PSP High Altitude

Spaceshot Project
System Requirements Review and Conceptual Design Review

December 2022

Contents

1 Introduction

This report corresponds to the System Requirements Review and Conceptual Design Review for the Spaceshot project of the Purdue Space Program High Altitude team. The purpose of this review is twofold: first, we seek feedback on our system requirements; we have a set of stakeholder requirements, which we have flowed down to the level of each vehicle subsystem, and we believe these requirements are complete and sufficient to define our project. Second, we present our conceptual design of the vehicle, and show that it is capable of meeting the requirements we have specified.

We believe the report is structured logically to achieve those objectives. In this section we offer a short introduction to the team and its finances. Next, Section 2 covers the stakeholder and functional requirements, and flows them down to the level of each subsystem. Section 3 begins discussion of the vehicle design with the initial sizing process, and our goals for future simulation. The propulsion system is next, in Section 4, followed by avionics in Section 5, mechanisms in Section 6, and structures in Section 7. Each of these component sections discusses the motivating requirements and the planned implementation that will satisfy them. Finally, Section 8 discusses our next steps with the project after this review, including aspects of the project we consider highest risk.

1.1 Purdue Space Program High Altitude

High Altitude (HA) is a project team within Purdue Space Program (PSP) that was formed in May 2021. High Altitude's objective is to fly a two stage, student developed rocket to the Kármán Line: 100 kilometers above mean sea level. The team was formed with the experience and leadership from the now-defunct PSP Solids team, which competed annually in the Spaceport America Cup. Over the course of the past year, High Altitude has continued to develop skills across the team in design iterations and flights as the team continues to move into more detailed work on the spaceshot rocket.

Since its formation, High Altitude has been involved in rapid iteration and prototyping of many smaller-scale rockets. Last year, the team conducted three launches that began developing experience for our team. This started with an initial L2 kit rocket, the Wildman Darkstar Extreme, and its launch in September 2021. The team's next launch was in December; it was fully designed and constructed by our team and made primarily out of carbon fiber. The third and most recent launch was a reflight of the Darkstar. After these launches, the team began work on the spaceshot project; this Design Review will conclude the first phase of that work.

1.2 Budget

The High Altitude team receives funding each semester from Purdue organizations including Purdue Engineering Student Council (PESC) and the Purdue Engineering President's Council (PEPC). These merit funds total up to $6,000 per semester. The team launched a successful crowdfunding campaign in the Spring of 2022 to raise over $3,000 and also participates in fundraising events through Purdue Athletics. In addition, we have applied for scientific research grants through organizations such as NASA to support the project's development. These research grants are limited by the type of research being completed by HA as there is not an experimental payload included onboard the rockets.

These funds are reallocated each semester to each technical team based on the current projects of each team. Currently, HA has about $8,000 with an expected addition of $3,000 from the PESC Merit Fund before the end of the year. For the Spring 2023 semester, the Avionics team will receive $2,000 for the research and development of a flight computer as well as the purchase of a commercial avionics board to be tested on an L1 kit rocket. The Mechanisms team will receive $500 to construct and test the de-spin mechanism as well as the recovery system. The Propulsion team will be designing and building a test stand at Zucrow Laboratories to characterize solid rocket motors; the cost of this project is dependent upon the involvement of other research groups. Structures will continue to finalize the design for spaceshot; prototyping, manufacturing, and testing the airframe is estimated to cost between $4,000 - $10,000 which will be allocated incrementally during the next few semesters. Future budgeting will involve attempting to obtain funding from companies, institutions, and foundations.

2 Spaceshot Requirements

This section includes our highest-level requirements for the spaceshot project. Throughout the rest of the report other sections will reference the specific requirement a particular design is satisfying, in order to motivate it. Each requirement ID in the rest of this report is a clickable hyperlink to the appropriate part of Appendix B, where all of our requirements are tabulated, with their derivations when appropriate.

Some of these requirements are dependent on whether or not we expend the first stage of the vehicle. Our team has decided to not include first stage recovery as an internal stakeholder requirement. However, there may or may not be external factors that require us to recover the stage. Currently, we have worked on designs for subsystems like avionics and recovery that will be included in the first stage, if and only if it is to be recovered. For the rest of this report, requirements that only exist if the first stage is to be recovered are marked with a $*$, and requirements that only exist if the first stage is able to be expended are marked with a †.

2.1 Internal Stakeholder Requirements

Our stakeholder requirements are derived from entities involved in the development, launch, or regulation of an amateur rocket. In this case, the customer of this rocket is the Purdue Space Program High Altitude team. These were decided in a team-wide planning meeting early in the vehicle design process.

2.2 External Stakeholder Requirements

These are requirements set by either PSP or PSPHA members.

Req. ID	**Requirement**
SR.1	The rocket shall reach 100 km mean sea level.
Our mission statement is to reach space, for which we use 100 km above sea level as the target height as that is widely regarded as the boundary between Earth and space.	

Req. ID	**Requirement**
SR.2	The rocket shall have two powered stages.
We want to learn from the complexity of the separation mechanism, develop valuable learning experiences, and become the first successful two stage space shot rocket built by a student team. Although 100 km is achievable with a single motor, mixing and casting a motor of this size introduces challenges as these processes are overseen by Zucrow Labs[1]. Additionally, the multistage design meets the team's vision and creates design challenges that the team wants to take on.	

Req. ID	Requirement
SR.3	The rocket shall have one or more motors created by students at Purdue Zucrow Labs.
Part of our vision is to involve as much student design as possible within the rocket. We have access to a propulsion lab and the equipment needed to mix our own solid rocket motor, which will allow us to fine tune our thrust profiles and not limit our designs to commercially available solid motors.	

Req. ID	Requirement
SR.4	The upper rocket stage shall be recoverable[2].
To be able to physically analyze the effects of high speed flight and verify any data recorded onboard.	

Req. ID	Requirement
SR.5	The rocket shall carry a payload non-essential to rocket performance.
We want to put an object inside the rocket that is meaningful to the team and launch it to space. It should not be a critical part of the vehicle.	

2.3 Functional Requirements

2.3.1 Flight-Critical Requirements

After considering all of our stakeholder requirements, we derived the high level requirements for our vehicle to achieve its mission. Functional requirements are more focused on what the overall rocket has to do and not how. It will also have the physical components stated for a rocket to be a rocket. Also, some of these requirements are dependent on whether or not we expend the first stage. Requirements that only exist if the first stage is recovered are marked with a $*$, and requirements that only exist if the first stage is expended are marked with a $\dagger$.

2.3.2 Recovery Requirements

Requirements for a successful recovery.

