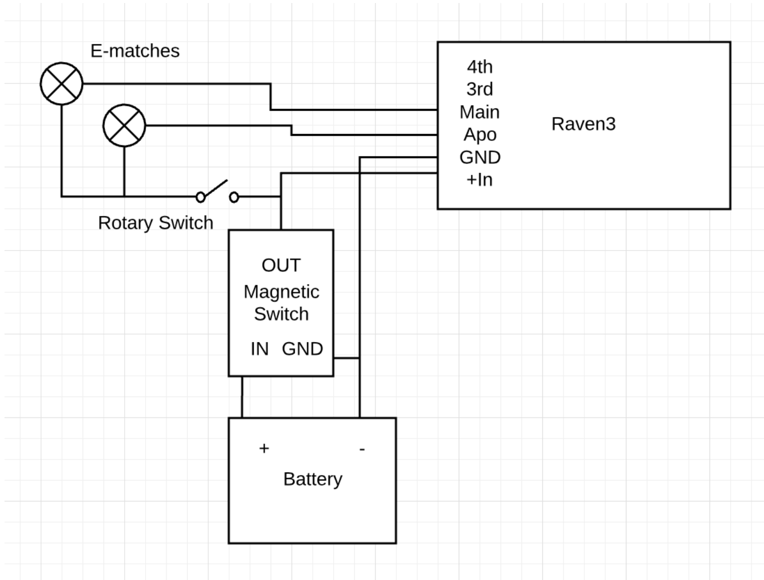

Figure 19: Circuit Diagram for Raven3 Altimeter

Figure 20 shows how the SL100 altimeter will be similarly connected. These systems will be integrated into the CRAM by soldering to solderable perfboards and screwing the boards into the CRAM core.

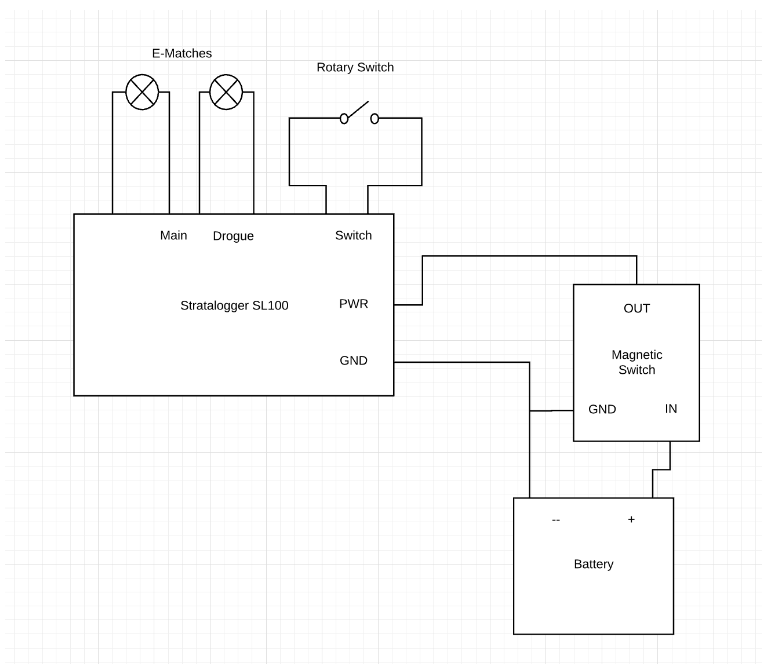

Figure 20: Circuit Diagram for Stratologger SL100 Altimeter

The switches employed in the system must conform to two requirements: they must be easy to access and use, and they must also resist any in-flight turbulence that might turn them off or tear them apart. Additionally, two switches are needed: one to activate the system and one as an external safety mechanism, as a second measure to prevent current flow to the e-match before the vehicle is ready for launch. Four types of switches were considered: buttons switches, screw switches, magnetic switches, and rotary switches; ultimately, the magnetic switch was selected as the system's main switch, while the rotary switch was chosen as the breaker. The button and the screw switch, despite their ease of use, were determined to be too vulnerable to accidentally closing if some component inside the CRAM broke. The magnetic switch takes up a minimal amount of space, while the screw switch would be embedded into the CRAM core and turned on from the outside of the rocket. Figure 21 shows the two models of switch that were selected.

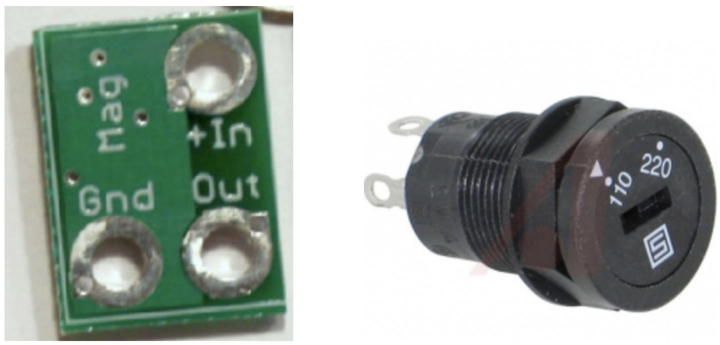

Figure 21: Switches Selected for use in Recovery system

Lithium polymer (LiPo) batteries will be used to power the recovery altimeters. A 3.7v, 170 mAh LiPo battery was chosen over the alternatives because of its small size, low weight, and rechargeability.

The systems will be integrated into the CRAM core using solderable perfboards. Two other methods were also considered: printed circuit boards (PCBs) and loose wiring. The former was eliminated from consideration due to repair difficulties and potentially long order times. The latter was eliminated from consideration due to organizational concerns; the tangle of wires in the CRAM could interfere with the placement of other components, resulting in a less efficient use of space within the module.

3.5.5 Recovery Harness

Shock cords are be used to connect the sections of the rocket after separation. These shock cords are 35ft long, 1 inch thick tubular nylon recovery harness rated for 4000lbs. Tubular Kevlar was also considered for the shock cord. Although rated for higher loads, the Kevlar shock cordage is significantly less elastic, which can cause higher G-forces on the vehicle during main parachute deployment. Figure 22 shows one such harness.

Figure 22: Potential full scale recovery harness

To connect the shock cords to the parachutes and eyebolts, 3/8 inch stainless steel locking quick-links are used. Stainless steel is less susceptible to corrosion due to repeated black powder ejections than galvanized or uncoated mild steel. Locking quick-links are used to ensure solid connection through the vibration of launch through the highly variable loads associated with parachute deployment.

To connect the shock cords to the bulkheads in the fore and aft sections of the vehicle and to the CRAM, 3/8 inch, stainless steel, forged construction eyebolts will be used. Stainless steel is used due to its corrosion resistance, and forged construction is used over bent construction due to much higher strength. Inside the CRAM, two eyebolts (one attached to the fore shock cord, one to the aft shock cord) are connected using a 3 inch, 3/8-16 steel coupling bolt. This ensures a solid connection between all of the sections of the rocket during descent.

3.5.6 Parachute Protection

Three protection methods were studied for parachute protection. A parachute deployment bag has been chosen primarily due to its control of the parachute inflation sequence and ease of ejection. The parachute deployment bag guarantees that the parachute lines exit the bag before the release of the parachute. This sequence prevents the lines from entanglement and thus ensures the inflation of the parachute. The deployment bag is made from Nomex, which also protects the parachute from the black powder explosion. A trade study has been performed for protection methods selection, as shown in 28. A Nomex blanket, when used to protect a large parachute, can increase the likelihood of entanglement, while a rigid piston can bind in the body tube. The deployment bag method has been determined to be the best option.

Table 28: Parachute Protection Trade Study

Criteria	Weight	Nomex Blanket	Piston	Deployment Bag
Protection Level	15%	7	10	10
Adaptability	5%	10	2	5
Availability	5%	10	5	10
Ease of Ejection	30%	10	4	10
Unfurling Sequence	40%	2	6	10
Ejection-Inflation Delay	5%	10	10	8
Total		6.35	5.95	9.65

3.5.7 Parachute Selection

A FruityChutes CFC-24 parachute is used for drogue deployment. It is a 24 inch elliptical parachute, constructed from 1.1oz rip-stop nylon and 220lb nylon shroud lines. It was selected due to its high drag coefficient (1.50) when compared to comparable flat parachutes (.75). This allows for a smaller, lighter parachute to slow the vehicle to the same velocity. In addition, the team already has a CFC-24 parachute in stock, and has had previous success with the parachute.

A FruityChutes Iris Ultra 120 Compact parachute will be used for main deployment. It is toroidal in shape, 10ft in diameter, and uses 400# Spectra cord for shroud lines. An alternate consideration was the 16ft Rocketman Standard parachute. The FruityChutes IFC-120-S was chosen over the 16ft Rocketman due to its lighter weight and much smaller packing size.

3.5.8 Electromagnetic Shielding

To prevent electromagnetic radiation causing unpredictable behavior in our altimeters, electromagnetic shielding will be used. After considering multiple alternatives, it was decided copper foil would be used to form a Faraday cage around the altimeters. The altimeters need to stay separate from the GPS units and any other circuit element that emits electromagnetic radiation. The reasons for selecting copper foil over an active shielding system or metallic paint were multifaceted. An active shielding system would entail a system that detected electromagnetic waves and then produced an opposing wave that would use destructive interference to cancel the signals. The active shielding system was eliminated due to the system being too heavy and expensive for the design constraints. Another option would be metallic paint, which could easily be applied to any container, but it would be difficult to regulate the consistency of the layers. Copper foil was primarily selected because of its low cost, low mass, low complexity and high effectiveness. Table 29 shows the results of the trade study performed.

Table 29: Electromagnetic Shielding Trade Study

Criteria	Weight	Copper Foil	Copper Paint	Active Shielding System
Mass	40%	10	10	3
Effectiveness	30%	8	6	2
Cost	10%	8	9	2
Complexity	20%	8	8	2
Total		8.8	8.3	2.4

3.5.9 Redundancy and Safety

Nearly every part of the parachute deployment system has redundancy. Each of the three altimeters are powered by its own battery. Each of the altimeters controls a drogue deployment charge and a main deployment charge, meaning each of the three altimeters are independently capable of deploying both parachutes and successfully recovering the rocket. In the extreme edge case of two separate altimeter or battery failures, both the main and drogue parachute will successfully deploy.

There are numerous safety concerns regarding the recovery system. One of the largest aspects is that the live black powder used to separate the rocket at apogee could potentially go off during setup or at an improper time. Part of this risk is reduced through careful and repeated testing of altimeters and consistent following of procedures, but other methods need to be in place in order to prevent potential injuries. The system uses two separate switches to fully activate each altimeter, a magnetic switch and a rotary switch. Using two switches to activate the system, one physical and one magnetic, significantly decreases the chance of accidental activation.

3.5.9.1 GPS and Telemetry

Competition rules require the vehicle to be able to transmit GPS coordinates. This year the GPS system shall transmit live data back to a ground station. Live mission data transmission from the vehicle more accurately mimics a NASA mission, which relies on live data from the launch vehicle to verify the status of the mission.

This self imposed challenge to provide more meaningful real-time data alongside the required GPS data is a new push by the team to involve more electrical engineers to develop custom integrated solutions rather than rely on commercial products for all systems. This provides flexibility for new capabilities and gives students improved experience and skills development. Because the decision to pursue this challenge was made very recently, detailed information about the telemetry system and its integration into the Recovery system are not available at this time. They will be included in the CDR report.

3.5.9.1.1 System Overview

As in past years, the team will keep the GPS system in the nose cone of the rocket. The nose cone will be 3D printed, which will allow for an RF transparent location that can be customized to allow for easy securing of the system. In order to receive meaningful position data of the rocket, a PIC33 microprocessor and a commercially bought accelerometer will be utilized.

In the event of developmental issues with the custom solution and ground station, The Eggfinder GPS Tracking System, manufactured by Eggtimer Rocketry, will be used. The team has used this device in past years, so the team is familiar with the system and it integrates easily.

3.5.9.1.2 RF Transmission

Data will be transmitted through ISM (Industrial, Scientific, and Medical) frequency bands, as the team can legally transmit on these bands, and components to transmit along these frequencies are readily available from manufacturers. Either a dipole or patch antenna will be used to transmit data from the vehicle to the ground station. Dipole antenna have the benefit of being the lighter option, and would also be more space efficient. However, a patch antenna would offer a more preferable radiation pattern.

3.6 Systems Integration

In this section the integration methods for each of the systems housed by the vehicle will be discussed as well as the integration of individual structural component in the launch vehicle. The goal is for integration to effectively secure each system from any linear displacement and rotation in the most cost effective and practical manner. Load bearing parts will be tested to ensure there is a safety factor of at least 2. The later testings and integration techniques will be discussed more in detail below.

3.6.1 Vehicle Integration

In order to connect the different parts of the launch vehicle the following integration techniques: bulkheads, couplers, centering rings, shear pins, and screw locks will be required. Each of these components has its own unique application within the launch vehicle. Bulkheads are circular disks secured into the body tube used to absorb loads created by parachute deployment. Although previous NDRT launch vehicles used plywood bulkheads, last year's launch vehicle used fiberglass. Being lighter than plywood and having successfully withstood the loads created by parachute deployment last year, this year's bulkhead decision will be made after solid testing on bulkheads is excecuted to further inform the team about maximum load each material can take. More about the testing can be found in section 3.5.6. The same material that is chosen from this process will be used for centering rings. Centering rings are used to attach the motor to the fin can. These will be sanded down to size to ensure a good fit. Couplers are sections of tubing that are used to connect 2 larger sections of tubing. In order for the launch vehicle's parachute to be deployed while in flight, the airframe's body tube sections will be separated using ejection charges. To prevent the body tube from separating ahead of time, the team will employ shear pins - small screws that keep the different body tube sections attached during flight and break under the shear stress of an ejection charge. Four nylon shear pins are needed for every separation point along the airframe - there will be a total of two separation points in the launch vehicle. Additionally, screw locks will be used to secure the bulkhead that sits directly in front of the Air Braking System (ABS) to the fincan. Screw locks are screws which go through the body tube and into the bulkhead. The screws must be of a large enough diameter as to not shear the body tube. This bulkhead is load bearing and will have the shock cord from one of the two recovery parachutes attached to it. Using screw locks is a method of securing a bulkhead within the airframe without using epoxy. This is necessary because the team should be able to access ABS from the in-flight separation point.

Additionally, NDRT will use the following epoxies to connect all the previous parts - all the epoxies were chosen based on NDRT's experience with them in previous launch vehicles. Great Planes Thirty Minute Epoxy for the fabrication of the subscale model; Glenmark RocketPoxy for all the carbon fiber and fiberglass parts; and JB weld for the motor mount - JB weld is an especially good choice for the motor mount because of its resistance to high temperatures. When applying epoxy to secure a component to the body tube, filets will be used to help transfer loads from the components to the body tube. Filets are small beads of epoxy applied at a contact point to strengthen the joint.

Lastly, there's sanding. Although sanding itself is not an integration technique, sanding is necessary to ensure that all the couplers, centering rings, tubes, motor mounts, and the nose cone fit together tightly. Sanding is also used to create a leading and trailing edge on the fins.

3.6.2 ABS Integration

For the integration of the ABS into the launch vehicle, it is important that the system is able to withstand the load force induced by the main parachute during deployment, that the drag tabs align well with their slots in the body tube, and that it is easily removable in order to make modifications and recharge batteries. Three methods were considered for integrating the ABS into the fin can. The team has historically used threaded rods, which are attached to a bulkhead at the bottom of the body tube housing the ABS, and extend through its length. In this method, four holes are drilled into each ABS platform, and the threaded rods are used to guide the holes into the body tube, where they rest on top of the bottom bulkhead. The benefits of this system are that it can endure a lot of force and is unlikely to be sheared or otherwise compromised, and that it requires minimal fabrication. One major drawback to this system is that it spans a greater length than the ABS itself, as the threaded rods have to extend beyond the point at which the ABS ends. Additionally, the rods tend to bend as the ABS is inserted and removed from the body tube, causing it to lock up.

Another option for integration is securing it with bulkheads and screws. In this method, the ABS would be secured by bulkheads on both ends. Its weight would rest on the permanent bottom bulkhead, while an additional removable bulkhead would be attached to the top of the ABS, and would be secured with screws to allow for easy removability of the system. The benefits to this design include compactness, minimal required fabrication, and the use of screws to withstand the loading induced by the deployment of the main parachute. However, one major drawback is the lack of a guide for accurately aligning the drag tabs to their corresponding slots in the body tube.

The final method of integration under consideration is a twist and lock mechanism, which would be 3D printed and incorporated into the bottom bulkhead of the ABS. This method would involve a 3D printed mounting ring with 4 equally spaced inner pegs, which would be epoxied to the inside of the fin can so that it can interface with a 3D printed platform attached to the ABS. The 3D printed platform would have guide slots cut into it, which would allow the pegs to slide in and twist to lock. The most significant benefit of this method is that the twist and lock mechanism would ensure that the drag tabs align properly with the slots in the body tube. Another benefit is that none of its features add length to the ABS. The biggest drawback is that the 3D printed pegs may not be able to withstand the stress induced by the loading force from the main parachute during deployment. CAD models of the twist and lock mechanism components are shown in Figure 23 and Figure 24, respectively.

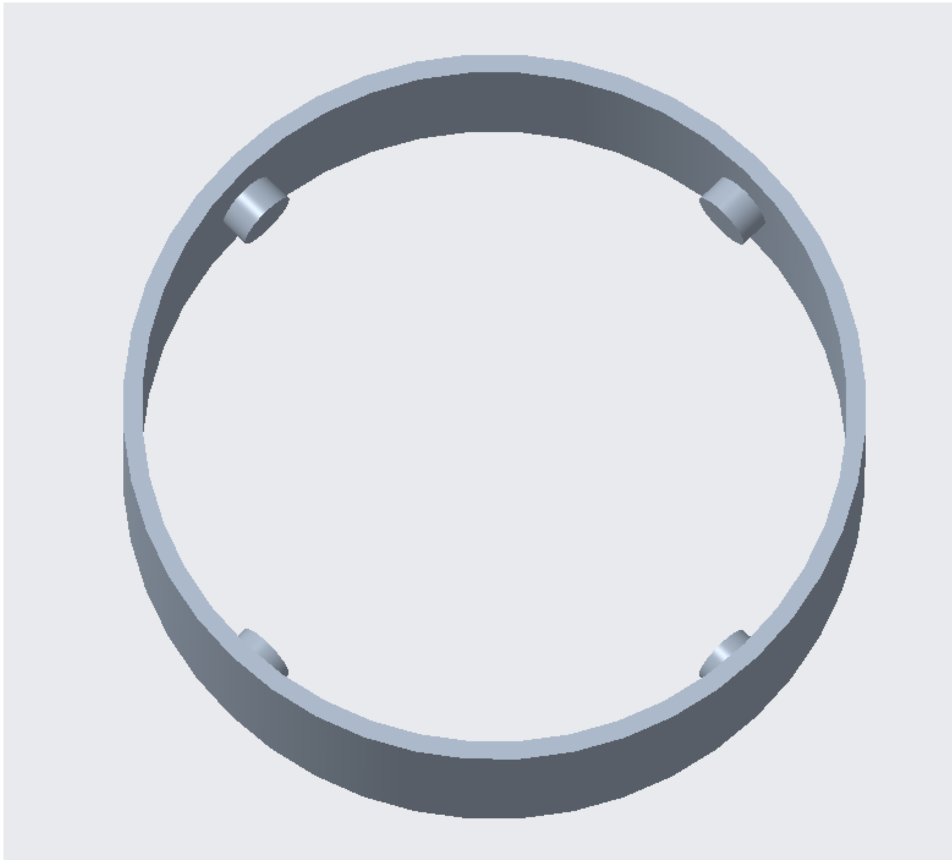

Figure 23: ABS twist and lock mounting ring

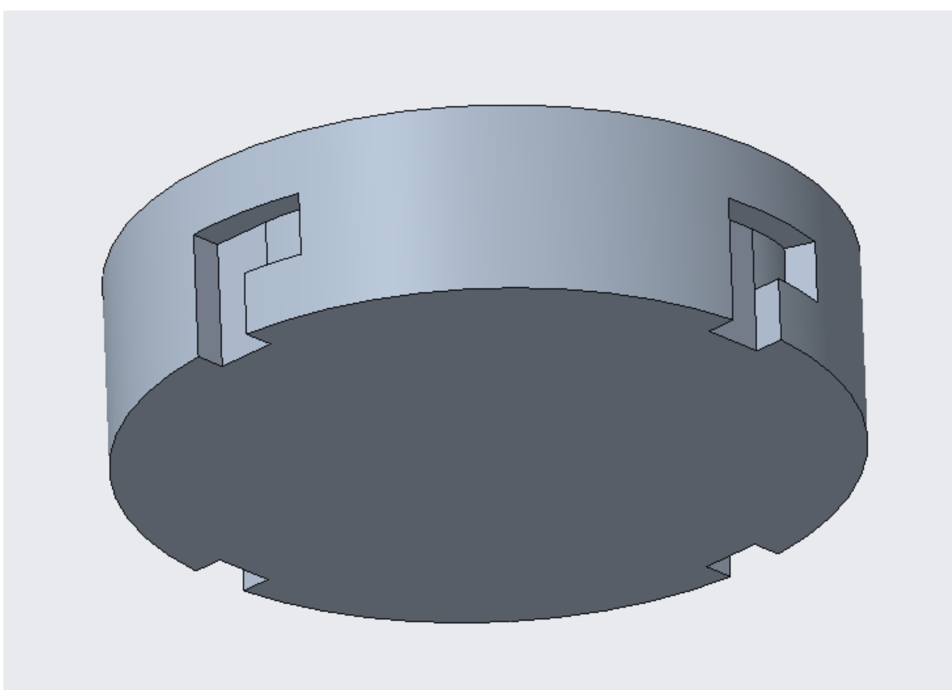

Figure 24: ABS twist and lock platform with guide slots

After consideration of these different integration methods for the ABS, the team decided to use a combination of the twist and lock mechanism and the use of screws in a removable bulkhead. The twist and lock will ensure that the drag tabs are properly aligned within the body tube, while the screws will ensure structural integrity of the system under loading from the main parachute.

3.6.3 Recovery Integration

The CRAM integrates with the vehicle body tube through a ring adapter epoxied place. Tapered protrusions from the ring adapter mate with matching cutouts in the CRAM body. This allows the CRAM body to be secured by simply inserting the CRAM in the body tube and rotating it 30^o. Screws are then inserted into the CRAM through the wall of the body tube, preventing the CRAM from backing out of the screw-to-lock mechanism. Other options that were considered include large bolts into the CRAM body through the exterior of the launch vehicle, and threaded rods embedded in a fixed bulkhead. The bolt-in CRAM was ruled out due to the protruding bolt heads in the side of the launch vehicle, potentially affecting the aerodynamics of the vehicle. The threaded rod method of integration was ruled out due to long installation time and potential for damage to the system due to bent threaded rods. Figure 25 shows the ring adapter that will be mounted in the vehicle body tube to accept the twist-to-lock CRAM.

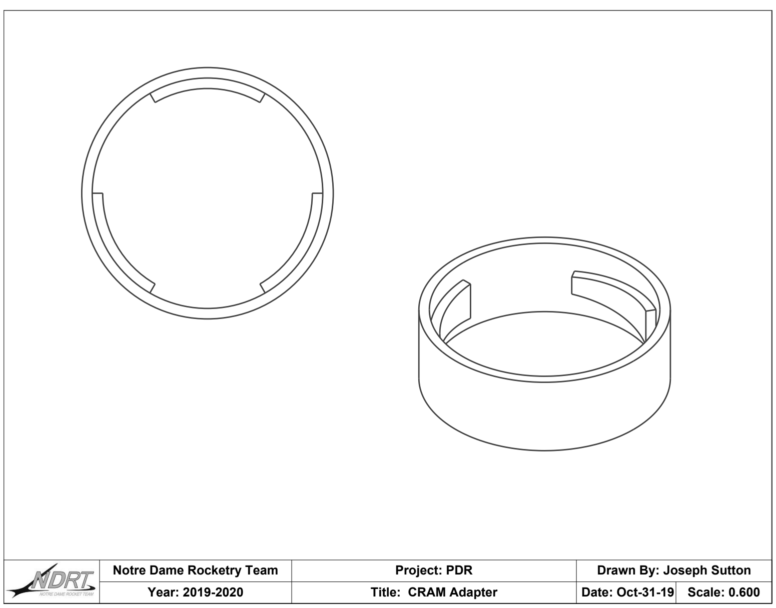

Figure 25: CRAM twist-to-lock adapter

3.6.4 GPS Solution Integration

The components of the GPS system shall be integrated into the nose cone through a twist to lock similar to that used by the CRAM. The system will then be secured by bolting the system to its housing in the nose cone. All integration and securing methods will be internal to the nose cone, so as not to interfere with the extremely sensitive aerodynamics of the vehicle at the nose cone. As such, this means that the system shall be almost exclusively internal to the vehicle body.

3.6.5 Integration Testing

To determine what materials to use for the bulkheads, the team has elected to do a number of integration tests in the labs on campus. In previous years, the bulkhead material was chosen based on calculated values available from sources and their costs; however, given the importance of the bulkheads as the load-bearing components inside the vehicle, it is necessary to be sure that the material chosen can indeed withstand the forces that will be exerted on it.

To carry out these tests, the team will be utilizing leftover carbon fiber couplers available from the previous year's project. The three options for the bulkhead material are plywood, carbon fiber, and fiberglass. Bulkheads of these three materials will be epoxied into the couplers in the same way that is done for the full scale model. The bulkheads will then be subject to a slow loading test to determine the force at which they will break. The goal will be to run between three to five tests for each material, which will allow for a mean force at which the bulkhead fails to be determined and a standard deviation. The total amount of tests done will depend on the amount of coupler that is available to use in this way.

Once the bulkhead material is chosen, more tests will be run on it. This time, impact testing will be done, to mimic the jolt of the parachute deployment. The data collected in these tests will allow the team to know the amount of force that will cause the bulkhead to fail at deployment.

Testing will also need to be done on the Air Braking System and the CRAM used for the recovery parachute. Because of the vehicle's design with two separations and two parachutes, both components will need to be load bearing as well as removable. This will allow the load bearing payload parts positioned behind them to be secured and accessible as well. Furthermore, with the ABS and the CRAM being removable and load bearing, there will be no need for more bulkheads or access points in the vehicle. In order to do so, a twist and lock system will be utilized. This system will consist of a n epoxied thread on the inside of the airframe and pin to lock it in fully in place. The material considered for this mechanism are plywood, abs plastic, and HDPE. This twist and lock system will need to be subjected to both the slow load and impact testing to ensure that it will be able to withstand the forces acting it when the parachutes deploy.

Integration testing will be done before the end of the semester to allow for time to analyze results, and decide on materials to be used and the techniques in which to secure the load bearing components.

3.7 Mission Performance Prediction

The designed variable diameter launch vehicle with a maximum length of 12 ft and 4 isosceles trapezoidal fins was modeled in OpenRocket and RockSim to further explore flight performance and stability margin. Aiming for a 4,444 ft apogee, the launch vehicle will use a Cesaroni L1395-BS-0 motor to overshoot the target apogee and allow for ABS actuation. Bellow, these simulations and stability calculations are described in greater detail.

3.7.1 Flight Profile Simulations

Simulations were conducted in OpenRocket and RockSim in order to predict flight performance. Simulations were performed for the selected motor in wind conditions ranging from 0 mph to 20 mph in 5 mph increments. Wind speeds above 20 mph were not considered, as this is the maximum wind speed allowed by NASA at the time of launch. The launch rail length for all simulations was assumed to be 144 in, and atmospheric conditions were set to International Standard Atmosphere. Table 30 below shows the average result from both simulations of the flight under all conditions.

Table 30: Flight Simulations

Motor	Wind Speed [mph]	Apogee [ft]	Max Velocity [ft/s]	Max Accelera [ft/s^2]	Ground Hit Velocity [ft/s]
Cesaroni L1395-BS	0	4997	591	222	13.9
	5	4985	591	222	13.9
	10	4953	590	222	14.1
	15	4894	590	222	14.7
	20	4853	589	222	14.2

3.7.2 Static Stability Margin

To ensure the stability of the launch vehicle, the center of pressure must be aft of the center of gravity to prevent the aerodynamic forces from creating a moment. According to the success criteria, the distance between the center of pressure and center of gravity must be greater than 2 calipers. The unloaded vehicle has a static stability margin of 3.64, and the loaded stability the selected motor is 2.62. The stability margin was calculated using CAD modeling, RockSim, and OpenRocket simulations. By ensuring that each payload complies with their allotted weight budget and that the material for the airframe is weighted to ensure correct material densities, the stability margin will not shift significantly. Historically, the simulated center of gravity has been very accurate when compared to the actual value, hence it gives us confidence that the stability margins predicted are accurate. Variables that affect the static stability include payload weights, length of the launch vehicle, and fin design will be carefully considered in our motor selection.

3.7.3 Kinetic Energy at Landing

Each separated section of the vehicle must have a kinetic energy of less than 75 ft-lbs at vehicle landing. Since all of the sections of the vehicle will be tethered together on descent, all of the sections will be descending with the same velocity. Therefore, the landing kinetic energy of the vehicle sections can be calculated from the equation,

$$KE = \frac{1}{2}mV^2 \tag{2}$$

where KE is the kinetic energy of the heaviest vehicle section on landing, m is the mass of the heaviest launch vehicle section (in this case), and v is the descent velocity of the vehicle under the main parachute.

Four calculation methods were used to determine the descent velocity of the vehicle: an OpenRocket simulation, hand calculations, and the FruityChutes Parachute Descent Rate Calculator. The OpenRocket simulation used a parachute diameter of 120 inches, and coefficient of drag of 2.20, which match the manufacturer specifications for the IFC-120-S parachute we will be using during terminal descent. Figure 26 shows one such OpenRocket simulated flight profile, in this case assuming the maximum 20 mph wind speed and a launch angle of 5 degrees.

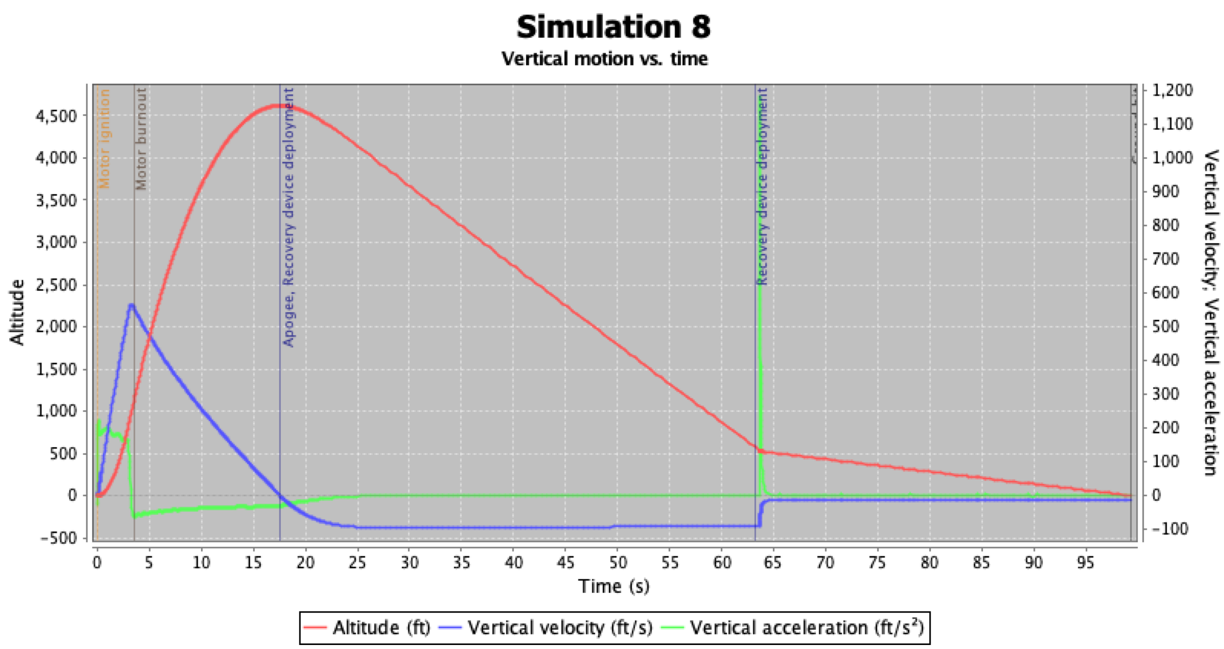

Figure 26: OpenRocket Simulation of Full Scale Flight, assuming 20 mph wind and 5 degree launch angle

The hand calculations of terminal velocity used the equation,

$$V = \sqrt{\frac{2W}{\rho C_d A}} \tag{3}$$

Where V is the terminal velocity of the vehicle, W is the total weight of the vehicle after motor burnout (45.7 lbf in this case), ρ is the density of the air, C_d is the drag coefficient of the parachute, and A is the effective area of the parachute.

