Rose Rocketry

Project Silverstein
Preliminary Design Review

Rose-Hulman Institute of Technology

November 1, 2021

5500 Wabash Ave, Terre Haute, IN 47803

Table of Acronyms 4

Summary of PDR Report 6
- Team Summary 6
- Launch Vehicle Summary 6
- Payload Summary 6
  - The RHIT Stuff 6
  - Payload Experiment 6

Changes Made Since Proposal 7
- Vehicle Criteria 7
- Payload Criteria 7
- Project Plan 7

Vehicle Criteria 8
- Mission Statement and Mission Success Criteria 8
- Overview of Vehicle Systems 8
- Airframe Design 9
  - Semi-Monocoque Aerostructure 10
  - Cylindrical Monocoque Airframe 10
  - Conical Monocoque Airframe 10
  - Sandwich Composite Aerostructure 11
  - Cylindrical Airframe Sizing Alternatives 12
  - Large-size Airframe 13
  - Small-size Airframe 13
- Aerodynamic Design 13
  - Nose Cone 14
  - Blunted Body 16
  - Aft Fins 16
  - No Fins 16
- Deployment Design 18
  - Dual-Point Separation 19
  - Single-Point Separation 20
- Altitude Assurance Design 20
  - Passive Mass Adjustment 22
  - Jettisonable Ballast Mass 22
  - Thrust Modulation 23
  - Drag-Producing Devices 24
- Propulsion System Design Alternatives 25
  - Disposable Motor System 26
  - Reloadable Motor System 26

Slow Burn Motor 26
Fast Burn Motor 27
Vehicle Systems Leading Alternatives and Justifications 27
Airframe Structural Design and Sizing 28
Aerodynamics 28
Deployment 28
Propulsion 29
Recovery Component Selection 29
Main Parachute Deployment Strategy 29
Tumble at Apogee then Main 29
Drogue Deploy at Apogee then Main Deploy 29
Streamer Deploy at Apogee then Main Deploy 30
Nylon 31
Disposable AA Cells 33
3.7v Lipo Battery 33
18650 Cell 33
Hybrid Supercapacitor 34
Summary of Design Redundancy 34
Mission Performance Predictions 35
Summary of Mission Performance Calculations 35
Flight Profile Simulation 36

Payload Criteria **37**
Experiment and Criteria for success 37
Overview of Payload Systems 37
Control System Design 38
Embedded Controller 38
Discrete Electronics 39
Embedded Linux 39
No Control System 39
Telemetry Design 40
Discrete Component RF Circuit 40
COTS Module 40
Sensor and Data Acquisition 40
Time Integration 41
Radar 42
RF passive 42
Visual Simultaneous Localization And Mapping 45
Power Delivery 45
Disposable Cell 46

Rechargeable Batteries 46
Hybrid Supercapacitor 46
Preliminary Payload Design 46
Leading Control System Alternative 46
Leading Telemetry Alternative 47
Leading Sensor and Data Acquisition Alternative 47
Leading Power Delivery System Alternative 48

Safety 49
Safety Officer 49
Personnel Hazard Analysis 51
Failure Modes and Effects Analysis (FMEA) 56
Vehicle System FMEA 56
Payload and Payload Integration FMEA 63
Payload and Payload Integration FMEA 66
Environmental Concerns 67
Project Risks 70

Project Plan 73
Derived Requirements 73
Budget 77
Line Item Budget 77
Material Acquisition Plan 82
Material Acquisition Plan 82
Timeline 84
Project Timeline Gantt Chart 84
Major Project Deadlines 85

Appendix 86
Citations 86

Table of Acronyms

Acronym	Definition
RR-SL	Rose Rocketry - Student Launch
BIC	Branam Innovation Center
KIC	Kremer Innovation Center
SL	Student Launch
NASA	National Aeronautics and Space Administration
FAA	Federal Aviation Administration
NAR	National Association of Rocketry
HPR	High Powered Rocketry
PPE	Personal Protection Equipment
PDR	Preliminary Design Review
TRA	Tripoli Rocket Association
LEO	Low Earth Orbit
LRR	Launch Readiness Review
FRR	Flight Readiness Review
CDR	Critical Design Review
CG	Center of Gravity
CP	Center of Pressure
RF	Radio Frequency
AGL	Above Ground Level
STEM	Science Technology Engineering and Math
SGA	Student Government Association
RSO	Range Safety Officer
GPS	Global Positioning System
IMU	Inertial Measurement Unit
APCP	Ammonium Perchlorate Composite Propellant
FIRST	For Inspiration and Recognition of Science and Technology
FRC	FIRST Robotics Competition
FMEA	Failure Modes and Effects Analysis
DSN	Deep Space Network

1. Summary of PDR Report

1.1 Team Summary

Table 1.1: Team Summary and Mentor Contact Information

Team Name	Rose Rocketry - Student Launch (RR-SL)
Mailing Address	5500 Wabash Ave, Terre Haute, IN 47803
Mentor Name	Gary Kawabata
Mentor Contact	rocketguy9914@gmail.com
Mentor Certifications	NAR 89092; TRA 3019; level 3
NAR/TRA Sections	Indiana Rocketry Group Tripoli #132 NAR Section #711
Hours Spent on PDR	275

1.2 Launch Vehicle Summary

Table 1.2: Launch Vehicle Summary

Official Target Apogee	5000 ft.		
Preliminary Motor Choice	Cesaroni Technology Inc. L2375WT-P.		
Recovery System	Rocketman 7ft. Pro Experimental Drogue Parachute SkyAngle CERT-3 Large		
Mass of Individual Sections	6 lb.	16.91 lb.	11.8 lb.

1.3 Payload Summary

1.3.1 The RHIT Stuff

The payload for this year's competition will be named "The RHIT Stuff," drawing inspiration from Tom Wolfe's book *The Right Stuff*.

1.3.2 Payload Experiment

The payload will determine the location of the landing site without using GPS. It must identify the grid cell that the vehicle landed in on a map of the launch field divided into a grid of squares that are 50m (164 ft) on each edge. Our payload will do this by measuring the time-of-flight of an RF signal to find the distance, together with integrating IMU data.

2. Changes Made Since Proposal

2.1 Vehicle Criteria

Since the submission of the Project Silverstein Proposal, no significant design changes to the Vehicle Systems have been decided.

2.2 Payload Criteria

Since the submission of the Project Silverstein Proposal, no significant design changes to the Mission Payload have been decided.

2.3 Project Plan

Since the submission of the Project Silverstein Proposal, the mission plan has accounted for the Indiana Rocketry Club 2021-22 timeline for planning of the Project Silverstein timeline. See Section 6.2 for more details. In addition, team derived requirements for the Vehicle Systems design and the Mission Payload design have been established.

3. Vehicle Criteria

3.1 Mission Statement and Mission Success Criteria

The objective of Project Silverstein is to design and fabricate a payload and launch vehicle that will ascend to a target apogee and report its landing location to our ground station. This is to be done to industry standards of reliability and in support of developing a team history of successful mission execution.

A successful mission meets all of the following criteria:

- All members abide by all safety regulations put into effect
- The launch vehicle is launched on a safe, stable, and predictable trajectory
- The payload can robustly locate the launch vehicle upon descent
- The launch vehicle is recovered in a state suitable for reuse

3.2 Overview of Vehicle Systems

Following the requirements outlined in the 2022 USLI handbook, the launch vehicle was divided into subsystems to perform systems-level design. These subsystems have been selected to present individual objectives and provide focus for an exploration of the potential vehicle design space. Table 3.1 shows a summary of these vehicle systems.

Table 3.1: Vehicle Systems Overview

Vehicle Subsystem	Section	Objective
Airframe	3.3	Provide sufficient structural housing for the vehicle components considering spatial constraints and vehicle mass. Heavy emphasis is put on reliability.
Aerodynamics	3.4	Support the vehicle in safe ascent considering constraints imposed on the propulsion system.
Deployment	3.5	Allow for separation of the vehicle to eject payload and/or recovery subsystems.
Altitude Assurance	3.6	Support the launch vehicle in achieving an altitude as close to the predicted apogee as possible.
Propulsion	3.7	Provides the necessary thrust for the rocket to reach a desirable range of apogees while considering structural load to the vehicle.
Recovery	3.9	Assures that the rocket returns safely to the ground without significant injury to itself and the payload during descent.

3.3 Airframe Design

The objective of the airframe system is to determine the external dimensions of the rocket, to rigidly locate each of the individual vehicle components relative to each other (such as recovery devices, payload, avionics, etc.), and to make sure that aerodynamic devices maintain their shape and orientation. Criteria considered for each design alternative include: rigidity, robustness, weight, cost, ease of construction, and spatial constraints on the vehicle components. Particular emphasis is put on the reliability of each design alternative, in order to maximize the chance of a successful flight. The design alternatives considered are summarized in Table 3.2, and discussed individually in the following sections.

Table 3.2: Airframe Structure and Sizing Alternatives

Airframe Designs	Pros	Cons	
Cylindrical Semi-Monocoque Airframe	- Can handle similar compressive loads with lower mass - Stresses on airframe can be modeled using beam-loading	- Structure must be custom made	
Cylindrical Monocoque Airframe	- Can be purchased commercially from multiple vendors - Purpose built airframe with a variety of material selection - Airframe sizes are standardized	- Thicker shell is required for compressive loads	✔
Conical Monocoque Airframe	- Imposes less size constraints towards the aft of the vehicle	- Larger surface area increases drag - Added complexity to constructing this geometry	
Cylindrical Sandwich Aerostructure	- Increased stiffness with minimal weight addition - Ease of customization - Increased flexibility in geometry	- Added complexity to manufacture an airframe	
Large-Dimension Cylindrical Airframe	- Less spatial constraints for internal vehicle components	-Greater material expense -Larger drag forces in flight	✔

Small-Dimension Cylindrical Airframe	-Lower drag forces in flight -Lower material expense	-Stricter spatial constraints	

3.3.1 Semi-Monocoque Aerostructure

One design decision considered for the airframe is to use a semi-monocoque structure shown in Figure 3.1, wherein structural reinforcement handles the load on the airframe and is protected by a non-structural skin. This would reduce vehicle mass in comparison to a monocoque airframe as less mass is needed to handle similar compression loads on the airframe. Additionally, the stresses endured by the airframe can be modeled using beam loading approximations [1]. This would allow for greater design optimizations in minimizing reinforcement mass. The drawbacks to this design is that a semi-monocoque structure would also need to be custom made using construction methods unfamiliar to the team as opposed to monocoque airframes that are readily available from rocketry parts vendors.

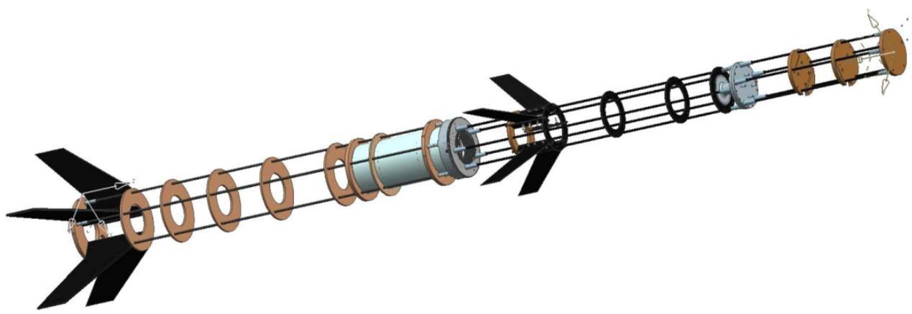


Figure 3.1: Example of a semi-monocoque structure without the skin

3.3.2 Cylindrical Monocoque Airframe

In traditional launch vehicles of our scale, a cylindrical monocoque airframe tends to be the default design choice. Because of this, these airframes tend to be sold from a variety of vendors and are offered in multiple selectable materials. Commercially available tube airframes are purpose-built for launch vehicles of our scale and are offered in standard sizes which streamlines the design process [2]. The drawbacks to this option is that monocoque airframes require thicker walls (and thus, greater airframe mass) to match the compressive strength of a semi-monocoque airframe [1]

3.3.3 Conical Monocoque Airframe

Another design alternative considered for the airframe system is to use a conical geometry. An example of a conical airframe projection onto a tube airframe is given in Figure 3.2.

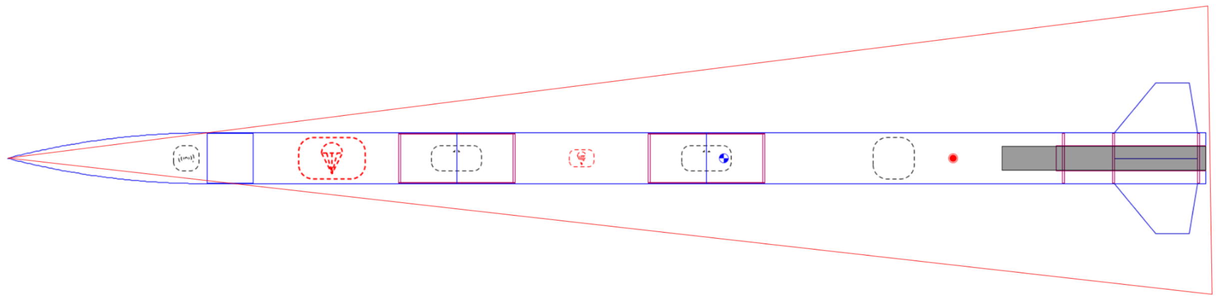


Figure 3.2: 2-D projection of a conical monocoque airframe

The advantage to this airframe design is that there is more space aft of the vehicle. It is not a great advantage given that there is not an obvious need for space in this direction of the vehicle. A similar design has been used in the NASA Delta-Clipper project [3]. The application of this design was for use in vertical take-off and landing operation hence the wider base at the aft end of the vehicle. Since there is no vertical take-off and landing operation in Project Silverstein, the wider base would have no advantage in this regard. The drawbacks to this design is that the larger surface area used by the cone increases drag forces on the vehicle [4] Additionally, this is a nonstandard airframe geometry, and using this shape would require developing a mold to form the structure.

3.3.4 Sandwich Composite Aerostructure

A hexcore, or honeycomb, airframe design shown in Figure 3.3 would comprise of two structural shells filled in by hexagonally structured material. In industry, this construction is sometimes referred to as “sandwich panel” construction. This design would allow for much greater stiffness of the airframe without drastically increasing the weight of the vehicle. Additionally, the geometry of the vehicle would have some flexibility, even when building up from a tubular core [5]. The drawbacks to this design are that construction would be difficult, and would require applying and aligning multiple layers to some sort of mold or rigid support to be formed to shape, and then applying pressure with a vacuum bag or balloon. Designing these kinds of structures is also more complex than working with simple tubes.

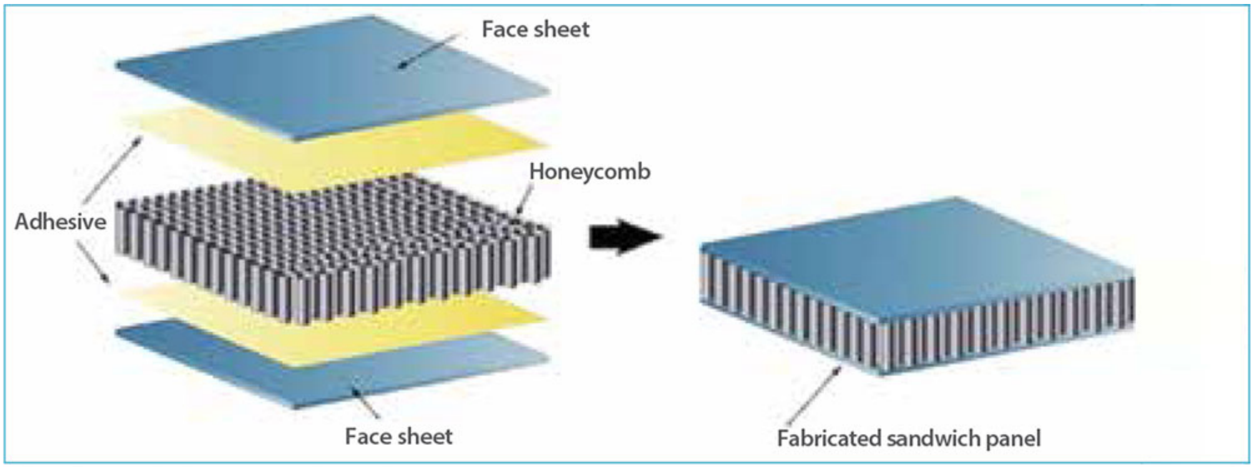


Figure 3.3: Sandwich Composite Aerostructure

3.3.5 Cylindrical Airframe Sizing Alternatives

[Discuss sizing limitations]

- The rocket has to fit an L2 motor (>4” diameter)
- Maximum diameter is 7” due to print bed limitations for the 3D printer

From the geometric airframe designs considered above, it appears to the RR-SL team that the most compelling designs are of cylindrical geometry. In justification for airframe sizing, a cylindrical airframe will be considered which applies to monocoque, semi-monocoque, and sandwich aerostructures. The team has derived sizing constraints for the airframe diameter and length. The minimum airframe diameter has been determined to be 4”. Per completion requirements, the maximum allowable impulse class of motors is an L class motor. To allow the vehicle to reach the extremities of L motor impulses, a 75mm motor mount is required. In order to avoid the design and construction of a minimum diameter vehicle, a 4” airframe is considered as a derived minimum airframe diameter. The maximum diameter airframe has been determined to be 6”. In consideration for the use of additive manufacturing in the design of vehicle components, It is desired to use an airframe diameter that may be 3D printed on a standard 200mm x 200mm FDM printer bed. This constrains the maximum diameter of the airframe to be 6” as commercially available airframes are sold in 2” diameter increments above the 4” minimum diameter. Since the team is actively avoiding constraints to construct large sections of the airframe, the airframe diameter is constrained to either 4” or 6”. The maximum airframe length was derived to be 12’. Assuming a reasonable thrust-to-weight ratio of 10:1 and a reasonable CG location of 60% the length of the airframe, a maximum airframe length can be derived from explicit competition requirements. The vehicle must be capable of achieving a rail exit velocity of 52 fps, where rail exit is defined as top rail button separation from the launch rail. If we assume that the top rail button is located at the CG and that the vehicle is launched off a maximum size 12’ rail, then the maximum derived airframe length can be determined from one-dimensional kinematics:

$$v_{exit}^{2} = 2\,g\,(TWR - 1)\,(L_{rail} - 0.6\,L_{vehicle,max})$$

Solving for $L_{vehicle,max}$ yields a maximum vehicle length of roughly 12’ and so this becomes a maximum dimension. A minimum dimension is derived from estimating the lengths of each subsystem internal to the airframe with a reasonable margin of error. The team considers minimum lengths for each of these systems to be the following:

- Payload: 1’
- Recovery: 2’
- Altitude Assurance: 0.6’
- Propulsion: 2’

Applying a tolerance of $\pm$10%, the minimum derived vehicle length has been determined to be 6’.