2.3.3 Non-Flight Critical Requirements

Requirements not necessarily required for the vehicle but fulfills a stakeholder requirement

2.4 System Requirements

Req. ID	Requirement
SR.6	The rocket development shall follow systems documentation.
This is a requirement meant to address some of the documentation shortcomings of our previous PSP rocket teams. Documentation tends to be lacking, and whenever a core member leaves the team, limited knowledge gets transferred, resulting in having to start certain research from the beginning. This will also standardize the explanation of the function of a system across the teams and pass on our knowledge to future teams and groups.	

Req. ID	Requirement	Rationale	Traced From
DEF.1.1	Rocket stages shall have fundamental flight articles.	These are the minimum components for a stage of our rocket to be considered a stage.	SR.1
DEF.1.1.1	The stage shall have an airframe.	Core structural part of a rocket that houses sub-systems.	SR.1
DEF.1.1.2	The stage shall have a motor.	Being a two stage powered rocket, all stages will have a motor.	SR.2
DEF.1.1.3†	The stage shall have a recovery system.	To safely recover the stage.	SR.4
DEF.1.1.3.1†	To be able to study the effects of high speed flight on all parts of the rocket on the ground.	The recovery system will be actively controlled for safety.	SR.4
DEF.1.2	The lower stage shall have the required flight articles to be the first stage.	Lower stage may contain components that are not required on other stages.	SR.1
DEF.1.2.1	The lower stage shall have fins.	Passively-stabilized rockets like ours usually require fins to remain stable throughout the flight.	SR.1
DEF.1.3	The upper stage shall have the required flight articles to be the first stage.	Upper stage may contain components that are not required on other stages.	SR.1
DEF.1.3.1	The upper stage shall have fins.	Passively-stabilized rockets like ours usually require fins to remain stable throughout the flight.	SR.1
DEF.1.3.2	The upper stage shall have a nosecone.	Rockets usually require a nose cone to remain stable throughout the flight.	SR.1
DEF.1.3.3*	The upper stage shall have a recovery system.	This stage travels to apogee and would be able to physically confirm height and performance.	SR.4
DEF.1.3.3.1*	Stages with a non-autonomous recovery system shall have an avionics system.	The recovery system will be actively controlled for safety.	SR.4
DEF.1.4	The vehicle shall have a staging mechanism between stages.	This allows the stages to separate.	SR.2
DEF.1.5	The vehicle shall ignite the upper stage motor.	The second stage motor is ignited by the rocket itself as there will be no external mechanism for rocket ignition.	SR.1, SR.2

Req. ID	Requirement	Rationale	Traced From
DEF.2.1	The upper[4] stage shall be recoverable.	The upper stage travels through the entire stage of the flight and records it.	SR.4
DEF.2.1.1	The upper stage touchdown velocity shall be less than 20 ft per second.	The stage must touch down slow enough to prevent significant damage.	SR.4
DEF.2.1.2*	The lower stage touchdown velocity shall be less than 20 feet per second.	The stage must touch down slow enough to prevent significant damage.	SR.4

Req. ID	Requirement	Rationale	Traced From
DEF.3.1	The vehicle shall have a payload.	Satisfies the payload requirement, and gained data is directly useful as visual proof of rocket location.	SR.5
DEF.3.2	The vehicle shall determine its apogee.	To confirm that the rocket has reached the target apogee.	SR.1
DEF.3.3	The vehicle shall identify its location.	For easier post launch recovery.	SR.4
DEF.3.4	The vehicle shall check its state before igniting second stage.	Implied required safety feature for any two stage rocket.	EX.1.1, EX.1.2, EX.1.4

3 Vehicle Sizing

3.1 Introduction

In order to verify that our vehicle is able to satisfy the mission requirements it is vital to have an understanding of what a proposed vehicle would look like in terms of its design aspects. Factors that include propulsive, structural, aerodynamic, thermal, and many others must be considered in order to properly assess if a contending vehicle design is viable in terms of matching requirements for the mission's success.

To find a vehicle design that met the individual requirements of each subteam, a large trade study was conducted in order to develop an idea of how different factors affected each other which accumulated in the final vehicle design. Since it was determined that the propulsion system of the launch vehicle had the most direct influence in a design's ability to match the mission requirements, the figure of merit analysis started with the propulsion design.

The launch vehicle sizing process started at the highest level possible, which then became increasingly refined and filtered until the end product resulted in a few point designs that satisfactorily met mission requirements. This process revolved around a 1-degree of freedom (1DOF) mathematical model that gave time history solutions for a set of input parameters that gave insight to factors about the point design including: propellant mass required, dynamic pressure experienced, burn out time for each motor, estimated inert mass from empirical propellant mass fractions, along with others. This data was then sifted through by the aerostructures team that looked at factors such as chamber pressure and temperature history, motor dimensions, inert mass, among others, to determine if designs were good candidates for further analysis. A point design was thrown out if the inert mass required to match the specified safety factors for structural stability were not achieved, and point designs were passed forward otherwise. In the final step of this process a comprehensive 6-degree of freedom (6DOF) mathematical model study was conducted that served a dual purpose: give a refined trajectory of the vehicle's mission, and verify the outputs from the 1-dimensional model were reasonable, which gave a sanity check to the entire process. Ultimately the 6DOF gave a finishing polish on the sizing process that gave the team confidence that the point designs that were chosen as viable candidates had a high probability of completing the mission requirements successfully.

The validity of this process in choosing viable vehicle designs that are able to satisfy all of the mission requirements is yet to be shown; however, with multiple checks and balances, namely the structural analysis fitting a particular mass budget from the 1DOF, the programs utilized were able to descope a large amount of cases to test. Also, the 1 DOF itself has been tested against known designs such as the University of Southern California's Traveler IV in order to get a sense of the accuracy of the code, which showed very similar results to published data on those examples. It is important to note, that the 1DOF model is best at predicting and sizing a launch vehicle for nominal conditions, meaning the results of the model in terms of the total change in velocity are likely to be an under prediction of what the true value may be. This is due to the lack of off-nominal events taking place within the math simulated, but this is exactly why later in the vehicle sizing process the 6DOF is utilized, which incorporates many more factors that build a more realistic and complete prediction to the overall size of system required for a launch vehicle that meets mission requirements. It wasn't

Parameter	Initial Run	Final Run
First Stage Diameter (in)	3.75, 4.0, 4.25, 4.5, 5.0	5.0, 5.25, 5.5
Second Stage Diameter (in)	3.0, 3.5, 4.0, 4.5	4.0, 4.25, 4.5, 4.75
Payload Mass (kg)	1, 3, 4, 5, 10	0.5
Desired Apogee Altitude (km)	100, 125, 150, 200, 250, 300, 400	100, 125, 150
First Stage ΔV Split	35%, 40%, 45%, 50%, 55%, 65%	35%, 42.5%
Propellant Mass Fraction (λ_p)	$\lambda_{p,1}$ = 0.85 $\lambda_{p,2}$ = 0.785	$\lambda_{p,1} = 0.7$ $\lambda_{p,2} = 0.6$
I_{sp} Efficiency (η_{isp})	0.925	0.9
Total Point Designs Tested	4200	72