The FruityChutes Parachute Descent Rate Calculator used the in-built settings for the IFC-120-S parachute and a launch vehicle weight of 45.7 lbs (41.4 lbs for the vehicle, and

4.3 lbs for the empty motor casing).

The various calculation methods were used to find the descent rates, and Equation 2 was used to find the associated terminal kinetic energies. Table 31 displays the results of the kinetic energy calculations.

Table 31: Terminal Kinetic Energy

Simulation	Kinetic Energy Prediction [ft-lbs]
OpenRocket	56.3
Hand Calculations	59.7
FruityChutes Calculator	61.5

All of the simulations predict kinetic energies below the maximum of 75 ft-lbs per section. The simulations are within 10% of each other, and all are more than two standard deviations below the requirement-mandated 75 ft-lbs.

3.7.4 Descent Time

The launch vehicle must descend from apogee to the ground in less than 90 seconds, necessitating the use of a drogue parachute in addition to a main. The launch vehicle will use a 24 inch CFC-24, deployed at apogee as the drogue, and a 120 inch IFC-120-S parachute, deployed at 600ft as the main. All the same simulation methods used in Section 3.7.3 were applied to the descent time calculation. Table 32 shows the results of these simulations.

Table 32: Vehicle Descent Time

Simulation	Descent Time Prediction [s]
OpenRocket	81.0
Hand Calculations	84.8
FruityChutes Calculator	83.4

All of the simulation methods predict descent times with the 90 second allotment, meeting the descent time requirement. The calculated descent times from all the calculation methods are within 5% of each other, and all are more than two standard deviations below the maximum 90 second descent time.

3.7.5 Drift Distance

The launch vehicle must stay within the confines of the launch area at all times, mandating a maximum drift distance of 2500 ft from the launch pad. The same OpenRocket and custom Matlab simulations were used from Section 3.7.3. For the hand calculations and derivation from the FruityChutes Descent Rate Calculator, the assumption was made that the launch vehicle's vertical velocity will be equal to the wind velocity. Table 33, below, describes the result of the drift calculations, assuming the worst-case scenario launch conditions (20 mph winds).

Table 33: Vehicle Drift Distances, assuming 20mph Winds

Simulation	Drift Distance Prediction [ft]
OpenRocket	2170
Hand Calculations	2498
FruityChutes Calculator	2446

All of the calculations made predict drift distances within the 2500 ft radius launch area, fulfilling the drift distance requirement. The differences in drift calculations are likely due to OpenRocket's consideration of weathercocking during ascent, while the hand calculations and calculation using the FruityChutes calculator assume an apogee directly above the launch pad.

3.8 Subscale Vehicle

In order to collect measurements on expected behavior of the vehicle before beginning construction, it is necessary to construct a scale model of the system, known as a subscale vehicle. Historically, a 40% scale has proven an effective substitute for the system when testing aerodynamics, both in simulations and launch of the model. The goal of building the subscale system is to accurately verify the design posed in this document, as well as to adjust the design for maximum efficiency. Two subscale vehicles will be constructed to allow for wind tunnel testing and flight testing. Details as to the design, construction, and testing of the subscale model can be found in the following section.

3.8.1 Comparison to Full Scale Vehicle

Two subscale vehicles will be made in order to test the design's performance. This year, the team decided to construct two subscale vehicles rather than one. One of the vehicles will be used for testing in a wind tunnel, and the other will be used for test launches. The scale of both subscale vehicles to the full scale vehicle will be 40%. The subscale vehicles will be constructed with kraft paper, plywood bulkheads, polypropene, and 3D printed ABS parts. These materials were selected with regard to cost effectiveness and because they have proven to be reliable when creating sub-scale vehicles in the past for the team. Additionally, the motor used for the subscale vehicle will be a G80BT-7.

The subscale recovery system will consist of a single parachute that deploys at apogee with a delayed charge from the motor. The subscale ABS will be tested using 3D-printed tabs on the launch vehicle body in a wind tunnel. The testing of ABS will be discussed in more detail below in Section 3.7.3. The payload will not be tested in the subscale launch vehicle.

Table 34 below provides a comparison between the subscale and full scale vehicle's materials and dimensions for each part.

Table 34: Subscale to full scale comparison

Part	Full Scale Material	Full Scale Dimension	Subscale Material	Subscale Dimension
Nose Cone	ASA plastic	L=24"; d=8"	Polypropylene	L=11.25"; d=3.1"
Payload Bay	Fiberglass	L=20"; d_{out}=8.005"; d_{in}=7.815"	Kraft paper	L=8"; d_{out}=3.1"; d_{in}=3"
Recovery Body Tube	Carbon fiber	L=5"; d_{out}=6"; d_{in}=5.888"	Kraft paper	L=14.4"; d_{out}=2.6"; d_{in}=2.55"
Fins	Carbon fiber	RL=6"; TL=3"; SL=1.5"	Plywood	RL=2.4"; TL=1.2"; SL=0.6"
Fin Can	Carbon fiber	L=12"	Kraft paper	L=12.8"
Transition Section	ASA plastic	L=5"; d_{fore}=8"; d_{aft}=6"	PLA (Polylactic acid)	L=2"; d_{fore}=3.1"; d_{aft}=2.6"
ABS	Various	L=12"	Various	L=4.8"

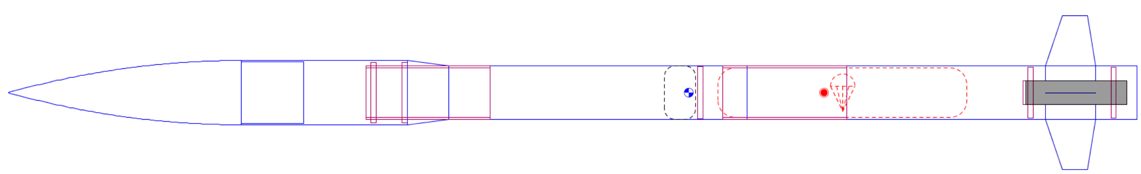

Figure 27: Subscale Vehicle Design

3.8.2 ABS Comparison to Full Scale

In order to determine the drag coefficient of the drag tabs for the ABS final design and ABS computer codes, three subscale wind tunnel tests will be performed with a 40% scale model of the drag tabs attached to the subscale launch vehicle. Three different subscale models of the drag tabs will be able to be secured to and removed from a coupler on the subscale model of the launch vehicle. The different models will represent the drag tabs at full extension, half extension, and fully retracted. By analyzing the resulting drag force on the entire launch vehicle at these different extensions, we will be able to calculate a drag coefficient for the drag tabs, and determine whether it is constant regardless of extension. The tabs will be machined out of Nylon 6/6, which is the same material that will be used in the final design, to ensure that the calculated drag coefficient values from the wind tunnel testing will be as accurate as possible, as the surface roughness will provide the same skin friction. CAD models of the subscale ABS tabs modelling full extension and half extension are included in Figure 29 and Figure 28 respectively.

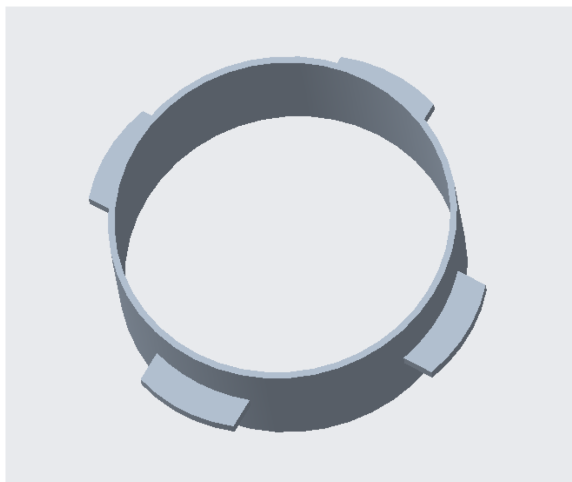

Figure 28: Subscale model of drag tabs at half extension

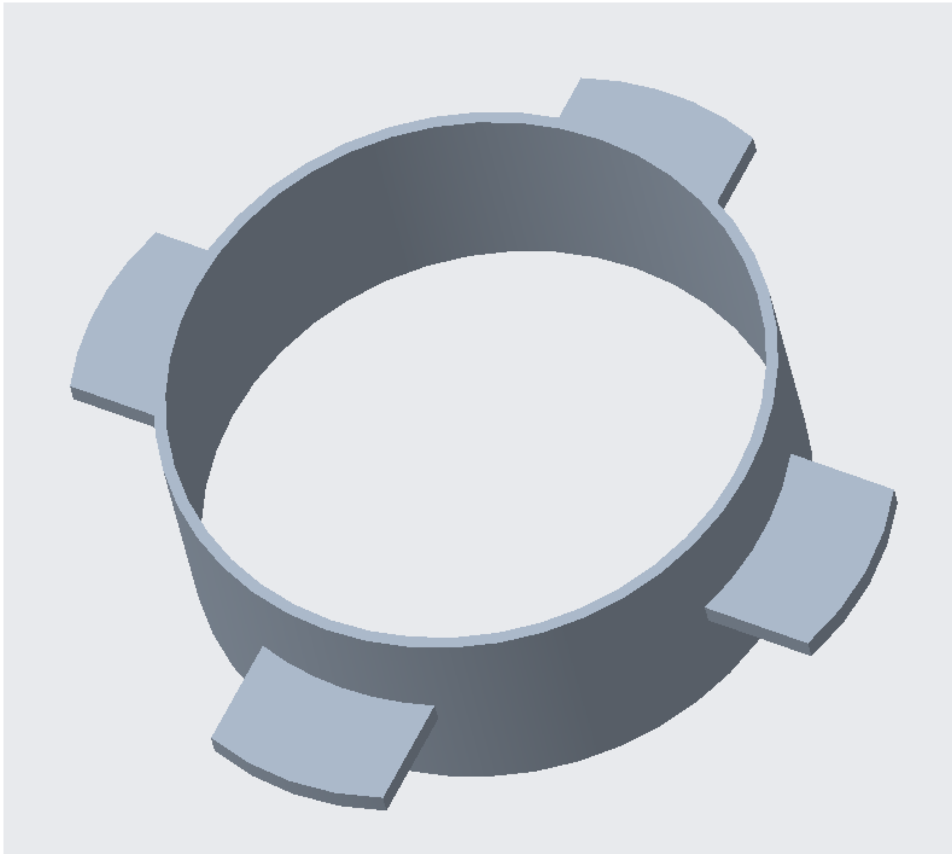

Figure 29: Subscale model of fully extended drag tabs

Two subscale launches will also be used to gather data regarding the ABS. For the first subscale test, the launch vehicle will include drag tabs that are fully deployed to scale with what the full extension on the final launch vehicle will be. For the second subscale test, the launch vehicle will have no tabs deployed. The selected barometer and accelerometer for the ABS will also be included on the subscale launch vehicle in order to gather data throughout the flight and test their functionality compared to recovery's sensors. The difference between the apogees of the two launches will be determined using the sensor data, and compared to the predicted apogee difference based on the drag coefficient measured from wind tunnel testing for further verification of its accuracy. Additionally, the sensor data from the subscale flight will be used to test the filtering and PID algorithms that will be used in the final ABS.

3.8.3 Subscale Testing

Subscale testing will consist of two phases, subscale launch and wind tunnel testing. Accordingly, two subscale vehicles will be constructed. Both will test the overall vehicle design and the effect of the ABS. Wind tunnel testing will consist of three tests in which the ABS tabs are either fully actuated, half actuated, or not actuated at all. This will be modeled with interchangeable parts that have the tabs in the varying positions as mentioned in the previous section. Furthermore, testing will be done with the vehicle positioned at varying angles from zero to fifteen degrees, given that per NASA requirements, the vehicle will be launched at an angle between zero and ten degrees. Wind tunnel testing is scheduled to be completed the week of November 11.

The subscale launch is an essential stepping point to verify that the vehicle's design achieves its purpose and that the ABS tabs affect the vehicle's apogee. It is scheduled to take place on November 10th. There will be two test launches in order to model the effect of the Air Braking System. The first flight will be a control flight, without ABS tabs, while the second will have a model of the tabs in order to show the decrease in apogee as part of the design. This will again be accomplished by interchangeable parts. Data from these two test launches will allow for design variations to be made for full scale construction.

4 Safety

4.1 Safety Officer

Brooke Mumma is the Safety Officer for the Notre Dame Rocketry Team for the 2019-2020 season. The primary responsibility of the Safety Officer is to ensure the safety of all team members, students, and members of the public involved with any activities conducted by NDRT. To ensure this, the safety officer shall ensure that the team abides by all requirements set for the NASA USLI Competition as defined in Section 5.3 of the NASA SLI Handbook in addition to team-derived safety procedures. The Safety Officer will also ensure the the team complies with NAR/TAR regulations.

4.2 Safety Analysis

Hazards are evaluated at a level of risk based on their severity and probability of occurrence. Risks will be evaluated at each subsystem level as well as the project management level. The Systems and Safety team will continue to re-evaluate the risks, mitigations, and verifications as the project continues. Probability of occurrence will be evaluated and designated with values 1 through 5, with 5 being that the event in question is almost certain to happen under present conditions, and 1 being that it is improbable the event occur. The criteria for this scoring is outlines in Table 35 below.

Table 35: Probability of hazard occurrence classification

Description	Value	Criteria
Improbable	1	Less than 5% chance that the event will occur
Unlikely	2	Between 5% and 20% chance that the event will occur
Moderate	3	Between 20% and 50% chance that the event will occur
Likely	4	Between 50% and 90% chance that the event will occur
Unavoidable	5	More than 90% chance that the event will occur

As mentioned, this probability is evaluated according to present conditions, meaning two assumptions were made. The first is that if the conditions change, the probability will be re-evaluated and changed accordingly. The second assumption is that all personnel involved in the activity will have undergone proper training and clearly acknowledged understanding of the rules and regulations outlined in safety documentation. This may include, but not limited to, the safety manual, compiled SDS document, FMEA tables, most recent design review, and lab manual if applicable. The evaluation of occurrence probability will also assume that proper PPE was used, all outlined procedures were correctly followed, and all equipment was inspected before use. Severity of the incident is evaluated on a scale of 1 through 4, where 4 is that the incident will prove catastrophic, and 1 is that the incident will prove negligible. Severity is evaluated according to the incident's impact on personal health and well-being, impact on mission success, and the environment. The score shall be based off of whatever the worst case scenario for the types of impacts being considered. These considerations will be re-evaluated anytime new hazards are identified. The criteria used to evaluate severity of each hazard is outlined are Table 36 below.

Table 36: Severity of hazard classification

Description	Value	Criteria
Negligible	1	Could result in insignificant injuries, partial failure of systems not critical to mission completion, project timeline or outcome possibly affected and might require corrective action, or minor environmental effects.
Marginal	2	Could result in minor injuries, complete failure of systems not critical to mission completion, project timeline or outcome affected and requires corrective action, or moderate environmental .
Critical	3	Could result in severe injuries, partial mission failure, severe impact to project requiring significant and immediate corrective action for project continuity, or severe and reversible environmental effects.
Catastrophic	4	Could result in death, total mission failure, complete failure of project rendering project unable to continue, or severe and irreversible environmental effects.

By combining the severity and probability values, a risk score will be assigned to each hazard. Risk scores will have a value from 1 to 20 where is lowest risk and 20 is the highest risk. Risk levels can be reduced through mitigating actions which will lower either the severity score or the probability score. Actions will be taken starting with the highest risk level hazards, and will continue through the lower levels until all hazards have been reduced as much as possible. All hazards pose a risk and will not be ignored, but the classifications help the Safety officer prioritize resources to those that require the most immediate attention. Mitigations can take the form of design considerations to reduce severity or probability of failure, verification systems created to ensure proper operating conditions, and better handling procedures to follow. Risk scores and the risk levels that correspond with each score are outlined in the risk assessment matrix shown in Table 37, and the description of each risk level is listed in Table 38.

Table 37: Risk assessment matrix

Probability Level	Severity Level			
	Negligible (1)	Marginal (2)	Critical (3)	Catastrophic (4)
Improbable (1)	1	2	3	4
Unlikely (2)	2	4	6	8
Moderate (3)	3	6	9	12
Likely (4)	4	8	12	16
Unavoidable (5)	5	10	15	20

Table 38: Description of Risk Levels and Management Approval

Risk Level	Acceptable Level/Approving Authority
High Risk	Highly Undesirable. Must be approved by Team Captain, Safety Officer, and supervising squad lead.
Medium Risk	Undesirable. Must be approved by Safety Officer and supervising squad lead.
Low Risk	Acceptable. Must be approved by supervising squad lead or Safety Officer.

In order to properly assess the risks facing the mission, key areas for assessment were identified: project risks, personnel hazards, failure modes and effects, and environmental concerns. Each one of these areas was then broken down further into more specific categories of interest and analyzed in the same manner. Each risk is assigned a risk value prior to mitigations and then a risk value after mitigations are in place.

4.2.1 Project Risk Analysis

Table 39: Project Rick Analysis

Hazard	**Cause**	**Outcome**	**Probability**	**Severity**	**Mitigations**	**Verification**
Complete destruction or loss of full scale or subscale vehicle	1. Uncontrolled descent 2. Energetics improperly contained	Team must build an entirely new vehicle causing project delays and doubling the costs of the project	Medium	High	1. All components will be tested individually prior to full-scale assembly 2. Construction procedures will be written prior to construction to ensure reliability of systems	1. Tests will be logged and documented. Multiple sources (calculations, simulations) and trials will be used to verify the results. 2. Construction procedures will be available prior to construction.

Failure to conduct subscale launch by January 10th full scale launch by March 2nd	1. Weather conditions 2. Construction is incomplete 3. Failure to find a date that works with both the team and mentor	Inability to participate in competition	Medium	High	1. Multiple dates will be chosen for a possible launch to provide the team with options. 2. The team will implement a Technology Readiness Level schedule to ensure that all the subsystems are meeting each deadline. 3. The team will push to meet the first available date for launch.	1. The team has chosen February 1st, 15th, and 22nd in order to meet the demonstration flight deadline. 2. The team has a chart to track the individual subsystems TRLs in order to identify any issues with meeting deadlines. 3. The team will begin full scale construction two weeks prior to the first available launch date.
Lack of funds/exceeding budget	1. Allocation of funds to a subsystem is insufficient 2. Parts are not properly sourced	Team takes on debt or funds from travel or other subsystems diminish	Medium	High	1. The allocation of funds are based off of previous years' spending and design. 2. Parts will be sourced to find the best quality at the lowest cost. Each part should be considered from at least three vendors if possible.	1. This years' budget has been set according to previous need and consultation with each design lead. 2. Team members must submit their receipts and add to the budget to ensure they are tracking their spending.

Delay in receiving parts/issues with vendors	1. Parts (especially custom) ordered have an anticipated arrival date that will not work with the team deadlines. 2. The part shipped by a vendor is incorrect or does not meet the needs of the team.	Project delays and/or mission failure	Medium	HIgh	1. Custom parts will be ordered early in order to avoid project delays and if they are critical the team will order an additional component in the case one is damaged. 2. NDRT has compiled a trusted vendor list to ensure quality of parts	1. Any custom parts will be ordered at least three weeks in advance of the start of construction and the design lead will determine whether or not multiples should be ordered. 2. All team members ordering parts will consult the trusted vendor document.
Team member leaves team	1. Injury or illness 2. Member has other commitments	Project delays and/or incomplete work	Medium	Medium	1. All tasks on the team will have multiple members assigned or at least multiple members aware of the details of the task	1. All designs and tests will be well documented in case someone should have to take over.

Safety violations	1. Insufficient PPE 2. Insufficient training	Injury to personnel and the potential for the workshop space to be revoked	Medium	High	1. PPE will always be stocked in the workshop and a part of the Systems & Safety budget. 2. All personnel that will be participating in construction must be certified in the Student Fabrication Lab according to university regulations.	1. The Safety Officer will check for PPE in the workshop prior to all construction. The Safety Officer will be notified when certain PPE items are almost out of stock. 2. Students must show their certification card before entering the workshop during construction.
Insufficient materials	1. Parts to complete the project are not ordered	Project delays	Medium	Medium	1. Personnel will make an itemized list of parts in their designs.	1. Construction procedures will provide a good check to make sure all the parts need for fabrication are ordered
Violation of FAA by exceeding approved altitude	1. Launch site does not have proper waiver for the team's altitude requirement	Potential legal action	Low	High	1. The team will not use any launch sites without the proper waiver	1. The NDRT leadership will confirm with prospective launch sites that they have the proper waiver for NDRT's selected altitude.

Improper testing equipment	1. Test equipment is faulty 2. Inability to use University resources for more complex testing	Incorrect data could lead to faulty analyses and/or design decisions	Low	Medium	1. The team will confirm all tests with calculated results and simulations. 2. The team will reach out to test facilities early to ensure lab time and comply with regulations at each facility.	1. All test results will be documented and shared with the team. 2. The team will reach out to test facilities at least three weeks in advance of the anticipated testing date

4.2.2 Personnel Hazard Analysis

4.2.2.1 Construction

Table 40: Personnel Hazard Analysis-Construction Operations

Hazard	Cause	Outcome	Probability	Severity	Pre	Mitigations	Verification	Probability	Severity	Post
Skin contact with strong adhesive materials, such as epoxy or glue	Not using proper gloves necessary for safe glue/epoxy application	Severe allergic reactions, severe irritation to skin, and damage to skin	3	2	6	Mandating safety gloves and safety training for all team members who will work with adhesives	1. Procedures for utilizing strong adhesives will be created and adhered to by all team members. 2. Procedures for proper use of gloves will created and adhered to by all team members.	1	2	2
Contact with the spinning bit of a portable drill or drill press	Improper technique	Severe damage to fingers and/or other body parts that include but not limited to cutting, scraping, breaking, or amputation	3	4	12	Mandatory safety training for all team members who will work with drills	Team members will be certified for proper use of drills through the review and signing of a safety form and hands-on training with members certified for drill use.	2	4	8

Not properly securing work materials when drilling, sanding, or cutting	Not securing part properly with vise, clamps, or hands during machine use	Blunt bodily damage, cuts, or impalement to the body	2	4	8	Mandatory general workshop safety training for all team members	Team members will be certified on properly securing workpieces through the review and signing of a safety form and hands-on training with members certified for fabrication activities	1	4	4
Contact with the spinning bit of a dremel	Improper technique and poor hand placement	Severe damage to fingers and/or other body parts that include, but not limited to, cutting, scraping, breaking, or amputation	2	4	8	Mandatory safety training will be conducted for all members who use the dremel	Team members will be certified for proper use of dremels through the review and signing of a safety form and hands-on training with members certified for dremels.	1	4	4
Contact with the cutting blade of a bandsaw or scroll saw	Improper sawing techniques, which includes footing, cut speed, and hand placement	Severe damage to fingers and/or other body parts that include, but not limited to, cutting, scraping, breaking, or amputation	2	4	8	Mandatory safety training will be conducted for all members who use the bandsaw	Team members will be certified for proper use of the bandsaw through the review and signing of a safety form and hands-on training with members certified for bandsaw use.	1	4	4

Contact with the sanding surface of a belt sander or a palm sander	Improper sanding techniques	Damage to fingers that include, but not limited to, scrapping, burning, and severe cuts.	3	3	9	Mandatory safety training will be conducted for all members who use the sanders	Team members will be certified for proper use of sanding equipment through the review and signing of a safety form and hands-on training with members certified for sanding equipment use.	1	3	3
Projectiles, shrapnel, or other hazardous materials launched into eyes	Not wearing protective eye gear at all times in the workshop	Temporary or permanent damage to eyes which may lead to future or immediate blindness or degradation of vision	4	4	16	All team members in the workshop will be required to wear safety glasses at all times	Team members will be required to wear safety glasses at all times whenever in the workshop.	2	4	8
Inhalation of airborne particulates resulting from cutting, machining, or sanding parts	Not wearing respirator when generating harmful airborne particulates	Temporary or permanent damage to the lungs which could cause intense pains and long-term health issues	4	4	16	Team members working with potentially harmful fumes will be required to wear proper protective breathing gear	Team members will be certified for proper sanding safety through the review and signing of a safety form and hands-on training with members certified for sanding of materials such as carbon fiber and fiberglass	2	4	8

Extended inhalation of toxic fumes from glue or epoxies	Not wearing protective breathing gear or taking too long when utilizing toxic chemicals	Damage to the lungs that could cause long or short term health effects	4	4	16	Team members working with potentially harmful fumes will be required to wear proper protective breathing gear	Team members will be certified for proper use strong epoxies through the review and signing of a safety form and hands-on training with members certified for strong epoxies	2	4	8
Baggy clothes getting caught in machinery and causing bodily harm	Baggy clothing that hangs too close to machinery when working on parts	Parts of the body could be pulled into machines, causing extensive bodily damage and potentially death	4	4	16	Mandatory general workshop safety training for all team members	Team members will be certified to work in the workshop through the completion of the lab safety quizzes and walkthrough for all of the required machinery for manufacturing.	2	4	8
Blunt bodily damage	Not wearing protective footwear and clothing to protect from falling objects that are blunt or sharp	Damage to the hands and feet that results in breakage or blunt damage	3	3	9	Mandatory general workshop safety training for all team members	Team members will be certified to work in the workshop through the completion of the lab safety quizzes and walkthrough for all of the required machinery for manufacturing.	1	3	3

Burns	Poor 3D printer operational procedures	Hands could receive painful burns that could lead temporary or lasting scarring	2	3	6	Mandatory general workshop safety training for all team members	Team members will be certified to work in the workshop through the completion of the lab safety quizzes and walkthrough for all of the required machinery for manufacturing.	1	3	3

4.2.2.2 Launch Operations

Table 41: Personnel Hazard Analysis-Launch Operations

Hazard	Cause	Outcome	Probability	Severity	Pre	Mitigations	Verification	Probability	Severity	Post
CATO	Imperfections in motor	Motor explodes causing personnel injury	2	4	8	The team mentor, Dave Brunsting, will inspect all motors prior to launch. Dave Brunsting will also install the motor prior to launch to ensire it is installed correctly.	Dave Brunsting will be the only individual to install any motor or energetics and will obey proper rocketry guidelines and procedures when doing so.	1	4	4
Vehicle impact wtih personnel	1. Rocket tips over towards peronnel during launch sequence 2. During recovery the rocket lands on personnel	Personnel injured by rocket impact	2	4	8	The launch platform will be built properly and checked to ensure structural integrity and stability of the rocket off the rail will be verified by simulations and testing. Personnel will be trained in launch proper procedures.	Shake test the launch platform and visually inspect to ensure stability. Results of simulations and tests are consistent and meet requirements. All launch personnel will attend a pre-launch training session.	1	4	4

High temperature of motor when ignited	1. Motor is still hot after landing 2. Personnel is too close to launch pad	Burns to personnel	3	3	9	Personnel will not touch the vehicle immediately after landing. Personnel will stand a safe distance as designated by the RSO at launch (at least 300 ft. as required by the NAR).	Only team leads and mentors are allowed to handle the vehicle after launch. Leads will be properly instructed on how to inspect and handle the vehicle after landing. The Safety Officer will visually confirm the personnel are a safe distance prior to launch.	1	3	3
Pinch-points	Pinch-points created during rocket assembly	Personnel is pinched/cut on their hands	4	1	4	The team leads will enforce the use of hand PPE.	The team will provide and keep hand PPE (gloves, etc) in stock.	2	1	2
Excessive Sunlight	Direct exposure to sun for an extended period of time	Sunburn, increased risk of skin cancer	5	2	10	The team leads will inform personnel attending the launch that they must wear proper clothes for long term exposure to inclimate weather.	Written announcements about potential weather hazards for team personnel will be sent in the full team email. The Safety Officer will provide a reminder during pre-launch training sessions.	2	2	4

Sharp tools for system assemblies	System assemblies may require pliers, scissors, and other sharp tools	Cuts to personnel	3	2	6	Enforcing the use of hand PPE and proper usage of all sharp tools, limiting who will be exposed to using sharp tools. All team personnel will be trained in proper tool handling.	The team will provide and keep hand PPE (gloves, etc) in stock. Leads will verify that personnel using tools has recieved training.	1	2	2
Car accident to/from the launch site	Bad traffic/road conditions to and from the launch site	Personnel injury	2	4	8	Only drivers who are properly certified will be allowed to drive personnel.	Leads will check driver certification before leaving for the launch.	1	4	4
Extreme cold	Inclement weather conditions	Hypothermia	2	4	8	Leads will inform all those attending the launch that they must wear proper clothes for long term exposure to inclimate weather	Leads will ensure that everyone leaving has proper attire.	1	4	4
Payload impact	1. Payload dislodged during launch 2. UAV falls during mission	Personnel injury via impact	2	3	6	NDRT members will be attentive during the launch and trained in proper launch procedures.	Enforce NDRT members using the ”finger pointing” technique to keeps all members’ eyes on the vehicle during launch and the UAV during the mission. Pre-launch training sessions will be conducted before each launch.	1	3	3

Battery chemical burn	Battery for payloads malfunctions during assembly	Personnel recieves chemical burn	3	3	9	Enforce the use of proper eye and hand PPE during the handling of chemical batteries, enforce the proper storage of chemical batteries to limit the damage to said batteries	Provide and keep in stock both hand and eye PPE, visually check to make sure all batteries are properly stored and that PPE is in use during handling. Store battery in fireproof container when it is not in use.	1	3	3

4.2.3 Failure Modes and Effects Analysis

4.2.3.1 Vehicles Flight Mechanics

Table 42: FMEA- Vehicles Flight Mechanics

Hazard	**Cause**	**Outcome**	**Probability**	**Severity**	**Pre**	**Mitigations**	**Verification**	**Probability**	**Severity**	**Post**
Deformation of vehicle	1. Vehicle is damaged due to improper procedures or methods during construction. 2. Vehicle is damaged during transportation from the workshop to the launch facility.	Vehicle does not perform as expected and endures possible damages or complete destruction. Property and people could be injured as a result of an unintended flight path.	2	3	6	Great care will be used when transporting the vehicle, and soft materials will be used to cushion each part of the rocket from potential damages. Vehicle will be inspected prior to flight to ensure it is in proper working condition.	Construction procedures will be written and approved by a safety team member, as well as team leadership. Additionally, all construction will abide by workshop safety protocols as outlines by the Notre Dame Aerospace and Mechanical Engineeering Department.	1	2	2