3.3.5.1 Large-size Airframe

A large-size airframe is considered for the launch vehicle for minimization of spatial constraints for subsystems internal to the airframe. This would allow for less project resources to be invested in spatial optimization of various subsystems. The drawbacks to a larger airframe is that greater drag forces would be incurred [53]. A larger airframe would also add greater cost to the project [18-19].

3.3.5.2 Small-size Airframe

A small-size airframe would benefit from lower material cost [18-19] and smaller drag forces to overcome in flight [53]. A consequence of this is that stricter spatial constraints would be imposed on vehicle subsystems internal to the airframe.

3.4 Aerodynamic Design

Table 3.3: Launch Vehicle Summary

Aerodynamic Design Alternatives	Pros	Cons	
Nose cone	- Lower drag in flight	- More spatial constraints on the fore airframe components - Higher cost of nosecones	✔
Blunted Body	- Less spatial constraints on fore airframe components - Simpler	- Higher drag in flight	

	construction and transportation		
Aft Fins	- Increased vehicle stability	- Added failure mode of fin damage or misalignment	✔
No Fins	- Eliminate fin cost - Airframe does not need modification to accept fins	- Decreased vehicle stability	

The objective of the aerodynamics system is to support a safe vehicle ascent while minimizing aerodynamic drag forces on the vehicle. In making decisions for the aerodynamic design, the stability of the vehicle and the aerodynamic drag forces on the vehicle are considered. The aerodynamic design aims to maximize vehicle stability by directing the center of pressure (CP) to the aft of the vehicle. The design also aims to minimize the drag forces on the vehicle to impose less design constraints on the propulsion system. These decisions consider cost as a factor in order to stay within our club's annual budget for RR-SL activities.

To that end, the major alternatives identified in the aerodynamic design of the launch vehicle are the use of a nose cone versus a blunted body on the front of the vehicle, and the inclusion or exclusion of aft fins. The pros and cons of each are discussed below.

3.4.1 Nose Cone

A more traditional design decision for the aerodynamics system is to integrate a nose cone onto the airframe. The most prominent advantage to using a nose cone is that the launch vehicle becomes more aerodynamic. According to Figure 3.4, the coefficient of drag for a rectangle is 2.05. Considering a rectangle to be a model for a 2D projection of a rocket without a nose cone, a launch vehicle would not be ideal in comparison to the nose cone design, whose 2D projection may be modeled as a triangle, coefficient 1.55, with its vertex facing into the wind. As shown in Table 3.4, more sophisticated simulation in OpenRocket further validates that a nose cone provides a significant reduction in the drag force experienced by the vehicle.

That being said, the primary disadvantage to this design is that a significant spatial constraint is imposed on the interior of the vehicle. Figure 3.5 below illustrates a high-powered rocket constructed by the team for a NAR Level 2 certification - on this particular vehicle, 25% of the total height of the vehicle was occupied by the nose cone. Unlike a cylindrical body, a nose cone has a more complex geometry that must be described with additional constraints - this makes construction of hardware that uses this interior space more complex, and it also complicates structural analysis by removing some of the geometric regularity.

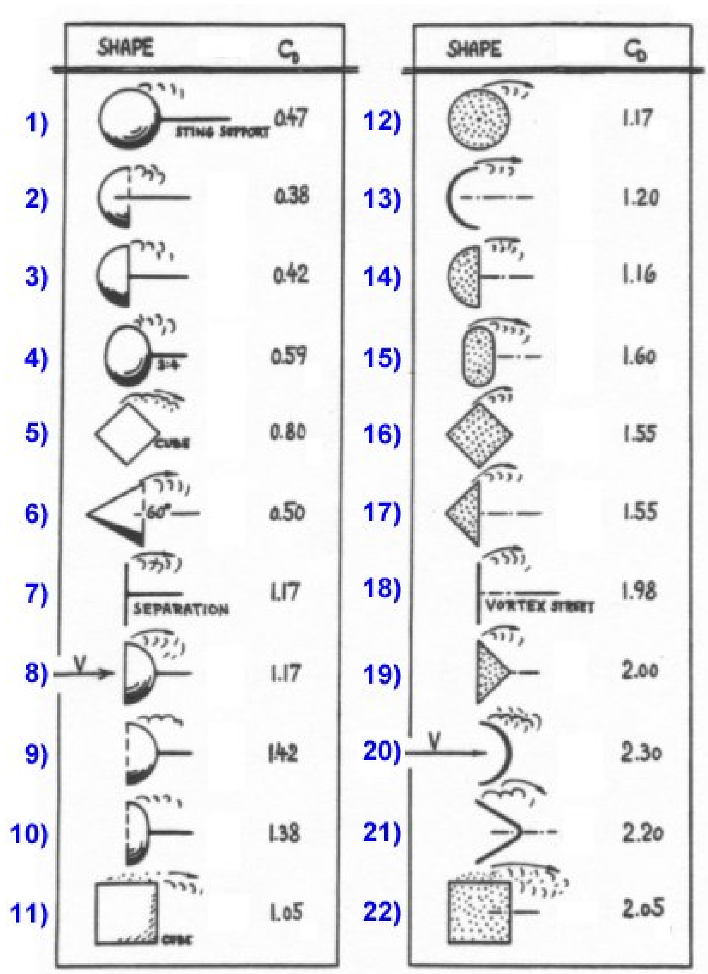


Figure 3.4: Examples of drag coefficients for varying shapes

Figure 3.5: Student-built Level 2 certification rocket

3.4.2 Blunted Body

One aerodynamic feature considered for the launch vehicle is to replace the nose cone found traditionally on most launch vehicles with a flat or hemispherical cap. Though this is unorthodox, there are benefits to adopting this aerodynamic configuration, namely in simplifying the vehicle. A nose cone adds another component that has to be fitted and secured, while a blunted cap simplifies installation and fitment requirements. This would have other positive impacts on pre-launch operations, since a smaller cap is less likely to be damaged during transit. Additionally, a blunted body does not occupy a significant amount of the launch vehicle's volume, while an extended cone geometry adds more significant spatial constraints to elements within the launch vehicle. The drawback of using a blunted body is that the vehicle would become less aerodynamic. This may be shown from kinematic simulations performed in OpenRocket. Table 3.4 summarizes the performance of a launch vehicle of identical parameters using a blunted body aerodynamic design with the performance using a nose cone design. This preliminary analysis suggests that a blunted body could reduce the vehicle apogee by 36%. Although this analysis is not precise, it demonstrates that the impact on aerodynamics is significant.

Table 3.4: OpenRocket Simulation Results

Aerodynamic Design	Apogee (m)	Max velocity (m/s)	Max acceleration (m/s^2)
Nose cone	1797	240	132
Blunted Body	1146	221	127

3.4.3 Aft Fins

An aerodynamic design using aft fins would allow for an increased stability of the launch vehicle in flight, by adjusting the center of pressure towards the rear of the vehicle [6]. Fins must be placed aft of the vehicle in order to meet the competition requirements stating that structural proteurbances must be located aft of the burnout CG. Aerodynamic stability is a further design requirement for the competition, and so the ability to freely modify the center of pressure is considered to be a strong design benefit. The main drawback to this design is that there is an added critical failure mode if the fins become damaged or misaligned.

3.4.4 No Fins

A launch vehicle design using no fins has the primary advantages of reduced weight and removal of critical failure modes. This design would also be simpler to construct, as no slots in the airframe would be required to accept any fins. The drawback to this design is that there is one less design degree of freedom to control the stability of the vehicle. That is, the only way to increase the stability of the vehicle is by adjusting the CG, which puts dramatic constraints on the mass distribution [6] Using no fins on a launch vehicle is a design choice that has been widely adopted by the industry, as shown in Figure 3.6 below. However, this is not necessarily relevant to the design goals of Project Silverstein. Consider the Falcon 9 rocket developed by Space Exploration Technologies Corp - this system is capable of orbital flight, and so the portion of its flight under heavy aerodynamic forces is not significant enough to warrant the added mass of fins. Instead, the Falcon 9 is able to gimbal its engines to actively control the thrust vector and adjust its trajectory when veering off course [7][8]. The team has adopted a derived requirement that energetic devices will only be controlled using COTS hardware, and so active thrust vectoring is not an acceptable stabilization strategy for Project Silverstein. Thus, passive aerodynamic stability is necessary for accomplishing the primary design goal of the aerodynamics system, and the only way to accomplish this is by adjusting the location of masses within the vehicle.

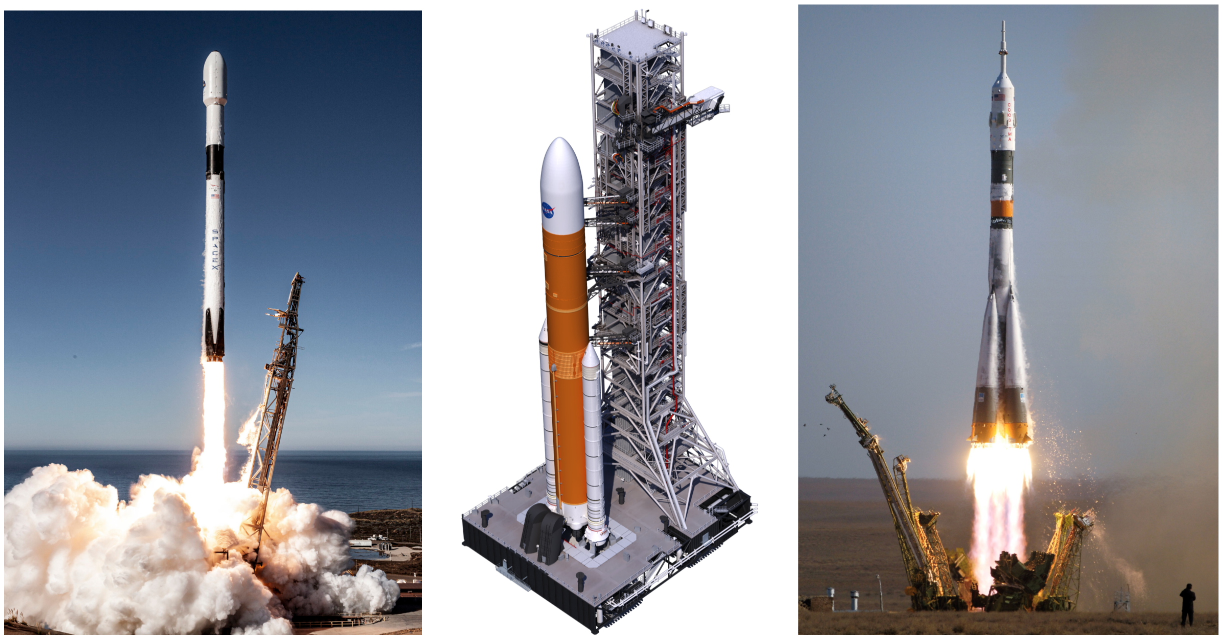


Figure 3.6: Falcon 9 launch showing no fins

3.5 Deployment Design

Table 3.5: Deployment Design Alternatives

Deployment Alternatives	Pros	Cons	
Payload Deploys	- Allows for both external and internal locating strategies	- More safety considerations - Added failure modes	
Fixed Payload	- Minimizes points of separation	- Restricted to internal locating strategies	✔
Dual-Point Separation	- Drogue and main chute reacted on by separate masses - Recovery charges 'push' independently on main and drogue chute - Nose cone weight allows for greater static stability margin	- Two points of separation, increasing potential failure modes - Altitude Assurance proximity to booster	✔
Single-Point Separation	- Reduced separation points, reducing potential failure modes	- Altitude Assurance is fore of massive components - Electronic Chute Release is necessary to control main chute deployment	

Early in the vehicle system design, it was decided that an alternative needed to be chosen for payload deployment. This alternative will govern the strategy for the payload interaction with the vehicle as well as constrain the other vehicle subsystem alternatives.

Considering the objective of the payload to locate the launch vehicle upon descent, many strategies for payload operation were considered. These strategies were divided into two groups of payload deployment needs: external locating and internal locating. External locating strategies benefit from the ability to expand volumetrically outside the space constraints of the vehicle. This would be useful in deploying the payload as an antenna or beacon. It was determined, however, that many strategies could be accomplished without the payload leaving the airframe. For example, a strategy employing phased-array signal steering would be feasible within the confines of the vehicle [9]. A non-deploying payload would then avoid the drawbacks of resource expense to additional safety considerations which would be mandated for a deploying payload. Additional failure modes (such as entanglement with recovery devices) are kept to a minimum with a simple, non-deploying payload. Because of this, it has been decided that the payload will be fixed, and will not deploy from the vehicle.

Below, various alternatives are explored for vehicle separation and energetic device configurations. In the following figures, a yellow component outline corresponds to a location of an energetic device and a red component outline corresponds to a recovery device.

3.5.1 Dual-Point Separation

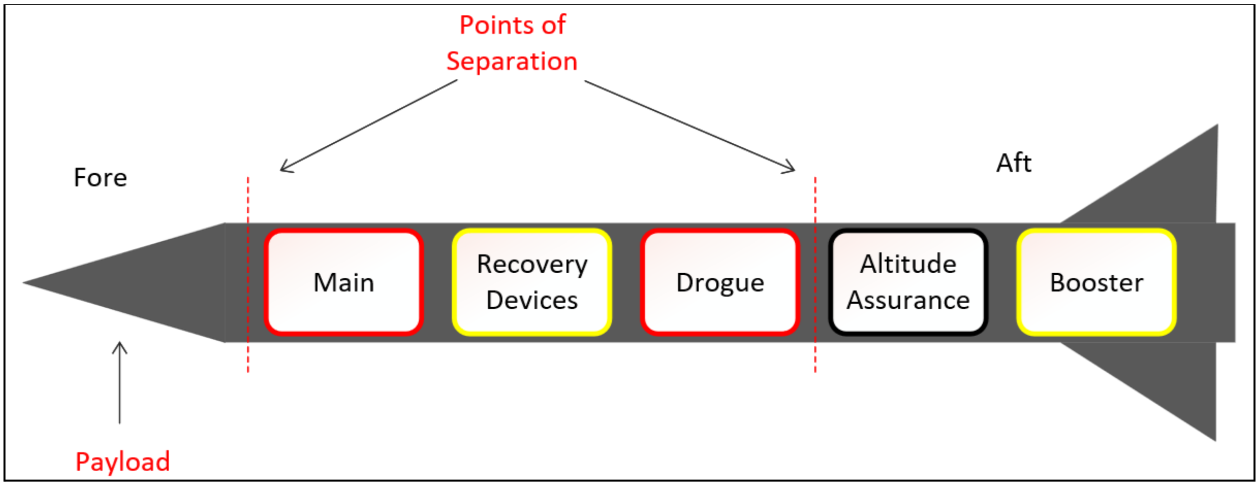


Figure 3.7: Rocket showing two points of separation

The deployment alternative shown in Figure 3.7 exhibits various advantages to recovery operation. The placement of the energetic recovery devices allows for a 'pushing' action upon the main and drogue chutes which ensures better recovery [10]. The attachment of the nose cone and aft airframe section are also able to pull on both recovery chutes to aid in the deployment process. One additional advantage to the vehicle aerodynamic performance that is depicted in this configuration (though not unique to dual-point separation) is that the main chute is located fore to the drogue in the airframe. This would aid in mass-distribution influence on the vehicle's static stability margin. Payload weight in the nose cone also aids in aerodynamic performance for the same reason [11]. The drawbacks to this design, however, are that multiple points of separation exist thus adding failure modes to the deployment subsystem. Similarly, the proximity of the booster to the Altitude Assurance subsystem raises the chance of failure by exhaust gas interference.