Table 1: Summary of Pareto analysis vehicle parameters

possible to start at the 6DOF for the sizing processes due to lack of particular expertise and lack of proprietary data for similar missions of this scope within our team, all of which lead to an initial starting point of having an under-defined problem. Assumptions made in the 1DOF gave a proper first step that allowed for reasonably accurate results, and the elimination of parameters that negatively affected the launch vehicle. Ultimately this process produces reasonably trustworthy results. It will be of great interest to see how well these predictions are after a launch attempt has been made to accomplish this mission. Before then, tests with the propulsion system have been planned in order to add corrective factors (accurate burn rate coefficients, true ISP efficiency, and true characteristic velocity) to the existing model which will result in an even better model prediction.

3.2 Figure of Merit and Pareto Analysis

The Pareto analysis, a formal technique which may be useful where many possible courses of action are competing for attention, was paired with a figure of merit analysis that allowed both methods to complement each other with the desired goal of finding how the multitude of input parameters affected the performance of a point design, and then show how the many point designs compared against each other. The figure of merit analysis generated a set of point designs for a possible launch vehicle with the parameters that were simulated summarized in Table 1, with every combination of parameters being a point design tested.

Each point design was evaluated using both the 1DOF model and the genetic algorithm, with the 1DOF model being the main computational engine in the sizing process. The genetic algorithm iterated on chamber pressure profiles for the first and second stage motors in order to maximize the following characteristic evaluation function.

$$
\begin{aligned}
CEF = W_1\left(1-\frac{t_{ref}}{t_{b1}+t_{b2}}\right) + W_2\left(\frac{m_{pl}}{m_{pl,ref}}\right) + W_3\left(\frac{h}{h_{ref}}-1\right) \\
+ W_4\left(1-\frac{m_{p,ref}-m_p}{m_{p,ref}}\right) + W_5\left(1-\frac{Q_{max,ref}}{Q_{max}}\right) + W_6\left(1-\left[\frac{L/D_{ref}-L/D}{L/D_{ref}}\right]^2\right)
\end{aligned}
$$

W_1	W_2	W_3	W_4	W_5	W_6
-0.4	0.05	0.35	-0.4	-0.2	0.6

t_{ref}	$m_{pl,ref}$	h_{ref}	$m_{p,ref}$	Q_{max}	L/D_{ref}
10 sec	5 kg	103.57 km	119.522 kg	200 kPa	19.5

Table 2: Characteristic evaluation function weights and reference values

This characteristic evaluation function was the backbone to the Pareto analysis, which included six metrics that were chosen to best represent the performance of a potential design. The metrics chosen were: burnout times for the first and second stage (t_b), payload mass (m_{pl}), altitude at apogee (h), mass of propellant for first and second stage (m_p), maximum dynamic pressure experienced (Q_{max}), and aspect ratio for the entire vehicle (L/D). Reference values were utilized in the characteristic evaluation function in order to normalize the data as best as possible, with the values used being summarized in Table 2. The weights chosen by our team as the values of W are designed to put emphasis on parameters deemed more impactful to the mission, and factors that have an unfavorable impact on the design have negative sign. In all, the characteristic evaluation function (CEF) is bound between -1 and 1, with a design performing the best with a score of 1. The overall distribution for point designs score of the CEF was modeled to be approximately normal. The reference values are modeled after Traveler IV, with the exception of t_{ref}, $m_{pl,ref}$, and Q_{max}, which were all chosen using a point estimation for the population mean of the point designs tested in the set. Parameters such as desired altitude, payload mass, and to an extent aspect ratio are all direct input parameters to the system, whereas the rest of the values are outputs from the 1DOF model.

A plot for an example batch of point designs that have been normalized within the set are shown in Figure 1, where the “value” is defined as the factors in the CEF that are positive, and the “cost” are the factors that are negative. The point designs that are colored in red are chosen as favorable designs since they have the best balance between the costs and value, whereas the blue labeled points have corresponding designs that may perform at the same value but with minimal cost. This method of screening was used for the initial selection process for viable point designs. This region of red dots is known as the Pareto Frontier, and the slope of the frontier shows a direct visual trade off of certain parameters of a point design to the overall performance.

The genetic algorithm used converged on a point design when the characteristic evaluation function was maximized for a given chamber pressure profile. In order to best prevent the program from settling on a local maximum, a few measures were implemented to find the solution which converged with the highest value which was the best approximation of a global maximum. The first generation that was run through the 1DOF model used a seed profile with a total of 7 “offspring” profiles, which were altered versions of the seed profile. Of the offspring, 4 were more conservative variations that were intended to refine the parent generation with minor adjustments, whose purpose was to converge on a local maximum of the characteristic evaluation function. The 3 other offspring profiles had much larger changes from the parent generation which are designed to bump the convergence from one local maximum to another. After the 8 profiles converged, the overall best score from the characteristic function was found, which then was chosen as the next parent