CATO	1. Imperfections or malfunctions of the motor cause and explosion during motor burn. 2. Motor is incorrectly secured to the vehicle and explodes during motor burn.	Complete mission failure and serious injury or death	2	4	8	All motors will be handled by Dave Brunsting.The construction of the of all components relating to the motor will be overseen by team leadership and will be abide by appropriate construction procedures, which will be written before construction occurs.	Dave Brunsting will examine the motor prior to launch to ensure there are no internal or external malfunctions of the motor.	1	4	4
Failure of motor to ignite	The ignitor and ignition system to not perform correctly and fail to ignite the motor.	The vehicle will not takeoff and the flight will be considered a failure.	2	3	6	All energetics will be handled by Dave Brunsting, who will ensure all components are appropriately in place for the ignition system to work correctly. The components of the systerm will be checked prior to intended take-off to ensure functional capabilities.	Dave Brunsting will examine the motor and ingition system prior to launch to ensure there are no internal or external malfunctions.	1	3	3

Vehicle fails to clear launch rail	1. Deformation of launch rail. 2. Insufficient motor burn result in a velocity smaller than that required to cleaer the rail. 3. Rail buttons deform or break during motor burn due to incorrect manufacturing.	Overall mission failure with potential dangers to propoery and people nearby who may endure injuries due to damages or destruction to the vehicle.	2	4	8	The launch rail and rail buttons will be inspected for deformities or damages prior to setting the vehicle up for launch. The motor will be selected to ensure that the vehicle exits the rail at a minimum of 52 ft/s.	The rail buttons will be tested prior to constructiuon, and will be attached to the vehicle according to construction procedures which will be written before conctruction begins. Dave Brunsting will examine the motor and ingition system prior to launch to ensure there are no internal or external malfunctions.	1	4	4

Failure of vehicle to reach sufficient velocity upon exiting launch rail	Motor does not function as intended, providing insufficient force to meet a velocity large enough to stabilize in flight.	Vehicle moves along an unintended line of motion, likely moving in a horizontal or downward vertical direction during motor burn, endangering property and people in the area. The vehicle will likely impact the ground during motor burn and be destroyed, or endure significant damages.	2	4	8	All motors will be handled by Dave Brunsting. Motor function will be ensured by Dave prior to insertion into the vehicle. The motor will be selected to ensure that the vehicle exits the rail at a minimum of 52 ft/s.	Dave Brunsting will examine the motor prior to launch to ensure there are no internal or external malfunctions of the motor. The team will select a motor that meets requirements.	1	4	4

Fin Flutter	1. Fins are not manufactured to specifications. 2. Fins are not made of the correct material. 3. Fins and fin can are not adequately secured to the vehicle due to failures of adhesives or load-bearing bulkheads.	Vehicle will move along an unintended flight path potentiallly damaging property and endangering bystanders. Vehicle may impact the ground in a nonoptimal fashion further endangering porperty and bystanders.	1	4	4	Fins will be composed of strong, lightwieght material. All adhesives and construction techniques will ensure the fins are secruely attached to the vehicle body. Fins will be selected to ensure a mininum stability margin of 2.0.	Construction and testing procedures will be written and approved by a safety team member, as well as team leadership. Fin design will meet requirements.	1	2	2
Airframe structural failure	1. Vehicle parts (i.e. fins, bulkheads) were attached with epoxy or othwer adhesives incorrectly. 2. Incorrect materials chosen for certain uses result in malfunction.	Vehicle damages or destruction occurs during flight, potentially injuring people or damaging property nearby. The mission would be considered a failure, and payload would likely suffer additionally damages.	2	4	8	Airframe will be inspected prior to launch, at least once in the workshop and at least once at the launch site, to ensure all pieces are connected securely and correctly.	Construction and testing procedures will be written and approved by a safety team member, as well as team leadership. Additionally, all constructiuon will abide by workshop safety protocols as outlines by the Notre Dame Aerospace and Mechanical Engineeering Department.	1	4	4

Premature separation of vehicle	1. Incorrect readings from altimeters and sensors. 2. Latch mechanism breaks and releases prematurely. 3. Restraining cords break during flight and release prematurely.	The laminar air flow around the body of the vehicle would be disrupted, leading to turbulent air during flight. As a result, the vehicle could change trajectory and the flight behavior would become unpredictable. There would be possibilities for damages to the vehicle, payload, nearby property, and people in the area.	2	4	8	Redundancy will be ensured by using several sensors for each reading to ensure one false reading does not trigger a separation. Additionally, materials and parts for the latch and parachute systems will be chosen carefully and tested according to procedures that will be written prior to any construction or testing. Prior to launch, the sensors will be tested to ensure accurate readings, the latch mechanism will be tested for proper functionality, and restraining cords will have to pass stress tests.	Construction and testing procedures will be written and approved by a safety team member, as well as team leadership. There will be three redundant black powder charges and altimeters within the CRAM.	1	4	4

4.2.3.2 Vehicles Structures

Table 43: Vehicles Structures

Hazard	Cause	Outcome	Probability	Severity	Pre	Mitigations	Verification	Probability	Severity	Post
Deformation of vehicle	1. Vehicle is damaged due to improper procedures or methods during construction. 2. Vehicle is damaged during transportation from the workshop to the launch facility.	Vehicle does not perform as expected and endures possible damages or complete destruction. Property and people could be injured as a result of an unintended flight path.	2	3	6	Great care will be used when transporting the vehicle, and soft materials will be used to cushion each part of the rocket from potential damages. Vehicle will be inspected prior to flight to ensure it is in proper working condition.	Construction procedures will be written and approved by a safety team member, as well as team leadership. Additionally, all constructiuon will abide by workshop safety protocols as outlines by the Notre Dame Aerospace and Mechanical Engineeering Department.	1	2	2

CATO	1. Imperfections or malfunctions of the motor cause and explosion during motor burn. 2. Motor is incorrectly secured to the vehicle and explodes during motor burn.	Complete mission failure and serious injury or death	2	4	8	All motors will be handled by Dave Brunsting.The construction of the of all components relating to the motor will be overseen by team leadership and will be abide by appropriate construction procedures, which will be written before construction occurs.	Dave Brunsting will examine the motor prior to launch to ensure there are no internal or external malfunctions of the motor.	1	4	4
Failure of motor to ignite	The ignitor and ignition system to not perform correctly and fail to ignite the motor.	The vehicle will not takeoff and the flight will be considered a failure.	2	3	6	All energetics will be handled by Dave Brunsting, who will ensure all components are appropriately in place for the ignition system to work correctly. The components of the systerm will be checked prior to intended take-off to ensure functional capabilities.	Dave Brunsting will examine the motor and ingition system prior to launch to ensure there are no internal or external malfunctions.	1	3	3

Vehicle fails to clear launch rail	1. Deformation of launch rail. 2. Insufficient motor burn result in a velocity smaller than that required to cleaer the rail. 3. Rail buttons deform or break during motor burn due to incorrect manufacturing.	Overall mission failure with potential dangers to propoery and people nearby who may endure injuries due to damages or destruction to the vehicle.	2	4	8	The launch rail and rail buttons will be inspected for deformities or damages prior to setting the vehicle up for launch. The motor will be selected to ensure that the vehicle exits the rail at a minimum of 52 ft/s.	The rail buttons will be tested prior to constructiuon, and will be attached to the vehicle according to construction procedures which will be written before conctruction begins. Dave Brunsting will examine the motor and ingition system prior to launch to ensure there are no internal or external malfunctions.	1	4	4

Failure of vehicle to reach sufficient velocity upon exiting launch rail	Motor does not function as intended, providing insufficient force to meet a velocity large enough to stabilize in flight.	Vehicle moves along an unintended line of motion, likely moving in a horizontal or downward vertical direction during motor burn, endangering property and people in the area. The vehicle will likely impact the ground during motor burn and be destroyed, or endure significant damages.	2	4	8	All motors will be handled by Dave Brunsting. Motor function will be ensured by Dave prior to insertion into the vehicle. The motor will be selected to ensure that the vehicle exits the rail at a minimum of 52 ft/s.	Dave Brunsting will examine the motor prior to launch to ensure there are no internal or external malfunctions of the motor. The team will select a motor that meets requirements.	1	4	4

Fin Flutter	1. Fins are not manufactured to specifications. 2. Fins are not made of the correct material. 3. Fins and fin can are not adequately secured to the vehicle due to failures of adhesives or load-bearing bulkheads.	Vehicle will move along an unintended flight path potentiallly damaging property and endangering bystanders. Vehicle may impact the ground in a nonoptimal fashion further endangering porperty and bystanders.	1	4	4	Fins will be composed of strong, lightwieght material. All adhesives and construction techniques will ensure the fins are secruely attached to the vehicle body. Fins will be selected to ensure a mininum stability margin of 2.0.	Construction and testing procedures will be written and approved by a safety team member, as well as team leadership. Fin design will meet requirements.	1	2	2
Airframe structural failure	1. Vehicle parts (i.e. fins, bulkheads) were attached with epoxy or othwer adhesives incorrectly. 2. Incorrect materials chosen for certain uses result in malfunction.	Vehicle damages or destruction occurs during flight, potentially injuring people or damaging property nearby. The mission would be considered a failure, and payload would likely suffer additionally damages.	2	4	8	Airframe will be inspected prior to launch, at least once in the workshop and at least once at the launch site, to ensure all pieces are conencted securely and correctly.	Construction and testing procedures will be written and approved by a safety team member, as well as team leadership. Additionally, all constructiuon will abide by workshop safety protocols as outlines by the Notre Dame Aerospace and Mechanical Engineeering Department.	1	4	4

Premature separation of vehicle	1. Incorrect readings from altimeters and sensors. 2. Latch mechanism breaks and releases prematurely. 3. Restraining cords break during flight and release prematurely.	The laminar air flow around the body of the vehicle would be disrupted, leading to turbulent air during flight. As a result, the vehicle could change trajectory and the flight behavior would become unpredictable. There would be possibilities for damages to the vehicle, payload, nearby property, and people in the area.	2	4	8	Redundancy will be ensured by using several sensors for each reading to ensure one false reading does not trigger a separation. Additionally, materials and parts for the latch and parachute systems will be chosen carefully and tested according to procedures that will be written prior to any construction or testing. Prior to launch, the sensors will be tested to ensure accurate readings, the latch mechanism will be tested for proper functionality, and restraining cords will have to pass stress tests.	Construction and testing procedures will be written and approved by a safety team member, as well as team leadership. There will be three redundant black powder charges and altimeters within the CRAM.	1	4	4

4.2.3.3 Air Braking System

Table 44: Air Braking System

Hazard	Cause	Outcome	Probability	Severity	Pre	Mitigations	Verification	Probability	Severity	Post
Power failure in electrical system	Broken circuits from poor construction/ launch forces or the use of under charged batteries	Shutdown of the electrical system and loss of control of ABS tabs causing an overshoot of target apogee	3	4	12	Checking of the battery, connections, and electronic components before launch time	1. Procedures for properly constructing/testing circuits will be created and properly adhered to by all members. 2. Procedures for properly charging and checking the batteries will be created and adhered to by all members. 3. Connections will be tested prior to launch with a multimeter.	2	4	8
Incorrect or unavailable sensor data	Improper installation and programming of the sensors, or loss of power to the electrical system	Improper data transmission to flight computer that causes improper deployment of air braking system	3	4	12	Testing the code and electrical components of the rocket before launch	Procedures for proper testing of ABS hardware and software will be implemented and utilized to verify the integrity of flight hardware before launch.	1	4	4

Improper commands signal from microcontroller	Improper coding of the electronic system or unexpected errors when computing live sensor data	ABS not fully deploying or partially deploying the flaps. Inability for ABS to function properly and slow down the rocket	1	4	4	Testing the code for the system and components before launch in a proper testing environment.	Procedures for proper testing of ABS hardware and software will be implemented and utilized to verify the integrity of flight hardware before launch.	1	4	4
Broken mechanical system for the Air Braking System	Damage from launch, improper construction techniques, or other outside damage to the ABS of the rocket	The ABS gets stuck open or closed and causes the rocket to not reach or pass the targeted altitude	2	4	8	Extensive hardware testing of the finalized Air Braking System to ensure the mechanical design functions properly	Procedures for proper testing of ABS hardware and software will be implemented and utilized to verify the integrity of flight hardware before launch.	1	4	4

Loss of structural integrity of drag tabs	Excessive exposure to launch forces or mechanical failure within tab deployment mechanism	Drag tabs are unable to deploy or break off the rocket, causing the complete loss of the drag system to control the final altitude	2	4	8	Load testing the final Air Braking System drag tabs to ensure that they can withstand forces up to and past those expected during launch and extensively testing the final mechanism of the ABS	1. Procedures for proper testing of ABS hardware and software will be implemented and utilized to verify the integrity of flight hardware before launch. 2. Load tests will conducted and compared to launch simulations to verify the ABS drag tabs will withstand the forces felt before their deployment after the rocket launch.	1	4	4
Failures in integration components that anchor the Air Braking System within the rocket	Improper construction or excessive forces on the rocket during its ascent that cause structural damage to the ABS	The ABS fails to properly deploy and potentially shifts within the body tube of the rocket, causing severe changes to the mass distribution of the rocket	2	4	8	Team members will work together to check each other's work and follow proper construction procedures and final rocket assembly procedures on launch day to assure the Air Braking System is properly integrated with the rocket	Procedures for the proper assembly and testing of the integration techniques for the ABS will be created and adhered to by all teams members during the construction of the rocket	1	4	4

4.2.3.4 Recovery

Table 45: Recovery

Hazard	Cause	Outcome	Probability	Severity	Pre	Mitigations	Verification	Probability	Severity	Post
Failure of vehicle to separate at apogee	1. Black powder charges are not powerful enough for separation 2. Avionics are not turned on, and black powder charges do not function.	Rocket will descend at higher than intended speeds, likely leading to damages of both the vehicle body, payload, surrounding property, as well as injuries to people in the area.	2	4	8	Redundancy of black powder charges, as well as avionics will be guaranteed before takeoff using appropriate launch procedures	Launch procedures will be written and followed to ensure all components are properly inspected and assembled.	1	4	4
Failure of vehicle to separate at main parachute deployment altitude	1. Altimeters do not record correct altitude readings. 2. Black powder charges act later than intended.	Vehicle will not slow down upon descent and will impact the ground at a speed large enough to damage the vehcile and components of the payload.	2	4	8	Redundancy of black powder charges, as well as avionics will be guaranteed before takeoff using appropriate launch procedures.	Launch procedures will be written and followed to ensure all components are properly inspected and assembled.	1	4	4

Failure of vehicle to separate at correct altitude	1. Altimeters do not record correct altitude readings. 2. Black powder charges act later than intended.	Vehicle will descend at maximum possible speeds until the main parachute deploys, which would likely result in a force great enough to damage components of the vehicle or payload.	1	4	4	Redundancy of black powder charges, as well as avionics will be guaranteed before takeoff using appropriate launch procedures.	Launch procedures will be written and followed to ensure all components are properly inspected and assembled.	1	4	4
Failure of parachute to adequately slow down the vehicle	1. Improperly sized parachute. 2. Parachute is deployed at an improper time. 3. Parachute is tangled and does not deploy correctly. 4. Black powder charges damage some or all of the parachute upon deployment at apogee.	Rocket descends at higher-than-intended speeds, leading to higher impact upon landing. There is potentail for damages to both the vehicle body and the payload, as well as potential in juries to bystanders of luanch parties.	2	4	8	All calculations and simulations will be verified by a secondary simulation or source. Parachute size will be determined by multiple analyses to ensure it is the appropriate size.	Calculations and/or simulations must be consistent with one another. At least two analyses must be performed to verify design choices.	2	2	4

Parachute separates from the vehicle upon descent	1. Broken component relevant to connected the separated pieces of vehicle (i.e. eyebolt or shock cord).	Sections of the vehicle descend at high speeds. Impact on the ground will result in damages to both the vehicle body and payload. Additional potential for injury to bystanders or launching parties.	2	4	8	All mechanical components of the recovery system will be either new or adequately tested to ensure functional success.	Testing procedures will be written and followed carefully by all individuals involved in testing.	1	4	4
Vehicle drifts further than expected during descent	1. Parachutre deploys earlier than expected. 2. Parachute is an improper size.	Vehicle could drift outside the launch field, unintentionally injuring people or damaging property not protected within the driuft radius. Payload mission success will be affected negatively when deploying further from the target.	3	2	6	Redundancy of black powder charges, as well as avionics will be guaranteed before takeoff using appropriate launch procedures.	All energetics will be handled by Dave Brunsting. All avionics and electronics within the recovery system will be inspected according to launch procedures written prior to the launch date.	2	2	4

Vehicle separates during motor burn	1. Incorrect readings from altimeters and sensors. 2. Latch mechanism breaks and releases prematurely. 3. Restraining cords break during flight and release prematurely.	The vehicle would shear, the interior components on the rocket (CRAM and payload) would be heavily damaged, and bystanders may be injured.	2	4	8	Redundnacy of avionics will be ensured so that one incorrect reading does not lead to mission failure. All mechanical components will be tested prior to construction and launch to ensure they are made of appropriate materials. All restraining cords will be tested prior to launch by the recovery squad.	All construction, testing, and launch procedures will be written prior to becoming necessary and will be followed and enforced by team leadership.	1	4	4
Avionics Module (CRAM) separates from rocket body	1. The material used to construct the CRAM is insifficient in supporting the necessay loads of both the main and drog parachutes	Parachutes and vehicle body sections would tangle and likely increase in descent velocity, impacting the ground at high speeds. Payload, vehicle components, and nearby property anf people would likely all be damaged.	2	4	8	The CRAM will be manufactured out of a material sufficient in bearing a load from parachutes on both sides. The CRAM will be secured using a twist-to-lock mechanism to restrict translation through the vehicle body.	All construction, testing, and launch procedures will be written prior to becoming necessary and will be followed and enforced by team leadership.	1	4	4

4.2.3.5 Payload Vehicles

Table 46: Payload Vehicles

Hazard	Cause	Outcome	Probability	Severity	Pre	Mitigations	Verification	Probability	Severity	Post
Fire	1. Lithium ion batteries on either the UAV or the rover vibrate too much during flight or impact and ignite as a result. 2. Wires within the UAV or rover systems short, causing a fire to ignite.	Payload vehicles are damaged, or destroyed, due to the fire. Additionally, the nose cone and payload bay would also be damaged as a result of an internal fire.	2	4	8	All lithium ion batteries will be checked and tested according to procedures that will be written prior to any testing or launching of the payload vehicles. All batteries will be secured to the payload vehicles to minimize vibrations. All wiring of the payload vehicles will be checked by at least three members, including team leadership.	A launch and safety checklist will be developed and followed to ensure all safety measured are taken prior to liftoff.	1	4	4

Nose cone does not spearate from the payload bay.	1. The UAV deploys prior to proper clearance from the payload bay. 2. The UAV deploys before the nose cone is cleared from the payload bay.	Propellers bare damaged or break. Additionally, the UAV may damage the rover's functional capabilities and parts of the deployment system.	2	4	8	Testing and construction procedures for the nose cone and payload systems will be written prior to these actions occuring. All procedures will be carefully adhered to.	Multiple members, including team leadership, will be involved in testing and construction of the payload system and nose cone removal mechanism.	1	3	3
UAV power switch not turned on before flight	Member of the team fails to turn the power on	UAV is not able to deploy or function, resulting in mission failure.	2	4	8	Pre-flight checks and procedures will be written prior to the launch date to ensure that the power switch is turned on prior to liftoff.	Multiple team members will be held responsible for the power switch of the UAV. Several members, including team leadership, will check that the switch is in the "on" before launch.	1	4	4
Rover power switch not turned on before flight	Member of the team fails to tuern the power on	UAV is not able to deploy or function, resulting in mission failure.	2	4	8	Pre-flight checks and procedures will be written prior to the launch date to ensure that the power switch is turned on prior to liftoff.	Multiple team members will be held responsible for the power switch of the rover. Several members, including team leadership, will check that the switch is in the "on" before launch.	1	4	4

UAV flight mechanism failure	1. Propellers do not rotate at sufficient speeds to carry UAV to the mission destination. 2. Wires on the UAV detach and disconnect the power supply. 3. UAV is unable to detach from the rover	UAV is not able to fly correctly and likely results in a mission failure because the UAV cannot locate the mission destination prior to failure.	1	4	4	All parts of the UAV will be chekced and tested prior to launch. All construction will follow strict procedures and guidelines to ensure all connections are accurate and all components are functional as intended. The UAV and rover connection and detachment suystem will be checked and tested several times prior to launch to ensure functional capabilities.	Construction, testing, and launch procedures will be created and adhered to. Multiple team members, including team leadership, will complete checklists to ensure each system is capable of functioning correctly.	1	3	3
Battery failure	1. UAV or rover battery does not have enough charge to complete mission. 2. UAV or rover battery is not capable of powering the the respective system for a long enough time period.	The UAV or rover will not be able to function for a long enough to complete the mission, resulting in a mission failure.	2	4	8	Batteries will be chosen carefully for their intended purpose, having a battery life longer than necessary for each payloaf vehicle. A fully charged battery will be guaranteed prior to launch by three or more team members.	Multiple members, including team leadership, will be involved in researching and purchasing an adequate battery. Launch procedures will be written to ensure batteries are handled properly and are capable of powering the payload vehicles for a long enough time.	1	4	4

Disruption in the transmission of radio signals from UAV to rover	1. Transmitters are functioning at an improper frequency and are disrupted by other nearby transmitters. 2. Transmitters lose signal due to a conductive material, such as carbon fiber, inhibiting signal transmissions.	UAV is unable able to become beacon for rover. The rover will not be able to correctly navigate to the mission destination and the mission will be a failure.	1	4	4	All transmitting frequencies will be carefully chosen to avoid overlap with other teams or nearby signals. All construction materials relating to transmitters within the payload system will be chosen carefully to ensure conductive materials are not used or are positioned so as to not interfere with transmitters.	Construction and testing procedures will be written to and adhered to by members of the team to ensure that transmitters work correctly and as intended. At least three members, including team leadership, will be involved in construction and testing. At least one member will have the proper licensing for radio transmitters.	1	2	2

Rover movement mechanism failure	1. Rover component breaks due to impact. 2. Wires on the rover detach and cause the rover to function incorrectly or completely hinder functional capabilities. 3. Rover is damaged by an environmental factor.	Rover is unable to function correctly for a long enough period of time to complete the mission successfully.	2	4	8	All components of the rover will be constructed and tested properly and thoroughly prior to launch. All considerations concerning rover capabilities will be incorporated into rover designs.	Construction, testing, and launch procedures will be created and adhered to. Multiple team members, including team leadership, will complete checklists to ensure the rover is capable of functioning properly in any possible environment to complete the mission successfully.	1	4	4

Sample retrieval mechanism failure	1. Sample retrieval components are damaged or break upon impact or due to a high-force event during flight (i.e. premature parachute deployment). 2. Sample retrieval mechanism is unable to locate the sample from the ground. 3. Sample retrieval mechanism is unablwe to support a sufficient load to complete the mission. 4. Sample retireval mechanism breaks due to fatigue during the mission.	Rover is unable to retrieve a sufficient amount of the provided sample, resulting in mission failure.	2	3	6	All components of the sample retrieval system will be constructed and tested properly and thoroughly prior to launch. All design considerations for the sample retrieval system will be incorporated into the design process and checked by team leadership to esnure a reliable system with minimal possible flaws.	Construction, testing, and launch procedures will be created and adhered to. Multiple team members, including team leadership, will complete checklists to ensure the sample retrieval is capable of functioning properly in any possible environment to complete the mission successfully.	1	2	2

4.2.3.6 Payload Deployment and Integration

Table 47: Payload Deployment and Integration

Hazard	Cause	Outcome	Probability	Severity	Pre	Mitigations	Verification	Probability	Severity	Post
Fire	1. Lithium ion batteries on either the UAV or the rover vibrate too much during flight or impact and ignite as a result. 2. Wires within the UAV or rover systems short, causing a fire to ignite.	Payload vehicles are damaged, or destroyed, due to the fire. Additionally, the nose cone and payload bay would also be damaged as a result of an internal fire.	2	4	8	All lithium ion batteries will be checked and tested according to procedures that will be written prior to any testing or launching of the payload vehicles. All batteries will be secured to the payload vehicles to minimize vibrations. All wiring of the payload vehicles will be checked by at least three members, including team leadership.	A launch and safety checklist will be developed and followed to ensure all safety measured are taken prior to liftoff.	1	4	4

Nose cone does not spearate from the payload bay.	1. The UAV deploys prior to proper clearance from the payload bay. 2. The UAV deploys before the nose cone is cleared from the payload bay.	Propellers bare damaged or break. Additionally, the UAV may damage the rover's functional capabilities and parts of the deployment system.	2	4	8	Testing and construction procedures for the nose cone and payload systems will be written prior to these actions occuring. All procedures will be carefully adhered to.	Multiple members, including team leadership, will be involved in testing and construction of the payload system and nose cone removal mechanism.	1	3	3
UAV power switch not turned on before flight	Member of the team fails to turn the power on	UAV is not able to deploy or function, resulting in mission failure.	2	4	8	Pre-flight checks and procedures will be written prior to the launch date to ensure that the power switch is turned on prior to liftoff.	Multiple team members will be held responsible for the power switch of the UAV. Several members, including team leadership, will check that the switch is in the "on" before launch.	1	4	4
Rover power switch not turned on before flight	Member of the team fails to turn the power on	UAV is not able to deploy or function, resulting in mission failure.	2	4	8	Pre-flight checks and procedures will be written prior to the launch date to ensure that the power switch is turned on prior to liftoff.	Multiple team members will be held responsible for the power switch of the rover. Several members, including team leadership, will check that the switch is in the "on" before launch.	1	4	4

UAV flight mechanism failure	1. Propellers do not rotate at sufficient speeds to carry UAV to the mission destination. 2. Wires on the UAV detach and disconnect the power supply. 3. UAV is unable to detach from the rover	UAV is not able to fly correctly and likely results in a mission failure because the UAV cannot locate the mission destination prior to failure.	1	4	4	All parts of the UAV will be chekced and tested prior to launch. All construction will follow strict procedures and guidelines to ensure all connections are accurate and all components are functional as intended. The UAV and rover connection and detachment suystem will be checked and tested several times prior to launch to ensure functional capabilities.	Construction, testing, and launch procedures will be created and adhered to. Multiple team members, including team leadership, will complete checklists to ensure each system is capable of functioning correctly.	1	3	3
Battery failure	1. UAV or rover battery does not have enough charge to complete mission. 2. UAV or rover battery is not capable of powering the the respective system for a long enough time period.	The UAV or rover will not be able to function for a long enough to complete the mission, resulting in a mission failure.	2	4	8	Batteries will be chosen carefully for their intended purpose, having a battery life longer than necessary for each payloaf vehicle. A fully charged battery will be guaranteed prior to launch by three or more team members.	Multiple members, including team leadership, will be involved in researching and purchasing an adequate battery. Launch procedures will be written to ensure batteries are handled properly and are capable of powering the payload vehicles for a long enough time.	1	4	4

Disruption in the transmission of radio signals from UAV to rover	1. Transmitters are functioning at an improper frequency and are disrupted by other nearby transmitters. 2. Transmitters lose signal due to a conductive material, such as carbon fiber, inhibiting signal transmissions.	UAV is unable able to become beacon for rover. The rover will not be able to correctly navigate to the mission destination and the mission will be a failure.	1	4	4	All transmitting frequencies will be carefully chosen to avoid overlap with other teams or nearby signals. All construction materials relating to transmitters within the payload system will be chosen carefully to ensure conductive materials are not used or are positioned so as to not interfere with transmitters.	Construction and testing procedures will be written to and adhered to by members of the team to ensure that transmitters work correctly and as intended. At least three members, including team leadership, will be involved in construction and testing. At least one member will have the proper licensing for radio transmitters.	1	2	2

Rover movement mechanism failure	1. Rover component breaks due to impact. 2. Wires on the rover detach and cause the rover to function incorrectly or completely hinder functional capabilities. 3. Rover is damaged by an environmental factor.	Rover is unable to function correctly for a long enough period of time to complete the mission successfully.	2	4	8	All components of the rover will be constructed and tested properly and thoroughly prior to launch. All considerations concerning rover capabilities will be incorporated into rover designs.	Construction, testing, and launch procedures will be created and adhered to. Multiple team members, including team leadership, will complete checklists to ensure the rover is capable of functioning properly in any possible environment to complete the mission successfully.	1	4	4

Sample retrieval mechanism failure	1. Sample retrieval components are damaged or break upon impact or due to a high-force event during flight (i.e. premature parachute deployment). 2.Sample retrieval mechanism is unable to locate the sample from the ground. 3. Sample retrieval mechanism is unablwe to support a sufficient load to complete the mission. 4. Sample retireval mechanism breaks due to fatigue during the mission.	Rover is unable to retrieve a sufficient amount of the provided sample, resulting in mission failure.	2	3	6	All components of the sample retrieval system will be constructed and tested properly and thoroughly prior to launch. All design considerations for the sample retrieval system will be incorporated into the design process and checked by team leadership to esnure a reliable system with minimal possible flaws.	Construction, testing, and launch procedures will be created and adhered to. Multiple team members, including team leadership, will complete checklists to ensure the sample retrieval is capable of functioning properly in any possible environment to complete the mission successfully.	1	2	2

4.2.3.7 Launch Support Equipment

Table 48: Launch Support Equipment

Hazard	Cause	Outcome	Probability	Severity	Pre	Mitigations	Verification	Probability	Severity	Post
Launch rail is at high angle with vertical	1. Launch equipment is improperly set up 2. Vehicle is improperly placed on launch pad	Vehicle does not reach apogee	2	3	6	1. Launch equipment will be set up according to NAR standards 2. The NDRT mentor and RSO reccomendatoins will be followed when setting up the vehicle	1. The RSO will verify that launch equipment is properly set up. 2. The vehicle set up will be verified by the team mentor before launch.	1	3	3
Launch controller fails to ignite motor	1. Wire connection or controller is faulty	Motor does not ignite	2	2	4	1. NDRT will use an official rocketry club's controllers	1. NDRT will ensure that the clubs the team launches with are reliable and have a good launch record	1	2	2
Launch ignition wires are live during set up	1. Launch controller unit is faulty	Premature motor ignition may injure personnel	2	4	8	1. All launch equipment will be inspected prior to use.	1. The NDRT mentor along with the local rocketry club will assist in inspecting equipment prior to set up	1	4	4

4.2.4 Environmental Hazards

4.2.4.1 Environmental Hazards to Vehicle

Table 49: Payload Deployment and Integration

Hazard	Cause	Outcome	Probability	Severity	Pre	Mitigations	Verification	Probability	Severity	Post
Rain	Local weather patterns	Potentially severe water damage to electrical circuits, batteries, payload, and the rocket motor	4	4	16	The team will follow the National Association of Rocketry Weather Safety Code, which states that rockets will not be launched in unsafe weather conditions or low cloud cover.	The team will not launch in rain or cloud cover.	1	4	4
Lightning	Local weather patterns/Heavy Rain	Can damage/short circuit the electrical components, batteries, payload, and change the course of the rocket after launch	2	4	8	The team will follow the National Association of Rocketry Weather Safety Code, which states that rockets will not be launched in unsafe weather conditions or low cloud cover.	The team will not launch in inclement weather.	1	4	4