3.5.2 Single-Point Separation

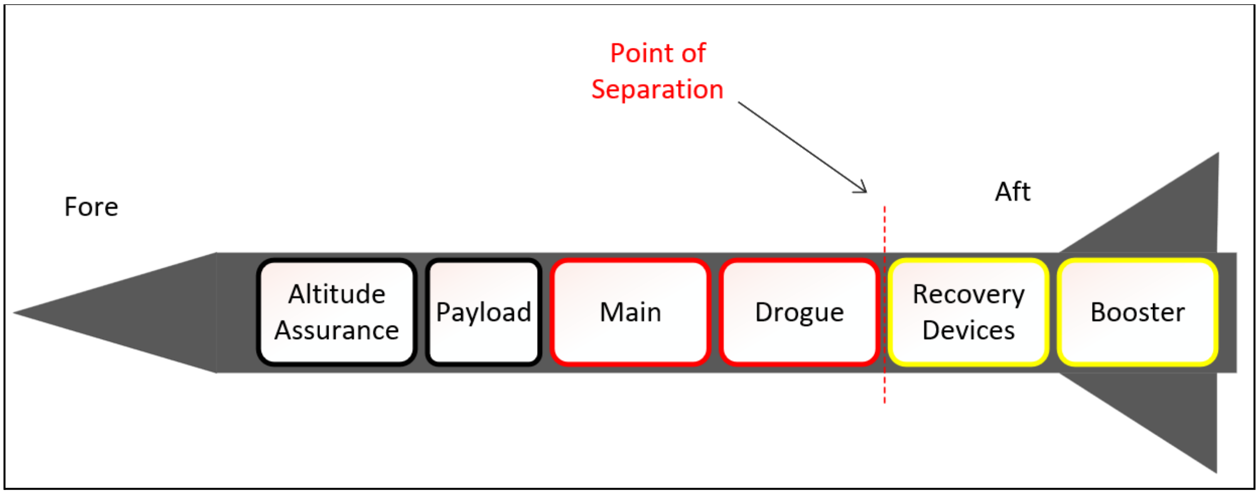


Figure 3.8: Rocket showing one point of separation

The deployment alternative shown in Figure 3.8 benefits from reduced points of failure in separation being just a single-point separation configuration. However, this introduces a significant disadvantage to the recovery process, in that an Electronic Chute Release is necessary to control the main chute deployment altitude [12] Otherwise, the main chute would deploy at the same time as the drogue chute which would result in hard deployment at low altitude separation or it would result in excessive recovery drift at high altitude separation. The operation of the Electronic Chute Deployment would add another possible failure mode thus counteracting the advantage of single-point separation.

3.6 Altitude Assurance Design

Table 3.6: Altitude Assurance Design Alternatives

Altitude Assurance Method	Pros	Cons	
Passive Mass Adjustment	- Simplicity - Failure-resistance	- Limited accuracy - No adaptiveness to environmental conditions - Requires precise measurement of vehicle mass	
Jettisonable Ballast Mass	- Active control - Minimal actuation force	- Low control authority after burnout - Risk of leaks with liquid ballast - Failure of system results	

		in either severe undershoot or severe overshoot - Safety concerns associated with dropping mass from the vehicle	
Thrust Modulation	- Large altitude range - Active control	- Short active time - High-temperature materials required - Risk of thrust deflection causing unsafe yaw - Potentially high actuation force - Difficult to test	
Drag-Producing Devices	- Active control - Good control authority over majority of flight - Can be tested to an extent on ground using wind tunnel - Can be tested on vehicles of various scales	- Requires devices external to airframe - Potentially high actuation force - Requires development of an accurate drag model	✔

The objective of the altitude assurance system is to support the launch vehicle in accurately reaching the target apogee. In making decisions for the altitude assurance system, control authority, feasibility of construction, safety to personnel and vehicle, and ability to recover from errors are considered. Control authority is prioritized, defined here as the ability of the system to change the final altitude of the rocket from a given point in a flight. For this section, preliminary control authority simulation was performed using a 3rd order accurate finite difference approximation with timestepping of 0.001s. The example rocket for this model is 144” tall, 6” in diameter, and uses an off-the-shelf 4:1 ogive nose cone. The simulation also assumes a liftoff mass of 14.6284 kg and uses the Aerotech L2200G motor as specified [13]. The ballast for passive adjustment and jettison systems obey the limitations defined in the competition guidelines, and a potential drag-producing device with assumed surface area of 0.010 m^2 at an angle of attack of 45°. Although this modelling makes many assumptions, it helps to compare the differences in control authority of different systems.

3.6.1 Passive Mass Adjustment

One possible method of altitude assurance is the addition of mass to the airframe in order to match a predicted ideal weight as determined by simulation, test flights, and environmental conditions measured prior to launch. This method has the advantage of lacking electronics or moving parts, meaning that several possible failure modes are eliminated. It also yields a reasonable level of control authority, with the simulation showing it to be able to reduce an unmodified apogee of 1550m to as low as 1450m (see Figure 3.9). However, this method requires very accurate simulations, along with numerous test flights to tune those simulations, to reach a high degree of accuracy. Additionally, the rocket's flight path cannot be corrected after launch to account for unexpected changes in atmospheric density, wind speed, or launch angle.

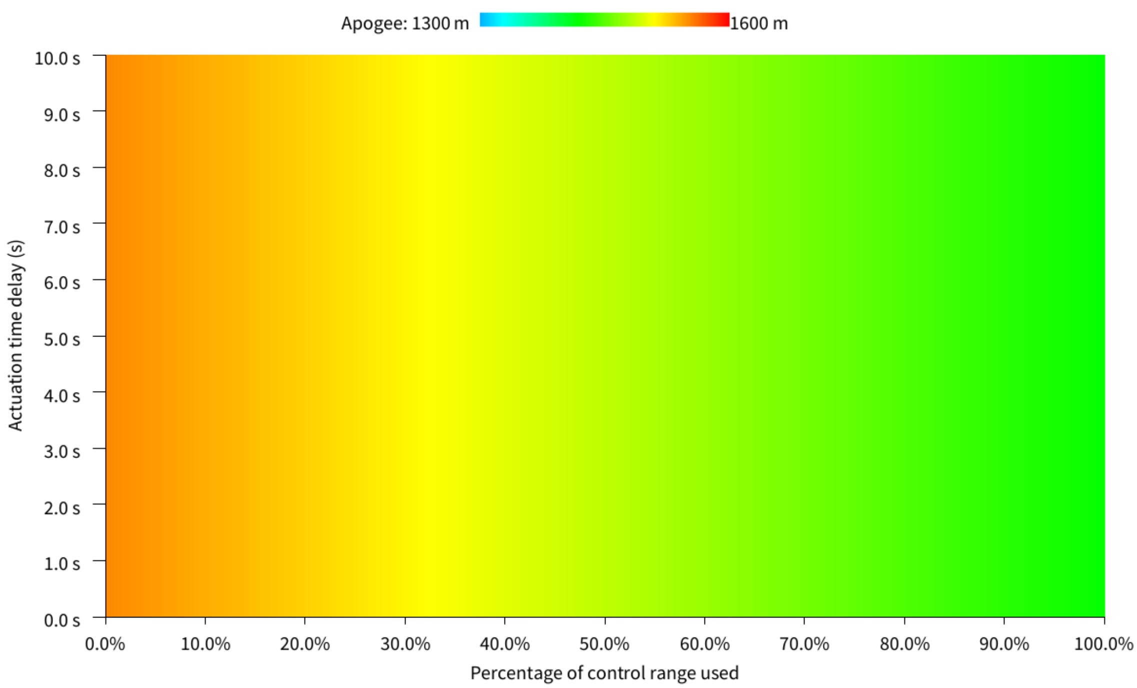


Figure 3.9: Heat map of control authority for passive ballast. Color represents apogee; "control range" is the percentage of allowable ballast used. Time axis included for comparing alternatives.

3.6.2 Jettisonable Ballast Mass

To improve the accuracy of tuned-mass methods, a liquid ballast can be used in place of solid mass, as this can be safely jettisoned during flight. This allows some degree of active control without requiring the addition of external devices. However, the overall control authority of such a method is minimal after the first few seconds of the flight, as seen in Figure 3.10, meaning that the vehicle would have little time to recover if a control error caused ballast to be incorrectly deployed or retained.

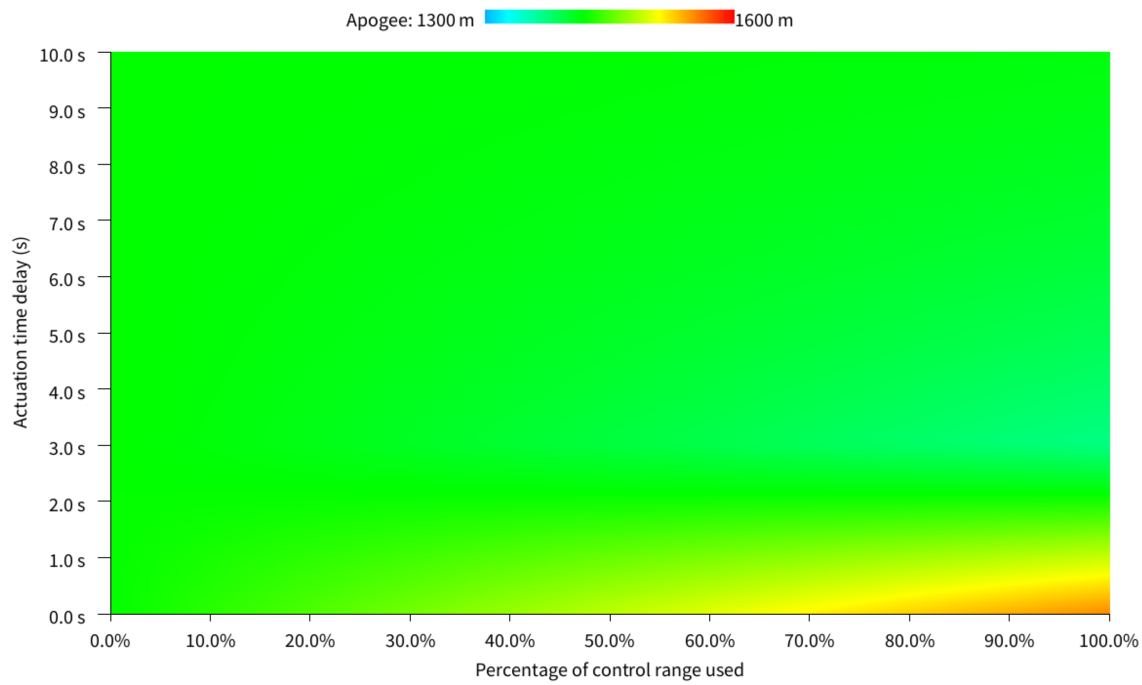


Figure 3.10: Heat map of control authority for active ballast. Color represents apogee altitude; control range represents percentage of maximum allowable ballast deployed (rocket is assumed to start with full ballast.) Time axis represents the time after launch the ballast is released.

3.6.3 Thrust Modulation

Another possible method of altitude assurance is adjustment of the motor's thrust in flight. Since a requirement of the competition is that the vehicle use commercially produced, unmodified motors, the only plausible method for achieving this is to obstruct the motor's exhaust in order to reduce its thrust. While this offers the greatest control authority if engaged initially, as shown in Figure 3.11, it is only active for the duration of the burn, it requires exotic materials to withstand the high temperatures of the exhaust stream, and presents the risk of an asymmetric failure causing thrust deflection and extreme instability. After consideration of the risks and challenges involved, the team has adopted a derived requirement that the launch vehicle will only interact with energetic devices using commercial-off-the-shelf components, disqualifying this alternative entirely.

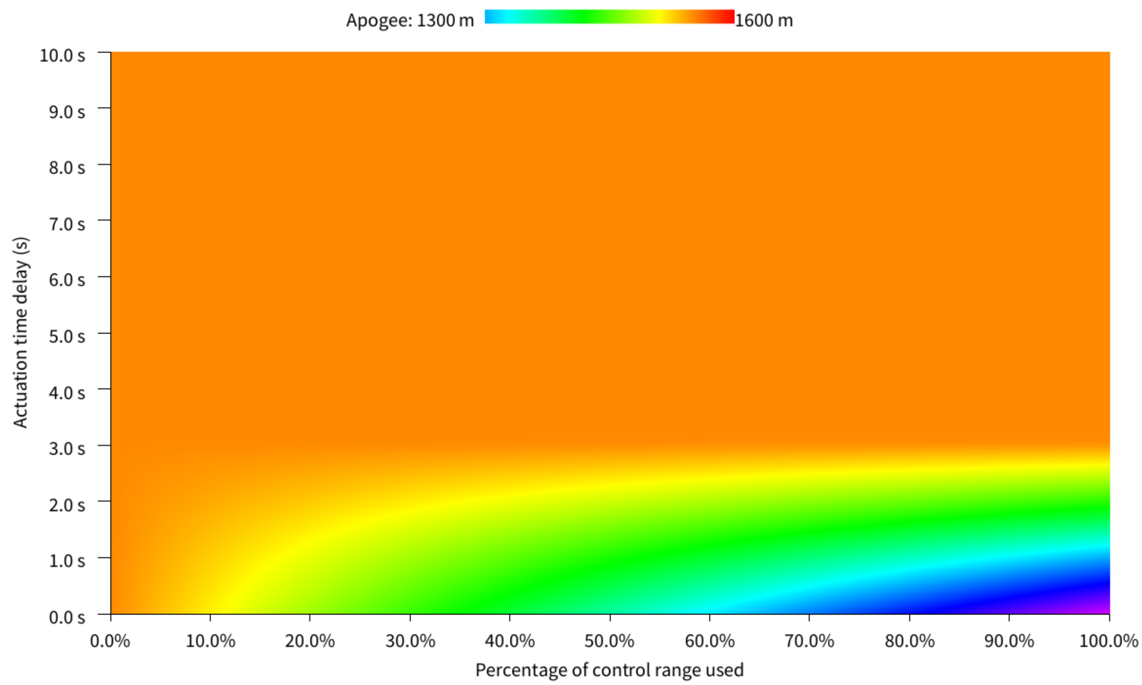


Figure 3.11: Heat map of control authority for thrust modulation. Color represents apogee altitude; control range represents percentage of thrust reduction used, with 100% on graph representing a total thrust reduction of 20%. Time axis represents the time after launch the modulation is applied.

3.6.4 Drag-Producing Devices

Another method of altitude assurance is to deploy external drag-producing flaps to slow the ascent of the launch vehicle. This method offers the greatest control authority after burnout allowing a 6.4% altitude reduction as late as 7 seconds into the flight (Figure 3.12). However, this solution is somewhat mechanically complex, and requires significant modifications to the exterior of the airframe in order to be implemented. Additionally, if not retracted properly, the flap mechanism may become tangled in the recovery harness, creating a risk of damage to the vehicle and/or unsafe landing.

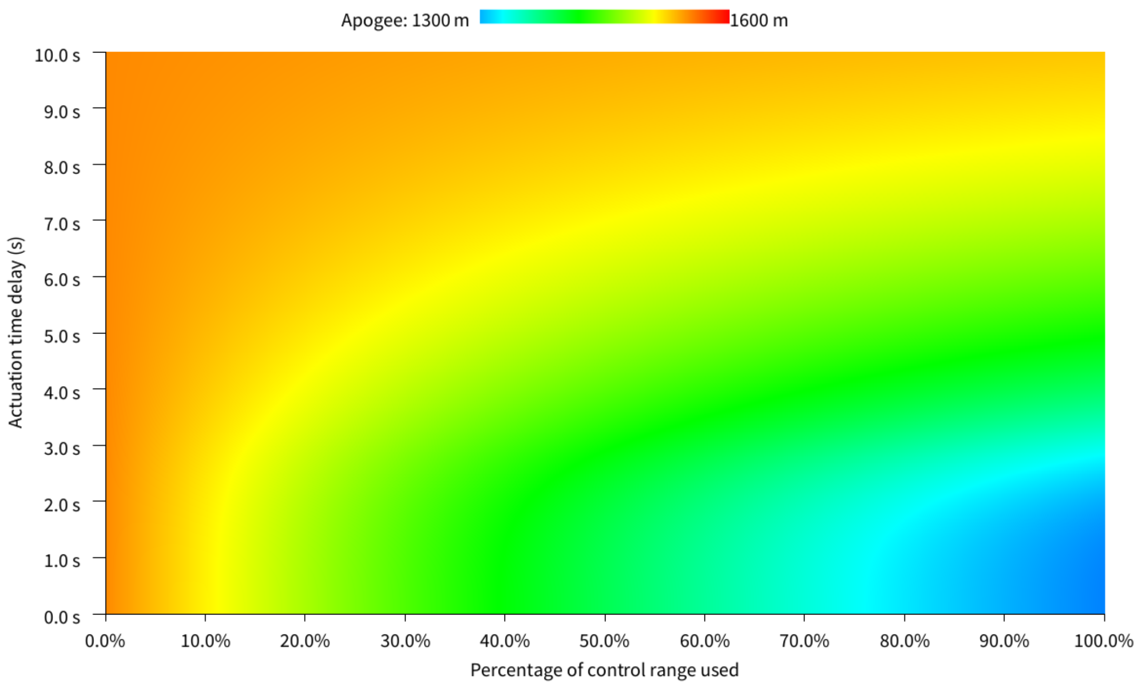


Figure 3.12: Heat map of control authority for drag flaps. Color represents apogee altitude; control range represents percentage of deployment used, to a maximum of 45 degrees for ideal flat flaps with a total area of 0.010 m^2 . Time axis represents the time after launch the flaps are deployed.