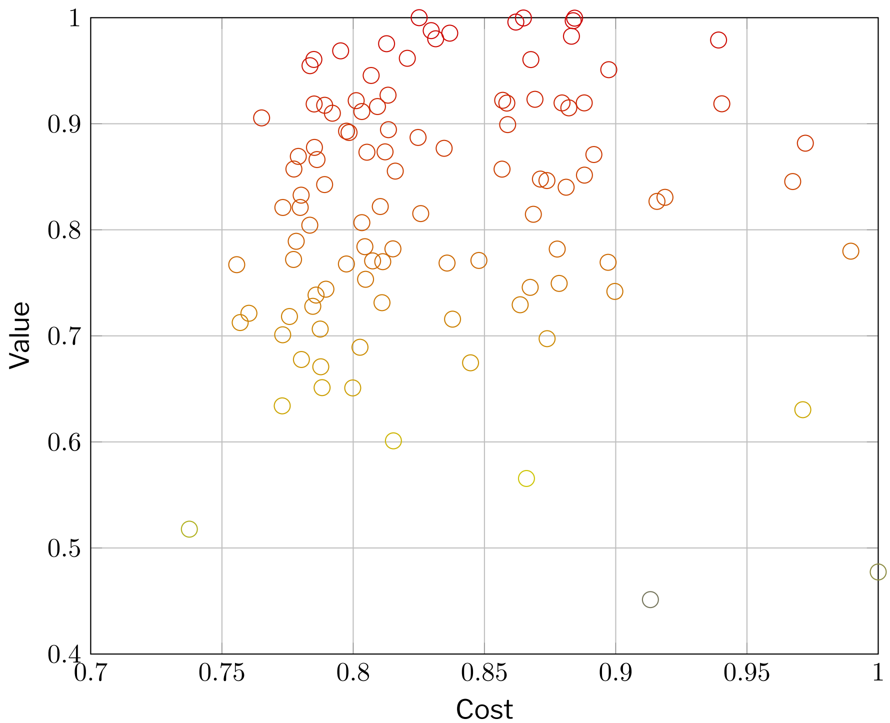

Figure 1: Pareto analysis results scatter plot

seed for the following generation. This process was repeated for the designated amount of generations by the user. A visual example of this process is shown below in Figure 2, where an example is shown after all schemes have been tested, and the darkened profile is selected as the parent profile for the next generation since it has the highest characteristic evaluation function score. The entire selection process is simplified in the flowchart in Figure 3.

3.3 One Degree of Freedom Analysis

The 1DOF utilized in this process started by initializing a few key parameters that were held constant in each subsequent point design, which included: propellant characteristics and composition, nozzle characteristics with expansion ratio, empirical estimations for propellant mass fractions, and empirical estimation for I_{sp} efficiency.

These aspects were held to be constant with one notable exception, the propellant mass fraction estimate. Using historical data provided in Figure 3.4 of "Rocket Propulsion" [4] a rough approximation was made for the propellant mass fraction, which relates the total mass of the motor to the mass of propellant. It was found later from the structures sub team, that these empirical estimations were giving values for acceptable inert masses to be less than what could be reasonably done, therefore these values had to be adjusted in order to give more inert mass to each stage so an appropriate aerostructure could be designed while fitting the designated inert mass budget. The propellant mass fraction was different for both the first and second stage, with the second stage having more inert mass accounted for, and likewise a lower inert mass fraction, due to an interstage and other factors due to the two stage nature of the vehicle.

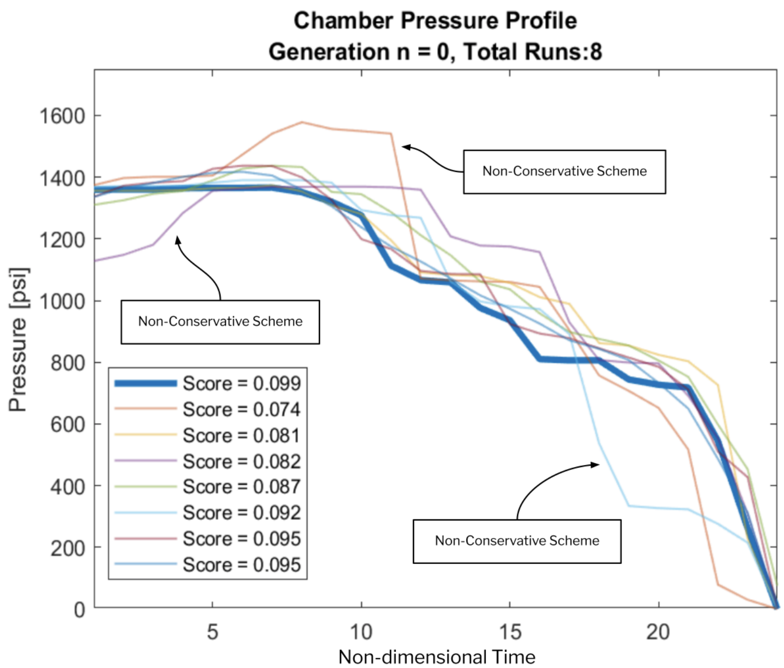

Figure 2: Example genetic algorithm run

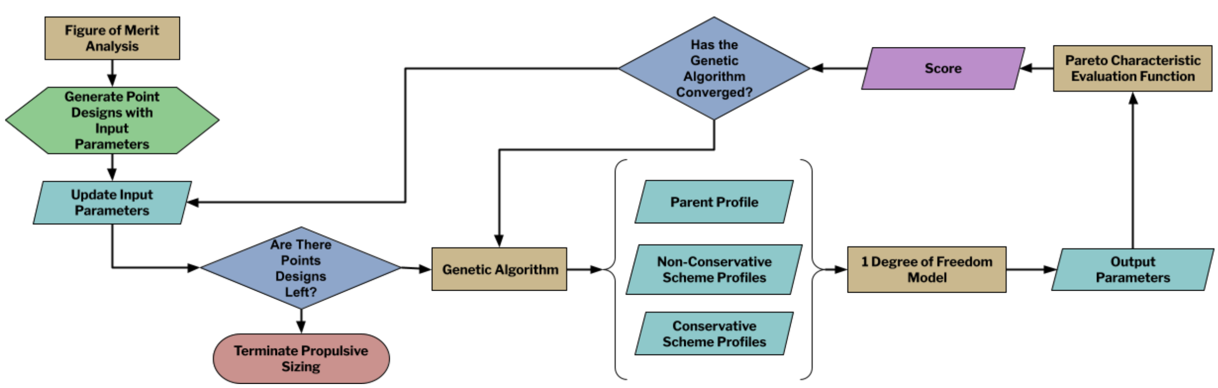

Figure 3: Genetic algorithm selection process flowchart

The propellant characteristics were not adjusted due to the limitation from faculty advisors to not develop our own proprietary propellant, which would allow for individual tailoring of different traits. This limited our total selection of propellants severely, and ultimately a propellant (TS - 78) derived from published literature from NATO was selected due to its high performance and relatively safe manufacturability. In further testing for the mission this input parameter of propellant performance will have the most direct impact on this model's results. Currently, the propellant is modeled using NASA CEA to give motor characteristics such as: exhaust velocity, chamber temperature, motor characteristic velocity, motor ISP, and exhaust static pressure. Further tests will give improved representations of these factors.

The nozzle was held constant since it added a few extra variables to the selection process, namely throat area, exit area, and therefore expansion ratio. It was decided that these factors could be adjusted after the selection process was done if need be.