High Winds	Local weather patterns	Potentially severe structural damage in the event of the rocket falling over, as well as launch trajectory issues with very powerful winds	3	4	12	The team will follow the National Association of Rocketry Weather Safety Code, which states that rockets will not be launched into winds exceeding 20 miles per hour.	The team will not launch if there are winds greater than 18 mph.	1	4	4
Snow	Local weather patterns	Potentially severe water damage to electrical circuits, batteries, payload, and the rocket motor	2	4	8	The team will follow the National Association of Rocketry Weather Safety Code, which states that rockets will not be launched in unsafe weather conditions or low cloud cover.	The team will not launch if there is snow or low cloud cover.	1	4	4
Extreme Temperatures	Local weather patterns	Potential damage to the battery and weakening of bonding materials within the rocket	2	4	8	The team will follow the National Association of Rocketry Weather Safety Code, which states that rockets will not be launched in unsafe weather conditions.	The team will not launch in extreme temperatures.	1	4	4

Low Cloud Cover	Local weather patterns	Turbulent air that could make launch and recovery operations difficult	3	2	6	The team will follow the National Association of Rocketry Weather Safety Code, which states that rockets will not be launched in unsafe weather conditions or low cloud cover.	The team will not launch if there is rain or low cloud cover.	1	2	2
High Humidity Levels	Local weather patterns	Could affect the bonding materials of the rocket as well as the rocket propulsion material (fuel)	4	4	16	All components that can be damaged by water will be housed in a waterproof casing.	Any casings will be tested to ensure reliability.	1	4	4
UV exposure from the Sun	No cloud cover over launch area	Potentially severe damage to the electronics and sensors within the rocket if significant exposure occurs	3	4	12	Any parts that can be damaged by UV exposure will be properly covered.	The team will test to any UV protective casings.	1	4	4

Freezing Rain	Local weather patterns	Potentially severe water damage to electrical circuits, batteries, payload, and the rocket motor	2	4	8	The team will follow the National Association of Rocketry Weather Safety Code, which states that rockets will not be launched in unsafe weather conditions or low cloud cover.	The team will not launch if there is rain or low cloud cover.	1	4	4
Hail/Sleet	Local weather patterns	Potentially severe water damage to electrical circuits, batteries, payload, and the rocket motor	2	4	8	The team will follow the National Association of Rocketry Weather Safety Code, which states that rockets will not be launched in unsafe weather conditions	The team will not launch if there is hail or sleet.	1	4	4
Local Terrain and Man-Made Structure Interference	Local terrain and the natural environment around the launch site	Could interfere with the course of the rocket and cause damage to the rocket and/or potentially a crash/destruction of the rocket	2	4	8	Closely monitoring local natural topography and man made structures near the launch area	Proper surveillance procedures will be created and implemented on launch by two or more members on the rocket for local terrain and structures in the area	1	4	4

Animal Interference	Local animal population in and around the launch site	Potential structural damage to the rocket and potentially lethal damage to the animal	2	3	6	Closely monitoring local animal movements and local species in the launch area	Proper surveillance procedures will be created and implemented on launch by two or more members on the rocket for local terrain and structures in the area	1	3	3

4.2.4.2 Vehicle Hazard to Environment

Table 50: Vehicle Hazard to Environment

Hazard	Cause	Outcome	Probability	Severity	Pre	Mitigations	Verification	Probability	Severity	Post
Fiberglass particulates (styrene gas)	Sanding of bulkhead or other fiberglass materials inside rocket	1. Cause skin, eye, and respiratory tract irritation to surrounding individuals 2. Emission of toxins depletes air quality.	3	3	9	The quanity of styrene gas emitted based on the amount of fiberglass utilized has negligable effects on the environment.	All members will wear proper PPE. Sanding will be conducting in ventilated area and the shop vaccum running.	3	1	3
Excessive CO2 emission	1. Motor produces CO2 emissions when ignited 2.The black powder charges in the recovery system produces CO2 emissions when ignited.	Contribute to greenhouse effect and increase global warming	5	1	5	CO2 emissions are negligible. All energetics will be tested in a large outdoor space and will be handled by team mentor, Dave Brunsting.	Motors and black powder charges will be inspected by the team mentor.	4	1	4

Hydrogen chloride emmission	Ammonium perchlorate motor produces hydrogen chloride	Reacts with water to form hydrochloric acid, contaminating water	4	1	4	Launches will take place away from water sources in order to prevent contamination.	Leads will survey the land to ensure the launchpad is placed away from water sources. Motors will be disposed of according to SDS and local standards.	2	1	2
Components come loose from vehicle	Improper retention of components	Wildlife could potentially ingest or be harmed by materials	3	3	9	Exterior and interior of rocket will be inspected prior to launch in order to ensure security of all components	Pre-launch checklists will be created and followed in order to ensure all components have been inspected and properly integrated.	1	3	3
Battery leakage	Defective batteries that fail to enclose the acid in its appropriate space.	1. Absorption of acid contaminates soil 2. Pollution of groundwater.	2	4	8	Batteries will be housed in a battery bag when not in use. Batteries will be inspected by a lead before and after each use for defects.	Batteries will be inspected by a team lead and batteries will be disposed of according to the SDS sheets and local regulation	1	4	4

Spray paint on vehicle	Use of spray paint to paint exterior of vehicle	Release of toxic emissions into the atmosphere	4	2	8	Spray painting will be executed within the confines of a ventilated area to reduce concentration of air contamination	Area will be well ventilated and only contain personnel participating in painting.	4	1	4
Plastic Waste	Plastic waste can be produced by 3D printing and other construction procedures	Wildlife could potentially ingest or be harmed by plastic	5	2	10	Plastics will be disposed of according to applicable SDS and local standards.	The workshop will contain a specific bin for recycling certain plastics in order to reduce waste.	4	1	4
Wire Waste	Excessive wire scraps as a result of electrical component construction	Wildlife could potentially ingest or be harmed by wire waste	5	2	10	Wire will be disposed of according to applicable SDS and local standards.	The workshop will contain a specific bin for recycling certain electronic components in order to reduce waste.	4	1	4
Soldering Material Waste	Excess materials improperly disposed of during the soldering of wires	Soldering releases toxic that can contaminate the air quality	4	2	8	As longer as proper ventilation is utlized, the release of toxins will be negligable on the environment.	Members involved in soldering will be certified. Disposal will be monitored according to SDS and local guidelines.	4	1	4

Grass fire	1. Motor burnout 2. Electrical components short circuit	1. Damage to surrounding grass 2. Damage to animals' natural habitats 3. Greenhouse emissions as a result of combustion	2	3	6	Wire connections and electronics inside of rocket will be inspected before launch. Approriate extinguishing devices will be on site of launch. Launch pad will be a safe distance above ground.	Pre-launch checklists will be created and followed in order to ensure all components have been inspected and properly integrated.	1	3	3
Damage to nearby property	1. High wind speeds knock vehicle out of expected trajectory 2. Recovery fails to deliver vehicle safely to the ground	Damage to private property and/or damage power lines or environment	3	4	12	Inspect launch equipement. Confirm stability of the vehicle through simulations and testing. Ensure redundant and reliable systems for recovery.	Multiple simulations and tests will confirm the launch vehicle's stability. The recovery system will employ three redundant altimeters.	2	4	8
Noise Impacts	Noise impact	Noise could harm wildlife, bystanders, and potentially vibrate structures.	1	4	4	The impact would be temporary and will not exceed the regulations set by the EPA	Personnel will stand a safe distance as designated by the RSO at launch (at least 300 ft. as required by the NAR).	1	2	2

4.3 Safety Manual

The team has developed a safety manual compiled in the previous year available on the NDRT website. The manual will be updated with current procedures and risks as the year progresses. The safety manual will continue to be updated with regards to

- Machine and Tool Use
- Personal Protective Equipment Use
- Construction
- Testing
- Launch
- Local, State, and Federal Law Compliance
- NAR/TAR Safety Code Compliance
- MSDS Purpose and Use

A physical copy of the Safety Manual shall be kept in the team's workshop, and will be updated to the most current version as revisions are complete.

4.3.1 Material Safety Data Sheets

A current MSDS binder is located in the workshop that accounts for the prospective materials under consideration. The MSDS will continue to be updated as the year continues and designs are finalized.

4.4 Procedures

Prior to construction, procedures will be written to ensure the safety of all personnel and quality of the project. The Systems and Safety Team will work closely with each subsystem squad to develop procedures specific to their design. A physical copy of each procedure shall be available in the workshop prior to construction.

4.4.1 Operation Readiness Reviews

Prior to all launches an Operation Readiness Review (ORR) will be conducted to ensure that all team members are aware of launch procedures and vehicle assembly. All members that intend on going to a launch must attend the ORR for that launch. Attendance will be taken to ensure that members are present. Each ORR will include a discussion of proper launch day safety including NAR/TAR regulations. They will also contain a walkthrough of the assembly of the vehicle and how to prep each subsystem for launch.

4.5 Sub-Scale Rocket Plan

Safety plans specific to construction and launch of the sub-scale are either already implemented or currently being implemented.

4.5.1 Construction

Construction of the sub-scale rocket will be performed solely by members of the team who have achieved at least a Level 1 safety certification from the University of Notre Dame through its Student Fabrication Lab. All necessary PPE, tools, and tool guards for construction have been acquired and implemented. Construction procedures have been developed for the construction. An ORR will be conducted prior to commencement of sub-scale construction

4.5.2 Launch

Launch of the sub-scale rocket will be performed solely by experienced members of the team who have prior experience of launches. All necessary PPE has been acquired for the sub-scale rocket launch, and the team has identified all hazards and failure modes posed by sub-scale launch has ensured that they pose as little threat as possible. The team will abide by the NAR Safety Code and the Launch Procedures outlined by the Michiana Rocketry Club. An ORR will be conducted prior to the sub-scale rocket launch. The team will develop a pre-launch checklist prior to the launch date.

4.5.3 NAR Safety Code Compliance

The Notre Dame Rocketry Team will be taking several steps to ensure compliance with the National Association of Rocketry High Power Rocket Safety Code that has been effective as of August 2012.

4.6 Systems Management

In order to better ensure risks are mitigated the Systems and Safety Team has designated members as a part of each subsystem team. The Systems and Safety Team has implemented a Technology Readiness Level (TRL) schedule with eight distinct levels in order to track each of the subsystems. This has allowed the team to better identify risks and issues that prevent a sub-team from meeting a defined deadline. The team will continue to use this throughout the project life cycle.

5 Technical Design: Payload

5.1 Payload Overview

The Notre Dame Rocketry Team will design, build, and test a payload system that will simulate retrieving lunar ice for the 2019-2020 NASA Student Launch Competition. The system will be comprised of Deployment, an Unmanned Aerial Vehicle (UAV), and a Rover.

5.1.1 Mission Success Criteria

The mission of the payload must accomplish 8 main tasks: (1) withstand forces experienced during vehicle flight and recovery, (2) activate remotely via a signal from the ground station, (3) orient and deploy, (4) locate the closest Future Excursion Area (FEA), (5) transmit the coordinates, (6) traverse to the sample area, (7) retrieve and secure a 10 milliliter lunar sample, and (8) transport the sample 10 feet away.

The mission will be evaluated successful if it meets all payload and safety requirements outlined in the 2020 NASA Student Launch Handbook and the following criteria:

1. The payload shall be powered off until the launch vehicle has safely landed and has been approved for remote-activation by the Remote Deployment Officer.
2. The payload shall remain retained inside the vehicle during vehicle flight and recovery.
3. The payload shall self orient to within 5° of its upright position for deployment.
4. The payload shall deploy from inside the launch vehicle from a position on the ground.

5. The UAV shall locate, fly to, and land at the closest FEA.
6. The UAV shall send its coordinates to the Rover and activate the Rover.
7. The Rover shall traverse to the UAV coordinates and locate the sample area.
8. The Rover shall recover and secure a 10 mL lunar ice sample.
9. The Rover shall move 10 linear feet away from the sample area.

5.2 System Level Trade Studies

5.2.1 Sample Retrieval Vehicle

The sample retrieval vehicle is a critical component of the payload system. The sample retrieval vehicle will be responsible for traversing the launch area terrain and transporting the sample retrieval system to the Future Excursion Area. The vehicles considered were a UAV and a Rover. These vehicles were primarily evaluated based on the traversing ability, durability, sensitivity to small design changes, and the impact they would have on the other designs in the payload system. Table 51 below shows a summary of the conducted trade study.

Table 51: Sample Retrieval Vehicle Trade Study

Criteria	Weight	UAV	Rover
Durability	10%	3	7
Terrain Traversing	20%	9	3
Control Algorithm	15%	4	9
Ease of Deployment	5%	4	8
Design Complexity	20%	4	7
Operating Time	20%	4	7
Adaptability to Design Changes	10%	3	7
Total		4.80	6.55

From table 51 it can be seen that the Rover scored higher than UAV. This design will be pursued as the sample retrieval vehicle due to its durability, adaptability, and greater operating time. Additionally, a Rover will allow for simpler deployment and sample retrieval than the UAV.

5.2.2 Sample Area Reconnaissance

Another key component for the success of the payload system is the sample area reconnaissance system. This system will be responsible for locating the closest Future Excursion Area and transmitting that location to the Sample Retrieval Vehicle. The designs considered were a reconnaissance drone with a camera, a camera placed on the nose cone that would analyse images taken during recovery of the launch vehicle, and a vertical telescope camera mounted on the Rover. These designs were primarily evaluated on their mobility, camera stability, and their field of view. Table 52 below shows a summary of the conducted trade study.

Table 52: Sample Area Reconnaissance Trade Study

Criteria	Weight	UAV with Camera	Camera on Nose Cone	Vertical Telescope
Controllable Mobility	20%	10	1	7
Height	15%	8	10	4
Power Source	10%	4	10	8
Durability	10%	8	9	3
Angle of Vision	15%	10	10	3
Camera Stability	20%	9	1	7
Complexity	5%	7	9	8
Cost	5%	2	9	6
Total		8.15	6.20	5.65

From table 52, it can be seen that the reconnaissance drone scored the highest of the three systems considered. The drone will be pursued due to the ability to control the camera location and the high stability offering clear imaging. Additionally, the drone will be able to survey a large area very quickly. While adding complexity to the payload system, the team is experienced with UAV design and it will provide a more realistic simulation of a lunar sample retrieval system.

5.2.3 Deployment Method

The deployment method of the reconnaissance drone and sample retrieval is an important design selection for the success of the payload system. Improper deployment can lead to incorrect orientation or even catastrophic failure of the payload system. Deployment methods that were considered were a jettison event of the payload system during recovery of the launch vehicle, ground deployment out of the nose cone of the launch vehicle, and ground deployment radially via a hinged door in the payload bay. These deployment methods were primarily evaluated on the complexity, dependability, and the load put on the payload system during deployment. Table 53 below shows a summary of the conducted trade study.

Table 53: Deployment Method Trade Study

Criteria	Weight	Nose Cone	Radial	Jettisoned
Complexity	20%	5	3	8
Dependability	20%	5	5	8
Mechanism Type	15%	7	7	4
Load on Payload	25%	9	9	2
Vehicle Modification	10%	5	2	6
Vehicle Stability	10%	9	7	3
Total		6.70	6.08	5.20

From table 53, it can be seen that ground deployment out of the nose cone scored the highest of the deployment methods considered and will be pursued as the deployment method for the payload system. This method offered a reliable deployment method that put minimal loading on the payload system. Additionally, it is a simpler ground deployment method than radial deployment due to internal orientation and does not require modification of the launch vehicle airframe.

5.2.4 Rover Translation Mechanism

The translation mechanism of the Rover is integral to the ability of the Rover to move across the terrain. The translation mechanism must be robust to handle adverse terrain conditions without immobilizing the Rover and must be capable of overcoming small obstacles. The mechanisms considered were a traditional 4-wheel drivetrain, a rotating belt with treads, and an eccentric crank design. The mechanisms were primarily evaluated on the ability to handle adverse terrain, power consumption, and weight. Table 54 below shows a summary of the conducted trade study.

Table 54: Rover Translation Mechanism Trade Study

Criteria	Weight	Eccentric Crank	Tank Treads	4-Wheels
Dependability on Rough Terrain	20%	6	4	2
Cost	15%	5	3	4
Weight	20%	6	3	4
Availability	10%	1	4	6
Complexity	15%	3	5	6
Power Consumption	20%	4	6	4
Total		4.50	4.20	4.10

From table 54, it can be seen that the eccentric crank mechanism scored highest and will be pursued as the translation mechanism utilized by the Rover. The unique motion of the eccentric crank provides superior maneuverability to the Rover compared to the other two mechanisms and the mechanism operates consistently despite the terrain condition. It utilizes the pre-existing frame to create translation of the Rover and consumes similar power compared to the traditional four wheel mechanism.

5.2.5 Flight Controller

The flight controller for the UAV is required to be bought from a commercial supplier. Given the constraints placed on the UAV, the flight controller is required to operate using Ardupilot software and interface with the other components on the UAV. The controllers considered were the Holybro Pixhawk 4, the Pixhawk 4 Mini, the Holybro Kakute F4, and the Airbot Omnibus F4 Nano V6. These controllers were evaluated on the size, mass, and cost of the controller. Table 55 below shows a summary of the conducted trade study.

Table 55: Flight Controller Trade Study

Criteria	Weight	Pixhawk 4	Pixhawk 4 Mini	Holybro Kakute F4	Omnibus F4 Nano
Size	20%	3	5	8	10
Mass	35%	3	5	8	10
Cost	25%	2	10	7	8
Extra Features	20%	6	6	10	8
Total		3.35	6.45	8.15	8.85

From table 55, it can be seen that the Omnibus F4 Nano scored highest and will be chosen as the flight controller for the UAV. The Omnibus was the smallest and lightest of the flight controllers analyzed and was the most cost effective given the performance in size and mass. Additionally, the Omnibus came with useful interface features with the other UAV components.

5.3 Payload Sub-Systems

The experimental payload is composed of three main sub-systems: Deployment, the UAV, and the Rover. Deployment will be responsible for retaining, orienting, and deploying the Rover and UAV. The UAV will be responsible for locating the closest Future Excursion Area (FEA) and transmitting its location to the Rover. The Rover will be responsible for retrieving and transporting the lunar ice sample. The following sections will go into detail about each sub-system.

5.3.1 Deployment System

The deployment system is an essential element to both the flight of the launch vehicle and the effectiveness of the payload. First, the retention subsystem must prevent the Rover and UAV from moving during flight. Longitudinal motion of the internal components of the launch vehicle can shift the location of the center of gravity, which would affect the static stability of the launch vehicle. This could lead to unstable and unsafe flight conditions. Radial motion of the internal components would affect the flight direction of the launch vehicle, which is undesirable when reaching target apogee. Furthermore, in the event that the nose cone is dislodged during flight or after parachute deployment, there must be a robust system to secure the payload inside the launch vehicle. Premature payload deployment will lead to catastrophic mission failure and more importantly, a safety hazard. The other deployment subsystems, namely deployment and self-orientation, are not required for safety concerns, but are certainly essential for payload mission success.

5.3.1.1 Deployment Mechanism

Given that the payloads are housed in the fore portion of the vehicle, just below the nose cone, there are two possible routes for exit from the launch vehicle. The first is an exit in the longitudinal direction of the vehicle, via nose cone removal. The second option is for the payload to leave the launch vehicle in the radial direction, via the opening of the payload bay. Two separate subsystems were considered for deployment.

The first method involves removing the nose cone and allowing the Rover and UAV to exit the payload bay. The main concern with this method is retaining the nose cone during flight, yet allowing for it to be removed upon landing. In order to solve this issue, a spring-locking system has been proposed. Two matching ring-shaped bulkheads would be epoxied in place, one in the opening of the nose cone, and the other in the foremost opening of the payload bay. Springs would be attached along the nose cone bulkhead. These springs would compress against the payload bay bulkhead as the nose cone is pushed on. While the springs are compressing, protruding metal rods on the nose cone bulkhead would slide into holes in the payload bay bulkhead. Once fully pushed in place, linear servo motors would be actuated to insert locking pins into protruding rods. This rod and pin system would retain the nose cone during flight. In order to remove the nose cone after landing, the linear servo would be actuated to pull the locking pin out of the metal protruding rods. The compression of the springs would produce the force needed to push the shoulder of the nose cone out of the payload bay tube. For the sake of redundancy and extra strength, the rod-pin system would be implemented twice, on opposite sides of the payload bay. The Rover and UAV would exit directly out of the body tube. If need be, the Rover could also push its way past the nose cone once detached. The Rover has a high enough torque to slide the nose cone along the ground once detached. A CAD model of this design is shown in Figure 30.

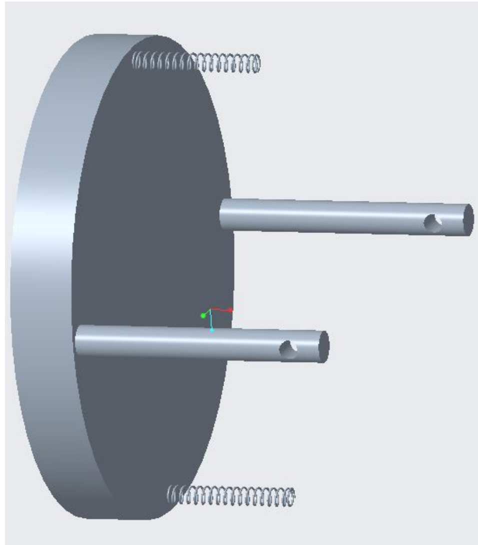

Figure 30: A CAD model showing the spring bulkhead mechanism.

The second method involves opening the payload bay using two hinged body pieces. Upon proper orientation, this would allow the Rover and UAV to exit the launch vehicle with the nose cone remaining attached. This method would involve using orientation correction for the entire payload bay section, as the payload bay opening must not be facing the ground. Furthermore, aerodynamic considerations must be made with this method, as there would be grooves between the body of the vehicle and the hinged doors. The second method is robust in that the nose cone would be permanently attached to a bulkhead throughout flight and recovery. Thus, there would be no nose cone retention concerns. However, this method could be difficult because it would require a rod to run through the entire transition section, to connect the nose cone to the aft body section. For proper orientation correction, several rods would extend radially out of the vehicle once the vehicle has successfully recovered. This would lift the vehicle off of the ground. Then, servo motors would unlock the rotating section of the payload bay. A gyroscope and servo motors would then spin the payload bay such that the opening is clear from obstructions. The doors would then open and allow the Rover and UAV to exit the vehicle. Torsion springs connected to the base of the platform will rotate the Rover and UAV 90°. Thus, the Rover and UAV would exit the launch vehicle in the upright orientation. No battery power would be required to orient the Rover and UAV.

After considering both methods in the trade study of Table 53, it was determined that the nose cone exit method is the best deployment system. Overall, the second method has fewer motors and sensors, and it is more lightweight and less complex. Furthermore, a rigid payload bay transition section will be more cost-effective and easier to construct.

5.3.1.2 Self Orientation System

Self orientation is crucial to the success of the Rover mission. If the vehicle is not oriented properly upon landing, it will be unable to exit the vehicle and complete the mission. On the other hand, the vehicle's rotational motion must also be completely restricted during flight, as it can affect the motion and stability of the vehicle. The self orientation system will consist of a bearing and a rotating cylinder, as a system that utilizes a planetary gear and gyroscope would be heavier. For this design, a small bearing will be embedded in the center of the aft bulkhead. A container will then be attached to the bearing such that the center of gravity is not directly aligned with the center of the bearing. During flight, a pin connected to a linear servo motor will be inserted into the container and will prevent the container from moving during flight. Upon landing, the pin will be removed, and the container will rotate freely about the bearing until it settles in the upright position, with the center of gravity of the container and Rover below the bearing axis.

5.3.1.3 Retention System

As previously stated, the retention system is essential for the payload subsystem. It allows for steady and safe flight of the vehicle from launch to landing. The retention system will not use motors. In order to retain the payload during flight without the use of motors, the container upon which the payload rests will constrain Rover and UAV motion in five directions. This will be done by designing side and top panels that are the exact height and width of the Rover and UAV. The Rover will be extended to its maximum height when stored in the container. Therefore, it will be unable to move in the vertical direction during flight. However, when the Rover begins to exit the payload, the tracks will move, decreasing the height of the Rover. Because the height decreases, it will be able to move directly out of the container. The only direction that will not be constrained by the container during flight is the longitudinal direction towards the nose cone. Motion in this direction during flight will be constrained by an inner bulkhead connected to the nose cone. Upon a safe landing of the launch vehicle, the nose cone will be removed, thus removing the fore Rover bulkhead. Next, the orientation correction system will spin to orient the Rover and container properly. Lastly, the Rover will be able to drive directly out of the nose cone and exit the container. A CAD drawing of the container and back bulkhead are shown in Figure 31.

5.3.2 UAV

The purpose of the UAV is to simulate a lunar orbiting satellite that provides location data to the Rover. Once the payload bay has been opened by the deployment system, the

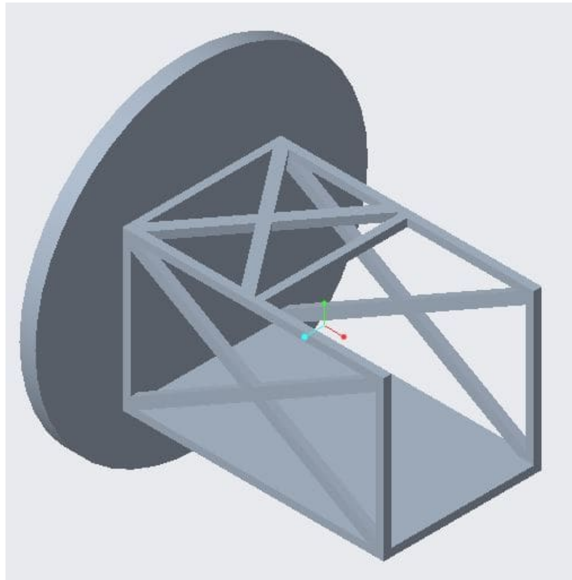

Figure 31: A CAD model showing the back bulkhead container.

UAV will fly from the vehicle. The UAV will ascend vertically up to 100 feet. It will observe the area below for any FEAs using computer vision and target detection algorithms. If an FEA is not found, the UAV will search radially until an FEA is found. The UAV will then descend to the FEA and land in the corner furthest from the Rover and payload bay. Upon landing, it will transmit the GPS coordinates of the FEA to the Rover.

5.3.2.1 UAV Mechanical Design

The purpose of the UAV frame is to provide a rigid structure to support the electronics and battery of the UAV while adding as little weight as possible to the UAV. To accomplish this, the frame must fit all components of the UAV in a compact layout and minimize the amount of mass required to provide the necessary support for the electrical components. The material the frame will be made out of is a key design selection and will determine the success of the UAV frame meeting the mass and strength requirements.

To begin, the design of the UAV frame is constrained in order for the UAV system to integrate well with the other subsystems of the vehicle. The frame was constrained to have a length and width dimension no larger than 4 inches. This constraint ensures the UAV will fit within the payload bay and not take up a large volume of space and thus interfere with the Rover. Additionally, the frame of the UAV is to weigh no more than 2.4 ounces. At this maximum weight, the UAV would still meet the flight time requirements with the other components mounted on the UAV. Lastly, since the UAV will be powered using a lithium-ion battery, the frame of the UAV must house and protect the battery to prevent any damage to the battery during flight operations. Damage to the battery can result in immediate failure of the UAV system and can be a safety hazard to nearby individuals.

To meet the constraints placed on the UAV frame, multiple designs were considered. The team benchmarked and researched frame designs from commercially available products. These consisted of video recording drones, experimental drones, and racing drones. In the end, the majority of the inspiration for the designs came from racing drones as they were designed to hold all necessary components and weigh as little as possible.

To minimize the amount of space the UAV uses in the payload bay, the team sought to build extending arms that the motors would be mounted on. This way, the UAV would have a compact orientation while inside the vehicle. The flight orientation, with extended arms, would occur after deployment from the launch vehicle. However, these designs were found to add too much weight and would require the frame to be very thin, making it fragile. The decision was then made to create a frame that was a rigid structure. This would maximize strength and would minimize the overall volume of the UAV.

Material selection was the next design factor considered. The material of the frame needs to be light yet strong. Additionally, the material needs to be affordable, commercially available, and easily manufactured into the desired shape of the frame. Materials considered were carbon fiber, PLA, and ASA. ASA and PLA were used in the deployable UAV for last year's competition and have been proven to be viable materials for frame construction. Additionally, additive manufacturing allows for utilizing complex designs that would otherwise be unavailable due to the restrictions of subtractive manufacturing. Carbon fiber, however, is a much stronger material that weighs about the same as both PLA and ABS. Since the frame will be no larger than 4 inches by 4 inches, there are commercially available carbon fiber sheets that can be milled using one of the CNC machines available in the Student Fabrication Lab. Additionally, the designs proposed could all be feasibly manufactured with subtractive manufacturing, thus negating the benefits of the additive manufacturing techniques capable with ASA and PLA. Therefore, carbon fiber has been selected as the material of the UAV frame due to its high strength and low weight. The current design of the UAV frame is shown below in Figure 32.

The UAV is designed as two decks joined by five standoffs with four rods under the bottom deck. Because it was decided to use 3 inch diameter propellers, the distance between the center shafts of the motors powering the props needed to be greater than 3 inches. Therefore, 3.1 inch spacing between motors will be utilized in order to prevent the propellers from colliding during flight. The top deck will house all of the electronics, and the bottom

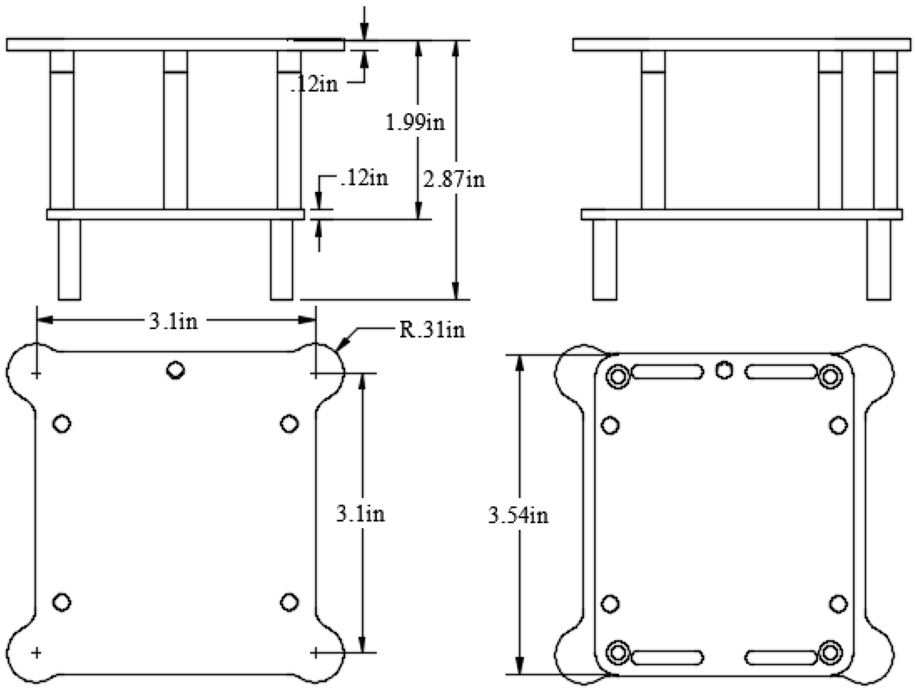

Figure 32: A CAD drawing showing the current design of the UAV frame.

deck will contain the battery. In this manner, the battery will be surrounded by the frame of the UAV and be protected from external damage. The motors will be mounted on the top deck with small spacers placed underneath each motor so that the propellers have sufficient clearance over the electronics. A pair of battery straps will retain the battery during flight. Both decks of the UAV will be made of carbon fiber. The standoffs between the decks will be made out of aluminum, and the landing gear rods will be made of nylon. This will keep the weight of the UAV's frame low while ensuring that the frame is capable of withstanding typical and atypical loads during flight.