3.7 Propulsion System Design Alternatives

Table 3.7: Propulsion System Design Alternatives

Propulsion System Alternatives	Pros	Cons	
Disposable Motor System	- Quality assurance from manufacturers - Minimize risk to personnel associated with motor reloading	- Motor system is uncommon with level 2 impulse class - High cost per flight	
Reloadable Motor System	- Low cost per flight - Motors for all impulse classes are widely available	- Little team experience with motor reloading - High starting cost associated with reload hardware - Personnel risk associated with motor reloads	✔

		- Variability in team motor reloads	
Slow Burn Motor	- Smaller structural load on the airframe and vehicle components - Higher apogee	- Lower rail exit velocity - Low airspeed may result in additional drift due to "weathercocking"	
Fast Burn Motor	- Higher rail exit velocity - Greater tolerance for additional weight	- Higher max vehicle velocity - Higher stress on airframe components	✔

The objective of the Propulsion system is to provide the launch vehicle with sufficient thrust to be able to achieve a desirable range of apogees. In making decisions for the propulsion system design, structural load on the airframe and maximum vehicle velocity are considered. The Propulsion design aims to accomplish its system design goals in a reliable manner with consideration to other vehicle systems. These design decisions are made in fiscal conscience to minimize the cost contribution to Project Silverstein.

3.7.1 Disposable Motor System

The propulsion system is considered to use a disposable motor system. The advantage to disposable motors is that the quality assurance from motor manufacturers offers reliability. Using a disposable motor also minimizes the risk to personnel associated with construction of the motor. The drawbacks to disposable motors is that they tend not to be made at the Level 2 impulse class which is mandated by the competition rules [14]. Disposable motors also cost more per flight than a reloadable motor system.

3.7.2 Reloadable Motor System

A reloadable motor system is considered for the propulsion system as it offers a low cost per flight. Additionally, motor impulse classes across the range of Level 1 to Level 3 are widely available. The drawbacks to reloadable motors is that experience reloading motors is not common within the team. These motors also have a high starting cost to entry as the motor hardware is an added cost. There is personnel risk associated with reloading these motors and variability in the quality of the reload.

3.7.3 Slow Burn Motor

When deciding the level of power that the propulsion system would use, a slow burn motor is considered. The advantages to a slow burn motor is a smaller structural load is imposed on the airframe and other systems due to lower thrust [15]. An accelerometer in the payload, for example, would be less easy to saturate. The drawbacks to a slow burn motor is that a lower thrust may result in an insufficient rail exit velocity which may put the vehicle at risk of breaking the competition requirements. Additionally, a low airspeed resulting from lower thrust may exaggerate the effects of weathercocking which would steer our vehicle away from the launch pad in windy conditions [16]

3.7.4 Fast Burn Motor

When deciding the level of power that the propulsion system would use, a fast burn motor is considered. The advantages to a fast burn motor is that a high rail exit velocity due to high thrust would put the vehicle at less risk of breaking competition requirements [16]. Additionally, a fast burn motor is able to bring a larger vehicle mass up to speed. The drawbacks to a high thrust motor is that it could bring the vehicle to a very large maximum velocity during the flight. This could break competition requirements if the maximum velocity is greater than or equal to Mach 1. Additionally, the greater thrust would put more load on the airframe and other systems.

3.8 Vehicle Systems Leading Alternatives and Justifications

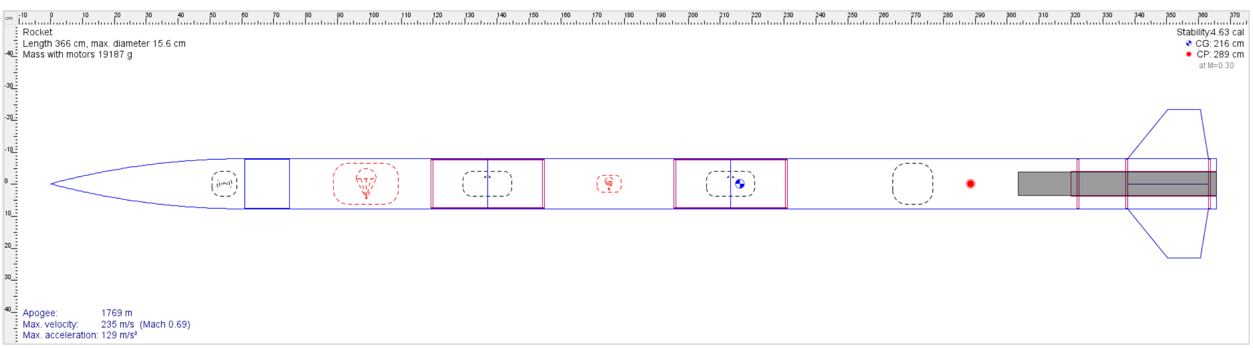


Figure 3.13: Preliminary OpenRocket Design

Table 3.8: Leading Alternatives

Vehicle Subsystem	Leading Alternative(s)
Airframe	Large-size Monocoque Airframe
Aerodynamics	Nose Cone
	Aft Fins
Deployment	Fixed Payload
	Staged Charges, Dual Separation

Altitude Assurance	Drag-Producing Devices
Propulsion	Reloadable Motor System
	Fast Burn Motor

3.8.1 Airframe Structural Design and Sizing

The leading design alternative for the Airframe Subsystem geometry is to use a cylindrical monocoque airframe of maximum derived length and diameter. The cylindrical monocoque structure was chosen primarily due to availability and simplicity. Because airframes of this type are commonly available from vendors and are purpose-built for launch vehicle airframes, variability is eliminated in construction methods and so we believe their usage in Project Silverstein would provide the mission with the highest possible outcome of success [17-19]. The team has decided to use the maximum derived sizing for the airframe since many complications of subsystem development are minimized in designing around more relaxed spatial constraints. This was deemed to be an acceptable tradeoff to drag concerns and added project costs.

3.8.2 Aerodynamics

The leading design alternatives chosen for the Aerodynamics Subsystem are use of a full nose cone and aft fins. The nose cone allows for a significant improvement to drag performance, and the decreased spatial constraints of using a blunted body are not considered substantial to the design of the vehicle. Placing fins to the aft of the vehicle allows for much more control over the stability of the vehicle in flight, and the constraints placed on ballast mass by the competition rules limit the stability that can be achieved without these aerodynamic features [20].

3.8.3 Deployment

The team has decided to use dual-point separation for the deployment system. The decision to use this type of deployment was simple: the team has no experience using a chute release mechanism required for single-point separation.

3.8.4 Altitude Assurance

The leading design alternative for the altitude assurance subsystem is utilizing drag flaps (dubbed "Rose Petals") to actively control deceleration. Specifically, the avionics system monitors its flight path via a combination of accelerometer and barometer data and compares its predicted apogee to the target apogee for the mission. If its projected altitude at any point in the flight differs from the target, it will adjust its aerodynamic drag to correct the discrepancy by deploying or retracting the Rose Petals. This method has been chosen as the leading alternative due to a large range of control authority in a variety of conditions, relative to the other alternatives considered.

3.8.5 Propulsion

The leading design alternative for the Propulsion Subsystem is to use a reloadable motor system at fast burn rate. The leading motor brand and designation is the Cesaroni Technology Inc. L2375WT-P.

3.9 Recovery Component Selection

3.9.1 Main Parachute Deployment Strategy

Table 3.9: Recovery System Design Alternatives

Recovery System Alternatives	Pros	Cons	
Tumble at Apogee then Main Deploy	- Simple design - Light weight	- Hard Main deployment - Potential tangles	
Drogue Deploy at Apogee then Main Deploy	- Low drift - Safe Main deployment	- More points of failure	✔
Streamer Deploy at Apogee then Main Deploy	- Light weight	- Hard Main deployment	

Per competition requirements, the vehicle must have a combination of two separate recovery system components, one to be deployed at apogee and the other to be deployed at a predetermined altitude during descent.

3.9.1.1 Tumble at Apogee then Main

A strategy often employed in mid-power rocketry is to simply separate the two sections of the rocket without deploying any parachutes, allowing the drag induced by the rocket's tumble to control its initial descent rate, then deploy or unfurl a large main parachute at a lower altitude. One popular solution for this is the Jolly Logic Chute Release[12] . This method allows for faster descent rates than a drogue/main or single-parachute architecture, reducing drift. However, it is difficult to ensure safe descent speeds when using this method with a larger vehicle, and the tumbling action of the components presents a risk of entanglement for the recovery hardware.

3.9.1.2 Drogue Deploy at Apogee then Main Deploy

A drogue chute and main chute deployment strategy is considered. The advantages to this design is that the vehicle is subject to low amounts of drift which gives the least possible outcome of breaking the competition maximum drift requirements. Deploying a drogue chute before a main chute also induces the smallest main chute recovery loads in comparison to all other competition-legal recovery strategies. The drawbacks to this design is that there are more points of failure associated with deploying two parachutes.

3.9.1.3 Streamer Deploy at Apogee then Main Deploy

A halfway point between drogue and drogueless dual-deploy strategies, a recovery method that has gained traction recently in high-power rocketry is to use a flat streamer as an initial drag device. This allows a fast descent rate while keeping the rocket in a stable orientation to avoid tangling. However, these strategies are fairly new to large rockets, and as a result, little literature is available on their application to heavy payloads, and much of the existing literature on small streamers is not accurate for larger ones [21]

3.9.2 Parachute Protection

Parachute Protection Alternative	Pros	Cons	
Chute Protector	Fewer components Analogous to smaller mid- and high-power rockets	Risk of tangling	
Deployment Bag	Reduced risk of tangling Standard in larger rockets	More complexity required	✔

3.9.3 Recovery Harness

Recovery Harness Alternative	Pros	Cons	
Nylon	- Elastic - Low Cost	- Subject to melting and charring	
Kevlar	- Strong - Burn resistant	- Inelastic - Higher cost	✔
Nylon/Kevlar	- Strong harness with	- Added point of failure in	

Dual Harness	elastic give - Section is resistant to melting and charring	Nylon/Kevlar knot	

3.9.3.1 Nylon

Nylon as a recovery harness material has the advantage of being more elastic [22] [23]. This allows the shock loading on structural components in the vehicle to be spread over a longer period of time and so the overall force to structural components is minimized [24] In addition, nylon makes for a lower cost option in comparison to kevlar [25]. The drawbacks to using nylon is that it is subject to charring and melting from deployment charges or other hot gasses [11]. As a result, an added failure mode exists in damaging the shock cord.

3.9.3.2 Kevlar

Kevlar is considered as a recovery harness material as it has a much higher tensile strength in comparison to Nylon [26] [27]. Because of this, kevlar is able to handle higher shock loading without failing. Kevlar is also burn resistant and so a failure mode in the shock cord due to hot gasses is removed entirely [28].

3.9.3.3 Nylon/Kevlar Dual Harness

In combining the best of each material, a Nylon/Kevlar dual harness is considered. This would have the benefit of providing an elastic shock cord that has a burn resistant section. In turn, this allows for burn failure mitigation for which the RR-SL team has experience using. The drawbacks to this combination is that an additional point of failure exists in the knot that interfaces the Nylon and Kevlar sections.

3.9.4 Altimeter Selection

In selecting an altimeter choice for the recovery system, it was decided that two altimeters were to be selected in order to promote dissimilar redundancy within the system. Additionally, the altimeter selected must be capable of dual-deployment as the competition rules prohibit the deployment of a single parachute. The altimeters must also not require the use of an FCC HAM radio license as there are no members on the RR-SL team that are licensed in HAM radio operation. Below are alternatives that meet these requirements from the Apogee Components vendor. Altimeters were selected from a singular vendor to streamline the final selection process.

Table 3.10: Altimeter Selection Alternatives

Altimeter	Pros	Cons	
Missileworks RRC3	- Low cost - Commonly used in high-power rocketry - Two-ended terminal blocks allow neater avionics assembly	- Low level of customizability	✔
Altus Metrum EasyMini	- Feature set suited for vehicle	- Low availability	✔
PerfectFlite StratoLogger CF	- Low cost option - Feature set suited for vehicle - Addable telemetry	- Low availability - Low level of customizability - Records at low sample rate	
Entacore AIM 3	- Mach delay avoids premature deployment in supersonic flight	- Records at low sample rate - High cost	

Due to low cost and prior team experience, the MissileWorks RRC3 and AltusMetrum Easy Mini were chosen to be the leading altimeter alternatives.

3.9.5 Battery Selection

Table 3.11: Battery Selection Alternatives

Power Delivery Design	Pros	Cons	
Disposable 9V?Cells	- Durable - Highest Energy Density	- Requires regular battery replacement - Additional mounting and securing considerations	
3.7v Lithium Polymer Battery	- High Energy Density - Use in high-power rocketry well documented	- Lower Cycle Count - Susceptible to critical burst failure - Dangerous if not handled correctly	✔

18650 Cell	- Durable - No Risk of Charge Imbalance	- Additional mounting and securing considerations	
Hybrid Supercapacitor	- Extremely Safe - Rapid Charging	- Unproven technology - Requires testing prior to implementation	

3.9.5.1 Disposable AA Cells

Disposable battery cells are standard equipment on high-powered rockets [29]. These batteries are widely-available and have higher practical energy density than rechargeable cells, as well as a longer shelf life. Additionally, the metallic casing of this type of battery is resistant to puncture, making it safer than prismatic rechargeable batteries. Another positive aspect of a disposable cell is that there is no recharging procedure; lack of appropriate charge can be a silent failure that causes critical mission failure. However, there are two major downsides of this type of battery. The first is that it requires replacement every flight, which presents a unique failure risk for the deployment system. The second is that the rigid cells require spring contact mounting, which is susceptible to momentary disconnection due to vibration, or, in extreme cases, failure of the mount and loss of power. This can be somewhat mitigated by the use of a clamshell mounting, but it is still an unnecessary risk for a mission-critical system [30].

3.9.5.2 3.7v Lipo Battery

Lithium-Polymer Batteries are commonly used in hobby radio-controlled vehicles and are standard equipment on high-powered rockets [29][31]. These kinds of batteries are widely-available, and are popular due to their high volumetric and mass energy density. The biggest downside to LiPo cells is their safety; if punctured, they can catch fire. The battery will also degrade if not charged or discharged properly, and could rupture due to improper charging.

3.9.5.3 18650 Cell

Rechargeable cylindrical cells present all of the benefits and drawbacks of disposable cells, except that they have somewhat shorter shelf life and also introduce the possible mischarging failure present in LiPo batteries [32]. When compared to prismatic LiPo cells, cylindrical lithium-ion batteries have higher mass energy density but lower volumetric energy density [33]. When compared to disposable cells, 18650 cells have a unique benefit in that they are intended for higher-power applications, and more robust mounting mechanisms are available that can maintain electrical operation under significantly higher shock loadings [34].

3.9.5.4 Hybrid Supercapacitor

Hybrid supercapacitors are an emerging technology that combines the behavior and benefits of both batteries and supercapacitors [35]. Hybrid supercapacitor technology is a theoretically ideal energy storage device for atmospheric rocket launches, because energy density is close to batteries, but power density is orders of magnitude greater [36][37]. The high current capacity of this type of energy storage relative to the amount of energy stored means that it does not have to be oversized in order to deliver reliable peak power, which could potentially save a significant amount of weight. Most importantly, hybrid supercapacitors produce much less heat than batteries when discharged, and have a very low risk of fire, even when used near energetic devices [38].

However, this kind of energy storage is extremely new. Eaton is one of the few companies with commercial hybrid supercapacitor offerings, and there are only 10 total parts offered, representing two lines of 5 capacities [39]. This product line was only introduced in 2020, and no information is readily available on industry applications. Eaton does not publish discharge curves, and so the cells would need to be tested as part of the electronics development, which presents added design complexity and an additional risk of hardware failure.

3.9.6 Summary of Design Redundancy

Redundancy in the recovery design exists in dissimilar redundancy of two different altimeter selections. Each altimeter will excite their own sets of deployment charges: one for the drogue chute and one for the main chute.

3.9.7 Main and Drogue Chute Sizing

Through iterative simulation using OpenRocket, it was determined that the SkyAngle CERT-3 Large would be sufficient to recover the vehicle. Similarly, iterative simulation using OpenRocket had determined that the Rocketman 7ft. Pro Experimental Chute would be sufficient to slow the vehicle in descent.