In order to properly have a solution from the 1DOF, iteration was relied on heavily to fully define the variables in the system. The main variable that was iterated upon was the mission's total change in velocity (ΔV), which trickled down to a few other variables. If a value for total ΔV was estimated, then using the ideal rocket equation with the propellant mass fractions and the stage specific impulse, a value for the vehicle's mass could be found broken up between first and second stage for its inert and propellant mass. Then, once propellant mass is found for each stage, burn time per motor can be iterated upon using the mass flow rate history of each stage (derived from the chamber pressure profile, throat area, and characteristic velocity) until the total mass accumulated is equivalent to the propellant mass found with the ideal rocket equation. Following this, an atmospheric model was utilized to find aerodynamic forces on the vehicle. The flight of the vehicle was modeled using a time stepping force balance that accounted for the mass exhausted from the motor and the thrust of the motor, drag, and gravity in order to find the acceleration at a designated time. Velocity was found from the integration of acceleration, and so forth for altitude. It is key to note that all thrust was modeled to be purely axial, all aerodynamic forces were axial, and likewise with body forces from gravity. After a motor had burnt out, a coasting period was modeled, if required by the input parameters of the mission, by the same procedure just without thrust of the motor. Once separation occurred, the inert mass of the first stage was subtracted from the vehicles total mass, and the second stage followed the same procedure to find acceleration on the vehicle. This process resulted in the time history data for vehicle mass, dynamic pressure, net force, acceleration, velocity, altitude, atmospheric conditions, Mach, along with others derived from these. It is important to note that the coefficient of drag used in this model was variable and had critical Mach numbers of 0.7 and 1.3, which affected the vehicle most as it was going through Mach 1. After all of these calculations took place, the final altitude was compared to the input parameter for the desired altitude, and if the margin of error was not met, the process would be repeated with an updated delta V for the mission. Therefore this method can be thought of as a modified ideal rocket solution, since at its core it uses the ideal rocket equation to find the mass of the vehicle, but it iterates on this value to find a solution that incorporates forces other than the vehicle's thrust.

3.4 Mass Estimation and Sizing System

Once the propulsion team generated point designs using ranges of parameters, it was decided that further steps were necessary to visualize and analyze the selected designs, as well as generate the inputs needed to evaluate the designs using the 6 degree of freedom (6DOF) model, which will be discussed in detail in Section 3.5. The goal of this script is to determine whether point designs can fit the propulsion analysis' inert mass requirement, while doing preliminary structural analysis to ensure these materials and geometries pass minimum safety factor requirements.

3.4.1 General Operation and Information

The program takes inputs from the Pareto analysis, as well as from preliminary mass and location estimates for the vehicle subsystems. Using the inputs gathered from these sources, the program then does the basic geometric layout of the rocket, generating lengths, wall thicknesses, and a design that can then be visualized with the help of tools such as OpenRocket and CAD software. The program also calculates key physical characteristics of the rocket, the center of mass and mass moment of inertia over the duration of the flight.

In order to simplify analysis, some assumptions about the rocket were made, which are covered in Table 3. The design generated contains the position and mass of all aerostructures and internal components from both stages. These include the nosecone, the sustainer airframe, the sustainer fins, the interstage, the booster airframe, and the booster fins. Internal components include the motor, the forward closure, the nozzle, the recovery subsystem, the despin subsystem, and the avionics subsystem.

The program also performs primary column buckling and local column buckling analysis on the sustainer airframe. This is done assuming the airframe is a fixed-free column; however this will change once the detailed design of the rocket is done. Currently this check is simply a “sanity check” so to speak, just as a qualifier for further analysis on each point design.

3.4.2 Analysis

The Pareto analysis provided the diameters of both stages, the maximum expected operating pressure (MEOP) of both motors, the mass of the motors as a function of time, the maximum drag force, and the maximum acceleration of the rocket. Additional inputs were also given in the form of material properties, and geometric properties.

The majority of the analysis was performed on the airframes, as the dimensions and stability of the airframe drive the viability of the point design.

Starting with the airframes, wall thickness of each stage was calculated using the MEOP of both motors and the thin wall hoop stress formula. The length of the motor is calculated using the propellant mass, the propellant density, and the internal diameter of the airframes. The forward closure was modeled as a uniformly loaded circular disk with clamped edges, and thickness was backsolved from the corresponding formula. The length was calculated by simply adding the length of the motor, the internals, and the bulkheads together, with some amount of room for error. The mass of the airframe and the bulkheads was calculated by calculating the volume of each and multiplying by material density. The script has three types of column buckling included, with two relevant criteria

Current Assumption	**Justification**
All point designs are sub-minimum diameter	The sub-minimum design, when compared to their minimum, and other counterparts for such high powered applications, offered more benefits in mass and space saving.
All point designs use metallic airframes	Sub-minimum diameter rockets require their airframes to be pressure vessels, and the team does not have the capability to make composite overwrapped pressure vessels in-house.
All point designs use four fins for each stage	This assumption was made in order to perform first-order analysis; in the future, fin characteristics will be optimized by the 6DOF and prior art.
All point designs have a 5:1 Von Karman nosecone	Based on prior art; for other high powered rockets the 5:1 Von Karman design was commonly used.
Point designs do not have igniters, RF-transparent sections, nozzles, fasteners, or couplers	These components are a part of the detailed design, as such to reduce complexity, they were left out of the analysis.
All point designs' internal rocket components' dimensions and masses are static	These components are a part of the detailed design, as such to reduce complexity, they were left static in this analysis
All point designs' internal components are modeled as cylinders	In order to simplify for the first order analysis, as well as generalize for the wide range of point designs generated, the internal subteams provided us with simplified representations of their sub-systems

Table 3: Mass estimation and sizing assumptions

Variable	Minimum Value	Maximum Value
Inert Mass	14.39 kg	18.39 kg
Propellant Mass	28.73 kg	34.58 kg
Stage 1 Diameter	4.5 in	5.5 in
Stage 2 Diameter	4.0 in	4.75 in
Burn Times	6.1 sec	9.1 sec
Inert Mass Fraction	63%	67%
Delta V Split	35%	42.5%
Max. Mach Number	6.5	8.0

Table 4: Design space determined by the structures sizing process

per airframe. The first, the primary buckling instability can be modeled by either the Euler or Johnson buckling formulas. The script chooses one over the other based on the slenderness ratio of the airframe and calculates the associated buckling stress. The third form of analysis is local buckling. The formula was obtained from NASA's SP 8007 manual [1], and used to calculate the critical local buckling stress.

To estimate inert masses, a simple method was used. For aerostructures, we calculated the volume of each component and multiplied by material density. For the internal components, the internal subteams were consulted on generalized masses and dimensions for each sub system that could be applied to all point designs. The motor mass was given by the Pareto analysis outputs, and is represented as a function of mass over the time duration of the flight.

The script calculates the center of mass of the rocket as a function of time, based on the motor mass time history, and was verified at the beginning and end of motor burns using OpenRocket. It also calculates the moment of inertia of the vehicle in its three principal dimensions. This was verified using SolidWorks models of point designs.