The spacers under the motors will be made out of PLA plastic and will be 3-D printed from the 3-D print lab on campus. The decks of the UAV will be manufactured using on of the CNC mills available in the Student Fabrication Laboratory. The nylon rods will be cut to size with a band saw and will have holes drilled using a drill press and will be tapped using a manual tap. The aluminum standoffs will be purchased from McMaster-Carr.

5.3.2.2 UAV Electrical Design

The UAV's greatest constraints are minimizing mass and volume while maintaining a range large enough so that the UAV is able to identify, fly to, and land on the closest FEA. The most important design specification is range, which is essentially flight time, and the second most important constraint is the weight of the UAV. Since the majority of the size and weight budgets of payload have been allocated to the Rover and Deployment systems, the UAV must be as small and light as possible. This presents a design challenge for the the electronics of the UAV because larger props and larger batteries will allow for a larger range, but will infringe on the other components within the payload bay.

When selecting components, the main factor considered was mass. Selecting lightweight components and batteries would allow for the use of a more efficient motor. A more efficient motor draws less power for the same thrust output so a lighter, smaller, lower-capacity battery could be implemented without losing any range. This would overall result in a lighter and smaller UAV capable of locating and flying to the closest FEA while occupying less space inside the payload bay and taking up less of the payload weight budget.

The process of selecting components began by setting design requirements. A desired flight time and a propeller size were chosen first because these had greater effects on the size of the UAV's frame and the other factors considered when choosing electronic components. A flight time of 15 minutes and a propeller size of 3 inches were selected, which would ensure that the UAV's frame will be small and mostly occupied by the battery. Therefore, the components selected were chosen because they were able to meet the UAV specific requirements while occupying a small volume, being lightweight, and consuming little power while operating.

The first component selected was the flight controller. The UAV requires software that is capable of flying an automated flight path, then use camera data to locate and approach the FEA. This requires special software that is not able to run on all flight controllers, so the chosen flight controller is required to be compatible with software capable of controlling the UAV in this manner. Research into what software is used in search-and-rescue drones, which perform a similar task and therefore have similar control requirements, revealed that Ardupilot is the most common program for such applications. Ardupilot is relatively easy to use, supported by many flight controllers, and free. For these reasons, Ardupilot was selected as the flight software which further restricted the selection of flight controllers to the list of compatible flight controllers on the Ardupilot website. This list includes the Holybro Pixhawk 4, the Pixhawk 4 Mini, the Holybro Kakute F4, and the Airbot Omnibus F4 Nano V6. A trade study comparing the different flight controllers resulted in the selection of the Omnibus F4 Nano. This guided the selection of sensors because all of the sensors needed to be able to interface properly with the Omnibus F4.

The next component to be selected was the camera. The camera needed to be small and light with a large field of view. Given that the considered cameras were intended to be used for drone racing and display live video to First Person View goggles, every camera's resolution would be sufficient to detect the FEA. Thus, the resolution was not a factor in the selection process. The Caddx Turbo EOS2 camera was selected due to its small size and mass, large field of view, and low cost. The mounting holes make securing the camera to the UAV simple.

The video from the camera must be processed to be useful in detecting the FEA, which requires a dedicated processing unit. Since an onboard image processor would increase the weight of the UAV and the current draw from the battery, the decision was made to do all processing using a processor at the ground station. Therefore, there needed to be a way to transmit the video feed to the ground station, which requires a video transmitter. The factors considered for the video transmitter were the size, mass, power consumption, and strength of the transmitted signal. Because the signal strength is tied to power consumption and the antenna's gain, the highest-gain antenna compatible with the video transmitter was another consideration. The highest-gain antenna series found was the Lumenier's AXII 2 line, which uses an SMA connector. Thus, the video transmitters considered all use an SMA connector. The TBS Unify Pro32 Nano 5G8 was selected because it offers 3 different signal power levels, allowing power consumption to be reduced if the maximum power output was not needed. Additionally, it has small form factor, low mass, and a 5.8 GHz signal frequency that allows for an increased baud rate compared to 2.4 GHz signals.

The motor, battery, and ESC selections are all connected so all three components were selected together. It was estimated that the battery was to be somewhere between 200 g and 300 g, which allowed for the selection of a combination of motors and propellers to lift the UAV most efficiently. The main concerns were mass and motor efficiency at hover thrust when using 3-inch bi-blade propellers. Three motors were considered: the EMAX RS1106II 4500 Kv, the RCX H1304 5000 Kv, and the RotorX RX1404V2 4000 Kv. These motors were all made by companies with reputations for high-quality, high-performance motors. All three motors are powerful enough to lift a quad-copter of the mass that was estimated with large margins for maneuverability. Furthermore, all three motors are similar in mass, cost, and current draw. The RCX 1304 5000 Kv motors were selected with RotorX 3020 propellers and an 11.1 V battery because it is the most efficient motor of the three considered despite the lower efficiency with large, high-pitch propellers suffers compared to the other two motors. Its efficiency in the mid-throttle range with the 3x2 inch propeller is also higher than the other motors considered. The maximum thrust is lower but because high-speed performance or agility is not desired, this is not a significant factor. All three motors are most efficient with 3x2 inch bi-blade propellers and are all close enough in mass that the difference is insignificant, so the decision was made to use the RCX 1304 5000 Kv because of the increased efficiency under the conditions in which the motors are expected to operate. The motor's thrust at peak efficiency allows for a battery with a mass around 210 g. The batteries considered were the TATTU 2700 mAh 3s LiPo battery, which has a mass of 195 g, and the Lumenier 3S2P 5000 mAh Li-Ion battery with a mass of 312 g. The increased capacity of the Lumenier 5000 mAh Li-ion battery results in an increased flight time despite the increased current draw, so the Lumenier 5000 mAh Li-ion battery was selected. Last, the ESCs were selected. Each motor draws a maximum of about 10 amps, so any ESC capable of supplying at least 12 amps continuously is sufficient. ESCs that were considered were the Tiger Motor S12A 2-4s 12A, the Lumenier 18A Silk, and the Airbot Ori32 25A 4-in-1. The Ori32 4-in-1 ESC was chosen because it is specifically designed to integrate with the selected flight controller. This only requires a single 8-pin cable rather than 4 separate 2-wire cables that must be soldered. It also simplifies the connection to the battery because only 2 components connect to the battery, and the 4-in-1 ESC is also lighter than 4 single ESCs.

Lastly, the UAV must be powered down in flight so there must be some way to control current flow. The flight controller and ESC are powered on as long as current is able to flow from the battery, so a switch is necessary to interrupt the circuit supplying power from the battery to the flight controller and ESC. This circuit-interrupting component can be triggered mechanically or electrically, so the options are either a relay or a toggle switch. A relay would either need to be active-on and powered on for the duration of the UAV's flight or be active-off and lose power when the UAV is ready to be deployed. A toggle switch would need to be flipped when the UAV is ready to be deployed. Any mechanism to power on the UAV would require physical interaction with the vehicle body because no electronic components will be powered on to control a relay. The three most practical options to interrupt and complete the battery circuit are (1) a button-cell battery mounted on the UAV that powers an active-on relay for the duration of the UAV's flight, (2) an active-off relay on the UAV powered by a battery mounted on the interior of the vehicle, and (3) a toggle switch on the UAV that is triggered by the payload deployment system. The active-on relay adds the most weight to the UAV because it requires a battery as well as a relay and risks the UAV losing power in flight, causing a crash. It also requires a mechanism for the payload deployment system to complete the relay circuit even while the UAV is in flight, and therefore eliminates the active-on relay from consideration. An active-off relay adds about as much weight to the UAV as a toggle switch, but it also requires a way to separate the battery from the relay. A toggle switch is the only purely mechanical solution considered, so it requires no additional electrical power. The active-off relay would be implemented by connecting the wires between the relay and the battery on the interior of the vehicle using quick-disconnect bullet connectors and securing the wires to the doors of the payload bay. When the nose cone is pushed away from the payload bay, the bullet connectors are pulled apart, the circuit powering the relay is broken, and the relay closes to provide power from the UAV's battery to the UAV's electronics. A toggle switch placed on the back right leg of the landing gear was chosen, with current flowing in from the battery's positive terminal and out to the flight controller and the ESC. The switch will be oriented so that its ”ON” position is upward and its “OFF” position is down.

5.3.2.3 Target Detection

The purpose of the UAV is to help guide the Rover to the target. In order to do this, the UAV must first locate the FEA on its own. The UAV will be able to search for the target using an onboard camera, and analyze the data with a ground station. The purpose of the Target Detection Subsystem is to take in the camera data and transform it into information about the location of the target. This information can then be used to guide the UAV into position above the target, which will then help the Rover reach the FEA.

5.3.2.4 Techniques Considered

In designing the Target Detection Subsystem, the team has examined several different techniques. These techniques fit broadly into two categories: data-driven approaches and feature-driven approaches.

The idea of data-driven approaches includes algorithms like convolutional neural networks. These algorithms can be very powerful, and provide insight into which features are important in an image, which may lead to connections missed by a human. One drawback of these algorithms is that they rely heavily on the training data provided to them. In order to properly train a robust model, large amounts of training data must be provided, and that data must be representative of the operating environment. If there are any flaws in the dataset provided, the algorithm could fail in unexpected ways. Another problem with convolutional neural networks is that they are computationally expensive to run, and would probably only give a few frames per second. This low frame rate would be an issue when trying to navigate to the sample extraction area under time pressure from the battery.

On the other hand, feature-driven approaches include several more traditional computer vision approaches. Several features can be examined, including color, texture, and geometry. With this approach, the team would gather video of a proxy yellow tarp meant to simulate the sample extraction area from onboard a UAV. The team could then go through a process of feature engineering in order to determine which characteristics of the image are actually important in detecting the target. This approach comes with the difficulty of coming up with the right combination of filters, but is overall a robust method which should run faster than a data-driven approach given the expensive nature of convolutional neural networks. Going forward, the team plans to pursue a feature-driven approach due to its faster runtime and increased robustness to incomplete training data.

5.3.2.5 Feature-Based Target Detection

The feature-based target detection subsystem can be thought of as a pipeline. It receives an image from the camera, runs image processing techniques, and outputs a direction in which the UAV should move in order to get closer to the sample extraction area. These image processing techniques can be further broken up into steps taken to find the target and steps taken to confirm the existence of the target.

In order to find the target, several transforms will be applied to the input image. One important transform to consider is the changing of color space. Images are usually stored in the RGB format. This format stores each pixel as three 8-bit numbers which each corresponds to the amount of red, green, or blue light in that pixel. However, while being the most commonly used color space, it is not necessarily ideal for identifying the target. A yellow tarp could be measured as having many different RGB values depending on factors like the brightness of the image. Other color spaces, like the hue-saturation-value (HSV) format, can provide a more stable option. In HSV, the color is mostly represented by the hue value. Because of this, changes in the brightness of an image could leave the hue relatively unaffected, allowing for a tighter decision boundary.

After the image has been transformed into a more suitable color space, a binary object map can be created. This is done simply by checking if each pixel in the image falls within a specified range. This range can be calculated by annotating the target in several example frames and examining the statistical distribution of the colors in the image. The goal is to find the color band that uniquely identifies the image while not falsely identifying the surrounding background. Once the object mask has been created, morphological operations can be applied to the image. These operations can both fill in any holes in the identified target and remove any potential false positives randomly scattered throughout the rest of the image. An example of this process can be seen in Figure 33 below:

Figure 33: Target detection example.

The image on the left is a picture taken from a Raspberry Pi camera mounted to a UAV. The image in the middle is the result of applying color thresholding and morphological transformations to the image on the left. Finally, the image on the right is the image on the left, annotated with a black square to represent the calculated location of the center of the target in the image.

Once transformations have been applied to the input image, steps can be taken to confirm the existence of the target in the image. Before any direction can be advocated by the system, it must first confirm that it has actually found the target and is not just finding random specks of dirt that happen to be a similar color to the target. In order to confirm the existence of the target, several geometric features can be analyzed. These features include things like aspect ratio (the width divided by the height of the bounding rectangle), extent (area divided by bounding rectangle area), solidity (area divided by float area), compactness (perimeter squared divided by area), eccentricity (major axis divided by minor axis), and the logarithm of the Hu Moments, a set of features which are transformation-invariant. A combination of these features can be used with a decision algorithm like the support-vector machine, which can determine if a set of geometric features are similar enough to past target images.

After all of these steps have been taken, several pieces of information should be present. The location of the average point of the object mask in relation to the center of the image can be interpreted as a direction in which the UAV should travel. Additionally, the output from the decision algorithm can be used to decide if the direction provided by the object mask is actually reliable. With this output, the target detection system can provide reliable directions to the rest of the UAV. Figure 34 provides a diagram of how the data will be processed within the system, transforming the input image into information about the existence of and directions to the target.

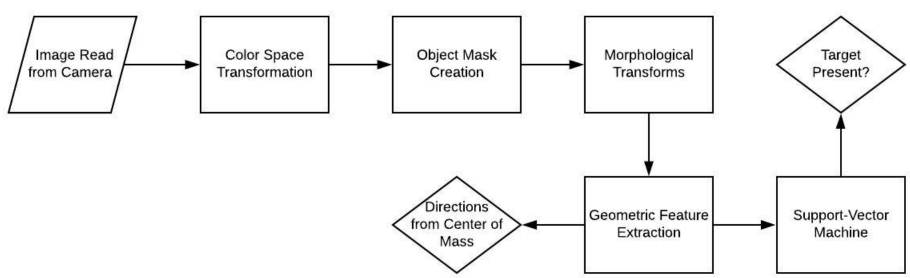

Figure 34: Target detection data flow.

5.3.3 Rover

The Rover will be the vehicle responsible for transporting the sample retrieval system to and from the FEA. The system will deploy once the UAV has landed and transmitted the GPS coordinates of the closest FEA to the Rover. Upon receiving the coordinates, the Rover will autonomously drive from the vehicle to the FEA using the received GPS coordinates. Once the Rover has arrived at the FEA, it will drive to the center of the sample area at which point the sample retrieval system will be initiated. Once the sample of lunar ice has been retrieved, the Rover will drive 10 linear feet from the sample area.

5.3.3.1 Rover Mechanical Design

The mechanical design of the Rover is a critical piece in the success of the Lunar Sample Retrieval System. The Rover body is required to withstand a variety of environmental conditions. To successfully accomplish the mission, the Rover must be able to withstand launch and landing conditions, deploy from the launch vehicle after landing, and travel a maximum distance of 2,500 feet to the sample site. It is also necessary for the Rover to be capable of holding the sample retrieval system, and allow for proper recovery once at the sample site. The Rover has an allocated weight of no greater than 40 ounces and a width of no greater than 6 inches in order to adhere to the payload mission success criteria. These requirements call for a robust system to ensure mission success.

The Rover mechanical design that the team is currently pursuing is an eccentric-crank Rover. An eccentric-crank Rover is a vehicle that is offset from the central axis of the four wheels of the system. A link is eccentrically pinned on the exterior of two wheels on each side of the Rover, 180 degrees out of phase of the body. The eccentricity of the Rover greatly changes the absolute motion of the vehicle, while still remaining a relatively simple design. The basic design of the eccentric-crank Rover is seen below in Figure 35.

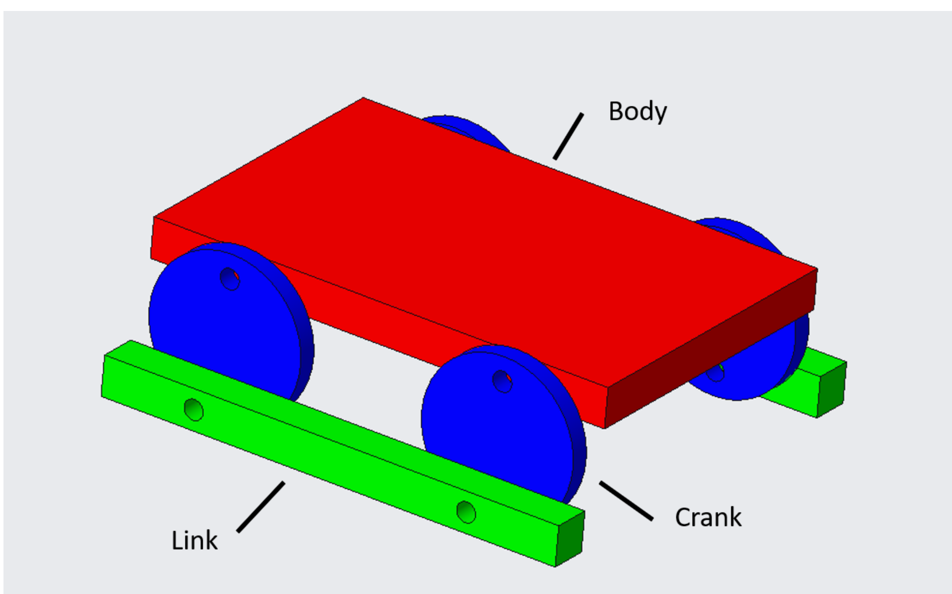

Figure 35: The three main components of the eccentric-crank Rover; the body, the two links, and the four cranks. This basic design was used to help the team visualize the motion of the Rover.

For the rest of this discussion, the wheels will be referred to as cranks, as depicted in Figure 35. When the four cranks are actuated, the body and the two links trace trochoid curves in their motion. Since the body and the links are completely out of phase, the two separate trochoid paths are also completely out of phase. This means that the components of the vertical motion cancels and the full system travels only in the horizontal direction. This enables the Rover to have an effective motion for climbing terrain without sacrificing its ability to reach the sample site. Another important feature of this system is that the weight of both links equals the weight of the Rover. This allows the system to have a constant center of gravity despite the rotation of the individual components. The constant center of gravity keeps the efficiency of the motion nearly as high as a regular four wheeled mechanism, as the system is doing no work in the vertical direction. The motion of the body and cranks is depicted below in Figure 36.

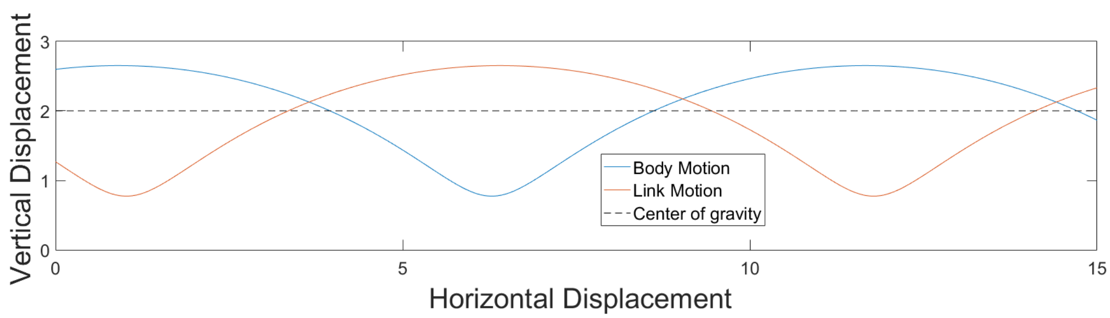

Figure 36: As depicted in this graph, if the Rover body, and two links remain out of phase, the Rover will be able to displace horizontally.

While the eccentric-crank Rover is the design that is currently being pursued, two other general designs were also considered: a traditional four wheeled Rover and a continuous track Rover. The main benefit of a traditional four wheeled vehicle is the simplicity of the drivetrain compared to that of a continuous track. Additionally, the low inertia of the system requires less torque to achieve higher accelerations. However, a four wheeled mechanism generally cannot navigate over hard obstacles that are taller than the radius of the wheel, because the wheel will dig into the terrain instead of going over it. The environmental barriers of launching in a cornfield made this design consideration of the utmost importance when selecting a preliminary design for a rover. The main benefit of a continuous track system is the ability to maneuver in difficult terrain. However, the complexity of the continuous tracks adds extra weight to the Rover, and therefore would significantly limit other aspects of the payload. This complexity also increases the chance of a critical component of the Rover breaking in flight or during operation. A failure in the drivetrain would result in a complete mission failure.

The eccentric crank Rover combines many of the benefits of the four wheeled and continuous track mechanisms, which make it a very suitable candidate for the payload. The eccentricity of the body and cranks give the Rover a significantly higher clearance height for a given crank radius. This allows the Rover to clear obstacles that would have been impossible with a traditional vehicle. The difference in clearance height for the eccentric-crank Rover and a traditional four-wheeled Rover is seen in Figure 37.

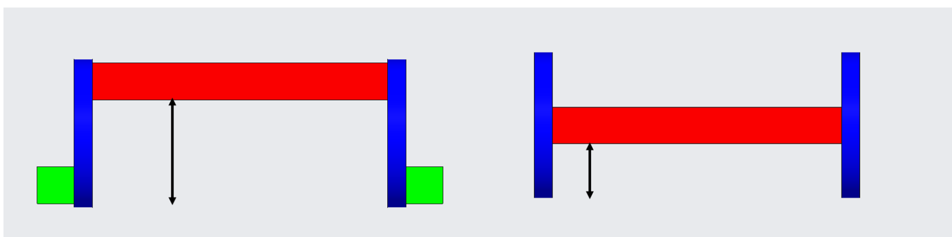

Figure 37: The eccentric crank Rover has significantly higher clearance height for a given crank radius, than a traditional four-wheeled mechanism as seen on the right.

The eccentricity significantly improves the Rover's ability to climb various types of terrain, even when compared to a continuous track vehicle. The rotation of the body and the links allow the vehicle to lift itself off the ground if the Rover encounters difficult terrain. This allows the Rover to climb terrain that is taller than its height. A basic depiction of how the eccentric-crank Rover climbs terrain is seen in Figure 38.

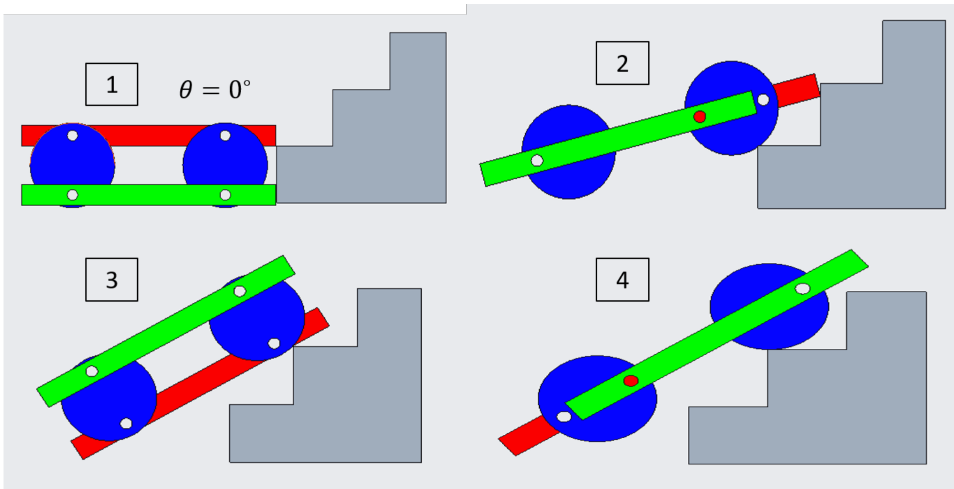

Figure 38: A simplified depiction of how the eccentric-crank Rover navigates over terrain. This figure shows the Rover at positions between $\theta = 0$ and $\theta = 270$.

The team decided to move forward with the eccentric-crank Rover because it is able to climb difficult terrain to the extent of a continuous track vehicle and because the drivetrain is as simple as that of a four wheeled vehicle. The system is controlled by four separate cranks, two on each side: one crank on each side is driven, and the other is passive. The simplicity of this drivetrain allows the eccentric-crank Rover to be accelerated at similar rates of a four wheeled vehicle. The traction of this vehicle is also significantly increased because the contact area is greater than the four wheeled and the continuous track vehicles. The links and body of the Rover both serve as contact surfaces, and will have treads to further increase contact area.

One detriment of the current design is the higher center of gravity compared to a continuous track vehicle; however, this is a tradeoff for the increased clearance height. The higher center of gravity makes this vehicle more unstable and and more likely to tip over. Due to the nature of the uneven terrain of the cornfield in which the Rover will operate, this aspect of vehicle will be optimized with prototyping. This design is also less maneuverable than most vehicle types, as the only effective way to turn is to run both of the cranks in the opposite direction. To turn the Rover, the links will be brought to 180 degrees out of phase with one another, and driven in the opposite direction. This will create a moment on the body that will allow the Rover to turn.

The overall design of the body will be optimized to be small and light enough to function with the launch vehicle, while still being able to carry out the task of collecting ten milliliters of lunar ice. The current plan is to have most components of the Rover be 3D printed out of ABS plastic. This serves not only to meet the weight requirements for the launch vehicle, but will also allow the team to rapidly prototype the design with available 3D printers. The cranks of the Rover will most likely be milled out of an aluminum alloy, as this component will see the highest stress of the entire design. The passively driven cranks will utilize PTFE journal bearings to reduce energy losses from friction. The overall mechanical design of the Rover is seen in Figure 39.

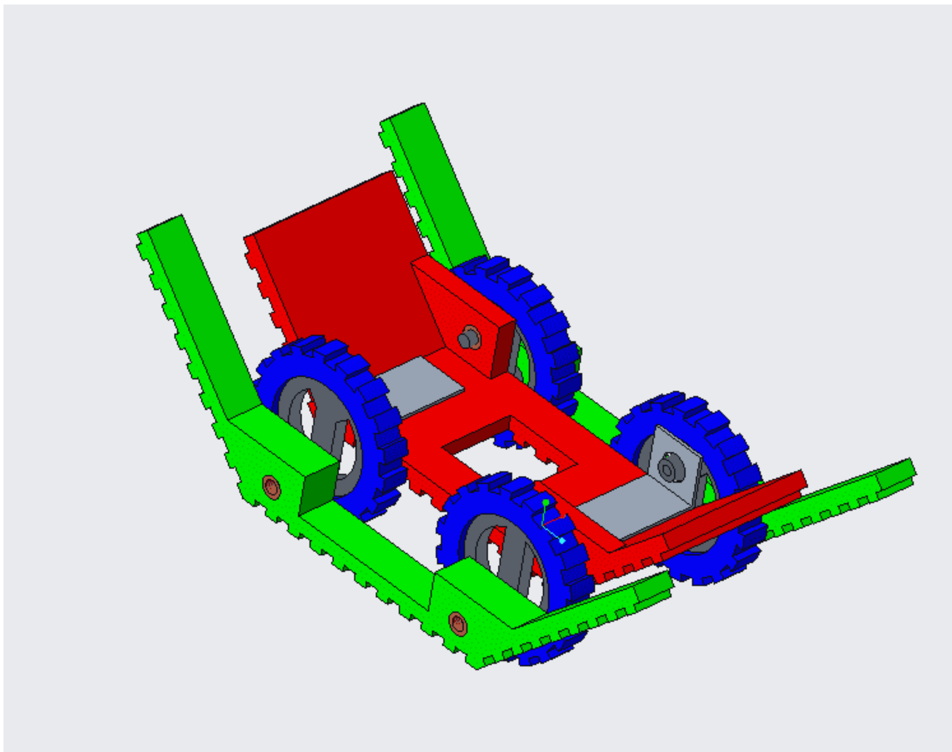

Figure 39: The current overall mechanical design for the Rover.

As seen in Figure 39, both the body and links have treads on the contact surface to increase traction. The front and back of both the body and the links are inclined from the ground; this is compared to being completely flat or vertical. The angle of these sections of the body and links will serve to both aid in the climbing of difficult terrain and the protection the electrical components of the Rover from the elements. The optimal angle of these components will be determined through various prototypes designed by the team. The crank of the Rover will consist of an aluminum inner hub, that will key into an outer ABS plastic wheel. This will increase the strength of the crank without sacrificing significant weight. There is also a large recess in the center of the body, which is to be used for the sample collection portion of the payload. Another large benefit of the eccentric-crank Rover is that the body contacts the ground every full revolution. This significantly eases the difficulty of sample collection, as the mechanism can be made flush with the bottom of the base plate for the duration of travel. Other designs would require the sample retrieval system to extend a significant distance to reach the ground, and therefore would require a more complex mechanism. The current recess design will allow all of the components of sample retrieval system to be stored securely, while not adding unnecessary weight.

The motor selection and placement are critical in the eccentric-crank Rover design, as weight and volume are the two largest design constraints of the Rover payload. The team considered using either one or two motors for the actuation of the cranks. The benefit of one motor is that the actuation of the links is much more precisely controlled. If two motors start driving the links out of phase, the Rover's motion will become significantly compromised. If one motor were to be used, one crank would be driven, and a timing belt would be attached to another crank to actuate the links in unison. The benefit of using two motors is that two links can be driven, and the need for any pulley system is removed. The team is confident in its ability to synchronize the two motors and that the pulley system would be an unnecessary addition. Another important aspect of the design is the motor placement and how they will drive the cranks. The current choice for motor placement on the Rover is as seen below in Figure 40.

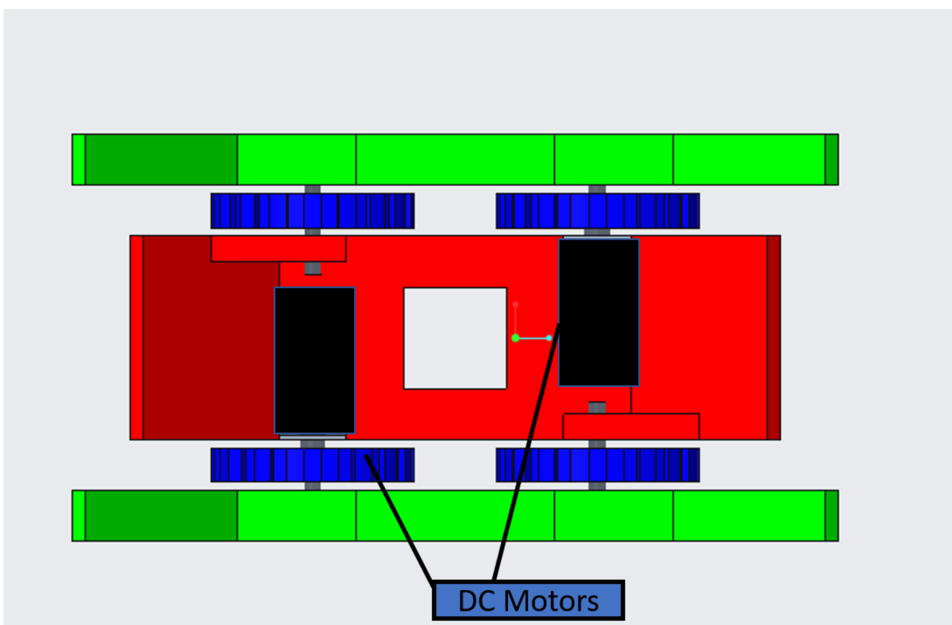

Figure 40: Motor placement on the Rover; motors are located on top of the body at diagonal cranks.