3.9.8 Recovery System Leading Alternatives

Table 3.12: Leading Alternatives

Component	Leading Alternative
Altimeters	MissileWorks RRC3 and Altus Metrum EasyMini
Recovery System	Drogue Deploy at Apogee then Main Deploy
Recovery Harness	Kevlar Harness
Power Delivery	3.7v Lithium Polymer Battery

3.9.8.1 Recovery Harness

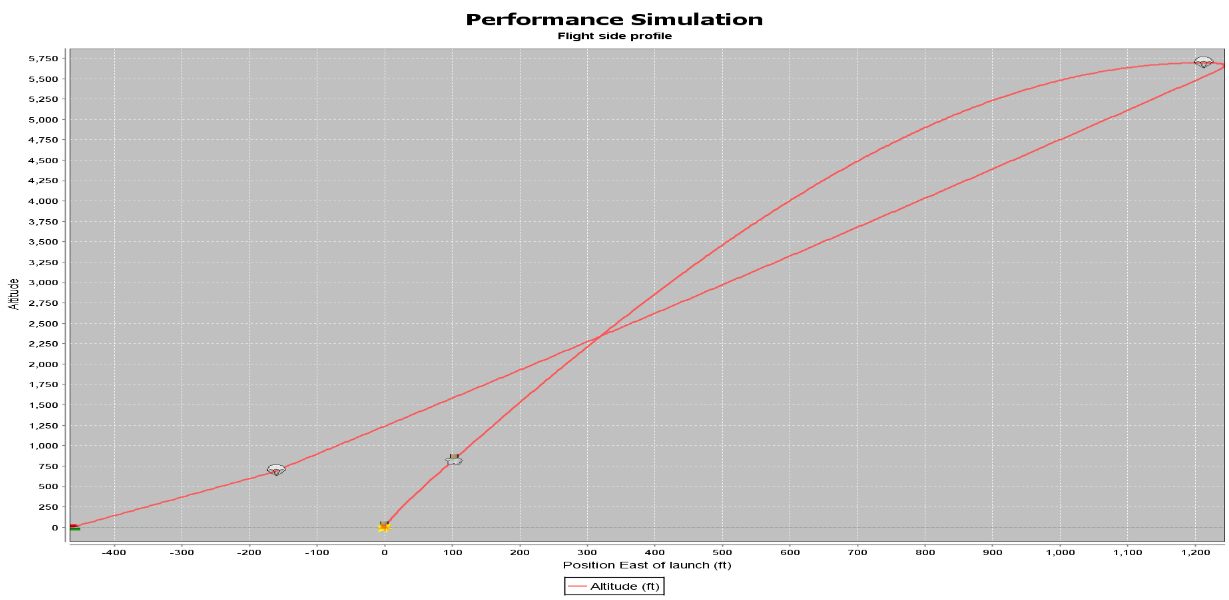
The team has decided to use Kevlar for a recovery harness. Where risk mitigation is paramount, Kevlar can minimize failure modes in the recovery process with its burn resistant properties and high tensile strength. The inelastic properties can be mitigated using various techniques to deliberately prolong the harness deployment such as using bundling the cord in sections [11]. The added cost of a Kevlar harness was deemed to be an acceptable tradeoff for the failure mode mitigation offered.

3.10 Mission Performance Predictions

3.10.1 Summary of Mission Performance Calculations

Official Target Competition Launch Altitude	5000 ft.
Landing Kinetic Energies	Section 1: 281.1ft-lbs Section 2: 652.7 ft-lbs Section 3: 554.3 ft-lbs
Expected Descent Time	202 s
Expected Maximum Drift	1740 ft.

3.10.2 Flight Profile Simulation



4. Payload Criteria

4.1 Experiment and Criteria for success

The vehicle payload will determine the location of its landing site. The launch field will be divided into a grid of square boxes 50m (164 ft) on an edge. The payload will determine which box the vehicle landed in, thereby exceeding the competition requirements. Our mission will be considered successful when, upon landing, our payload autonomously returns the correct (verified with GPS) gridded position of its landing location to the team. This operation should be performed independently of a successful landing.

4.2 Overview of Payload Systems

Similar to the vehicle design process, the payload has been divided into individual subsystems with a specific purpose. Listed below are the payload subsystems and their objectives.

Table 4.1: Payload Systems Summary

Payload Systems	Objective
Control System	Process the data collected from the Sensor and Data Acquisition system, determine the location of the payload, and transmit that location to the ground station via the telemetry system.
Telemetry	Report back the payload's position determined by the Control System, along with other useful data for logging and verification purposes, to the team's ground station.
Sensor and Data Acquisition	Collect all information required to determine the payload's position. Store all data for post-flight validation and mission statistics.
Power Delivery	Store and deliver power to all other subsystems within the payload.

4.3 Control System Design

Table 4.2: Control Systems Alternatives

Control System Design	Pros	Cons	
Embedded Microcontroller	- Proven within other competition teams - Flexibility and Robust GPIO Support - Wide Industry Adoption	- Limited Computation - Highly Specialized Programming - Low level software	
Discrete Electronics	- Speed - No sampling of analog signals - No Toolchain	- Complex design - Relatively inflexible - Hard to debug - Larger footprint	
Embedded Linux	- Linux tools - Performance - High Level Languages - Native Development - Code Portability - Prior Experience	- Size - High Power Draw - Complex Software	✔
No Control System	- Simple - Reliable - Low-Cost	- Sensors must directly determine position	

4.3.1 Embedded Controller

One design alternative considered for the Control System is an embedded microcontroller. This design implementation makes use of proven technology used in a variety of industrial sectors to control everything from car functionality and medical equipment to aircraft control [40]. In addition to being proven in industry, other teams within Rose-Hulman have current working and successful implementations of embedded microcontroller technology in their competition vehicles, namely the Grand Prix Engineering team and Combat Robotics. Using a microcontroller provides the team with a wide range of general-purpose I/O (GPIO) and peripheral interfaces [41]. to interact with the other payload systems. However, programming an embedded microcontroller requires moderate prerequisite knowledge in low level programming and device hardware, creating a potential barrier to some team members. Lastly, due to their more limited memory and processing power, the utilization of a microcontroller could create constraints on payload locating strategies.

4.3.2 Discrete Electronics

Discrete electronic circuits offer a speed advantage when compared to other control logic implementations. This speed is demonstrated in devices like the HCTL-2020 quadrature decoder [42]. In addition to speed, discrete electronics remove the need for a software toolchain (a compiler, linker, software development kit, etc). And, they remove the delay and approximations created by converting analog signals to digital signals to be processed by a microcontroller or microprocessor. But, these benefits come at the cost of significantly increasing both development time and complexity. Additionally, our team does not have any prior experience with designing discrete digital logic.

4.3.3 Embedded Linux

One control system considered is the use of an embedded linux computer, such as a Raspberry Pi or a Beaglebone Black computer. An embedded linux board provides the team with the benefit of the linux operating system a vast ecosystem of open source software and tools written for it. Complementing this ecosystem of software tools, linux allows the team to use higher level programming languages, such as python, typically unavailable on embedded microcontrollers. This is attractive because it allows the team to take advantage of existing high level signal processing libraries, such as Scipi [43] and run GNURadio natively on the control system hardware [43], a possibly useful development tool. Additionally, all programming and development of the payload control system can be completed natively on the control system hardware. The combination of native development and high level programming languages dramatically lowers the barrier of entry to team members wishing to contribute to this system. However, programming in high level languages with multiple software tools interacting adds complexity to the software architecture. And, embedded linux boards, due to their more powerful processors and computational ability, require larger power delivery and storage systems. Additionally, embedded linux boards are typically sold only as a completed board, not a single integrated circuit (IC), possibly increasing the footprint of the control system.

4.3.4 No Control System

The design alternative of no control system is advantageous because it would completely remove a system from the payload. With the absence of this system, many failure modes are eliminated, development resources can be reacoleted to other systems, and the overall payload is simplified. However, with no control system, the Sensor and Data Acquisition system must directly measure the payload's location, violating the spirit of the competition.

4.4 Telemetry Design

Table 4.3: Telemetry Alternatives

Telemetry Design	Pros	Cons	
Discrete Component Radio Frequency (RF) Circuit	- Mechanical robustness - No software failure	- Relatively inflexible - Difficult to test - Requires extensive background knowledge	
Commercial Off the Shelf (COTS) Module	- Easy to develop and use -Proven technology -Experience within the team -Smaller Footprint	-Software Failure	✔

4.4.1 Discrete Component RF Circuit

Similar to building a discrete digital control system, building a discrete component RF circuit offers a robustness advantage over digital signal processing (DSP) techniques. This is a result of the system design containing little to no fragile ICs and mainly passive circuit components. Additionally, software failure is completely removed from the system. However, similar to a discrete digital system, designing and building a discrete RF circuit would require extensive prerequisite knowledge and create a barrier of entry to many students on the team. (DSP First, mark yoder). Additionally, this circuit would be difficult to test with the tools currently available to the team and not be flexible to any mission changes.

4.4.2 COTS Module

Using a commercial off the shelf (COTS) RF module provides the team with an easy to use and proven platform to transmit payload data back to the ground station. This is an effect of the COTS modules handling all of the RF circuitry and processing, requiring the team to only send serial data to the module's onboard computer in order to receive the data back at the ground statio [44]. Additionally, the team has experience using COTS modules and many are well documented [45]. Although, the team will need to dedicate time to developing robust software. And, this introduces a point of failure during the mission.

4.5 Sensor and Data Acquisition

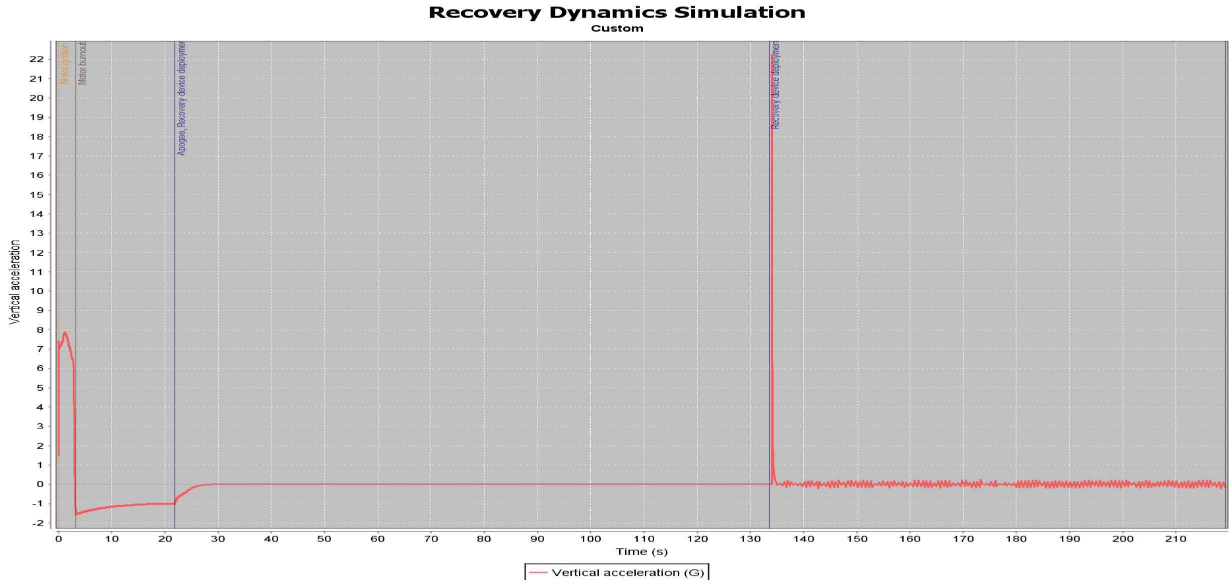
To fulfill both explicit and implicit requirements, sensing method alternatives were limited to technologies that could either be implemented on deep-space probes, or are currently in use. For example, the Deep Space Network handles all communication with science instruments above Low Earth Orbit (LEO) [46]. The DSN is made up of 3 locations on Earth separated by approximately 120 degrees. Each location is made up of a high-gain, parabolic, steerable antenna. At any point, a DSN location will be pointed at a given probe. The DSN can determine location and velocity. The distance to the spacecraft is found using a variation of Time-of-Flight, and velocity is determined by measuring the Doppler Effect [47]. Since the goal of the competition is to mimic communication with another planet, the methods that are currently used to do so were considered.

Sensor and Data Acquisition Design	Pros	Cons
Time Integration	- No equipment external to the vehicle is required - Easy to compute - Easy to test	- Complex signal processing required to filter data - Possibly large amounts of error
Radar	- No additional hardware on rocket	- Not in spirit of competition
Passive Rf Field	- Accurate - Significant prior art - No error accumulation	- Large hardware investment - Computationally complex - Needs hardware at ground station to work
Visual Simultaneous Localization And Mapping (vSLAM)	- Can correct accumulated error - No external hardware is needed	- Complex signal processing - Computationally expensive

4.5.1 Time Integration

In this method, the acceleration of the vehicle is measured with an Inertial Measurement Unit, and the acceleration is double integrated with respect to time to determine the position as the vehicle moves, relative to its starting position [48]. This method is useful because it is computationally easy to calculate the position. The main drawback to this method is that error can accumulate as it is integrated twice, due to noise with each measurement. This error accumulates because the algorithm can only find the position relative to the position at the previous measurement [48]This is of particular concern during parachute deployment, as the acceleration at this point is high and can lead to accelerometer saturation, wherein the acceleration is larger than the maximum recordable value of the accelerometer. Thus, the magnitude of the acceleration will be under-recorded, which will lead to incorrect estimation of velocity, and thus extreme error in final position. This can be demonstrated with a simulated recovery event. Using

OpenRocket, a simulation of a similarly scaled launch vehicle's acceleration over the course of a flight is shown below.



The simulated flight recovery is similar to the recovery plan of the Project Silverstein launch vehicle wherein a drogue chute deploys at apogee and the main chute deploys at a safe altitude. From the results above, the forces of recovery induce a large acceleration of around 22G at the main chute deployment. This level of acceleration can saturate a high resolution, low range accelerometer used in time-integration analysis.

4.5.2 Radar

Radar works (usually) by using one transmitter and receiver at the same location; a signal sent out will be reflected by the object of interest and the transmitter will receive it. The angle and distance to the rocket could be found using this method. Radar is currently in use to detect space debris, but there is no documented use of it to determine location of deep space objects [49]. State-of-the-art radar tracking systems are capable of resolving meter-scale objects in geostationary orbit which is one order of magnitude closer than the moon, and three orders of magnitude closer than the minimum distance to mars [50]. With limited prior art, it is unlikely that radar would work to determine location from earth to a probe intended to land on another celestial body. While this method would be theoretically viable for the payload mission, it would not be in the spirit of the competition, which is a disqualifying drawback.

4.5.3 RF passive

The following section is a discussion of our considerations in the passive RF system. We identified two components of this system: angle determination and distance determination. These can be performed separately or simultaneously. The angle and distance measurement will be combined to find the location of the payload on the gridded launch site image.

4.5.3.1 Finding Angle in RF Passive system

- Rotating Antenna

We can mount a swiveling directional antenna in the ground station, and rotate the antenna while transmitting a signal to the payload. The payload will receive the strongest signal when it is aimed directly at the current location of the rocket. With the direction from the ground station obtained from this, and the distance known from one of the other methods suggested above, we can determine the exact location of the rocket relative to the ground station.

- Phased Array Antenna

Phased array antennas are made of an array of antenna elements which can be controlled by shifting the phase difference between the elements. The signals produced by the antenna elements interfere to produce a directional signal which can be electronically steered by shifting the phase of the inputs into the elements. This can be used with the same fundamental principles of the rotating directional antenna method to sweep through an area and determine the direction where the signal is the strongest, and the payload will be in that direction. Because the direction is controlled electronically, the angle can be measured faster and more precisely. The downside is that the phased array antenna requires additional electrical hardware which increases complexity and price [51]

4.5.3.2 Finding Distance in RF Passive System

- Frequency (Fourier Transform)

If we transmit a signal that changes frequency linearly over time, then we can measure the time of flight from the difference in frequency between the signal being transmitted and the signal being received. When a signal is sent to the payload, retransmitted by the payload, and travels back to the ground station, there is a delay that is proportional to the distance between the station and payload plus the delay of the retransmitter. Since the frequency of the signal is increasing linearly, we can compare the signal being generated at the ground station to the signal being at the ground station from the payload and the difference in frequency will be constant and proportional to the delay. To easily measure the difference in frequency of these high-frequency signals, we can input the generated signal and the received signal into an RF mixer cascaded with a low-pass filter to produce a signal whose frequency is equal to the difference in frequencies. This method could make it easier to accurately measure the short time it takes for the signal to travel between the payload and ground station. There may be difficulties if the delay of the retransmitter is significant, or the retransmitter causes a feedback loop with itself.

- Clock-Synchronized Time of Flight

The process of the Time of Flight method is to synchronize two clocks between a source signal and a remote receiver, and then measure the time it takes for the signal to travel from the source to the remote location. This competition would use this method by having the rocket send out a signal (after already knowing the angle to the ground station) and measuring the time it would take for the ground station to receive and retransmit it. The time will be directly proportional to the distance. One problem with this method is that, for the purposes of the competition, the time between transmission and receiving of the signal, so our measuring devices must be precise.

4.5.3.3 Discussion of Possible Systems

In this subsection, two more methods are discussed that are similar to, or build on systems discussed in 4.5.3.1 and 4.5.3.2.