3.4.3 Results

After the Pareto analysis determined a significant number of viable point designs, the structures script was used to further narrow down this number. Most point designs were discarded because they could not meet inert mass requirements, while others were deemed to have motors that would be beyond our team's manufacturing capabilities. We used the 1DOF model to simulate all designs with an altitude ceiling of 150 km, however, the 6DOF model was not ready to validate or update this parameter. 6DOF validation of these point designs will likely bring down their flight ceiling significantly, and the extra altitude provides a mass budget excess that can be used in detailed design.

The materials and geometries used in the conceptual designs are primarily outlined in Section 7, but the materials (aluminum, titanium, steel) were chosen for their availability, manufacturability, thermal resistance, and strength, while geometries were chosen based on prior art and manufacturability. Following the structures analysis and its simplifying assumptions, the design space determined viable point designs were found within the bounds in Table 4.

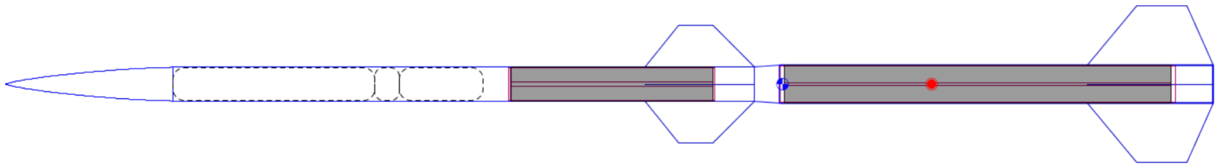

Figure 4: The 4"-4.5" design, shown in its expendable configuration

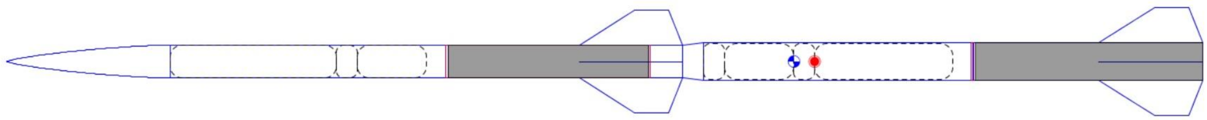

Figure 5: The 4.75"-5.5" design, which can reasonably be flown with either a recoverable or expendable first stage

As stated in SR.4, the team wishes to have an expendable first stage. This appears viable examining prior spaceshot launches, but confirmation is needed. Sizing has established that viable point designs exist with first stages that are not recoverable, such as the 4.5 to 4 inch rocket in Figure 4. Additionally, viable point designs exist for fully recoverable rockets, such as the 5.5 to 4.75 inch rocket in Figure 5.

Throughout the rest of this report the work required to arrive at this design space will be presented. All of the work requiring dimensions and masses was done with the 4 to 4.5 inch expendable rocket, as this design represents the team's Minimum Viable Product (MVP). Once the 6DOF can evaluate the design space, then additional analysis will be performed to ensure that the assumptions and Pareto analysis still hold true.

3.5 Trajectory Analysis Model and Statistical Methods (6DOF)

4 Propulsion

4.1 Introduction

The key responsibility of the propulsion system is to propel the rocket to the Kármán line utilizing two separate stages, each having their own individual motor (PRO.1 and PRO.2). Quantitative requirements that each motor will need to fulfill are being determined through Pareto analysis and the 6DOF model. The two point designs and the results of our analysis determine the specific aspects of the propulsion system. The team will thoroughly verify the ability of the propulsion system to complete the mission through simulation and testing. Each stage will contain its own ignition motor made of the same formulation as the main motor. While the first stage ignition will be manually activated by the mission control room, the second stage will be ignited by the avionics system on the second stage.

The formulation for each of the rocket motors and igniters is based on NATO propellant research [5] and will be carefully mixed and manufactured at Purdue University's propulsion laboratory — Maurice J. Zucrow Laboratories. Through direct coordination with Zucrow, the propulsion team will construct a robust test stand in order to evaluate the performance of propulsion mechanisms and their interactions with other systems.

Throughout the research, design, manufacturing, and testing process, the team recognizes that it is paramount that safety is placed first, and is taking proper precautions to ensure this. We realize the inherent dangers of working with solid propellants and will work with experienced researchers to ensure the process is as safe as possible.

4.2 Performance

Two different point designs are currently being considered for the two-stage propulsion system based on system integration between the first and second stages. Currently, the primary difference between each design is the motor diameter. Variations and optimizations of the fuel grain geometry, the shape of the burning surface of the solid motor, will be determined after a point design is selected. Each point design was simulated based on each motor utilizing a BATES grain geometry (shown in Figure 6) with sub-minimum motor diameter. The quantitative requirements of performance for each design to achieve its mission were found through the thrust profiles from the 1DOF model and ΔV outputs from the Pareto analysis. The first design considered has a first stage with an external diameter of 4.5 inches and a second stage with an external diameter of 4 inches. The second design consideration has a first stage with an external diameter of 5.5 inches and a second stage with an external diameter of 4.75 inches. The performance of each design is displayed in.

4.3 Ignition

4.4 Manufacturing

4.5 Testing

4.6 Analysis and Simulation

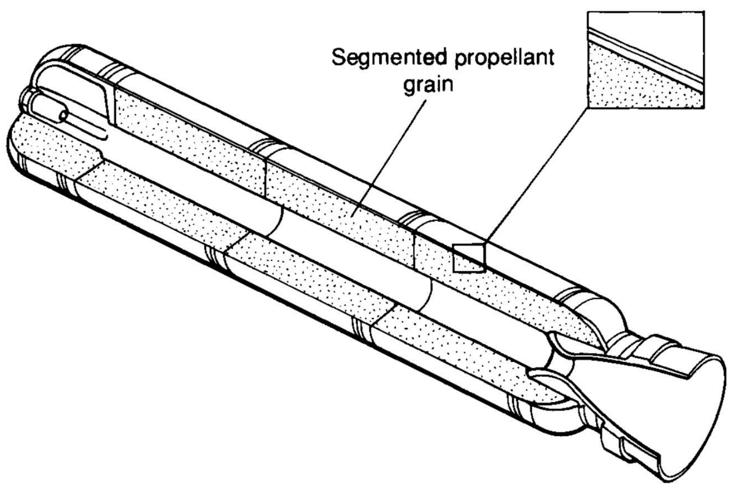

Figure 6: Section view of a segmented BATES grain (from [7])

5 Avionics

5.1 State Estimation and Apogee Determination

Considering the team's overarching goal of reaching the altitude of 100 km, an important responsibility of the avionics system is to verify the apogee reached by the vehicle. The objective of the flight will not be achieved without high confidence that the vehicle has reached the target altitude. Being able to verify the rocket's apogee (AVI.1) will be achieved through a culmination of different data in order to account for the highly dynamic environment throughout the spaceshot. During the flight, the avionics system will record several different datastreams, and an algorithm will combine all the data collected to estimate the state of the rocket (AVI.1.1). This data will be stored onboard and recovered to allow a more accurate apogee verification after the flight (AVI.1.2).