The motors will be placed on the top of the Rover for protection from the elements. This placement is a trade-off with a higher center of gravity. Motor placement on the bottom of the Rover would also lessen the traction of the Rover, as the contact area would mainly be on the motors. It was also determined that the best motor configuration would be to drive the cranks diagonally, as this allows for larger sized motors on the body. If motors were placed directly across from one another, the length of the motors would need to be significantly reduced.

5.3.3.2 Rover Electrical Design

The electrical design for the Rover component of the payload competition is being designed around providing the appropriate communication protocols, controls, and communications necessary for the chosen methods of sample retrieval and system requirements. Major design components for the electrical system will include the drive system, sample retrieval, sensors and controls, communications, batteries, and processor, which shall be integrated into a custom printed circuit board and programmed by members of the team. As part of the challenge this year, the team is aspiring to implement an autonomous system for driving the Rover to the sample to recover by processing data such as GPS sent from the UAV to the Rover to determine the direction to drive. This system will be supplementary to a manual controller used in the event of issues with the autonomous drive or safety concerns expressed by the team or the RSO. A preliminary decision flowchart for the autonomous Rover is shown below in Figure 41.

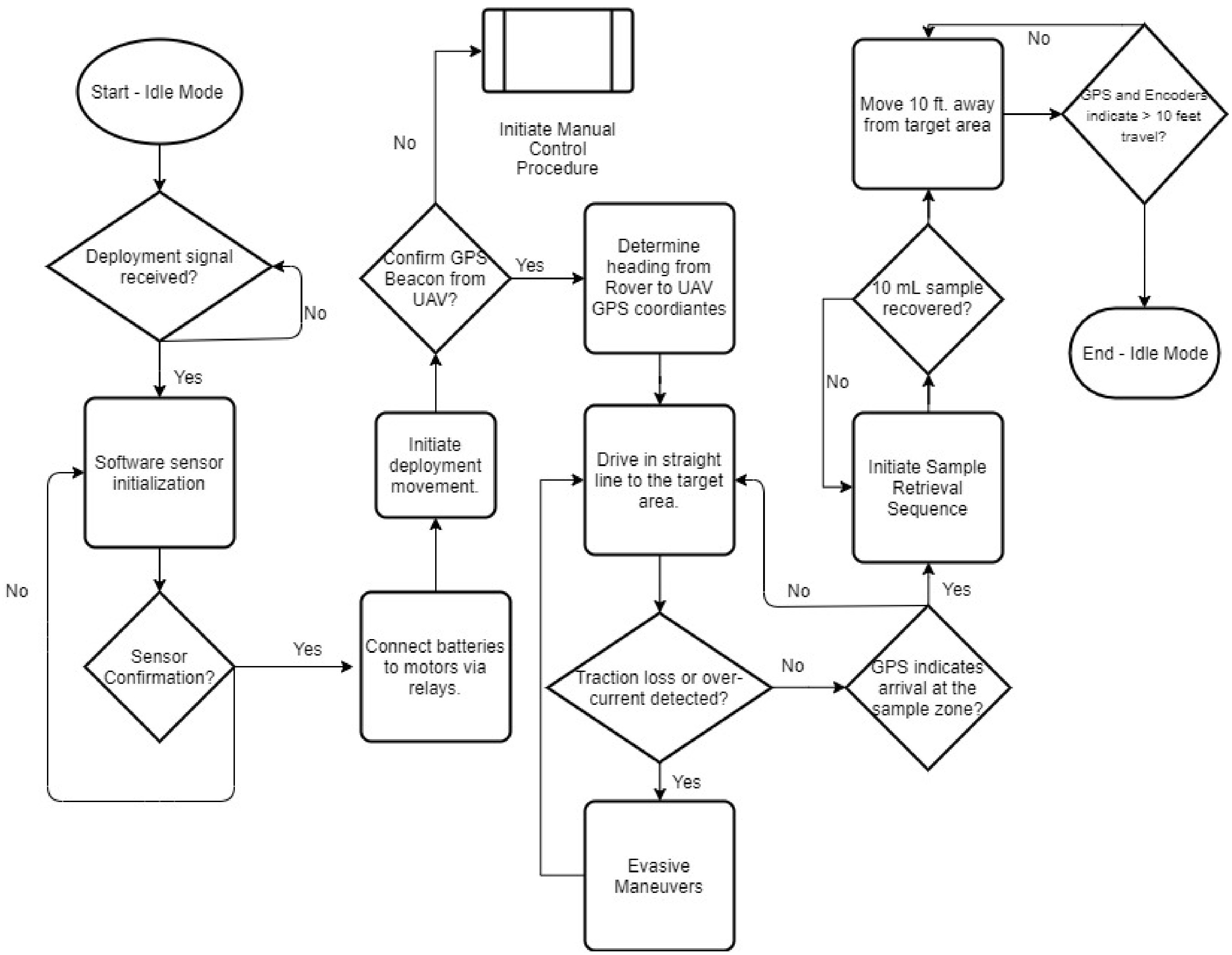

Figure 41: Preliminary Flowchart for Rover Autonomy

5.3.3.2.1 Drive Motor Selection

The primary constraints in selecting a drive motor are torque, weight, and size, particularly the length of the motor along the axis of the shaft. Two motors will be used to provide power to the drive-train. Based on preliminary Rover mechanical designs, the approximate specifications needed for each motor are summarized in table 56 below. These values were used in looking for a motor with similar specifications. The Actobotics 98 RPM Econ Gear Motor was selected as a preliminary choice for the purpose of system prototyping. This motor meets the necessary specifications while being readily available from supplier ServoCity at a low cost, making it ideal for prototyping and final production if deemed sufficient at the time of CDR. This motor also provides a lower maximum current draw, reducing the current rating needed from the on board battery which will allow for a weight reduction when selecting a battery. The specifications are included below in table 56.

Table 56: Motor Design Specifications

Characteristic	Desired Value	Actobotics Econ Motor
Torque [oz-in]	100	524
Nominal Speed [RPM]	60	98
Weight [oz]	3	3.25
Length [in]	2	2.25
Voltage [V]	≤ 12	12
Stall Current [A]	< 10	3.8

Motor driver circuitry will be included on the final system printed circuit board (PCB) in order to properly control the motors using a PWM signal.

5.3.3.2.2 Sensors

Various sensors will be utilized to characterize and control the behavior of the Rover. In order to measure the response of the Rover, motor encoders will be used to measure the rotation of the motor and Rover treads. Additionally, an accelerometer will be used to compare the motion indicated by the encoders with the measured movement which can be used to determine if the Rover is stuck or lost traction in the mud. A GPS chip will be used to determine the location of the Rover, and will be compared with coordinates transmitted by the UAV in order to determine the direction to autonomously drive the Rover to get within range of the sample recovery area. A compass or magnetometer will be used in order to determine the current heading of the Rover.

Inertial Measurement Unit

In order to provide the acceleration and compass data for the Rover, a single inertial measurement unit (IMU) chip will be used. The BNO055 IMU has been a reliable choice in the past and provides sufficient specifications for measuring the desired values, with an acceleration range of ± 8 g's and accuracy of 0.3 m/s^2, and a magnetometer accuracy of 0.3 uT. The compass heading of the Rover can be calculated by taking the arc-tangent of the y and x axis magnetometer data, after accounting for any calibration outlined in the BNO085 datasheet.

Motor Encoder

Motor encoders on the drive motors of the Rover will be used to determine any issues with the Rover being stuck or losing traction during movement. There are two primary approaches to selecting a motor encoder. One option is to choose a motor with an encoder already built in. The Actobotics motor included in this preliminary design also comes in a variation with an encoder included in the motor encasing. This has the benefit of simplifying the number of components, but with the drawback of increasing the length of the motor. The second option is to choose an encoder that mounts on the body and encases the motor shaft to gather an encoding value. This provides the benefit of having more flexibility in placing the encoder.

Due to the size constraints of the Rover, the preliminary choice is to use a mounted encoder. The ENC-AMT10 - Capacitive Modular Encoder has been identified as a good choice for the encoder at a reasonable cost of $23.00. One benefit is that the capacitive encoding will allow it to work despite dirt that would block optical encoders. Additionally, this encoder comes with 9 shaft diameter sleeves to quickly adapt to a change in the design. The encoder has 16 programmable resolutions ranging from 48 to 2048 PPR providing plenty of resolution for the system's motor speed. As such, the ENC-AMT10 encoder is the preliminary choice for the motor encoder.

GPS

A GPS module will be included in the Rover circuitry to compare the position of the Rover with the position of the UAV at the sample recovery site, and determine the heading the Rover needs to take in driving autonomously to the site. The GPS chip is desired to provide a low power consumption on a 3.3 V logic to match the micro-controllers considered for the system. Additionally, the module is required to have an accuracy within 3 meters in order to ensure ability to get within the 10 feet target tarp.

The STMicroelectronics Teseo-LIV3F Tiny GNSS module has been identified as a low cost on-board module which meets these requirements. The specifications of the module are included below in table 57. This module provides a very small standby power draw and low tracking power draw of 75 mW. A strong advantage of this chip is the accuracy of 1.5 meters which should be more than sufficient for getting within the recovery area. The module is readily available from suppliers such as Digi-Key for a low cost of $14.93.

Table 57: STMicroelectronics Teseo-LIV3F GPS

Characteristic	Value
Max Tracking Power [mW]	75
Standby Power [uW]	45.5
Accuracy [m]	1.5
Package [mm x mm]	9.7x10.1
Voltage [V]	3.3
Tracking Sensitivity [dBm]	-163

5.3.3.2.3 Communications

Radio Communication Radio communication shall be used to transmit and receive data from the UAV such as the GPS location of the sample area. Because the rover will be traveling at a relatively slow speed, a high refresh rate of the received data is not necessary. As such, the range and frequency band of the transceiver are the primary constraints. A couple frequency bands are under primary consideration. The 900 MHz band is a popular long range frequency band with the benefit of being open to use without an FCC operator license. The 2.4 GHz band is another option under consideration especially due to preliminary plans to use a 2.4Ghz transceiver on the UAV, which could offer room to bypass the use of a mediator ground station. The 900 MHz band provides some benefits for outdoor usage, including flexibility in placement of the antenna.

A couple on-board transceiver modules are under consideration for communicating via an antenna and sending data to the controller over UART communication. For the 900 MHz band, the Microchip RN2903 module is the primary selection. This module runs on 3.3 V and has a low 2.8 mA idle current draw and 13.5 mA receiving current draw with a transceiver power of 70 mW, under the 250 mW limit. It provides more than enough range at over 9 miles in ideal line of sight conditions. For the 2.4 GHz band, the Semtech SX1280 is a 2.4 GHz long range transceiver module ideal for low data-rate applications such as this, and provides similar specifications to the RN2903 with a lower transmission power of 18mW. As the design process progresses, the frequency band will be determined and a module will be selected. Additionally a simple whip antenna will be used to transmit and receive information for the module.

Manual Control In order to provide manual control, two technical solutions are considered. The first option is to trigger manual control over the radio communication link and send the manual commands over this link. This allows to operate on a single frequency band and radio link, but adds complexity to the data processing on that line.

A second consideration is to include a bluetooth module on the rover circuit and connect a bluetooth controller such as a Playstation dual shock controller to control the rover. This would allow the information conveyed from the UAV to be processed separately from the bluetooth link, which would monitor for an interrupt to trigger manual control mode, and then receive those controls over the same bluetooth link until triggered to return to autonomous mode by the manual operator. Similar to the radio communication modules, the decision on which option to pursue will depend heavily on the finalized design of the UAV frequency band and a possible ground station linking them. If bluetooth is selected, a module such as the Microchip RN4871 on-board module will provide bluetooth integration into the board for programming a connection to a wireless controller. Alternatively, commercial solutions exist from reputable vendors such as RobotShop that provide simplified bluetooth controller solutions at a low cost. One example is the Lynxmotion PS2 Controller V4 which simply plugs into a UART or other serial interface. One downside to this approach is size constraints of the receiver included and less control over the hardware.

5.3.3.2.4 Microcontroller

In order to accommodate the various serial protocols for the system's sensors and actuators, the micro-controller selected must provide for the planned interfaces and provide overhead for changes during the development process. A block diagram of the preliminary component connections and their protocols are included below in Figure 42. The outputs of the system are the drive motors and sample retrieval motors. Additionally the design allocates overhead for additional outputs such as solenoids that might be used for securing the sample retrieval components.

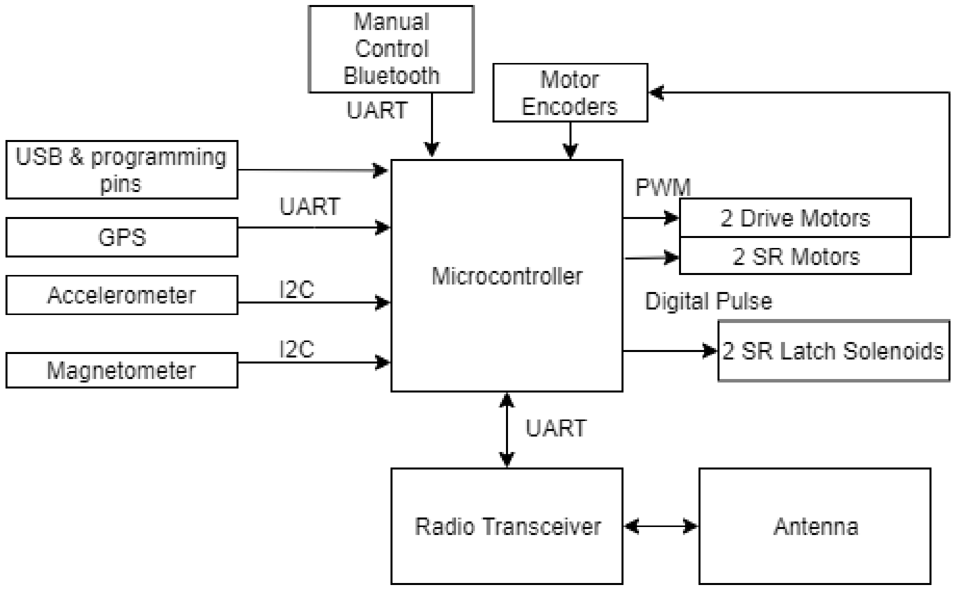

Figure 42: Preliminary Electronic Interfaces

Based on the diagram, the micro-controller it has been determined that the chosen micro-controller should provide two I2C and four UART buses. Additionally, two SPI buses would be beneficial for leaving overhead for peripheral expansion or changes to the chosen interface for components that can use different protocols. The controller must also be capable of producing multiple pulse-width-modulation (PWM) signals for the motor control, and should have a minimum of 64 pins in order to allow for overhead GPIO pins for other components such as status LEDs.

Two micro-controllers are under serious consideration. The first is the Microchip PIC32. The second option is the STMicroelectronics STM32. The primary characteristics of the controllers are shown below in 58. Many characteristics are dependent on the selected package variation of the processor, so two mid-range options offering the required serial protocols have been selected for comparison.

Table 58: Microcontroller Comparison

Microcontroller	PIC32MX170F512H	STM32F407VGT6
Clock Frequency [MHz]	50	168
Program Memory [KB]	512	1024
SRAM [KB]	64	192
Architecture	MIPS	ARM
UART	5	2
I2C	2	3
SPI	4	3
Program Environment	MPLAB X	Keil
Cost [$]	4.88	11.71

Though the chips presented here have differing specifications, it is also possible to acquire different package models for different specifications in a given chip. In cases like this it is also important to consider the importance of a given specification, as the higher clock frequency of the STM32 may not provide significant benefits in this application when offset by the typically higher costs of the STM32.

When it comes to software development, the PIC32 is a well known industry partner with stable processor performance and community development support. Comparatively, the STM32's ARM based architecture is more aligned with industry trends and provides some more productive software development applications as a result.

At this time, the preliminary selection is the PIC32. This is because the PIC32 would provide sufficient specifications for a lower cost both financially and in training time, as many members of the team have experience developing for PIC processors. Additionally the engineering department at the University of Notre Dame supplies a number of PIC based programming and hardware tools for lab learning and course development, so this would better align with the resources available to the team.

In order to integrate all of the components with the microcontroller, a custom printed circuit board (PCB) is under development to provide all the sensor serial interfacing, output control, and power distribution. The PCB will be developed using a student license with

Autodesk's EAGLE software. Additionally the team is considering using free development tools such as EasyEDA for improved design and simulation on a cloud based platform which provides benefits for team collaboration without directly sharing files.

5.3.3.2.5 Power Supply

In order to power the drive system, sample recovery motors, and the computing architecture for the rover, two batteries will be used. As determined by the mechanical design of the rover, the batteries can be a maximum of 1lb total weight. Based on the motors under consideration, a 12V nominal voltage was selected. There will also be a voltage regulator to convert to 12V to 3.3V that can power the microchip and other peripheral components.

The first option considered is using two 12V Nickel Metal-Hydride (NiMH) batteries of 1600mAh each. The second option is a 12 V Lithium Polymer (LiPO) battery of 2200 mAh which provides a better factor of safety in maximum current discharge to avoid risk of getting near the limit, but presents raised fire risk factors. The specifications are shown in Figure 59. Note that the specs are for one battery and 2 will be needed in parallel, and the estimated run time is based on a current draw of 4A from each battery which represents motors running at near stall conditions.

Table 59: Battery Options

Battery	NiMH Battery	LiPO Battery
Capactiy [mAh]	1600	2200
Voltage [V]	12	12
Max Current [A]	16	77
Weight [oz]	8	6.4
Dimensions LxWxH [in]	3.40 x 0.68 x 2.30	4.33 x 1.37 x 0.9
Cost [$]	22.95	27.49
Estimated Run Time [minutes]	54	66

Based on the higher capacity, lower weight, and higher factor of safety on maximum current discharge the LiPO batteries are considered the preliminary choice.

In order to regulate voltage from the 12 V motor batteries to a 3.3 V appropriate for the microcontroller and peripherals, a voltage regulator will be used. Many options are available and under consideration. The primary concern is selecting a voltage regulator that can provide a high enough output current for all the peripherals supplied; otherwise, multiple voltage regulators may be necessary. One regulator under consideration is the Analog Devices ADP7105ACPZ-3.3-R7 which provides an output current of 500 mA and operates down to -40 degrees Celsius which is sufficient for the potentially cold launch conditions expected.

In order to isolate the batteries from the motors to avoid idle discharge prior to the mission, relays shall be used to control the connection between the motors and the batteries. The relay under consideration is the TE Connectivity OJE-SH-112HM which is rated for a 12V contact and up to 10 A current, with a 37.5 mA contact current. In order to supply the 37.5 mA to the relay which the microcontroller cannot supply, the relay will be closed using the 3.3V from the voltage regulator and will be switched on with an optoisolator such as the Broadcom Limited ASSR-4118-503E. When a signal is applied from the micro-controller, this device will allow current to flow to the relay that will allow the motors to begin operating.

5.3.3.2.6 Rover Software

The rover will be programmed to provide the option of both an autonomous mission to meet the UAV at the sample site, or manual control as a safety backup. The diagram in Figure 41 will serve as an architecture for developing the software for autonomous missions. Additionally the flowchart in Figure 43 represents how the program cycle can be programmed to account for interrupts that would trigger the system to transition to manual control or evasive functions for traction control based on the comparisons of sensor data.

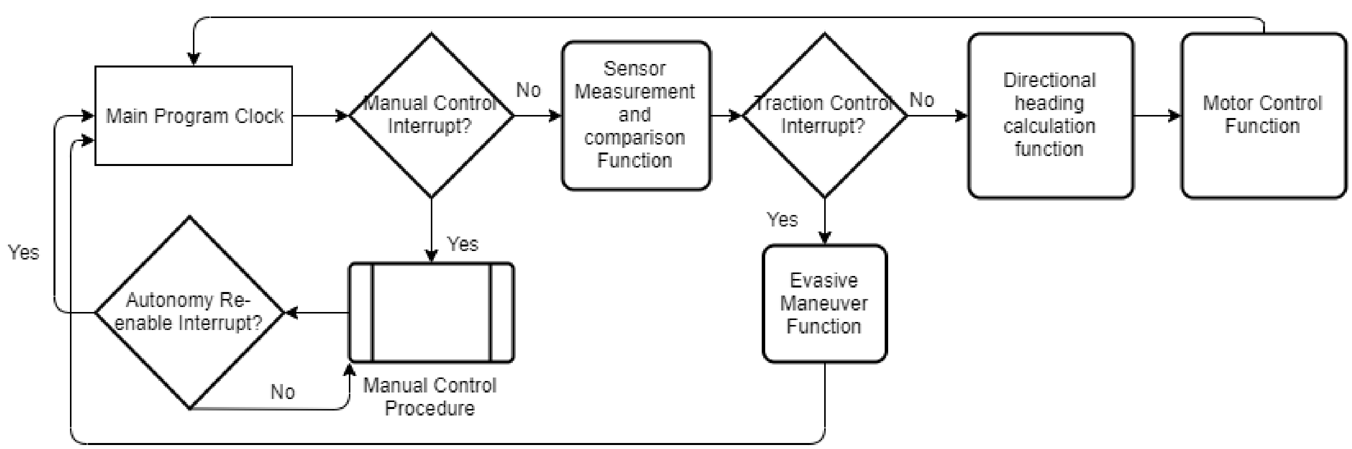

Figure 43: Preliminary Flowchart for Rover Programming

If the PIC32 is the selected microcontroller, the programming can be done in C and compiled using the MPLAB X IDE. For simple systems, relying on the basic IDE functionality for programs and libraries is sufficient. Additionally, the team is considering learning to adopt a real time operating system (RTOS) which could provide benefits in managing the system clock and various events within the program architectures presented here in a way more in line with professional embedded systems development. Various free options such as FreeRTOS are available and compatible with the PIC32 and STM32, so the option is available in a free and open source format. However, this would also represent a major technical challenge in learning the skills for RTOS programming and adds a layer of complexity to the software system. The team will continue exploring the different options available for constructing the full programming architecture including RTOS and applying test-driven and Agile development strategies.

5.3.3.3 Sample Retrieval

In order to accomplish this year's mission of sample retrieval from an unexplored planetary environment, the team has considered three systems capable of collecting a 10 milliliter simulated lunar ice sample. In addition to collecting the aforementioned sample volume, the sample must be stored and transported at least 10 linear feet from the recovery area. The constraints on the sample retrieval system, as of now, are a 3x3 inches footprint and a weight of about 6 oz for the final design. The primary goal is to be comfortably above the 10 milliliter sample requirement.

Design A: Mechanical Bucket

The first design considered was a mechanical bucket attached to the Rover at a fixed point. It would use the Rover's eccentric crank motion to collect the sample quickly without having to add additional motors. After reviewing several clips of the model Rover's motion, there was not a convenient location identified for the bucket. As the Rover is limited in terms of width because of the launch vehicle, it would not be convenient to place it on the side of the Rover. Additionally, it would be difficult to place the bucket on the bottom of the Rover because it touches the ground every cycle; constant contact with the ground might damage the equipment. A viable location for the bucket is the front of the Rover. For the most part, there is a significant amount of space that would serve to fit a motor and the collection box. The mechanical bucket idea is still being looked into, but as of now other courses of action are being pursued.

Design B: Archimedes Screw

The second design considered was the Archimedes screw. The team came up with this design through team input and research The team found the screw to be a viable, secure option considering that similar designs have been used extensively throughout history. Although Archimedes screws have been most commonly used with fluids, it is believed that the small substance grains will act in a fluid-like manner and therefore adhere to similar upward movement. There are several advantages to an Archimides screw including its reliability due to one moving part, the ease of prototyping and modeling, and the ability to optimize the buckets to collect a certain volume. Additionally, the optimization of the buckets can minimize the number of blades and revolutions required for sample retrieval. Lastly, the Archimedes screw parallels real-world solutions to similar problems.

Along with the advantages of the Archimedes screw, there are issues that need to be resolved in order to use it. The first idea was to place the Archimedes screw at a position parallel and level to the Rover's base plate. Within this plate, there would be a small slot where the screw would lie when not activated. The tip of the screw would align with reference to the geometric center of the Rover's plate. When the target was located, the center of the plate would align over the center of the sample, therefore resulting in an accurate alignment of the screw. Following the Rover's alignment with the sample, a servo would pivot and deploy the screw at a certain angle to come into contact with the sample. This angle would have to calculated in advance based on each component's optimal orientation and the size of the final screw. Then, a motor would rotate the screw, forcing the sample to travel up the body and be deposited in a small collection site. Since the bottom of the Rover touches off the ground every cycle due to its eccentricity, it would be best to avoid the bottom of the plate to minimize contact with the rough terrain and the risk of damage.

The next proposed location is to mount the Archimedes screw vertically and at the required angle to meet the sample on top of the building plate. This could prove to be more advantageous since the pre-existing vertical space of the Rover would be utitilized rather than generating an entirely new location. Instead of having a servo change the angle, all that would be necessary for deployment would be a piston to push the screw out of its containment and then a motor to rotate the screw. Once the sample collection is finished, the screw will be brought back into its original position. After considering the Archimedes screw, it was decided to design and 3D print a screw to pursue more physical inquiries. The first design we created is shown below in Figure 44. This design will continue to be developed and printed to test the efficacy.

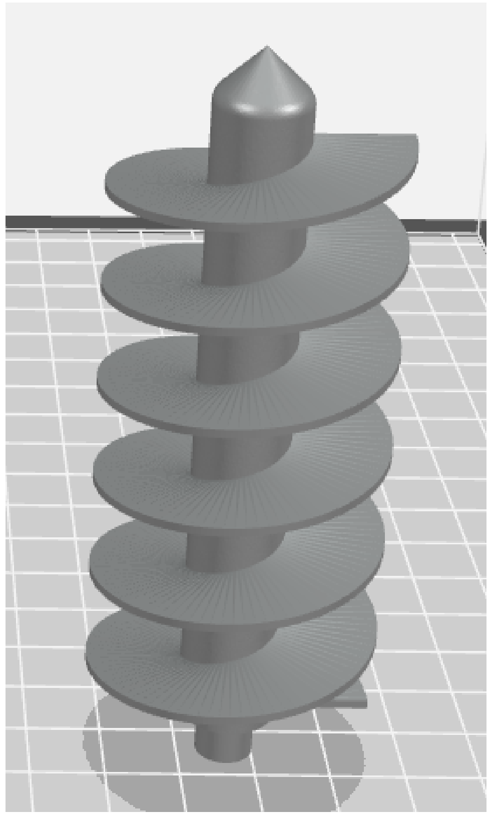

Figure 44: Preliminary CAD of the Archimedes screw.

Design C: Piston Core

The third design considered was a piston core sampling method suggested by Dr. Clive Neal, a Notre Dame professor and geologist who specializes in the science of lunar exploration and excavation. The advantages of the piston core are that it would require less complex machinery and a simpler design. This would simplify the integration to the Rover as it requires less materials and it can be easily experimented with. On the other hand, the disadvantages are that it only works efficiently if the sample is found underground and if the terrain found under the sample is consolidated. The thickness of the sample layer and how deep it is must also be known.

The deployment process would be similar to the Archimedes screw in that the piston would be positioned above the center of the Rover with a small hole where the piston would push out the sealed drive point. Once the 10 milliliter sample was collected, the sealed drive point would retract along with the sample. The process of transporting the sample would be easier since it would not have to be moved along the Rover; the sample would stay confined within the barrel. Additionally, the location of the piston core is versatile. There is less machinery involved, so it can be positioned in the front or back as well. The piston core design being pursued is shown in Figure 45 below.

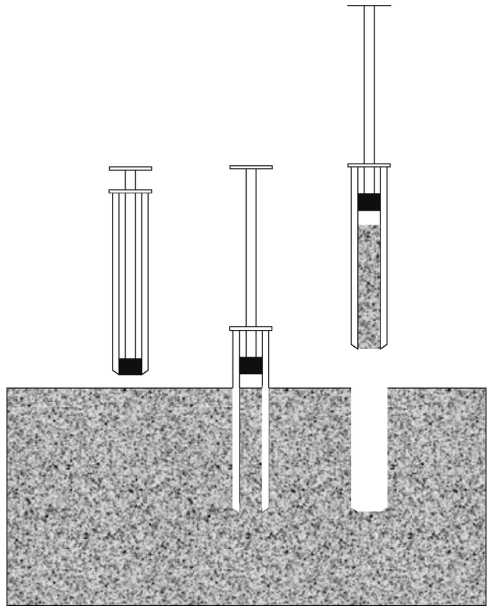

Figure 45: Preliminary CAD of the Archimedes screw.

In the end, the design that is being pursued is Design B, the Archimedes screw, due to its reliability, ease of development, and process of deployment that only requires a start and stop sequence. Although the other designs considered are promising, they either require additional manipulation of the Rover body, as in the case of the mechanical bucket, or depend on conditions of the sampling. Designs will continue to be developed and tested designs to ensure that the goal of a 10 milliliter sample collection is achieved through the most efficient means possible.

6 Project Plan

6.1 Requirements and Verifications

The requirements for the project are broken into NASA provided requirements for the system and the team derived requirements that further guide the design process. The NASA requirements are listed in the order that they appear in the SL Handbook and include the Verification Method and Plan the team has deemed sufficient for meeting the requirement.