- Coordinated Clock Angle Sweep

Two clocks, one at the ground station, and one on the rocket, are synced. A rotating antenna at the ground station sends out an RF signal containing the time of transmission, and the angle of the rotating antenna at the time of transmission, at regular time intervals. The payload of the rocket will have a receiving antenna which records the time stamp when a signal is received. Using the time difference between the time of receiving the signal, and the time the signal was sent, the distance between the ground station and the payload is calculated by multiplying the time difference by the speed of light. The angle from the ground station on which the payload is located, is received with the signal. Thus, the exact position of the payload is ascertained by the payload. The problem with this method is that highly precise clocks in sync will be required, which can be very expensive.

- Trilateration

Trilateration works through using three locating-sites that each determine the distance to an object. This is the same way that GPS works; 3-4 GPS satellites each determine the distance to the object using coded frequencies. By finding the one intersection point of all the spheres, GPS can find the location of the object.
We would use trilateration by using three separate antennas within our ground station that could each determine the distance to the object by either measuring time or frequency. With three separate distances and therefore 3 circles, we could determine the location of the rocket. It would be fast as no additional communication would be needed to determine angle or distance, but the distances between the antenna would be small since our ground station is small which would require precise instrumentation.

In space, trilateration would need 4 locations to locate (since they will all be spheres). The competition encourages no existing technologies on the planet, so all the stations would have to be on earth or around earth. There is no prior art for locating probes on celestial bodies using trilateration, but it is possible that the large distances would cause a large amount of computation to find the intersection of the spheres.

4.5.4 Visual Simultaneous Localization And Mapping

The vehicle can track its position during flight using images taken from an onboard camera using a Simultaneous Localization And Mapping algorithm. The algorithm would identify features in the captured images, match features between images, determine the relative positions between where the images were taken, and construct a map of feature points along with the vehicle's trajectory through the map. This method is advantageous because the locating system is self-contained without a need to communicate with the ground station. It can also compare data taken at the end of the flight directly with data at the beginning of the flight to circumvent accumulated errors in intermediate data points. This method may encounter issues because it requires real-time video processing, which is highly computationally expensive.

4.6 Power Delivery

The goal of the payload power delivery subsystem is to store and deliver power to all other subsystems within the payload, independently of successful operation of the rocket. The payload power system must be isolated from the rest of the power system, so that electrical failure of the payload will not affect the other critical electronics, such as recovery and altitude logging.

Power Delivery Design	Pros	Cons	
Disposable Cell	Durable Highest Energy Density Use in high-power rocketry well documented	Requires regular battery replacement Additional mounting and securing considerations	
Rechargeable Battery	High Energy Density Use in high-power rocketry well documented	- Lower Cycle Count - Susceptible to critical burst failure - Dangerous if not handled correctly	✔
Hybrid Supercapacitor	Low fire risk Rapid charging Extremely high power density	Unproven technology Poor charge stability compared to batteries	

4.6.1 Disposable Cell

As mentioned in Section 3.9.6, disposable battery cells are standard equipment on high-powered rockets [29]. From a systems design perspective, disposable cells have the highest available energy density of any battery technology and do not need to be recharged. By establishing a single-use-and-disposal procedure, undercharged battery risks can be efficiently mitigated at the expense of added component waste. However, this requires additional maintenance procedures that are undesirable for the reliability of the system

4.6.2 Rechargeable Batteries

As mentioned in Section 3.9.6, rechargeable Lithium-ion and lithium-polymer batteries are commonly used in hobby radio-controlled vehicles and are standard equipment on high-powered rockets [29]. These kinds of batteries are widely-available, and are popular due to their high volumetric and mass energy density. However, these batteries have a shorter shelf life and less energy density than disposable battery cells [32]. Rechargeable batteries also introduce the possibility of mischarging failure, which can lead to damage, explosion, and battery fire. Additionally, the use of rechargeable batteries necessitates developing a charging and charge verification procedure, and charging equipment must be maintained as part of launch operations.

4.6.3 Hybrid Supercapacitor

Again, as initially mentioned in Section 3.9.6, hybrid supercapacitors are a new class of energy storage that is theoretically ideal for rocket launches due to good energy density and high current delivery at low stored energy [37]. Hybrid supercapacitors have low shelf stability but charge orders of magnitude more quickly than batteries, and so the maintenance procedure would simply be to charge the system directly prior to launch operations, which has fewer failure modes than alternatives, as it is much more difficult to place the system in a state of partial charge. However, this technology is so new that offerings are limited, and datasheets do not have a full set of information that would be critical for application [52]. Additionally, without any test data, it is unclear if these supercapacitors are stable enough to fulfill the competition requirement that the vehicle can launch successfully after 2 hours on the launchpad.

4.7 Preliminary Payload Design

4.7.1 Leading Control System Alternative

The team has chosen an embedded linux computer as our leading payload control system. No control system was quickly ruled out due to it being overly restrictive on the payload design, requiring the sensors to directly measure position. This is something the team could not find a viable solution to without going against competition rules and using GPS.

A control system designed using discrete electronics was ruled out due to its complexity and extensive prerequisite knowledge required. An embedded microcontroller was considered due to their real world uses, extensive GPIO, and readily available support from other Rose-Hulman competition teams. However, the team forseas ourselves utilizing complex signal processing algorithms that would be better carried out in a high level programming language unavailable on embedded microcontrollers. Additionally, the low barrier to entry provided by being able to develop natively on the embedded linux board is something we value, especially as a rookie team.

4.7.2 Leading Telemetry Alternative

To transmit the data collected and processed by the Control System, the current leading alternative telemetry system is a commercial off the shelf radio transmitter. This COTS module was chosen for its simplicity, reliability, ease of use, and prior team experience.

4.7.3 Leading Sensor and Data Acquisition Alternative

For location data acquisition and computation, two methods will be used instead of one. This is done to gain a more accurate location, as well as a method to achieve mission goals in case of a critical failure of one of the payload sensor systems. In this payload, the two systems being deployed are the passive RF system and time integration method.

The time integration system has been selected as its major drawbacks discussed in 4.5.1 can be solved using one of two feasible solutions. The first is using two accelerometers, one with a high range of acceleration magnitude measurement, and one with a low range. For the majority of the flight, the low range accelerometer is used. Prior to parachute deployment, data acquisition is switched to the high range accelerometer which does not get saturated by the acceleration induced, and switches back to the higher sensitivity accelerometer when the magnitude of acceleration stabilizes.

The second solution is to use mechanical shock absorbers to dampen the acceleration measured by the IMU and prevent saturation.

The RF passive system has been selected as it will be able to correct drift in the IMU. We plan to measure the distance with a clock synchronized Time Of Flight method and the angle with the rotating directional antenna. We have selected the time of flight method because it directly measures the distance between the ground station and the vehicle, so it can correct the bias that accumulates from integrating the IMU measurements. The rotating directional antenna has been chosen because it is simpler and more cost effective than the phased array antenna for transmission.

If each of these systems yields different values of calculated position, the team will use trajectory prediction to determine which method produces the more reasonable result.

4.7.4 Leading Power Delivery System Alternative

The leading design alternative for the power delivery subsystem is to use a rechargeable LiPo battery, which is a standard rechargeable battery used for high-powered rocketry. This was selected over disposable cells because it was felt that recharging a battery prior to launch operations was a preferable maintenance operation to replacing disposable batteries. Battery charging can be performed without fully removing the payload subsystem from its location in the nose cone, while replacement of disposable batteries would be more invasive. Hybrid supercapacitors would likely be the preferred option due to their weight efficiency, space efficiency, preferable charging characteristics, and low fire risk, but the novelty of the technology is a massive downside.

5. Safety

5.1 Safety Officer

The Safety Officer for this year's competition season is Donald Hau and the responsibilities include:

a. Be familiar with all BIC/KIC, Rose-Hulman, NASA USLI, and team safety policies.
b. Enforce all BIC/KIC, Rose-Hulman, NASA, and team safety policies.
c. Monitoring team activities with an emphasis on safety during:
    i. Design of vehicle and payload
    ii. Construction of vehicle and payload components
    iii. Assembly of vehicle and payload
    iv. Ground testing of vehicle and payload
    v. Subscale launch test(s)
    vi. Full-scale launch test(s)
    vii. Launch Day
    viii. Recovery activities
    ix. STEM Engagement Activities
d. Assist in the writing and development of the team's hazard analyses, failure modes analysis, and procedures.
e. Manage and maintain current revisions of the team's hazard analyses, failure modes analysis, procedures, and MSDS/chemical inventory data.

Table 5.1: Probability of Event

Category	Value	Description
Improbable	1	Less than 10% chance
Unlikely	2	10-35% chance
Possible	3	35-65% chance
Likely	4	65-90% chance
Probable	5	Greater than 90% chance

Table 5.2: Severity of Event

Category	Value	Human Impact	Equipment Impact	Mission Impact
Negligible	1	Minor or none	Minor or none	No disruption
Marginal	2	Minor injury	Minor damage	Proceed with caution
Moderate	3	Moderate injury	Repairable equipment failure	Flight delayed until event resolved
Critical	4	Serious injury	Partially irreparable equipment failure	Flight does not proceed until system removed
Catastrophic	5	Life threatening or debilitating injuries	Failure resulting in total loss of system or equipment	Flight canceled or destroyed

Table 5.3: Mapped Risk Assessment Matrix

Category	Negligible	Marginal	Moderate	Critical	Catastrophic
Improbable	1	2	3	4	5
Unlikely	2	4	6	8	10
Possible	3	6	9	12	15
Likely	4	8	12	16	20
Probable	5	10	15	20	25

5.2 Personnel Hazard Analysis

Identified Hazard	Causes	Effects	Mitigations
Fire	- Open flames - Mishandling of equipment - Improper wiring	- Severe burns - Loss of part or project - Death	- Store flammable substances in flammables cabinet, fire extinguisher placed nearby, no open flames, test circuitry before use
Airborne particle exposure	- Sanding dust - Metal shavings - Paint - Aerosols	- Skin laceration or irritation - Eye damage - Respiratory distress	- Proper use of PPE and safety training, use paint booth and ventilated workspace where necessary
Electric Shock	- Improper wiring - Device failure - Test equipment misuse	- Extreme personal injury - Hardware damage/loss - Mission delays	- Members will not work alone and will be trained on use of high-voltage electrical equipment
Entanglement with machines	- Improper use of machinery - Machinery failure	- Severe lacerations - Crushed limbs - Fatal injuries	- Use PPE, follow dress codes in machine shops, adhere to required safety training
Epoxy Contact	- Surface contamination - Broken PPE - Resin spill	- Skin irritation - Eye irritation - Epoxy sensitivity	- Discard broken PPE, limit exposure, wear proper PPE, limit use to specified working surfaces
Eye Irritants	- Solder and epoxy fumes - Flying debris - Airborne particles	- Possible temporary vision loss - Eye irritation - Blindness	- Wear proper PPE, document irritants and limit exposure, use workspace ventilation booth,

			locate and train on use of eyewash station for every team member
Falling tools or materials	- Mounting failure - Improper use of storage racks	- Tool damage - Storage rack damage - Personal injury	- Store frequently used tools in easy to access locations, adhere to 5S standards of lean production
Fiberglass Contact	- Airborne particles created during fabrication - Fiberglass skin irritation	- Skin irritation - Respiratory Issues - Splinters	- Wear N95 respirators during fabrication, only sand fiberglass in sanding booth
Flying debris	- Improper use of machinery - Machinery failure	- Blunt force trauma - lacerations	- Maintain a safe distance from machines under operations, ensure those working on machinery are properly certified by the BIC
Exposure to Hazardous Fumes	- Working with inadequate ventilation - Improper soldering and welding practices - Epoxy handling - Activities from other teams in shared workspace	- Eye irritation/damage - Lung irritation/damage - Lightheadedness - Shortness of breath and nausea - Possible nerve damage	- Maintain proper PPE when working with fuming materials or maintain a safe distance from fuming materials in a well-ventilated environment
Hazardous Waste Contact	- Chemical spills - Incidental contamination	- Skin contact may cause rashes to burns - May require hospitalization	- Follow hazardous waste disposal techniques set by BIC/KIC
Exposure to Unsafe Noise Levels	- Use of BIC/KIC machine shop - Loud power tools - Other BIC/KIC teams	- Increased rate of higher frequency hearing damage	- Use proper PPE, maintain a safe distance from active machinery

Improper use of tools	- Use of BIC/KIC machine shop - Soldering irons	- Damage to equipment is unlikely - Injury may range from deep lacerations - Burns to lost fingers	- Ask BIC/KIC personnel or team Safety Officer before using high-risk tools, attend BIC safety training
Soldering or Welding Injuries	- Worker inattentiveness - Distractions during fabrication - Lack of fixturing equipment	- Second or third-degree burns - Hardware damage due to reflex response	- Only solder and weld during work hours and in predefined locations, make sure all personnel are aware when work is being performed, use sufficient fixturing equipment
Tripping	- Carrying unsafe loads - Unclean workspace - Worker inattentiveness	- Equipment damage - Sprains and bruises - Fractured bones, concussion, death (unlikely)	- Maintain well lit work areas. Adhere to 5S workspace standards of organization. Maintain walking areas.
Contact with Launch Vehicle Debris	- Faulty parachute ejection - Severe winds	- Blunt damage to the rocket or payload - Concussion - Fractured skull - Death	- Keep a close eye on the vehicle or have someone spot the vehicle for those who are unable
Launchpad Fire	- Flammable debris blown across launch pad - Flammable fuel spilled	- Heat damage to parachute - Motor - Electronics	- Remove brush, dry debris, and other flammables around the launch pad area and have a fire extinguisher on hand
Personnel Injury from Terrain	- Uneven footing, potholes, nails, etc.	- Sprained or broken ankles - Small puncture wounds	- Watch footing around terrain, travel in groups, maintain cell phone contact

Airborne Debris	- High wind speeds - Systems on the rocket breaking mid-flight	- Blunt force trauma - Lacerations	- Maintain a reasonable and safe distance from energetic devices
Contact Burns	- Contact with motor after flight - Standing too close to the launchpad	- Mild to severe burns	- Proper handling of the rocket will be used
Heat Stroke	- Prolonged exposure in a high-temperature environment	- Possible hospitalization	- Ensure team members limit exposure to dangerously high temperatures
Hypothermia	- Failure to wear appropriate clothing	- Possible hospitalization	- Ensure team members limit exposure to dangerously low temperatures
Dehydration	- High environment temperature - Low fluid consumption	- Fatigue - Dizziness - Confusion - Immediate medical treatment	- Ensure access to cool drinking water at team events - Provide shaded areas available for rest

Identified Hazard	Pre - Mitigation Risk (Probability/Severity/Total)			Post - Mitigation Risk (Probability/Severity/Total)		
Fire	2	5	10	2	4	8
Airborne particle exposure	3	3	9	2	2	4
Electric Shock	2	4	8	2	3	6
Entanglement with machines	3	5	15	2	5	10
Epoxy Contact	4	2	8	2	2	4
Eye irritation	3	4	12	2	4	8
Falling tools or materials	2	4	8	2	2	4
Fiberglass Contact	3	3	9	1	2	2
Flying debris	2	4	8	2	1	2
Exposure to Hazardous Fumes	4	3	12	1	3	3
Hazardous Waste Contact	2	3	6	2	2	4
Exposure to Unsafe Noise Levels	3	3	9	3	1	3
Improper use of tools	3	3	9	1	2	2
Soldering or Welding Injuries	4	2	8	3	1	3
Tripping	2	3	6	2	2	4

Contact with Launch Vehicle Debris	1	5	5	1	3	3
Launchpad Fire	2	3	6	1	3	3
Personnel Injury from Terrain	2	2	4	1	2	2
Airborne Debris	3	3	9	3	2	6
Contact Burns	1	4	4	1	3	3
Heat Stroke	3	3	9	2	2	4
Hypothermia	1	3	3	1	2	2
Dehydration	3	3	9	2	2	4

5.3 Failure Modes and Effects Analysis (FMEA)

5.3.1 Vehicle System FMEA

Identified Hazard	Causes	Effects	Mitigations
Structural Failure Under Intended Loading	- Inadequately-designed structure - Not all failure modes considered during analysis - Material defects during construction	- Unpredictable competition performance - Vehicle cannot be reflown - Falling debris exceeds competition limits for kinetic energy upon landing	- Design airframe to withstand compression load at a safety factor of 2
Airframe Overloaded During Launch	- Motor improperly packed - Loose components cause local shock loading - High winds	- Falling debris exceeds competition limits for kinetic energy upon landing	- Multiple checks to internal packing - System testing with a variety of parameters

	- Improper parachute deployment		
Hidden Structural Damage Prior To Launch	- Accidental damage during transportation or construction	- Falling debris exceeds competition limits for kinetic energy upon landing	- Check for cracks and material inconsistencies during construction
Structural Damage During Landing	- Miscalculation of landing energy or improper parachute deployment	- Significant repairs needed	- Test recovery system extensively
Bond Line Failure	- Lack of checks to bond line Rushed construction	- Falling debris exceeds competition limits for kinetic energy upon landing	- Multiple checks to bond lines
Component Mounting Failure During Launch	- Failure to utilize correct mounting techniques	- Launch failure - Destruction of component	- Multiple checks to mounting - Tests of mounting techniques
Structural Failure Of Deployment Systems	- Improper design of deployment subsystem - Construction errors	- Falling debris exceeds competition limits for kinetic energy upon landing	- Multiple checks of deployment systems during launch - Tests of deployment systems
Structural Failure During Deployment	- Insufficient damping in parachute attachment - Construction errors - Jammed structures	- Mission failure	Same as above
Aerodynamic Instability	- Location of masses change within the vehicle	- Vehicle exceeds competition limits for kinetic energy on landing	- Static stability margin is measured as part of preflight checklist