5.1.1 System Architecture

The avionics system will consist of both Commercial Off-The-Shelf (COTS) and Student Researched and Developed (SRAD) components. There will be both a primary and backup flight computer for redundancy on the vehicle. The current plan is to use the SRAD flight computer as the primary flight controller, with one or more COTS computers as backups. This setup is preferred because the SRAD computer allows for more customization of hardware and software to better match the overall profile of the flight. The SRAD computer will gather and store data from multiple sensors, estimate the rocket's current state, trigger staging and recovery events, and possibly transmit data to the ground. It will also allow all the raw sensor data to be stored for analysis on the ground. However, extensive development and testing of the SRAD flight computer is still in progress. It may be determined after testing that higher customization of the SRAD board does not outweight the relaibility and flight heritage of commercially available options. In this case, the SRAD system will primarily record data for post-flight analysis, while a COTS computer will be the primary controller for the flight.

One COTS flight computer option under consideration is the AltusMetrum Telemega[5]. This computer includes a GPS, Inertial Measurement Unit (IMU), and barometer. It has the ability to trigger staging and recovery events, based on altitude and attitude data, only triggering flight events when the programmed conditions are met. It also can transmit telemetry to the ground. However, it does have limitations. Due to COCOM regulations[6], its GPS will provide data at altitudes greater than 50 km or velocities greater than 500 m/s. While this computer has Kalman filtering and could work on a 100 km flight, it has many limitations, as is further discussed in Section 5.1.2. Finally, its telemetry downlink antenna only has a range of about 12 km, which is much less than the distance traveled throughout the flight.

By using the SRAD flight computer, GPS data could be obtained up to 80 km[7], and a more precise accelerometer and barometer could be used. It would also have custom software to match the mission of the rocket. This includes more specific conditions for triggering events, an improved

[5] The Altus Metrum Owner's Manual can be found at this link.

[6] Due to government regulations, commercially available GPS units will not provide data while they are above 60,000 ft (18 km) MSL and traveling faster than 1000 knots (500 m/s). Some manufacturers enforce both limits, while others only use one, and some use the speed limit with varying height limits.

[7] The ublox MAX-M10S is capable of this altitude, per its data sheet.

Kalman filter that uses the predicted mass and aerodynamic profile of the rocket, and storing all the raw sensor data and GPS satellite pings for post-flight analysis.

The discussion until now has been about the avionics system on the second stage. If the first stage of the vehicle is expended, it will require no avionics system, but if is recovered, it will have its own avionics bay, which would likely be entirely COTS components to reduce complexity. A COTS system onboard the first stage would serve the simple purpose of parachute deployment. Because of the lower altitude, flight speed, and reduced flight complexity, an SRAD system would be unnecessary, and only serve to increase complexity and chance of failure. Whether or not the first stage is expended, the stage separation and ignition will be controlled by the second stage avionics system. Therefore, regardless of which design is used for the space shot, the design of the second stage avionics system will be unaffected.

5.1.2 Data Fusion Algorithm

In order to provide telemetry and trigger flight events, the SRAD flight computer must know the current state of the rocket throughout the flight. “State” includes the rocket’s position, velocity, and attitude on all 3 axes. To do this, the computer will take measurements from three different sensors: a 6-axis IMU, a barometric pressure sensor, and a GPS. These sensors all have their own benefits and limitations, and their individual reliability will change throughout the flight. While GPS might provide the most accurate measurement of position, it will not be available for the entirety of the flight. Due to COCOM regulations, the GPS receiver will not provide data when moving faster than 500 m/s and/or above a set altitude, dependent on the specific module. Even without these restrictions, a GPS may have trouble obtaining a lock during the highest-speed portions of the flight. Additionally, there could be radio frequency (RF) interference from parts of the rocket, limiting the GPS signal transmission. We are aware of rockets operating in similar flight regimes experiencing these issues; USC’s Traveler IV spaceshot did not have a GPS lock for the majority of the flight, and Derek Deville’s Qu8k, despite carrying four different GPS modules, failed to maintain a lock.

The other two data streams are flawed as well. A pressure sensor, sampling the ambient air pressure, can be used to estimate the rocket’s current altitude. This requires adding holes to the airframe to equalize the pressure in the avionics bay with the external ambient pressure. However, shock waves from traveling at supersonic speeds can interfere with the ambient pressure readings. Additionally, pressure readings will become ineffective once the atmospheric pressure decreases below 10 mbar (25-30 km MSL). The 6-axis IMU measures acceleration and rotational rates. This data will need to be integrated to get a measurement of state, and this causes error to build up over time. Finally, all of these sensor measurements will have some level of noise which will decrease their precision.

In order to obtain the best state estimate from multiple measurement sources, a sensor fusion algorithm is needed. This sensor fusion algorithm will be some version of a Kalman filter, likely an Extended Kalman Filter (EKF). A Kalman filter combines a measurement with a predicted value, and outputs the estimated state along with a measurement of the uncertainty. An EKF works for non-linear systems, which include systems with multiple measurement inputs. Although some COTS computers (such as the Telemega) use Kalman filtering, they are limited since they do not have any information about the flight profile or the aerodynamic characteristics of the rocket, so they fall back on a simpler physical model. An EKF run on the SRAD computer can provide a more accurate estimate, because the state update function can account for the thrust, mass, and drag profile of the rocket. This sensor fusion algorithm is currently being developed and tested using MATLAB and Simulink, as discussed in the testing section.

In addition to the onboard state estimation, the raw sensor data and GPS satellite pings from the flight will be saved for post-flight analysis on the ground. The data will be analyzed with more accurate methods that may be too computationally expensive to run during the flight. This will allow the apogee to be estimated with more precision.

5.1.3 Testing

The flight computer's software algorithms are currently being developed in MATLAB Simulink, and will go through software in the loop testing in Simulink to validate them prior to flight. First, the 6DOF simulation will provide a theoretical flight path of the rocket. State data from this flight path is then converted into sensor data that would be generated by the computer's sensors. This includes adding noise to the sensor outputs and matching the frequency and range of the sensor. This simulated data is then sent to the Simulink flight computer prototype. The flight computer's state output can then be compared to the simulation output to judge the accuracy of the state estimation algorithm. This allows changes to be made to the algorithm to find optimize the estimate. This method will also be used to test the flight logic. Many potential failure modes can be tested to verify that the flight computer makes the correct decision in each scenario.