6.1.0.1 NASA Requirements

Table 60: NASA General Requirements

ID#	**Requirement**	**Verification Method**	**Verification Plan**	**Status**
1.1	Students on the team will do 100% of the project, including design, construction, written reports, presentations, and flight preparation with the exception of assembling the motors and handling black powder or any variant of ejection charges, or preparing and installing electric matches (to be done by the team's mentor). Teams will submit new work. Excessive use of past work will merit penalties.	Inspection, Demonstration	The Notre Dame Rocket Team is completely student led. Team officers will delegate all work to student members and verify students conduct all activities except those that mentors are required to conduct (i.e. assembling motors, handling ejection charges).	Complete
1.2	The team will provide and maintain a project plan to include, but not limited to the following items: project milestones, budget and community support, checklists, personnel assignments, STEM engagement events, and risks and mitigations.	Demonstration	Team captains are actively maintaining a project plan including a gantt chart for scheduling milestone targets, team budget, and software such as Trello and Slack for organization and task delegation.	In Progress

1.3	Foreign National (FN) team members must be identified by the Preliminary Design Review (PDR) and may or may not have access to certain activities during launch week due to security restrictions. In addition, FN's may be separated from their team during certain activities on site at Marshall Space Flight Center.	Inspection	Design team leads will collect team member information, inform Foreign Nationals of the launch week restrictions, and ensure all Foreign Nationals attending launch week are properly registered in time to attend available activities.	In Progress
1.4.1-1.4.3	"The team must identify all team members attending launch week activities by the Critical Design Review (CDR). Team members will include: Students actively engaged in the project throughout the entire year, one mentor, and no more than two adult educators.	Inspection	Team members, mentors, and educators will be required to express interest in attending launch week prior to CDR submission.	Incomplete

1.5	The team will engage a minimum of 200 participants in educational, hands-on science, technology, engineering, and mathematics (STEM) activities, as defined in the STEM Engagement Activity Report, by FRR. To satisfy this requirement, all events must occur between project acceptance and the FRR due date and the STEM Engagement Activity Report must be submitted via email within two weeks of the completion of the event. A sample of the STEM Engagement Activity Report is on page 35.	Demonstration	An Educational Outreach officer has been elected and will communicate outreach activities with community partners and team members. Educational Engagement Activity Reports will accurately describe outreach activities and community impact.	In Progress
1.6	The team will establish a social media presence to inform the public about team activities.	Demonstration	A Social Media Manager has been elected and will maintain the team's online presence and interaction with the public.	In Progress
1.7	Teams will email all deliverables to the NASA project management team by the deadline specified in the handbook for each milestone. In the event that a deliverable is too large to attach to an email, inclusion of a link to download the file will be sufficient.	Inspection	Team Captains will confirm deliverables are delivered via email by the deadline and will confirm receipt with the NASA project management team.	In Progress

1.8	All deliverables must be in PDF format.	Inspection	Documentation will be prepared using Overleaf and Google Suite products accessed via an academic license. All documentation shall be compiled into a PDF format.	In Progress
1.9	In every report, teams will provide a table of contents including major sections and their respective sub-sections.	Inspection	Documentation prepared using Overleaf will use code tags to create a table of contents and update sections automatically to ensure accuracy.	In Progress
1.10	In every report, the team will include the page number at the bottom of the page.	Inspection	Documentation prepared using Overleaf will be formatted to include the page number at the bottom of the page.	In Progress
1.11	The team will provide any computer equipment necessary to perform a video teleconference with the review panel. This includes, but is not limited to, a computer system, video camera, speaker telephone, and a sufficient Internet connection. Cellular phones should be used for speakerphone capability only as a last resort.	Inspection, Demonstration	The Notre Dame Rocket Team maintains a set of teleconferencing equipment and will verify its functionality prior to each presentation. The team will reserve a room in the college of engineering two weeks prior to each presentation.	In Progress

1.12	All teams will be required to use the launch pads provided by Student Launch's launch services provider. No custom pads will be permitted on the launch field. At launch, 8-foot 1010 rails and 12-foot 1515 rails will be provided. The launch rails will be canted 5 to 10 degrees away from the crowd on launch day. The exact cant will depend on launch day wind conditions.	Demonstration	The launch vehicle shall be designed to launch with the required launch pads and rails as provided by the launch service provider.	In Progress

1.13	Each team must identify a “mentor.” A mentor is defined as an adult who is included as a team member, who will be supporting the team (or multiple teams) throughout the project year, and may or may not be affiliated with the school, institution, or organization. The mentor must maintain a current certification, and be in good standing, through the National Association of Rocketry (NAR) or Tripoli Rocketry Association (TRA) for the motor impulse of the launch vehicle and must have flown and successfully recovered a minimum of 2 flights in this or a higher impulse class, prior to PDR. The mentor is designated as the individual owner of the rocket for liability purposes and must travel with the team to launch week.	Inspection	The Notre Dame Rocket Team works with a mentor who meets all requirements.	Complete

Table 61: NASA Launch Vehicle Requirements

ID#	**Requirement**	**Verification Method**	**Verification Plan**	**Status**
2.1	The vehicle will deliver the payload to an apogee altitude between 3,500 and 5,500 feet above ground level (AGL). Teams flying below 3,000 feet or above 6,000 feet on Launch Day will be disqualified and receive zero altitude points towards their overall project score.	Demonstration, Analysis	Accurate simulations of the vehicle design will be created in RockSim and OpenRocket to project the vehicle apogee and ensure the vehicle will be within the required range and projected to hit the set apogee target. Test flights will be performed to demonstrate this.	In Progress
2.2	Teams shall identify their target altitude goal at the PDR milestone. The declared target altitude will be used to determine the team's altitude score during Launch Week.	Inspection, Analysis	Analysis of the preliminary vehicle and payload design and dimensions will be used to set the final target altitude. The target altitude will be declared in the PDR report.	Complete

2.3	The vehicle will carry one commercially available, barometric altimeter for recording the official altitude used in determining the Altitude Award winner. The Altitude Award will be given to the team with the smallest difference between their measured apogee and their official target altitude on launch day. This altimeter may also be used for deployment purposes (see Requirement 3.4)	Inspection	The team will select a commercially available barometric altimeter and verify with team mentors and launch managers that the selected altimeter is a reliable selection. The team will be using three altimeters for deployment redundancy, so one altimeter will be identified to the launch managers as the scoring altimeter.	In Progress
2.4	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.	Demonstration, Testing	The vehicle will be designed to be reusable. Extensive ground testing of recovery and payload systems will be conducted to ensure written procedures allow for a recoverable and reusable vehicle and payload. This will be verified during full scale flight tests.	In Progress

2.5	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.	Inspection	The team will verify during the design and fabrication phases of development that the vehicle has a maximum of 4 independent sections.	In Progress
2.5.1	Coupler/airframe shoulders which are located at in-flight separation points will be at least 1 body diameter in length.	Inspection	Team will verify that coupler/airframe shoulders at in-flight separation points are at least 1 body diameter in length.	In Progress
2.5.2	Nosecone shoulders which are located at in-flight separation points will be at least $\frac{1}{2}$ body diameter in length.	Inspection	Team will verify that nosecone shoulders at in-flight separation points will be at least 1/2 body diameter in length.	In Progress
2.6	The launch vehicle will be capable of being prepared for flight at the launch site within 2 hours of the time the Federal Aviation Administration flight waiver opens.	Demonstration	Systems and Safety team will prepare launch day procedures which shall be fully practiced (with the exception of arming any energentics) prior to the first launch day. Full scale test flights will be used to ensure the vehicle is prepared within 2 hours.	In Progress

2.7	The launch vehicle and payload will be capable of remaining in launch-ready configuration on the pad for a minimum of 2 hours without losing the functionality of any critical on-board components, although the capability to withstand longer delays is highly encouraged.	Testing, Analysis	During the design phase analysis will be conducted on the power draw of system components and available capacity of on-board batteries. Testing of the vehicle and payload systems will be performed to ensure they are capable of remaining in a launch-ready configuration for at least 3 hours while still having enough capacity to perform the maximum length of the mission without risk of losing power.	In Progress
2.8	The launch vehicle will be capable of being launched by a standard 12-volt direct current firing system. The firing system will be provided by the NASA-designated launch services provider.	Inspection, Demonstration	The vehicle will be designed to launch with a standard 12-volt direct current firing system. The team will work with our launch manager to ensure compatibility prior to demonstration flights.	In Progress

2.9	The launch vehicle will require no external circuitry or special ground support equipment to initiate launch (other than what is provided by the launch services provider).	Inspection, Demonstration	Team will work with our launch manager to ensure compatibility without external circuitry.	In Progress
2.10	The launch vehicle will use a commercially available solid motor propulsion system using ammonium perchlorate composite propellant (APCP) which is approved and certified by the National Association of Rocketry (NAR), Tripoli Rocketry Association (TRA), and/or the Canadian Association of Rocketry (CAR).	Inspection	The team will review NAR, TRA, and CAR certifications to ensure the selected motor is in compliance.	In Progress
2.10.1	Final motor choices will be declared by the Critical Design Review (CDR) milestone.	Inspection	Final motor selection will be declared in the CDR report.	In Progress
2.10.2	Any motor change after CDR must be approved by the NASA Range Safety Officer (RSO) and will only be approved if the change is for the sole purpose of increasing the safety margin. A penalty against the team's overall score will be incurred when a motor change is made after the CDR milestone, regardless of the reason.	Inspection, Analysis	All motor changes requested after the CDR milestone will be requested with accompanying analysis demonstrating a safety derived reasoning. The team accepts a penalty regardless of the reasoning if the change is approved.	In Progress

2.11	The launch vehicle will be limited to a single stage.	Inspection	The team shall design the vehicle as a single stage with a motor in accordance with Req. 2.10	In Progress
2.12	The total impulse provided by a College or University launch vehicle will not exceed 5,120 Newton-seconds (L-class). The total impulse provided by a High School or Middle School launch vehicle will not exceed 2,560 Newton-seconds (K-class).	Inspection	As a Univeristy launch team, the team shall select a motor providing a total impulse which does not exceed 5,120 Netwon-seconds (L class).	In Progress
2.13	Pressure vessels on the vehicle will be approved by the RSO and will meet the following criteria:	Inspection	The pressure vessels will be inspected and approved by the RSO prior to launch.	In Progress
2.13.1	The minimum factor of safety (Burst or Ultimate pressure versus Max Expected Operating Pressure) will be 4:1 with supporting design documentation included in all milestone reviews.	Analysis	The team shall design all pressure vessels on the vehicle with a minimum factor of safety of 4:1 with supporting analysis ensuring the requirement is met.	In Progress

2.13.2	Each pressure vessel will include a pressure relief valve that sees the full pressure of the tank and is capable of withstanding the maximum pressure and flow rate of the tank.	Analysis	All pressure vessels will include pressure relief valves. Analysis will be performed to ensure the valve sees the full pressure of the tank and is capable of withstanding maximum pressure and flow rates.	In Progress
2.13.3	The full pedigree of the tank will be described, including the application for which the tank was designed and the history of the tank. This will include the number of pressure cycles put on the tank, the dates of pressurization/ depressurization, and the name of the person or entity administering each pressure event.	Inspection	Documentation shall be maintained including information about the tanks application, pressure cycles, dates of the pressurization/depressurization, and the name and signature of the entity administering each event.	In Progress
2.14	The launch vehicle will have a minimum static stability margin of 2.0 at the point of rail exit. Rail exit is defined at the point where the forward rail button loses contact with the rail.	Analysis	The team shall analyze the vehicle design using software such as OpenRocket and RockSim to verify a static stability margin of 2.0 at the point of rail exit.	In Progress

2.15	Any structural protuberance on the rocket will be located aft of the burnout center of gravity.	Inspection, Testing, Analysis	All structural protuberance on the vehicle including but not limited to an Air Braking System shall be located aft of the burnout center of gravity as determined by analysis and center of gravity testing.	In Progress
2.16	The launch vehicle will accelerate to a minimum velocity of 52 fps at rail exit.	Demonstration, Analysis	Vehicle design softwares OpenRocket and RockSim shall be used to ensure the vehicle will accelerate to a minimum velocity of 52 fps at the rail exit. This will be demonstrated at full scale launches by analyzing recorded flight data.	In Progress
2.17	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.	Demonstration	The team shall launch and recover a subscale model of the rocket prior to CDR.	Incomplete
2.17.1	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.	Inspection	The subscale model shall be designed to be as accurately resembling the full scale model as possible, and shall be a separate vehicle from the full scale.	In Progress

2.17.2	The subscale model will carry an altimeter capable of recording the model's apogee altitude.	Inspection	The subscale model shall be designed with a payload section for carrying the same altimeter selected for scoring purposes in the full scale rocket.	In Progress
2.17.3	The subscale rocket must be a newly constructed rocket, designed and built specifically for this year's project.	Inspection	Team leaders will ensure that the subscale rocket is newly constructed based on this year's design.	In Progress
2.17.4	Proof of a successful flight shall be supplied in the CDR report. Altimeter data output may be used to meet this requirement.	Inspection	A post launch assessment with test results and altimeter data shall be supplied in the CDR report.	Incomplete
2.18	All teams will complete demonstration flights as outlined below	Demonstration	Team shall complete demonstration flights under the supervision of team launch manager Dave Brunsting and the RSO.	Incomplete

2.18.1	Vehicle Demonstration Flight - All teams will successfully launch and recover their full-scale rocket prior to FRR in its final flight configuration. The rocket flown must be the same rocket to be flown on launch day. The purpose of the Vehicle Demonstration Flight is to validate the 8 launch vehicle's stability, structural integrity, recovery systems, and the team's ability to prepare the launch vehicle 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 the intended lower altitude, functioning tracking devices, etc.). The following criteria must be met during the full-scale demonstration flight:	Demonstration	The full scale vehicle shall be launched and safely recovered prior to FRR to verify the listed vehicle metrics. The rocket flown shall be the final fight configuration and all major vehicle or payload changes shall be approved by the NASA Student Launch team and require a reflight in accordance with the vehicle demonstration deadlines.	Incomplete
2.18.1.1	The vehicle and recovery system will have functioned as designed.	Demonstration	The vehicle and recovery system shall function safely as designed and meet the relevant launch requirements as determined by collected flight data.	In Progress

2.18.1.2	The full-scale rocket must be a newly constructed rocket, designed and built specifically for this year's project.	Inspection	Team leaders shall ensure that the full-scale rocket is newly constructed, designed and built for this year.	In Progress
2.18.1.3	The payload does not have to be flown during the full-scale Vehicle Demonstration Flight. The following requirements still apply:	Inspection	The team shall inspect whether the payload is flight-ready prior to the full-scale demonstration flight.	Incomplete
2.18.1.3.1	If the payload is not flown, mass simulators will be used to simulate the payload mass	Demonstration	If the payload is not flown, an appropriate mass simulator will be secured in the same section as the payload to simulate payload mass.	Incomplete
2.18.1.3.2	The mass simulators will be located in the same approximate location on the rocket as the missing payload mass.	Demonstration	Mass simulators shall be secured in the same approximate location as the payload.	Incomplete
2.18.1.4	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 Vehicle Demonstration Flight.	Demonstration	All payload systems which alter the external surfaces of the rocket or manage the total vehicle energy shall be active during full-scale demonstration flights.	Incomplete

2.18.1.5	Teams shall fly the launch day motor for the Vehicle Demonstration Flight. The team may request a waiver for the use of an alternative motor in advance if the home launch field cannot support the full impulse of the launch day motor or in other extenuating circumstances (such as weather).	Inspection	Team shall fly the selected launch day motor for the demonstration flight. The team shall request a waiver for using an alternative motor well in advance of the flight if the launch field is unable to support the selected motor or other extenuating circumstances arise. Team shall consult with launch manager Dave Brunsting prior to making any such request.	Incomplete
2.18.1.6	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.	Demonstration	The vehicle shall be flown in its fully ballasted configuration during the full-scale test flight. Additional ballast will require an approved re-flight of the full-scale launch vehicle. Additionally, the team shall seek to minimize the amount of ballast required during the design and launch preparation phases.	Incomplete

2.18.1.7	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).	Inspection	Systems and Safety officers shall enforce requirements that the launch vehicle and its components are not handled or modified by team members following flight without the approval of the NASA or local launch site RSO.	In Progress
2.18.1.8	Proof of a successful flight shall be supplied in the FRR report. Altimeter data output is required to meet this requirement.	Demonstration	A post launch assessment with test results and altimeter data shall be supplied in the FRRreport.	Incomplete
2.18.1.9	Vehicle Demonstration flights must be completed by the FRR submission deadline. No exceptions will be made. If the Student Launch office determines that a Vehicle Demonstration Re-flight is necessary, then an extension may be granted. THIS EXTENSION IS ONLY VALID FOR RE-FLIGHTS, NOT FIRST TIME FLIGHTS. Teams completing a required re-flight must submit an FRR Addendum by the FRR Addendum deadline	Inspection	The team shall conduct a vehicle demonstration flight prior to the FRR deadlines. The team acknowledges that no exceptions shall be made and extensions shall only be considered for re-flights seeking to demonstrate improved vehicle safety and payload functionality.	Incomplete

2.18.2	Payload Demonstration Flight - All teams will successfully launch and recover their full-scale rocket containing the completed payload prior to the Payload Demonstration Flight deadline. The rocket flown must be the same rocket to be flown on launch day. The purpose of the Payload Demonstration Flight is to prove the launch vehicle's ability to safely retain the constructed payload during flight and to show that all aspects of the payload perform as designed. A successful flight is defined as a launch in which the rocket experiences stable ascent and the payload is fully retained until it is deployed (if applicable) as designed. The following criteria must be met during the Payload Demonstration Flight:	Inspection	The team shall complete a payload demonstration flight prior to the Payload Demonstration Flight deadline.	Incomplete

2.18.2.1	The payload must be fully retained until the intended point of deployment (if applicable), all retention mechanisms must function as designed, and the retention mechanism must not sustain damage requiring repair.	Inspection, Demonstration	The payload shall be designed to be fully retained until the intended point of deployment and all retention mechanisms must function as designed without sustaining damage requiring repair inhibiting the reusability of the payload and vehicle, in accordance with Req. 2.4. This will be demonstrated during the full-scale flight tests.	In Progress
2.18.2.2	The payload flown must be the final, active version	Inspection	The team shall fly the final active payload. Any changes to the payload following the flight will require NASA Student Launch team approval and re-flight in accordance with the demonstration flight deadlines.	Incomplete

2.18.2.3	If the above criteria are met during the original Vehicle Demonstration Flight, occurring prior to the FRR deadline and the information is included in the FRR package, the additional flight and FRR Addendum are not required.	Inspection	The team shall review the requirements and flight performance following the Vehicle Demonstration flight and determine if an additional flight is required.	Incomplete
2.18.2.4	Payload Demonstration Flights must be completed by the FRR Addendum deadline. NO EXTENSIONS WILL BE GRANTED	Inspection	The team shall complete payload demonstration flights prior to the FRR Addendum deadline. The team acknowledges no extensions will be granted.	Incomplete
2.19	An FRR Addendum will be required for any team completing a Payload Demonstration Flight or NASArequired Vehicle Demonstration Re-flight after the submission of the FRR Report.	Inspection	The team shall complete an FRR Addendum for any payload demonstration or vehicle demonstration re-flights after the FRR deadline.	Incomplete
2.19.1	Teams required to complete a Vehicle Demonstration Re-Flight and failing to submit the FRR Addendum by the deadline will not be permitted to fly the vehicle at launch week.	Inspection	The team shall complete a vehicle demonstration re-flight and FRR addendum by the deadline as necessary or forfeit the permission to fly at launch week.	Incomplete

2.19.2	Teams who successfully complete a Vehicle Demonstration Flight but fail to qualify the payload by satisfactorily completing the Payload Demonstration Flight requirement will not be permitted to fly the payload at launch week.	Inspection	The team shall complete a successful payload demonstration flight prior to the Payload Demonstration Flight deadline. If the payload demonstration is not complete, the payload will not be permitted to fly at launch week even if the vehicle is permitted to do so.	Incomplete
2.19.3	Teams who complete a Payload Demonstration Flight which is not fully successful may petition the NASA RSO for permission to fly the payload at launch week. Permission will not be granted if the RSO or the Review Panel have any safety concerns.	Inspection	If the payload demonstration flight is not fully successful, the team shall assess the failures and petition the NASA RSO for permission to fly the paylaod at launch week by preparing documentation about the failures, their risk analysis, and steps that can be taken to resolve the failures safely prior to launch week.	Incomplete

2.2	The team's name and launch day contact information shall be in or on the rocket airframe as well as in or on any section of the vehicle that separates during flight and is not tethered to the main airframe. This information shall be included in a manner that allows the information to be retrieved without the need to open or separate the vehicle.	Inspection	The team shall include team information including name and contact information on the external of the vehicle by incorporating the information into the vehicle paint or applying external labels.	Incomplete
2.21	All Lithium Polymer batteries will be sufficiently protected from impact with the ground and will be brightly colored, clearly marked as a fire hazard, and easily distinguishable from other payload hardware.	Inspection	The Safety and Systems team shall verify that all lithium polymer batteries in the vehicle are sufficiently protected from impact with the ground and shall be clearly labeled with bright colors as a fire hazard. Additionally the team shall use fire-proof lithium polymer battery carrying cases for transporting and storing batteries before and after the flight.	Incomplete

2.22	Vehicle Prohibitions	Inspection	The listed vehicle prohibitions shall be inspected prior to all flights to ensure the vehicle is in compliance.	Incomplete
2.22.1	The launch vehicle will not utilize forward canards. Camera housings will be exempted, provided the team can show that the housing(s) causes minimal aerodynamic effect on the rocket's stability	Demonstration, Analysis	The vehicle will not utilize forward canards. If camera housings are used the team shall provide computational fluid dynamics (CFD) analysis and a subscale launch demonstrating the housing does not affect vehicle stability.	In Progress
2.22.2	The launch vehicle will not utilize forward firing motors.	Inspection	The vehicle will not utilize forward firing motors.	Complete
2.22.3	The launch vehicle will not utilize motors that expel titanium sponges (Sparky, Skidmark, MetalStorm, etc.)	Inspection	The vehicle motor documentation shall be inspected to verify it does not expel titanium sponges. This shall be verified with the approval of team launch manager Dave Brunsting.	Incomplete
2.22.4	The launch vehicle will not utilize hybrid motors.	Inspection	The vehicle shall not utilize hybrid motors.	In Progress
2.22.5	The launch vehicle will not utilize a cluster of motors.	Inspection	The vehicle shall not use a cluster of motors.	In Progress
2.22.6	The launch vehicle will not utilize friction fitting for motors.	Inspection	The vehicle shall not utilize friction fitting for motors.	In Progress

2.22.7	The launch vehicle will not exceed Mach 1 at any point during flight	Demonstration, Analysis	The launch vehicle shall not exceed Mach 1 at any point during flight as determined by OpenRocket and RockSim analysis, and demonstrated by analyzing the recorded flight data.	In Progress
2.22.8	Vehicle ballast will not exceed 10% of the total unballasted weight of the rocket as it would sit on the pad (i.e. a rocket with an unballasted weight of 40 lbs. on the pad may contain a maximum of 4 lbs. of ballast).	Inspection	The vehicle ballast will not exceed 10% of the total unballasted weight of the rocket as it would sit on the pad.	In Progress
2.22.9	Transmissions from onboard transmitters will not exceed 250 mW of power (per transmitter).	Testing, Analysis	Transmissions from oboard transmitters shall not exceed 250 mW of power as determined by the specifications of on-board transmitters and relevant testing.	In Progress
2.22.10	Transmitters will not create excessive interference. Teams will utilize unique frequencies, handshake/passcode systems, or other means to mitigate interference caused to or received from other teams.	Inspection, Testing	Transmitters shall not create excessive interference and shall be utilize unique frequencies or other means to limit interference and shall be tested prior to launch week.	In Progress

2.22.11	Excessive and/or dense metal will not be utilized in the construction of the vehicle. Use of lightweight metal will be permitted but limited to the amount necessary to ensure structural integrity of the airframe under the expected operating stresses.	Inspection	Excessive and/or dense metal shall not be utlized in the construction of the vehicle unless approved by the NASA Student Launch team and team launch manager Dave Brunsting limited to the amount necessary to ensure structural integrity.	In Progress

Table 62: NASA Recovery Requirements

ID#	Requirement	Verification Method	Verification Plan	Status
3.1	The launch vehicle will stage the deployment of its recovery devices, where a drogue parachute is deployed at apogee, and a main parachute is deployed at a lower altitude. Tumble or streamer recovery 10 from apogee to main parachute deployment is also permissible, provided that kinetic energy during drogue stage descent is reasonable, as deemed by the RSO.	Inspection	The rocket will contain two separate parachute bays, one for the drogue parachute and one for the main. Each of the altimeters will be programmed to eject the drogue parachute at or shorty after rocket apogee, and the main parachute at a lower altitude.	In Progress

3.1.1	The main parachute shall be deployed no lower than 500 feet.	Demonstration, Testing	All of the recovery altimeters will be programmed to eject the main parachute at an altitude of 500ft or greater. Requirement shall be verified with simulated flight tests and vehicle demonstration flights.	In Progress
3.1.2	The apogee event may contain a delay of no more than 2 seconds.	Inspection	All of the recovery altimeters will be programmed to eject the drogue parachute at apogee or between 0 and 2 seconds after apogee.	In Progress
3.1.3	Motor ejection is not a permissible form of primary or secondary deployment.	Inspection	All recovery ejection charges will be controlled by commercial altimeters.	In Progress
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.	Testing	Before any rocket launch by the team, a ground separation test will be performed on the rocket using an ejection charge of the same type and size to be used for the launch. Passing condition: Complete separation of the rocket at the intended separation points, and no significant damage to the parachtue, shroud lines, or shock cords	Incomplete

3.3	Each independent section of the launch vehicle will have a maximum kinetic energy of 75 ft-lbf at landing.	Demonstration, Analysis	The main parachute will be apropriately sized such that the largest rocket section falls with a kinetic energy under 75 ft-lbf under main parachute. The expected descent velocity will be simulated using OpenRocket and a custom Matlab simulator. Descent velocity data will also be taken during test launches to confirm simulation accuracy and ensure compliance with descent kinetic energy requirements.	In Progress
3.4	The recovery system will contain redundant, commercially available altimeters. The term "altimeters" includes both simple altimeters and more sophisticated flight computers.	Inspection, Testing	The recovery ejection charges will be controlled by three Featherweight Raven3 altimeters. In simulated launch environment, two of the recovery altimeters will be intentionally disabled. Passing condition: The remaining recovery system functions such to effectively initiate ejection of both the drogue and main parachutes	In Progress

3.5	Each altimeter will have a dedicated power supply, and all recovery electronics will be powered by commercially available batteries.	Inspection	In simulated launch environment, one of the commercially-produced altimeter powering batteries will be intentionaly disconnected. Passing condition: Both main and drogue parachute are successfully deployed by the remaining altimeters and batteries.	In Progress
3.6	Each altimeter will be armed by a dedicated mechanical arming switch that is accessible from the exterior of the rocket airframe when the rocket is in the launch configuration on the launch pad.	Inspection	Rocket assembly dry run: the rocket will be assembled (without energetics), placed in the upright position, and an attempt will be made to activate the altimeters. Passing condition: Altimeters are successfully activated.	Incomplete
3.7	Each arming switch will be capable of being locked in the ON position for launch (i.e. cannot be disarmed due to flight forces).	Inspection, Testing	A shake test will be performed on the recovery electronics bay, with altimeters active but without energetics. Passing condition: Altimeters remain active for full duration of shaking.	Incomplete

3.8	The recovery system electrical circuits will be completely independent of any payload electrical circuits.	Inspection	In simulated launch environment, the full function of the recovery electronics will be tested without the payoad present to ensure independence from the payload.	In Progress
3.9	Removable shear pins will be used for both the main parachute compartment and the drogue parachute compartment.	Inspection	The same number and configuration of shear pins will be used to secure the parachute compartments during ground separation tests and full launches.	Incomplete
3.1	The recovery area will be limited to a 2,500 ft. radius from the launch pads.	Demonstration, Analysis	The drift distance of the rocket after apogee will be simulated in both OpenRocket and a custom Matlab simulator at a variety of wind speeds up to 20 mph. In addition, the drift distance of the rocket after apogee during the test launch will be recorded to ensure accuracy of the simulations and compliance with competition rules.	In Progress

3.11	Descent time will be limited to 90 seconds (apogee to touch down).	Demonstration, Analysis	The descent time after apogee will be simulated using OpenRocket and a custom Matlab simulator. The descent time after apogee will also be measured during test launch to ensure accuracy of simulations and compliance with competition rules.	In Progress
3.12	An electronic tracking device will be installed in the launch vehicle and will transmit the position of the tethered vehicle or any independent section to a ground receiver.	Inspection, Demonstration	The rocket willl contain an active GPS transmitter during test launch, which will be used to track the location of the rocket after launch.	Incomplete
3.12.1	Any rocket section or payload component, which lands untethered to the launch vehicle, will contain an active electronic tracking device.	Inspection	All sections of the rocket will be tethered together, with a single dedicated GPS transmitter in one of the sections.	Incomplete
3.12.2	The electronic tracking device(s) will be fully functional during the official flight on launch day.	Demonstration	The GPS transmitter wil be tested for functiionallity before the official flight on launch day.	Incomplete

3.13	The recovery system electronics will not be adversely affected by any other on-board electronic devices during flight (from launch until landing).	Demonstration, Testing	The rocket will be flown with all electronics active during a test launch before competition. Altimeter data will be inspected afterwords for any evidence of adverse effects.	Incomplete
3.13.1	The recovery system altimeters will be physically located in a separate compartment within the vehicle from any other radio frequency transmitting device and/or magnetic wave producing device.	Inspection	The recovery electronics will be mounted in a recovery bay separate from the payload and any RF or EM transmitters or receivers.	Incomplete
3.13.2	The recovery system electronics will be shielded from all onboard transmitting devices to avoid inadvertent excitation of the recovery system electronics.	Inspection	A conductive Faraday cage will encase the recovery altimeters to prevent interference by any outside transmitters.	Incomplete
3.13.3	The recovery system electronics will be shielded from all onboard devices which may generate magnetic waves (such as generators, solenoid valves, and Tesla coils) to avoid inadvertent excitation of the recovery system.	Inspection	A conductive Faraday cage will encase the recovery altimeters to prevent interference by any internal magnetic wave producing devices.	Incomplete

3.13.4	The recovery system electronics will be shielded from any other onboard devices which may adversely affect the proper operation of the recovery system electronics.	Inspection	A conductive Faraday cage will encase the recovery altimeters to prevent interference by any internal transmitters and other electronics.	Incomplete

Table 63: NASA Payload Requirements

ID#	**Requirement**	**Verification Method**	**Verification Plan**	**Status**
4.2	College/University Division – Teams will design a system capable of being launched in a high power rocket, landing safely, and recovering simulated lunar ice from one of several locations on the surface of the launch field. The method(s)/design(s) utilized will be at the teams' discretion and will be permitted so long as the designs are deemed safe, obey FAA and legal requirements, and adhere to the intent of the challenge. An additional experiment (limit of 1) is allowed, and may be flown, but will not contribute to scoring. If the team chooses to fly an additional experiment, they will provide the appropriate documentation in all design reports so the experiment may be reviewed for flight safety	Inspection	The team shall be competing in the university division and shall design a payload which meets the listed requirement in order to recover a simulated lunar ice from designated locations in the launch field. The team shall have discretion to design the payload but shall work with team mentors to verify the design is safe, meets FAA requirements, and adhere to the requirements of the challenge.	In Progress

4.3	Lunar Ice Sample Recovery Mission Requirements	Inspection	The payload shall be designed in adherence with the mission requirements listed.	In Progress
4.3.1	The launch vehicle will be launched from the NASA-designated launch area using the provided Launch pad. All hardware utilized at the recovery site must launch on or within the launch vehicle.	Inspection	The launch vehicle will be launched from the NASA designated launch area using the provided launch pad. All hardware utilized at the recovery site shall be launched on or within the vehicle.	In Progress
4.3.2	Five recovery areas will be located on the surface of the launch field. Teams may recover a sample from any of the recovery areas. Each recovery site will be at least 3 feet in diameter and contain sample material extending from ground level to at least 2 inches below the surface.	Demonstration	The team shall design the payload to be capable of travelling to one of the recovery areas and recover a sample extending at least 2 inches below the surface.	In Progress
4.3.3	The recovered ice sample will be a minimum of 10 milliliters (mL).	Demonstration	The payload will be designed to be capable of recovering an ice sample with a minimum volume of 10 mL.	In Progress
4.3.4	Once the sample is recovered, it must be stored and transported at least 10 linear feet from the recovery area.	Demonstration	The payload will be designed to be capable of transporting the recovered sample at least 10 linear feet from the recovery area.	In Progress

4.3.5	Teams must abide by all FAA and NAR rules and regulations.	Inspection	The team shall abide by all FAA and NAR rules and regulations. The team shall conduct a review with the team launch manager prior to the launch day to verify all regulations are met.	In Progress
4.3.6	Black Powder and/or similar energetics are only permitted for deployment of in-flight recovery systems. Any ground deployments must utilize mechanical systems.	Inspection	The team shall not utilize energetics for the ground depoloyment of the payload. The payload ground deployment shall utilize a mechanical system.	In Progress
4.3.7	Any part of the payload or vehicle that is designed to be deployed, whether on the ground or in the air, must be fully retained until it is deployed as designed.	Demonstration, Testing	The payload shall be designed to be fully retained until it is deployed as designed. This shall be verified in tests prior to launches and demonstrated during the demonstration flights.	In Progress
4.3.7.1	A mechanical retention system will be designed to prohibit premature deployment.	Analysis	The mechanical system designed to prohibit premature deployment shall be designed and analyzed using methods such as Finite Element Analysis (FEA) to determine forces on the system to avoid premature deployment.	In Progress