	- Dynamic instability due to drag flaps		- Final vehicle configuration is tested at Vehicle Demonstration Flight - Drag flaps will command closed if high vibrations are detected
Electronics Failure Of Deployment Systems	- Parts dead on arrival - Insufficient charge of battery - Damage from aerodynamic forces	- Unpredictable competition performance - Vehicle does not separate - Vehicle exceeds competition limits for kinetic energy upon landing - Personal injury	- Remove-before-flight tag arms vehicle - Dissimilar redundancy in altimeter selection - Test altimeters upon arrival and before flight
Electronics Fire	- Overcharge of battery - Short circuit wiring	- Vehicle and/or falling debris exceeds competition limits for kinetic energy upon landing	- Teach all members the proper handling of the batteries and wiring - Multiple checks for proper wiring

Battery Depletion During Launch	- Unintended draw on electronics - Battery is not charged prior to launch	- Deployment electronics not functional - Flight altimeter not functional for scoring - Vehicle exceeds competition limits for kinetic energy upon landing	- Tests of battery under launch conditions - Potential redundant battery systems
Failure Of Airframe To Separate	- Over-tight fitting tolerances between airframe components - Unintended mechanical locking between airframe components	- Vehicle exceeds competition limits for kinetic energy upon landing	- Tests of airframe separation
Internal Hardware Damaged During Separation	- Damage to internal electronics	- Failure to successfully calculate and to test the recovery system	- Test the recovery system multiple times
Recovery Hardware Does Not Eject	- Damage to airframe, electronics, and possible damage to property	- Vehicle exceeds competition limits for kinetic energy upon landing	Same as above
Damage To Parachute	Same as above	Same as above	Same as above
Parachute Does Not Open	Same as above	Same as above	Same as above
Excessive Vehicle Drift During Recovery	- Failure to test and successfully simulate recovery system	- Vehicle exceeds competition limits for recovery drift	Same as above
Altitude Assurance Initialization Failure	- Failure to test, successfully simulate, and properly construct altitude assurance	- Flaps do not actuate, apogee overshoot	- Extensively test, validate simulations, and carefully construct altitude assurance

		- Flaps actuate before burnout, destabilization	
Altitude Assurance Control Scheme Failure	- Excessive loads jam control mechanism - Faulty control logic - Incorrect apogee prediction model	Same as above	- Final vehicle configuration is tested at Vehicle Demonstration Flight
Altitude Assurance Does Not Halt At Apogee	Same as above	Same as above	- Final vehicle configuration is tested at Vehicle Demonstration Flight
Mechanical Failure Of Altitude Control Hardware	Same as above	Same as above	Same as above
Structural Failure Of Altitude Control Hardware	Same as above	- Falling debris exceeds competition limits for kinetic energy upon landing	- Altitude Control Structure will be designed with a factor of safety appropriate for critical systems.
Uneven Deployment Of Drag Flaps	- Failure to test and successfully simulate drag flaps	- Aerodynamic instability of launch vehicle - Failure to deploy recovery systems - Vehicle exceeds competition limits for kinetic energy upon landing	- Testing and successfully simulating drag flaps

Motor Cannot Ignite	- Faulty product or packing of motor - Faulty igniter installation	- Vehicle fails to launch - Failure to compete with all other systems	- Test motor packing and ensure product is in good condition - Multiple sign-offs on motor assembly and installation Igniter retention using support rod
Motor Does Not Provide Design Thrust	- Faulty product or packing of motor	- Vehicle fails to reach 4000 ft	- Altitude Assurance actively adjusts flight trajectory if too much thru
Motor Explodes	- Imperfections in motor grain packing cause localized high pressure regions	- Mission fails	- Test motor and check datasheets for verification
Motor Retention Mechanism Breaks	- Imperfections in motor grain packing cause localized high pressure regions	- Falling debris exceeds competition limits for kinetic energy upon landing	Same as above
Motor Misalignment	- Poor construction quality of motor mount	- Unpredictable vehicle trajectory	Same as above
Motor Damages Internal Components	- Heat conduction through structure - Failure of bulkhead	Same as above	Same as above

Identified Hazard	Pre - Mitigation Risk (Probability/Severity/Total)			Post - Mitigation Risk (Probability/Severity/Total)		
Structural Failure Under Intended Loading	2	3	6	2	2	4
Airframe Overloaded During Launch	2	4	8	2	2	4
Hidden Structural Damage Prior To Launch	1	4	4	1	2	2
Structural Damage During Landing	3	3	9	2	3	6
Bond Line Failure	3	4	12	2	3	6
Component Mounting Failure During Launch	2	4	8	1	3	3
Structural Failure Of Deployment Systems	3	4	12	2	2	4
Structural Failure During Deployment	3	3	9	2	2	4
Aerodynamic Instability	4	3	12	3	3	9
Electronics Failure Of Deployment Systems	2	4	8	2	2	4
Electronics Fire	1	5	5	1	3	3
Battery Depletion During Launch	2	4	8	2	2	4
Failure Of Airframe To Separate	4	5	20	3	4	12
Internal Hardware Damaged During Separation	2	3	6	1	3	3
Recovery Hardware Does Not Eject	3	5	15	2	4	8

Damage To Parachute	2	4	8	1	4	4nmj
Parachute Does Not Open	3	5	15	2	5	10
Excessive Vehicle Drift During Recovery	2	2	4	2	1	2
Altitude Assurance Initialization Failure	2	2	4	2	1	2
Altitude Assurance Control Scheme Failure	2	2	4	2	1	2
Altitude Assurance Does Not Halt At Apogee	3	2	6	2	2	4
Mechanical Failure Of Altitude Control Hardware	3	4	12	2	3	6
Structural Failure Of Altitude Control Hardware	3	2	6	2	2	4
Uneven Deployment Of Drag Flaps	2	4	8	2	3	6
Motor Ignition Incapability	1	4	4	1	3	3
Motor Does Not Provide Design Thrust	2	4	8	1	3	3
Motor Explodes	1	5	5	1	4	4
Motor Retention Mechanism Breaks	1	4	4	1	3	3
Motor Misalignment	2	4	8	1	3	3
Motor Damages Internal Components	2	4	8	1	3	3

5.3.2 Payload and Payload Integration FMEA

Identified Hazard	Causes	Effects	Mitigations
Mounting Failure During Flight	- Rushed implementation or lack of training	- Damaged payload bay	- Multiple checks
Mounting Failure During Landing	Same as above	Same as above	Same as above
Hardware Misassembly	Same as above	Same as above	- Bench test payload prior to launch
Faulty Control Logic	- Oversight or lack of checks	Same as above	- Multiple checks from multiple people to ensure correct logic
Failure to Arm Electronics	- Oversight or lack of checks	- Mission Failure	- Embed a master switch to enable the electronics for the vehicle - Train students to enable switch when not enabled
Failure to Detect Landing	- Failure to test sensors - Incorrect wiring	- Premature determ ination of vehicle location	- Testing of sensors under multiple conditions
Wiring Failure Between Controller and Hardware	- Oversight or lack of checks - Improper placement of electronics bay	- Electronics fire - Effects range from small burnout on pins to explosion mid flight	- All electronics will be checked by all students before the launch

	- Loose or misassembled components		
Telemetry Transmission/Reception Failure	- Interference - Parachute Interrupts Telemetry	- Miscommunication with other sensors and main controller	Same as above
Sensor Hardware Failure	- Parachute covers sensors - Aerodynamic effects influence barometric readings - Mismounting or misalignment of	- Bad readings to determine location	Same as above
Battery Depletion Prior to Data Transmission	- Lack of testing	- Loss of the sensor data Failure of payload competition	- Test the battery under launch conditions

Identified Hazard	Pre - Mitigation Risk (Probability/Severity/Total)			Post - Mitigation Risk (Probability/Severity/Total)		
Mounting Failure During Flight	2	5	10	2	4	8
Mounting Failure During Landing	3	3	9	2	2	4
Hardware Misassembly	2	4	8	2	3	6
Faulty Control Logic	3	5	15	2	5	10
Failure to Arm Electronics	4	2	8	2	2	4
Failure to[;.. Detect Landing	2	4	8	2	2	4

Wiring Failure Between Controller and Hardware	3	3	9	1	2	2
Controller Hardware Failure	2	4	8	2	1	2
Telemetry Transmission/Reception Failure	4	3	12	1	3	3
Sensor Hardware Failure	2	3	6	2	2	4
Battery Depletion Prior to Data Transmission	3	3	9	1	2	2

5.3.3 Payload and Payload Integration FMEA

Identified Hazard	Causes	Effects	Mitigations
Debris	- Debris not removed from launch site	- Interference with the launch vehicle causing a postponed launch to mission failure	- Clear area before the launch
Premature Deployment	- Deployment charge self-ignites - Deployment electronics trigger charge early	- Vehicle exceeds competition drift limit	- Testing of the launch vehicle and verification of simulations
Late Deployment	- Failure to successfully calculate and to test the recovery system	- Vehicle exceeds competition limits for kinetic energy upon landing	Same as above
Failure To Arm Electronics	- Oversight of electronics arming	- Vehicle exceeds competition limits for kinetic energy upon landing	- Remove-before-flight tag arms vehicle - Electronics arming is made explicit in pre-flight checklist

Identified Hazard	Pre - Mitigation Risk (Probability/Severity/Total)			Post - Mitigation Risk (Probability/Severity/Total)		
Debris	1	3	3	1	2	2
Premature Deployment	2	2	4	2	1	2
Late Deployment	2	2	4	1	2	2
Failure To Arm Electronics	2	4	8	2	2	4

5.4 Environmental Concerns

Identified Hazard	Causes	Effects	Mitigations
Launchpad fire	- Dry environment - Flammables near launchpad during motor ignition	- Grass fire - Charred launch field	- Launch pad cleared as part of pre-flight checklist
Fire at landing site	- Dry environment - Unintentional motor ejection	- Launch field fire	- Motor will not protrude past aft end of vehicle
Collision with spectator drones	- Launch environment carelessness	- Possible complete mission failure	- Visually verify safe launch conditions prior to ignition, and coordinate with range safety officers to verify conditions at time of launch

Vehicle Fouled by Foreign Objects	- Unclean team preparation area	- Cascaded mission hazards	- Vehicle and payload inspection as part of pre-flight checklist
Inclement Weather	- Poor launch planning	- Component material embrittlement	- Independently measure launch conditions, and/or coordinate with other teams and range safety officers to verify conditions at time of launch
Wet Launch and Landing Sites	- Prior inclement weather effects present launch conditions	- Component material weathering	- Design vehicle to withstand wet environments
Components overheat on launchpad	- Overexposure to sun - High temperature launch day conditions	- Component material melting or failure	- Ensure proper protection of mission components on launch day as part of launch day guidelines
Launch debris left on site	- Rocket ejects debris during flight - Failure to collect waste generated during mission operations - Catastrophic mission failure	- Littering during launch operations	- Track waste generated during launch operations and provide trash bags for immediate disposal - Design vehicle to fail in minimal independent sections - Construct external vehicle components from materials that can be visually identified at the launch site

			- Visual environmental inspection as part of post flight checklist
Vehicle lost on recovery	- Recovery subsystem failure - Vehicle destruction	- Failed mission - Littering during launch operations	- Ensure redundancy in recovery design
Team equipment left on site	- Negligence of launch day operations	- Equipment must be repurchased	- Post flight checklist
Launch vehicle stuck in tree	- Unintended collision trajectory	- Potential vehicle and payload loss	- Do not perform test launches at sites with trees.
Launch vehicle collision with structures	- Unintended collision trajectory - Wind turbines and buildings present at launch fields	- Launch vehicle and payload destruction - Potential damage to structures	- Evaluate launch day conditions with special consideration to intended vehicle trajectory as part of pre-flight checklist

Identified Hazard	Pre - Mitigation Risk (Probability/Severity/Total)			Post - Mitigation Risk (Probability/Severity/Total)		
Launchpad fire	3	4	12	2	3	6
Fire at landing site	2	4	8	1	2	2
Collision with spectator drones	2	4	8	1	4	4
Vehicle Fouled by Foreign Objects	1	3	3	1	2	2
Inclement Weather	1	5	5	1	1	1

Wet Launch and Landing Sites	2	2	4	1	2	2
Components overheat on launchpad	3	3	9	2	3	6
Launch debris left on site	2	3	6	1	3	3
Vehicle lost on recovery	3	5	15	2	5	10
Team equipment left on site	2	3	6	1	3	3
Launch vehicle stuck in tree	2	5	10	1	5	5
Launch vehicle collision with structures	2	5	10	1	5	5

5.5 Project Risks

Identified Hazard	Causes	Effects	Mitigations
Time	- Poor time management - Improper delegation of tasks - Students shifting focus away from competition	- Document Writing/Vehicle Fabrication is rushed - Failure to meet deadlines	- Establish a reasonable timeline and adhere to it - Evenly distribute tasks among students
Miscommunication	- Students not requesting help - Poor attitude towards people and leadership	- Project requirements are completed incorrectly - Project requirements are not completed because they are assigned to no one	- Have a good relationship with the team - Foster a friendly and inviting atmosphere
Scope	- Failure to maintain focus on core design	- Project becoming infeasible due to complexity	- Stay on track of project plan

	- Adding too many features that may deviate from requirements		- Regularly reevaluate our design requirements
Resource	- World-wide shortages - Equipment breaking down - Students unable to participate	- Insufficient resources to complete project	- Order parts as early as possible
Budget	- SGA not providing us enough funding - No sponsorships	- Insufficient funds to finish vehicle advancements	- Request for funding early on in the process to avoid late delivery
Performance	- Wrong motor type or poor selection of vehicle aerodynamics	- Not enough thrust to reach desired apogee - Overshooting the vehicle beyond 6000 feet	- Testing in environments similar to launch site

Identified Hazard	Pre - Mitigation Risk (Probability/Severity/Total)			Post - Mitigation Risk (Probability/Severity/Total)		
Time	5	5	25	4	2	8
Communication	3	3	9	2	2	4
Scope	2	3	6	2	2	4
Resource	3	4	12	2	4	8
Budget	4	4	16	4	3	12
Performance	3	4	12	3	2	6

6. Project Plan

6.1 Derived Requirements

System	Requirement	Justification	Verification
Airframe	Material failure calculations or simulations will be required to justify the deliberate use of metallic materials in each structural component.	The competition rules prohibit the use of excessive and/or dense material in the construction of the vehicle per Req. 2.23.10. This is to ensure that the vehicle is constructed with minimal use of metallic materials.	The vehicle design will be audited for metallic material use by the submission of the Project Silverstein CDR.
	The airframe design and construction must be able to accommodate multiple internal arming switches.	Per Req. 3.6, altimeters must be activated by a dedicated arming switch. Per Req. 3.7, these arming switches must not be able to be disarmed during flight. Internal arming switches for altimeters and other electronics must be accessible by the airframe to protect these switches from aerodynamic manipulation.	An audit for electronic arming switches in the airframe design will be performed by the submission of the Project Silverstein PDR.
	The airframe will be restricted from designs utilizing asymmetric geometry and/or mass distribution along the length of the vehicle.	Significant asymmetry of the airframe would cause non-negligible deviations from the projected trajectory.	The vehicle design will be reviewed by the submission of the Project Silverstein CDR.

Altitude Assurance	Altitude Assurance System will be restricted to extending drag-producing devices aft of the burnout CG.	Extended drag-producing devices that are a part of the altitude assurance system are classified as structural protuberances by the RR-SL team. Per Req. 2.16, these devices may only act aft of the burnout CG.	Burnout CG will be calculated to determine placement of the altitude assurance system by submission of the Project Silverstein CDR.
	The Altitude Assurance System must be capable of decreasing launch vehicle apogee by 1700 ft.	-Performance cals, petal performance, margin, req 2.1	Verification of agreement between more than 1 performance simulation will be required to finalize component level design at the Project Silverstein CDR.
	The altitude assurance system must be designed to fail at minimum half deployment actuation throughout the vehicle ascent.	The altitude assurance must be able to fail open in order to avoid overshooting our	The final vehicle configuration will be tested by the FRR.
Recovery	All energetic devices must be handled using COTS electronics.	The team is not experienced in experimenting with energetic devices. Handling energetic devices with COTS electronics will remove variability	COTS electronics will be incorporated into the critical recovery and propulsion design.
	Each independent section of the launch vehicle will have a maximum kinetic energy of 65 ft-lbf at landing.	This requirement derives from Req. 3.3. The team has determined that a 10 ft-lbf error margin will be required to ensure that this requirement is met with reasonable variation.	Using descent velocity simulated, independent section KE at landing will be calculated and verified by FRR deadline.

Propulsion	Energetic devices may only be manipulated, triggered, or reacted on by COTS hardware.	The team is not experienced in experimenting with energetic devices. Handling energetic devices with COTS hardware will remove variability	COTS hardware will be incorporated into the critical recovery and propulsion design.
Payload	The method we use for locating the rocket will strictly applicable to communication with a probe on another planet	Derived from Req. 4.1, the team determined that the phrasing “adhere to the intent of the challenge” as indication that our solution should be viable on another planet with no existing technology	The team kept this in mind when researching possible solutions to locating the rocket, and stopped considering solutions that would not work on a primitive planet. Research was also done on current methods of deep space communication in order to understand the constraints.
	The payload experiment must fully fit inside the nose cone	The vehicle team has concluded that the payload must fit entirely inside the nose cone	The final payload design will be reviewed by the submission of the CDR.
	The payload must not deploy from the launch vehicle	The vehicle team has determined that the added safety and mission risk caused by payload deployment are not necessary to successfully complete this year’s mission	The final payload design will be reviewed by the submission of the CDR.
	Payload telemetry must not require a HAM license	Currently, no team members possess the requirements necessary to operate on HAM bands. Obtaining such a license would add additional time constraints and failure modes to the project.	The final payload design will be reviewed by the submission of the CDR.

6.2 Budget

6.2.1 Line Item Budget

Preliminary Component Level Budget				TOTAL:	$16,328.00
Item	**Price**	**Quantity**	**Shipping**	**Total**	**Vendor**
Equipment					
Voron 2.4	$940.00	1	$0.00	$940.00	3d Printers Bay
Voron 0.1	$493.00	1	$0.00	$493.00	3d Printers Bay
Lipo Battery Charger	$80.00	1	$0.00	$80.00	Hobby King
Lipo Battery Bag	$5.00	2	$0.00	$10.00	Hobby King
Soldering And Rework Station	$200.00	1	$0.00	$200.00	Amazon
Wire Brush	$15.00	1	$0.00	$15.00	Amazon
Electrical Vise	$30.00	1	$0.00	$30.00	Amazon
Solder Hands	$25.00	1	$0.00	$25.00	Amazon
Hand Clamp	$8.00	2	$0.00	$16.00	Amazon
Bar Clamp 4 Pack	$16.00	1	$0.00	$16.00	Amazon
Cobalt Drill Index	$200.00	1	$0.00	$200.00	Amazon
Pliers, Wrenches	$94.00	1	$0.00	$94.00	Amazon
			Section Total:	**$2,119.00**	
General Consumables					
Solder	$25.00	1	$0.00	$25.00	Amazon
B/W/R 22 Gauge	$12.00	3	$0.00	$36.00	Amazon

B/R 18 Gauge	$10.00	2	$0.00	$20.00	Amazon
Gf30 Nylon 3d Printer Filament	$185.00	1	$15.00	$200.00	3dxtech
Pla Plus Filament	$25.00	3	$0.00	$75.00	Amazon
Epoxy	$172.00	1	$0.00	$172.00	Total Boat
Fine Adjustment Cable Ties	$17.00	1	$8.00	$25.00	Mcmaster Carr
Electrical Tape	$4.00	6	$0.00	$24.00	Amazon
Solo Cups	$5.00	1	$0.00	$5.00	Amazon
Rail Buttons	$8.00	4	$5.00	$37.00	Rail Buttons
M2/M3/M4/M5 Bolts	$25.00	2	$0.00	$50.00	Amazon
Popsicle Sticks	$4.00	1	$0.00	$4.00	Amazon
Duct Tape	$13.00	1	$0.00	$13.00	Amazon
Aluminum Wide Rivets	$13.00	1	$3.00	$16.00	Mcmaster Carr
Aluminum Narrow Rivets	$10.00	1	$3.00	$13.00	Mcmaster Carr
Aluminum Billet	$146.00	1	$0.00	$146.00	Mcmaster Carr
Protoboard	$12.00	1	$0.00	$12.00	Amazon
			Section Total:	**$873.00**	
Rocket Body					
G12 Body Tube	$46.00	10	$27.00	$487.00	Wildman Rocketry
Nosecone	$150.00	1	$15.00	$165.00	Wildman Rocketry
Mica Insulation Sheets	$85.00	1	$14.00	$99.00	Mcmaster Carr
Spray Paint	$6.00	3	$0.00	$18.00	Amazon
14" Coupler	$78.00	2	$14.00	$170.00	Madcow Rocketry
G10 Sheet	$18.00	4	$20.00	$92.00	Wildman Rocketry

		Section Total:		**$1,904.00**	
Altitude Assurance					
2 Ft X 1/4" Diameter Uhmwpe Rod	$3.00	1	$11.00	$14.00	Mcmaster Carr
Ptfe Film Tape	$15.00	1	$0.00	$15.00	Amazon
16mmx75mm Air Cylinder	$12.00	2	$0.00	$24.00	Amazon
2-Way Solenoid Valve	$17.00	2	$0.00	$34.00	Amazon
Altimeter	$10.00	2	$10.00	$30.00	Adafruit
Control Computer	$15.00	2	$8.00	$38.00	Digikey
Absolute Position Encoder	$8.00	6	$12.00	$60.00	Sparkfun
		Section Total:		**$215.00**	
Motor					
Motor Case	$560.00	1	$20.00	$580.00	Wildman
Motor	$350.00	3	$40.00	$1,090.00	Wildman
75mm Motor Tube	$40.00	1	$7.00	$47.00	Madcow Rocketry
75mm Motor Retainer	$65.00	1	$7.00	$72.00	Wildman Rocketry
		Section Total:		**$1,789.00**	
Subscale					
54mm Motor Retainer	$31.00	1	$0.00	$31.00	Wildman Rocketry
Motor Reload Kit 38mm 720 Case	$104.00	1	$0.00	$104.00	Wildman Rocketry
Centering Ring	$7.00	3	$0.00	$21.00	Madcow Rocketry
Motor	$120.00	1	$40.00	$160.00	Wildman Rocketry
54mm Motor Tube	$30.00	1	$7.00	$37.00	Madcow Rocketry
4" Airframe Tube	$272.00	1	$23.00	$295.00	Madcow Rocketry

4" Coupler	$29.00	1	$16.00	$45.00	Madcow Rocketry
4" 4:1 Ogive Nose Cone	$38.00	1	$18.00	$56.00	Madcow Rocketry
			Section Total:	**$749.00**	
Payload					
Raspberry Pi 4 Kit	$120.00	2	$0.00	$240.00	Amazon
Cots Telemetry Modules	$80.00	2	$10.00	$170.00	Sparkfun
750 Mah 4s Battery	$38.00	2	$0.00	$76.00	Getfpv
Sd Cards	$9.00	4	$0.00	$36.00	Amazon
Mountable Xt60 Plugs	$12.00	1	$0.00	$12.00	Amazon
22awg Silicone Wire	$15.00	1	$0.00	$15.00	Amazon
18awg Silicone Wire	$15.00	1	$0.00	$15.00	Amazon
Accelerometer	$20.00	3	$10.00	$70.00	Adafruit
Altimeter	$10.00	4	$10.00	$50.00	Adafruit
750 Mah 4s Battery	$38.00	1	$0.00	$38.00	Getfpv
			Section Total:	**$722.00**	
Recovery					
Rrc3 Altimeter	$74.00	2	$7.00	$155.00	Wildman
Rocket Locator				$0.00	
Recovery Harness	$72.00	2	$7.00	$151.00	Wildman
Avionics Bay	$50.00	2	$10.00	$110.00	Madcow Rocketry
750 Mah 4s Battery	$38.00	1	$0.00	$38.00	Getfpv
Hybrid Supercapacitor	$11.00	2	$4.00	$26.00	Digikey
Nylon Shear Pins	$4.00	2	$5.00	$13.00	Apogee Rockets

Skyangle Cert-3 Large	$139.00	1	$13.00	$152.00	Madcow Rocketry
Drogue Chute	$86.00	1	$29.00	$115.00	The Rocket Man
Mica Insulation Sheet	$85.00	1	$14.00	$99.00	Mcmaster Carr
			Section Total:	**$859.00**	
Travel					
Mileage Rebimusmnet (4 Per Car)	$415.00	5	$0.00	$2,075.00	N/A
Student Hotel (4 Per Room)	$135.00	20	$0.00	$2,700.00	N/A
Mentor Hotel	$135.00	4	$0.00	$540.00	N/A
Meals (Per Person)	$15.00	40	$0.00	$600.00	N/A
			Section Total:	**$5,915.00**	
Branding					
Stickers (Bulk Order)	$100.00	1	$4.00	$104.00	Sticker Mule
Team Presentation Polos	$18.00	20	$0.00	$360.00	Bagnoche Sports
Team Event T-Shirts	$10.00	20	$0.00	$200.00	Bagnoche Sports
			Section Total:	**$664.00**	
Outreach					
Vehicle Mileage (3 Events)	$11.50	6		$69.00	N/A
Meals (10 People, 3 Events)	$15.00	30		$450.00	N/A
			Section Total:	**$519.00**	

6.2.2 Material Acquisition Plan

The Rose Rocketry Student Launch Team receives all funding through the Rose Hulman Student Government Association (SGA), Rose Hulman Branam Innovation Center (BIC), and sponsor donations.

Each year every competition team inside the BIC submits a budget, which is later awarded in full or adjusted. This academic year, the BIC received a 40% budget cut by school administration, in effort to make up for ongoing COVID-19 expenses. As a result, every BIC team also received a budget cut. Rose Rocketry's BIC budget is $3000 for the 2021-2022 academic year.

In a similar process to the BIC, every club on campus submits a budget to SGA. These budgets are then reviewed, edited, and awarded. However, this process only applies to clubs fully approved and recognized by SGA. Due to miscommunication, unclear instructions, and contradictory SGA policies, Rose Rocketry is not a fully recognized SGA club. Instead, Rose Rocketry currently holds a probationary club status and is ineligible for a full budget. This means we do not have any funds set aside by SGA for the team and no dollar amount we expect to receive. To receive SGA funding, the club must submit special One Time Funding Requests (OTFR). This is a lengthy process which can take anywhere from one week minimum to 4 weeks maximum to unlock funding for the requested items. This places a unique risk on the team of not having funding for parts ordered any less then a month or more in advance. However, due to a majority of club activities and competitions being cancelled last academic year, SGA has a surplus of funds and is able to support the setup and operational cost of Rose Rocketry, so long as OTFRs are submitted in a timely manner.

In addition to BIC and SGA funding, Rose Rocketry has received a $1000 donation from an anonymous donor to support team efforts.

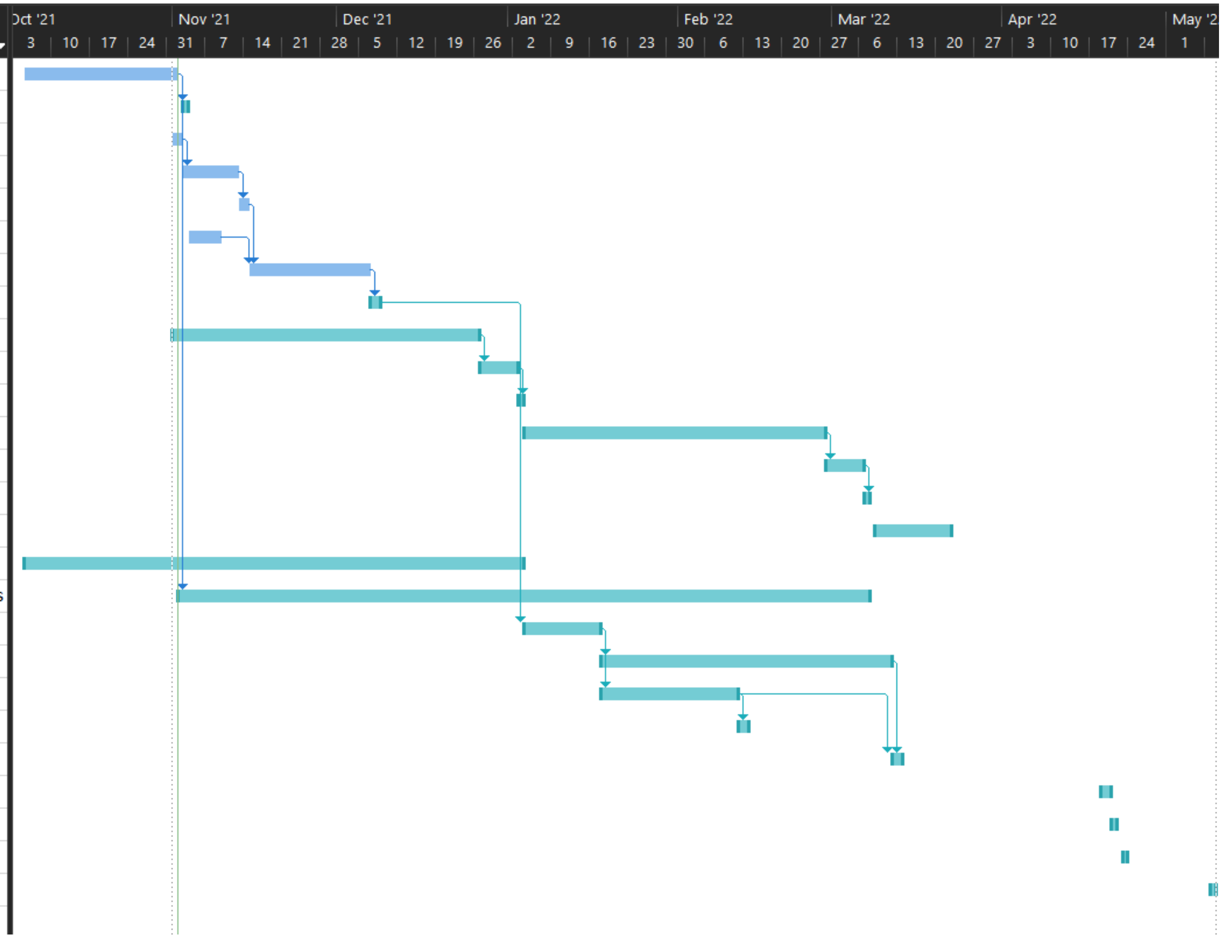
6.2.3 Material Acquisition Plan

Due to the timeline issues laid out above with SGA and the ongoing global supply chain issues, the team forseas the ordering and receiving of parts to be one of the biggest challenges faced this competition season. In order to be better prepared for competition, the team has added additional milestones throughout the season, such as the launching of a level 2 fiberglass kit in November to gain experience before building and launching the subscale rocket. Although these additional milestones will benefit the team, they add an additional timeline constraint to an already tight timeline. We have already run into problems with parts being out of stock, such as the RRC3 altimeter, and SGA taking weeks to release funding, such as not having funding for the first 5 weeks of the school year. In order to ensure the team has everything required to complete competition and derived milestones, the team is ordering components as soon as possible and prioritizing discussions of component funding at team meetings. Because we do not have a specified budget from SGA, at any point a component is considered to be a leading contender in a leading design alternative, that component will be submitted through an OTFR and ordered. This is done due to the high likelihood that by the time a system component is finalized, there may not be enough time left to submit an OTFR, wait for approval, wait for shipping, and add the component to its respective system.

6.3 Timeline

6.3.1 Project Timeline Gantt Chart

	Task Name	Duratio
2	Write PDR	28 days
3	PDR Video Teleconference	1 day
5	Order L2 Parts	2 days
6	Construct L2 Rocket	10 days
7	L2 Rocket Test	2 days
9	Order Subscale Parts	6 days
10	Construct Subscale Rocket	22 days
11	Subscale Flight	2 days
13	Write CDR	56 days
14	Revise CDR	7 days
15	CDR, Presentation, Flysheet DUE	1 day
17	Write FRR	55 days
18	Revise FRR	7 days
19	FRR Report, Presentation, Flysheet DUE	1 day
20	FRR Video Teleconferences	14 days
21	Design Subsystems	91 days
22	Test Subsystems	126 days
23	Order Parts For Final Construction	14 days
24	Construct Subsystems	53 days
25	Construct Full-Scale Vehicle	25 days
26	Full Scale Test Flight	2 days
27	Payload Integration Test Flight	2 days
29	Travel	2 days
30	Official Kickoff, LRRs, Activities	1 day
31	Launch Day & Awards	1 day
32	PLAR Due	1 day



6.3.2 Major Project Deadlines

NASA + Indiana Rocketry Schedule + Rose Rocketry Deadlines

- November 1 - PDR, presentation slides, flysheet due
- November 12 - L2 Rocket Completion Deadline
- November 13-14 - High Power Launch
- December 10 - Subscale Rocket Completion Deadline
- December 11-12 - High Power Launch
- January 3 - Subscale Flight Deadline
- January 3 - Completed gridded map due
- January 3 - CDR, presentation slides, flysheet due
- February 11 - Full-Scale Rocket Construction Deadline
- February 12-13 - High Power Launch
- March 7 - Vehicle demonstration flight deadline
- March 7 - Flight Readiness Review (FRR) report, presentation slides, and flysheet due to NASA project management team by 8:00 a.m. CST.
- March 11 - Payload Integration Deadline
- Saturday & Sunday, March 12-13, High Power Launch
- April 4 - Payload Demonstration Flight and Vehicle Demonstration Re-flight deadlines
- April 4 - FRR Addendum Due
- April 19-20 - Travel to Huntsville, AL
- April 20-24 - Competition Week
- May 9 - Post-Launch Assessment Review (PLAR) Due

7. Appendix

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