4.3.7.2	The retention system will be robust enough to successfully endure flight forces experienced during both typical and atypical flights.	Demonstration, Testing	The retention system shall be subjected to shake tests to ensure the system is capable of enduring typical and atypical flight forces while still being reusable per Req. 2.4.	Incomplete
4.3.7.3	The designed system will be fail-safe.	Inspection, Testing	The designed system shall be designed to be fail-safe to ensure that the failure of any system components does not result in the payload being damaged or released prematurely. Additionally the system shall be designed with redundancy to avoid failures.	In Progress
4.3.7.4	Exclusive use of shear pins will not meet this requirement.	Inspection	The team ackownledges that shear pins shall not meet the fail-safe requirement.	Complete
4.4	Special Requirements for UAVs and Jettisoned Payloads	Inspection	The team shall follow all requirements for UAVs and Jettisoned payloads.	In Progress
4.4.1	Any experiment element that is jettisoned during the recovery phase will receive real-time RSO permission prior to initiating the jettison event.	Inspection	Any element that is jettisoned during the recovery phase will receive real-time RSO permission prior to initiating the event.	Incomplete

4.4.2	Unmanned aerial vehicle (UAV) payloads, if designed to be deployed during descent, will be tethered to the vehicle with a remotely controlled release mechanism until the RSO has given permission to release the UAV.	Inspection, Demonstration	Any components deployed during descent shall be tethered to the vehicle with a remotely controlled release mechanism until the RSO gives permission to release the UAV and shall be demonstrated during test flights.	Incomplete
4.4.3	Teams flying UAVs will abide by all applicable FAA regulations, including the FAA's Special Rule for Model Aircraft (Public Law 112-95 Section 336; see https://www.faa.gov/uas/faqs).	Inspection	The team shall abide by all FAA regulations and shall carefully review the regulations during each step of the development process (design, testing, pre-flight review etc.)	In Progress
4.4.4	Any UAV weighing more than .55 lbs. will be registered with the FAA and the registration number marked on the vehicle.	Inspection	Any team UAV weighing more than 0.55 lbs will be registered with the FAA and the registration number marked on the vehicle.	Incomplete

Table 64: NASA Safety Requirements

ID#	Requirement	Verification Method	Verification Plan	Status
5.1	Each team will use a launch and safety checklist. The final checklists will be included in the FRR report and used during the Launch Readiness Review (LRR) and any launch day operations.	Inspection	The team shall write and maintain a launch and safety checklist which shall be included in the FRR and LRR reports. The Safety and Systems team shall lead development and enforcement of these safety procedures.	In Progress
5.2	Each team must identify a student safety officer who will be responsible for all items in section 5.3.	Inspection	The team has elected Brooke Mumma to serve as the safety officer who will lead the Safety and Systems team. As such she shall be responsible for all safety matters in accordance with section 5.3.	Complete
5.3	The role and responsibilities of the safety officer will include, but are not limited to:	Inspection	The safety officer shall manage the responsibilites listed.	In Progress

5.3.1.1-5.3.1.9	Monitor team activities with an emphasis on safety during: Design of vehicle and payload, construction of vehicle and payload components, assembly of vehicle and payload, ground testing of vehicle and payload, full-scale launch test(s), subscale launch test(s), launch day, recovery activities, STEM engagement activities	Inspection	The safety officer shall monitor all listed team activities during the full development cycle of the team throughout the year. The safety officer shall focus on the safety of the team and shall have the discretion to maintain enforcement methods for handling safety violations.	In Progress
5.3.2	Implement procedures developed by the team for construction, assembly, launch, and recovery activities.	Inspection	The Safety and Systems team shall manage the design teams in writing procedures for construction, assembly, launch, and recovery activities and shall ensure the procedures meet safety requirements following a standardized format set by the team.	In Progress
5.3.3	Manage and maintain current revisions of the team's hazard analyses, failure modes analyses, procedures, and MSDS/chemical inventory data.	Inspection	The Safety and Systems team shall maintain the team's hazard analyses, failure mode analyses, procedures, and MSDS inventory data. The team shall conduct frequent revision meetings.	In Progress

5.3.4	Assist in the writing and development of the team's hazard analyses, failure modes analyses, and procedures.	Inspection	The Safety and Systems team shall lead writing and development of the analyses and procedures listed.	In Progress
5.4	During test flights, teams will abide by the rules and guidance of the local rocketry club's RSO. The allowance of certain vehicle configurations and/or payloads at the NASA Student Launch does not give explicit or implicit authority for teams to fly those vehicle configurations and/or payloads at other club launches. Teams should communicate their intentions to the local club's President or Prefect and RSO before attending any NAR or TRA launch.	Inspection	The Safety and Systems lead, team captains, and team launch manager shall communicate with the local RSO to ensure the vehicle meets all local configuration requirements and address any safety concerns of the local RSO.	In Progress
5.5	Teams will abide by all rules set forth by the FAA.	Inspection	The team shall abide by all FAA rules and regulations and will conduct frequent reviews to ensure continued compliance.	In Progress

6.1.0.2 Team Derived Requirements

In order to further define the scope and detail of the system design, the team has derived additional requirements for Vehicle Design, Recovery Subsystem, and Lunar Ice Sample Recovery Payload. Requirements are given in the subsequent tables.

Table 65: Derived Launch Vehicle Requirements

ID#	**Description**	**Justification**	**Verification Method**	**Verification Plan**	**Status**
V.1	The vehicle will have two in-flight separation points to allow for a drogue and main parachute deployment and an additional access point for ABS integration.	Drogue parachute necessary to slow vehicle and decrease drift	Inspection	The design of the rocket will have three separation points, two covered by bulkhead for recovery and a third for ABS integration	In Progress
V.2	The weight distribution throughout the vehicle will be kept the closest possible to constant.	Decrease parachute size	Inspection	An updated weight budget for the launch vehicle will be kept updated at all times	In Progress
V.3	The vehicle must have a fully designed and integrated ballast area at the rocket's Cg to diminish ballast's effect in the vehicle's stability. Ballast area must hold up to 10% of total vehicle weight.	In the case that payloads are under weight budget ballasting will be necessary to meet target apogee	Inspection	Ballast area will be designed to fit in the area closest to the rocket's Cg	In Progress

V.4	The vehicle is designed to reach a 4,123 ft altitude.	Target apogee must be set by the team	Testing, Analysis	Simulation software will be used to verify vehicle designs reach a 4,100 ft. apogee in a simulated environment, and full scale test flights shall be used to verify the accuracy of the simulation and completion of the requirement.	In Progress
V.5	The payload bay shall be a fiber glass body tube with an 8in OD and 20 in length.	Payload bay must be radio transparent for signals to payload	Inspection	The team will design the rocket to provide the required dimensions for the payload system.	In Progress
V.6	ABS must be secured to the rest of the vehicle and fill the full aft diameter of the rocket.	Avoid force unbalance due to movement of payload	Inspection	ABS will be designed for ideal integration into the aft part of the rocket	In Progress
V.7	The vehicle shall not exceed a maximum length of 12 ft	Vehicle must be easily transported	Inspection	The total length of the full scale rocket will be measured when construction material is delivered to us	Incomplete

V.8	The vehicle shall not exceed a maximum weight of 70 pounds	Vehicle must be able to achieve target apogee	Demonstration	The rocket will be weighted with all of the systems before launch	Incomplete
V.9	The vehicle must house a camera that looks downward with an angle of visibility that includes ABS	Allows view of ABS tab extension and retraction.	Inspection	A housing area will be integrated and a securing mechanism will be designed to safely hold the camera in place	In Progress
V.10	The stability margin of the vehicle with the motor must be between 2 and 3 calibers	Avoid any possibility of vehicle tilting into the wind	Demonstration, Analysis	Flight simulation applications will be used to design for a 2-3 caliber stability margin and before test flights the actual Cg will be measured to calculate the actual vs predicted stability margin	In Progress
V.11	The motor selection must tend towards overshooting rather than undershooting the target apogee.	Allow use of ABS	Demonstration, Analysis	The motor selection will be based on flight simulations and test flights will determine predicted vs actual apogee	In Progress

V.12	Epoxied bulkheads must be able to hold the load of drogue and main parachute deployments	Load bearing bulkheads must not break under max load	Testing	Solid testing will be designed to test max force that an epoxied bulkhead can take	In Progress
V.13	The recovery body tube will not exceed a maximum length of 48 in	Length budget to fulfill V.7	Inspection	Recovery body tube will be designed under that length and constant communication with Recovery squad will make sure it is not exceeded	In Progress
V.14	ABS will not exceed 10 inches in length	70 oz. in weight and length budget to fulfill V.7 and weight budget to fulfill V.8	Inspection	Various checkpoints in the design process will be used to verify ABS design is meeting this requirement, and measurements will be made during fabrication to confirm the requirement is met.	In Progress

V.15	The ABS drag tabs must extend at a location no greater than 3 inches from the CP	This will ensure that the induced drag force does not result in destabilizing moments	Inspection	The integration design of ABS will focus on the location of the tabs in relation to the CP. Measurements will be made during fabrication to confirm that this requirement is met.	In Progress
V.16	Removable bulkhead attached to ABS must be able to withstand the load of drogue and main parachute deployments	Failure of the bulkhead or the screws holding it in place during deployment would prevent recovery's ability to execute a safe and successful landing	Analysis	Analysis of the stresses experienced by the bulkhead and screws during deployment will help determine material and dimensional requirements to ensure these components will not fail.	In Progress

Table 66: Derived Recovery Requirements

ID#	**Description**	**Justification**	**Verification Method**	**Verification Plan**	**Status**
R.1	Recovery ejection charges shall be capable of being "safed," such that at least 2 independent actions are necessary before the altimeter is fully armed.	To ensure the safety of the team, extra steps must be taken to ensure that the black powder ejection charges do not ignite early and injure personnel.	Demonstration	Two switches of different types will be placed in series with the powering battery such that both switches need to be closed before the altimeter fully arms.	Incomplete
R.2	The parachutes, shroud lines, and shock cordage shall be protected from potential damage due to the ejection charges.	The ejection charges can burn the parachute, reducing its ability to successfully slow down the rocket.	Demonstration	A Nomex deployment bag is be used to contain the folded main parachute and protect it from the ejection charges. The drogue parachute will be protected by a Nomex blanket. Ground separation tests will be performed to ensure adequate parachute protection before launch.	Incomplete

R.3	The altimeter bay shall be removable such that rocket apogee/flight data can be quickly retrieved after successful recovery. (Preference should be given to systems that can be removed from the rocket with minimal tools)	A removeable altimeter bay allows for quicker assembly and retrieval, allowing for quicker launch turnarounds.	Testing	The recovery altimeters will be contained in the CRAM, which will be easily removed from the rocket via a twist-to-lock mechanism. Altimeter retrieval demonstration shall be performed such that the altimeters are removed from the landed, separated rocket in 5 minutes or less.	In Progress
R.4	System shall be redundant such that any 2 component failures (such as altimeter malfunction, battery disconnect, or defective E-match) does not compromise the ability to safely recover the vehicle and complete the mission.	Failure to deploy parachute is a major safety issue, and redundant components increase the reliability of the recovery system and decrease the likelihood of parachute deployment failure	Demonstration	Three independent altimeters are used to control parachute deployment, with each altimeter fully capable of deploying both parachutes at the proper times	In Progress

R.5	The altimeter compartment shall be sealed off from the parachute compartment, to prevent the ejection charges from damaging the electronics	Unless the altimeter compartment is sealed off from the parachute bay, the ejection charges can damage the altimeters and hinder main parachute deployment, which is a major safety hazard.	Demonstration, Testing	1. Ground separation testing. An improperly sealed altimeter compartment will allow gas to escape out the altimeter ports, which will be visible during test. 2. Altimeter data will be analyzed after test flight looking for sudden dips in altitude just after apogee, which is indicative of the ejection charge gasses entering the altimeter bay	Incomplete

R.6	Recovery system shall be capable of being "safed" after landing, in the event that an ejection charge has failed to deploy.	In the rare case that an ejection charge does not ignite in flight, it becomes a safety hazard to personnel recovering the rocket. A method of external safety allows for safe retrieval of the rocket in the case of a live deployment charge.	Demonstration	In a simulated launch environment, an attempt will be made to initiate the ejection charge with one of the stops in place. Passing condition: The ejection charge does not activate.	Incomplete

Table 67: Derived Payload Requirements

ID#	**Description**	**Justification**	**Verification Method**	**Verification Plan**	**Status**
S.1	The Rover must not have an overall width larger than 6 inches	Constraining the Rover to a 6 inch maximum width gives the other subsystems a dimension to design around	Inspection	All rover body designs will be constrained to a width of 6 inches	In Progress

S.2	The Rover must be able to overcome small obstacles such as rocks, corn stalks, and crop rows	The terrain where the launch will be conducted is not flat and easy to navigate, so the Rover should be able to overcome any obstacles it may encounter	Demonstration, Testing	The translation mechanism will be tested traversing multiple types of obstacles	Incomplete
S.3	The Rover must be able to traverse through mud, puddles, corn stalks, and corn fields	The state of the terrain is variable and the rover should be designed to overcome any terrain it may experience.	Demonstration, Testing	The rover will be tested traversing through various terrains that may be present at the launch including mud, puddles, and high cut corn	Incomplete
S.4	The Rover must hold and protect the electronics in a water proof container	Making the Rover water resistant will enable it to travel through puddles rather than going around them and wasting time.	Inspection, Demonstration, Testing	All containers that will house electronics will be water tested to ensure there are no leaks and they function as intended	Incomplete
S.5	The Rover must not weigh more than 40 ounces	Constraining the Rover to a maximum weight will prevent the payload from going over weight	Inspection	An up to date weight budget of all components will be maintained to keep track of the weight of the systems	Incomplete

S.6	The Rover must have a minimum operating time of 20 minutes	A 20 minute operating time will provide adequate time for the Rover to traverse to the closest FEA	Demonstration, Testing	Operating time calculations will be conducted at various design milestones to verify the selected components will enable the Rover to operate for a minimum of 20 minutes	In Progress
S.7	The Rover must have a manual override switch	A manual override enables the operator to take control of the Rover should an error occur in the control code	Inspection, Demonstration, Testing	All control software will be required to have a manual override built into the code	In Progress
S.8	The Rover will remain dormant until receiving the initiation signal from the UAV	A low power mode will conserve the battery life of the Rover prior to deploying	Demonstration, Testing	Various testing will be conducted with the Rover in the low power mode to ensure that no external force or signal will bring the Rover out of the dormant state	Incomplete

S.9	The UAV must be no larger than 4 in x 4 in	This constraint enables the UAV to fit inside the payload bay without the need for moving arms	Inspection	All UAV frame designs will be constrained to a width of 6 inches	In Progress
S.10	The UAV frame must protect the battery	Damage to the battery can result in catastrophic failure so the risk for damage should be mitigated	Inspection, Demonstration	All UAV frame designs will be required to have no moving parts and all components will need to be statically secured	In Progress
S.11	The UAV must weigh under 2.4 ounces	Constraining the UAV to a maximum weight will prevent the payload from going over weight	Inspection	An up to date weight budget of all components will be maintained to keep track of the weight of the systems	In Progress
S.12	The UAV must have a minimum flight time of 10 minutes	A 10 minute flight time will provide adequate time for the UAV to search the area around the Rover	Demonstration, Testing	Flight time calculations will be conducted at various design milestones to verify the selected components and the selected frame design will enable the UAV to fly for a minimum of 10 minutes.	In Progress

S.13	The UAV must use a commercial flight controller	Using a commercially available flight controller expedites the flight software development process	Inspection	Only commercial flight controllers will be accepted when reviewing proposed electrical designs for the UAV	In Progress
S.14	The UAV must have a manual override switch	A manual override enables the operator to take control of the UAV should an error occur in the flight code	Inspection, Demonstration, Testing	All flight software will be required to have a manual override built into the code	In Progress
S.15	The Sample Retrieval system must recover a minimum sample size of 15 mL	Having a sample size target over the required sample size will ensure the retrieval of a 10 mL sample	Demonstration, Testing	All sample retrieval designs will be required to hold a 20 mL sample. The system will be extensively tested to ensure it consistently retrieves a sample no smaller than 15 mL.	In Progress

S.16	The Sample Retrieval system must be able to correctly orient itself for retrieval operations	A self orienting sample retrieval system will allow the rover to be in an position when the sample retrieval system is operating	Demonstration, Testing	The retrieval system will be extensively tested to verify it can correctly orient itself to perform the retrieval operations consistently and reliably	Incomplete
S.17	The Sample Retrieval system must retain and protect the recovered sample from spillage and contamination	Securing the sample once it is collected will ensure successful deliver of the sample from the FEA	Inspection, Demonstration, Testing	The sample container will be water tested to ensure no contaminants can leak into the container and the container will be tested through sample retrieval simulations to ensure no amount of sample can spill out of the container during the translation of the rover	Incomplete

S.18	The Sample Retrieval system must interface with the Rover electronics	This will reduce system complexity and reduces the risk of failure	Demonstration, Testing	The sample retrieval team will communicate regularly with the rover electronic team to ensure that the retrieval system can integrate into the electronic system of the rover	In Progress
S.19	The Sample Retrieval system must be easily integrated with the Rover frame	This will reduce system complexity and reduces the risk of failure	Inspection, Demonstration	The team is utilizing Fusion 360 and cloud based models to ensure all assemblies use up to date models and all systems integrate together	In Progress
S.20	The Deployment system must have multiple failsafes	Multiple failsafes will ensure system success despite a component failure within the system	Demonstration, Testing, Analysis	All designs of the deployment system will include a minimum of two redundant locking mechanisms for restricting motion of components in the bulkhead of the vehicle	In Progress

S.21	The Deployment system must be able to correctly orient the Rover and UAV regardless of the landing position of the upper section of the vehicle	The orientation of the Rover is paramount to mission success and must operate successfully	Demonstration, Testing	The orientation system will be extensively tested with the bulkhead section of the vehicle to ensure that it consistently and reliably orients the Rover and UAV for multiple orientations and landings of the bulkhead section of the vehicle	Incomplete
S.22	The deployment system must restrict motion of the Rover and UAV in all directions until the deployment sequence is initiated	Flight stability of the vehicle is dependent on all components in the payload bay remaining locked in place	Demonstration, Testing	All designs of the deployment system will be required to restrict motion of the Rover and UAV in the X, Y, and Z directions. Additionally, all motion restricting designs will be extensively tested to verify proper functionality	In Progress

S.23	The target detection system must correctly identify the closest FEA	This will minimize travel time and distance for the Rover	Demonstration, Testing	The target detection software will be tested to consistently locate the closest FEA during multiple simulations in which fluorescent material will be placed on multiple types of terrain.	Incomplete
S.24	The target detection system must identify the corner of the FEA that is furthest from the Rover	This will reduce the risk of the Rover driving over the UAV	Demonstration, Testing	The target detection software will be tested to correctly and reliably identify the corner of the FEA that is furthest from the Rover.	Incomplete

Table 68: Derived Systems and Safety Requirements

ID#	**Description**	**Justification**	**Verification Method**	**Verification Plan**	**Status**
S.1	Prior to any launch, team members shall be briefed and tested about safety and procedures in accordance with NAR/TAR and NDRT regulations.	To ensure the safety of team personnel, members must be informed of the hazards at launch and proper procedures.	Inspection	Attendance will be taken at pre-launch briefings and any members not in attendance will not be eligible to attend the launch. Additionally members failing to pass the safety quiz will not be eligible to attend the launch.	Incomplete
S.2	Prior to construction of subsection components and the full assembly, schematics and procedures shall be published to ensure correct and safe manufacturing and assembly techniques	Construction procedures and schematics provide clarity which makes the construction process safer and more efficient.	Inspection	Schematics will be created based on finalized 3D models and available in the workshop prior to any construction	In Progress

S.3	The team shall maintain updated records of the team's hazard analyses, failure modes analyses, procedures, and MSDS/chemical inventory data and will use this information to drive design, construction, and testing decisions	Updated information allows the team to make safer and improved decisions.	Inspection	Documentation will be available, reviewed, and updated on a regular basis. The most current version will be available on the team shared drive.	In Progress
S.4	Each NDRT member participating in construction shall be certified on the machines and tools used in accordance to the Notre Dame Student Fabrication Lab standards	Requiring certifications for workshop tools ensures that members learn the proper technique and are informed of workshop hazards.	Inspection	Members will receive a card that indicates which tools they are certified on. Each team member must present this card to a team officer before working on any construction.	In Progress

S.5	Each subsection of the vehicle and payload shall be tested individually before the full scale test.	Component testing allows the team to identify and correct errors prior to full-scale testing, increasing probability of a successful mission.	Inspection	The Safety and Systems Team will work with each design team to develop testing plans and rigs prior to conducting tests. The physical copy of the testing plan will be used for running the test, and the test results will be filed digitally.	In Progress
S.6	The team will develop detailed test procedures at the component and full-scale level to ensure that the designs are robust and reliable.	Testing procedures will allow for a standardization of documentation and streamlined communication. Thorough procedures also ensure that members go into testing fully prepared.	Inspection	A generic test procedure format will be available to the technical leads to modify. Each subsystem will present their testing results prior to full scale assembly.	In Progress

6.2 Project Budget

The Notre Dame Rocketry Team has budgeted $20,753 for the competition this year. The primary funding for this project comes from The Boeing Company and Pratt & Whitney. Additionally, various smaller sources of funding have added to the revenue supporting the team. The break down of NDRT's corporate sponsorship and school-sourced funding is shown in Table 69.

Table 69: Notre Dame Rocketry Team Funding Sources

Funding Source	**Amount**
Carryover (2018/19)	$2,722.00
Team Merchandise	$160.00
ND Day Fundraising	$671.00
The Boeing Company	$10,000.00
Pratt & Whitney	$5,000.00
ND EE Department	$1,000.00
Jim Lampariello	$1,000.00
General Electric	$200.00
TOTAL	**$20,753.00**

The current sourced funds from corporate outreach this year total $16,671 and are sufficient for covering the costs of this year's project. While Table 69 covers the monetary funds provided by the team's sponsors, there are also relationships being forged with companies to provide materials desired by the team. Currently, NDRT is pursuing a relationship with the company 4PCB in order to provide the team with printed circuit boards. Going forward, the team hopes to continue building on its primary revenue stream and increase fundraising to support Research and Development costs for the program. The team plans to continue reaching out to new companies as the year progresses, in order to continue building new corporate relationships. After considering historic spending for the project and initial materials sourcing, a projected budget was established and funds were allocated to each of the major program categories. The budgeted amounts are given in Table 70 along with the current amounts expended for the project.

Table 70: Notre Dame Rocketry Team Funding System Allocations

Allocation	**Amount**
Vehicle Design	$5,000
Recovery Subsystem	$1,500
Lunar Ice Sample Recovery Payload	$2,000
Air Braking System	$1,300
Systems & Safety	$650
Telemetry Subsystem	$500
Rocket Subtotal	**$10,950**
Educational Engagement	$300
Competition Travel	$9,000
Miscellaneous	$500
TOTAL	**$20,750**

The largest expenditures for the team are the overall launch vehicle construction and traveling to competition. This budget currently only has an overrun margin of $3, but the team predicts this margin to grow as more companies are contacted for sponsorship throughout the competition year.

The material acquisition plan for the team this year has relied heavily on vendors the team has partnered with in the past, such as Apogee Components. Additional sources for procuring components have been researched to reduce both cost and lead time on materials after being ordered. One final avenue, is to leverage the team's relationship with corporate sponsors, such as Boeing, to purchase excess composite materials from the company at a discounted rate. This is something the team is actively pursuing and will take into consideration for the competition vehicle. A detailed breakdown of the itemized budget organized into allocation categories for the project is shown in Table 71.

Table 71: Itemized Budget

Recovery System Components	Vendor	Description	Qty	Price per Unit	Total Cost
3.7V 170mAh Lipo	Wing Deli Storefront	Rechargeable Battery Pack	1	7.48	7.48
		TOTAL COST			7.48
		Budget Allocation			1500.00
		Margin			1492.52
Systems & Safety Components	Vendor	Description	Qty	Price per Unit	Total Cost
Vinyl Gloves (200)	Walmart	PPE	1	11.98	11.98
Face Masks (5)	Walmart	PPE	4	0.97	3.88
Lysol Wipes	Walmart	Cleaning	1	2.98	2.98
		TOTAL COST			18.84
		Budget Allocation			650.00
		Margin			631.16
Vehicle Components	Vendor	Description	Qty	Price Per Unit	Total Cost
RockSim Licenses	Apogee Components	General	4	20.00	80.00
G80T-7 Motors	Apogee Components	Subscale	3	35.30	105.90
Motor Retainer	Apogee Components	Subscale	1	10.00	10.00
Nose Cones 11.25" long	Apogee Components	Subscale	2	22.19	44.38
Payload Bay (3" tube)	Apogee Components	Subscale	1	11.17	11.17
Rest of rocket (66mm tube)	Apogee Components	Subscale	1	13.00	13.00
Balsa Sheet	Apogee Components	Subscale	1	1.76	1.76

Couplers	Apogee Components	Subscale	5	16.75	83.75
Motor Mount (29mm tube)	Apogee Components	Subscale	1	4.99	4.99
Epoxy Clay	Apogee Components	Subscale	1	14.95	14.95
Taxes + Shipping	Apogee Components	Subscale	1	86.48	86.48
		TOTAL COST			456.38
		Allocation			5000
		Margin			4543.62
Lunar Ice Recovery System Components	Vendor	Description	Qty	Price Per Unit	Total Cost
Raspberry Pi 3	CanaKit	Pi 3 with 2.5A USB Power Supply	1	49.62	49.62
16 GB Memory Card	SanDisk	Ultra microSDHC Memory Card with Adapter	1	5.79	5.79
		TOTAL COST			55.41
		Allocation			2000
		Margin			1944.59

6.3 Project Timeline

In order to meet the deadlines set by the NASA Student Launch Management Team, the Notre Dame Rocketry Team has committed to the following timeline shown in Table 72 and project overview Gantt chart shown in Figure 46.

Table 72: Notre Dame Rocketry Team 2019-2020 project overview.

Date	Task
November 2019	
01	Complete sub-scale construction; PDR report, presentation slides, and flysheet submitted to NASA
10	Sub-scale test flight with Indiana Rocketry Inc.
11-15	Sub-scale wind tunnel testing
16	Backup sub-scale test flight with Michiana Rocketry
20	PDR video teleconference

25	Critical Design Review (CDR) Q&A
December 2019	
02-13	Full-scale parts ordered
January 2020	
05	CDR report, presentation slides, and flysheet submitted to Dr. Jemcov
10	CDR report, presentation slides, and flysheet submitted to NASA
13-17	Final purchases for full-scale completed; Testing completed at the Materials Tensile Properties Lab with parts that were delivered over winter break
13–28	CDR video teleconferences
20-31	Full-scale construction
31	Flight Readiness Review (FRR) Q&A
February 2020	
01	Early date for #1 Full-scale test flight (potentially with Indiana Rocketry Inc.)
03-14	Continued full-scale construction and testing
15	Full-scale test flight #2 with Michiana Rocketry (if applicable)
16	Full-scale test flight #2 with the Wisconsin Organization of Spacemodeling Hobbyists (given that there are poor weather conditions in Three Oaks, MI, if applicable)
22	Full-scale test flight #3 (potentially with Indiana Rocketry Inc., if applicable)
24	FRR report, presentation slides, and flysheet submitted to Dr. Jemcov
March 2020	
02	Vehicle demonstration flight deadline
02	FRR report, presentation slides, and flysheet submitted to NASA
02-06	Full-Scale and Payload Testing (if applicable)
06–19	FRR video teleconferences

14	Full-scale final test flight for payload demonstration with Michiana Rocketry (if applicable)
20	FRR Addendum submitted to Dr. Jemcov
23	Payload demonstration flight and vehicle demonstration re-flight deadlines
23	FRR Addendum submitted to NASA (if applicable)
26	Launch Week Q&A
April 2020	
01	Teams travel to Huntsville, AL
01-03	Official launch week kickoff, Launch Readiness Reviews (LRRs), and launch week activities
04	Launch Day
04	Awards Ceremony
05	Backup launch day
23	Post Launch Assessment Review (PLAR) submitted to Dr. Jemcov
27	PLAR submitted to NASA

WBS	Task description	Start date	Finish date	Progress
3	PDR Drafting	9/22/2019	10/27/2019	100%
4	Sub-Scale Part Ordering	10/14/2019	10/18/2019	100%
5	PDR Review and Completion	10/27/2019	11/1/2019	100%
6	Sub-Scale Construction	10/28/2019	11/1/2019	100%
7	Sub-Scale Wind Tunnel Testing	11/11/2019	11/15/2019	25%
8	CDR Drafting	11/25/2019	1/5/2020	0%
9	Full-Scale Part Ordering	12/2/2019	12/13/2019	0%
10	CDR Review and Completion	1/5/2020	1/10/2020	0%
11	Materials Tensile Properties Testing with Delivered Full-Scale Parts	1/13/2020	1/17/2020	0%
12	CDR Video Teleconferences	1/13/2020	1/28/2020	0%
13	Full-Scale Construction	1/20/2020	1/31/2020	0%
14	Full-Scale Flight Window	2/1/2020	3/2/2020	0%
15	FRR Drafting	1/31/2020	2/24/2020	0%
16	Full-Scale Construction (if needed) and Payload Testing	2/3/2020	2/14/2020	0%
17	FRR Review and Completion	2/24/2020	3/2/2020	0%
18	Full-Scale and Payload Testing (if needed)	3/2/2020	3/6/2020	0%
19	FRR Addendum Drafting (if needed)	3/2/2020	3/20/2020	0%
20	FRR Video Teleconferences	3/6/2020	3/19/2020	0%
21	FRR Addendum Review and Competion (if needed)	3/20/2020	3/23/2020	0%
22	Launch Week	4/1/2020	4/5/2020	0%
23	PLAR Drafting	4/5/2020	4/23/2020	0%
24	PLAR Review and Completion	4/23/2020	4/27/2020	0%

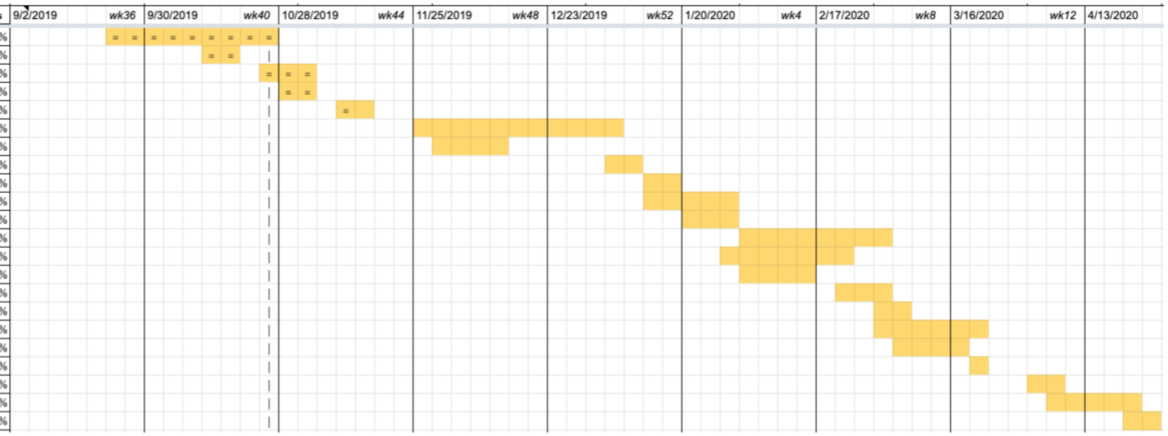

Figure 46: Project Gantt chart