For verifying firmware functionality, we will unit test all sensor and hardware drivers extensively using fakes, which are essentially simplified software models of the corresponding devices. Unit tests will be written both for native desktop hardware as well as the target boards to balance ease of development and accuracy. Unit tests will use a hardware abstraction layer that calls into faked models of the hardware to allow seamless testing without hardware being available. We are also currently investigating doing hardware in the loop testing. However, due to the complexity of implementing such a system and the diminishing increase in failure scenarios being tested, we intend to pursue this only if we have extra time, and will not make this a limiting goal on our timeline.

Finally, the flight computer will be tested as a passive data recorder on many flights before it is the primary computer on any rocket. Its state estimation will then be compared to data from other COTS flight computers on the rocket to determine its accuracy.

5.2 In-Flight Events

As previously discussed, the spaceshot vehicle will have two stages with a recoverable upper stage. During the flight, this requires a stage separation and ignition followed later by a recovery system deployment. The avionics system will be in charge of sending the activations for these flight events (AVI.2 and AVI.3). The activation conditions of these events need to be carefully verified before the rocket is permitted to proceed to the next stage of flight (AVI.2.1, AVI.3.1, MEC.3, and PRO.2.5), and this process will be completed by the avionics' onboard state estimation algorithm. After this verification occurs, the signal will be sent to activate the relevant subsystems on the rocket (AVI.2.2 and AVI.3.2).

In order to ensure the flight events happen in the correct order, the SRAD flight computer will keep track of the current phase of flight. These phases can only happen in the intended order. Transitions between phases will be triggered when conditions using the rocket's position or acceleration data are met. These phases include: on the pad; first stage burn; first stage coast; second stage burn; second stage coast; and descent. There are specific conditions that must be met for transitioning between phases and triggering events which are based on altitude, acceleration, and/or attitude. For example, the second stage can only be ignited when the rocket is pointing less than a set angle from vertical. If this condition isn't met, there is an option for a second stage abort scenario. A more detailed summary and flowchart of phases and transition criteria is available in Appendix D. This flight sequence is modeled using Simulink and Stateflow, which can be compiled into C and run on the flight computer.

5.2.1 Safety

To ensure the safety of everyone working on the rocket, we intend to employ several subsystems that will ensure that energetic materials are not inadvertently triggered in unsafe situations or before they are expected to.

In order to prevent triggering of the motor or recovery hardware while the rocket is assembled, all avionics hardware will have a hardware power cutoff controlled by switches that will not be enabled until the rocket is vertical. We are currently exploring two primary cutoff schemes. The first is a WiFi switch, which can be opened and closed over a wireless connection. This is a device we have used in the past, and it has performed very reliably. However, the switch does not automatically give an indication of its state. A simple solution might be to wire a buzzer in-line. The second option that we are exploring is a mechanical pin, which arms the avionics when pulled out of the rocket. This might require a larger mechanism, along with a hole in the airframe, but it is conceptually simpler than the WiFi switch, and it gives a clear visual indication of its state.

Two systems to arm the flight computer that we are not pursuing are magnetic switches and key switches. On previous launches, we have used magnetic switches, but they have shown consistent reliability issues. Key switches embedded in the body of the rocket come with many of the benefits of the pin, but there is more structural complexity. Additionally, if the keyway protrudes from the vehicle at all, there are aerodynamic concerns.

We also plan to include software or hardware lockout timers that will ensure that staging or recovery doesn't occur during early stages of the flight where potentially anomalous sensor data is expected (high acceleration and transonic regimes). On SRAD flight computers and programmable COTS boards, we intend to use time based software lockouts as we believe that the primary points of failure are the state estimation algorithms, which a software lockout at the flight logic level should be able to mitigate.

Finally, a similar system will be used to disable send stage motor ignition after a certain time period. This is to ensure that in the case of a second stage abort, it is safe to later approach and recover the rocket without fear of the second stage motor (which would still be loaded) accidentally being ignited. To minimize complexity, we intend to have an electromechanical system trigger a digital timer on launch that then cuts off a relay or transistor after the prescribed time (i.e. the system is prevented from igniting the second stage motor after five minutes have passed beyond the detected launch).

5.3 Downlink

It is critical that the team is able to locate the rocket after the flight is completed and the rocket has touched down (AVI.4). In order to achieve this, the rocket will use GPS data so the recovery crew can have accurate coordinates of the landed vehicle (AVI.4.1). Due to potential GPS limitations, the rocket will transmit the GPS data throughout the flight as it is received (AVI.4.2 and STR.7). This will allow the recovery crew to have accurate GPS readings of position and velocity for as long as possible, being able to estimate the final ground location if needed.

While it would be ideal to have live telemetry throughout the entire flight, it is most important to have a signal on the descent portion so that the rocket can be tracked for recovery. One significant challenge of live telemetry is getting a RF signal out of the airframe, which is likely to be metal. To solve this problem, the avionics bay will be inside an RF transparent section of the airframe. The length of this section of airframe will likely need to be at least the length of any antennas inside of it. This will also allow the GPS receivers to receive GPS signals. Even with this section, there may be other sources of interference to the antennas during flight. Also, the orientation of the antennas may affect their signal range. To ensure a strong signal on the descent, an antenna may be attached to the parachute or shock cord so that it leaves the body of the rocket after parachute deployment.

There are multiple COTS GPS modules that can transmit telemetry to the ground. One option is the Multitronix Kate-3 GPS Tracking System. This system apparently does not have a GPS altitude limit, but it does still have the COCOM speed limit of 500 m/s. It also includes an accelerometer, but this has a limit of 50G, which may be less than the forces experienced during flight. Because of this and its lack of a barometer, it is currently not planned to be used for triggering staging and recovery events. However, it may be used as the primary telemetry system since it can transmit telemetry to a range of 150 km.

On the other hand, an SRAD solution could be more adaptable to our specific needs. On the rocket, we could transmit telemetry using simple FM transmitter modules like the popular Radiometrix HX1 (but likely at higher frequencies) used in high-altitude balloons, or higher power XBee modules. We could experiment and research a wider range of frequencies, protocols, and data rates in order to find a solution that maximizes range and reliability. This would also give us the options to test different antennas to maximize range while decreasing space use. Antennas will need further research and guidance as they have many tradeoffs, especially in the rocket. We want to try to maximize range, but the rocket's antennas should also be omnidirectional to allow reception in any position the rocket may land. Additionally, we should try to limit the size of the RF transmissible airframe section, since it will decrease the overall structural integrity of the vehicle. On the ground we could use a combination of omnidirectional and directional antennas to allow flexibility in how we track the rocket. This could include a simple custom ground station with a software defined radio (SDR). This custom ground station would make it easier to use different antennas and also allow us to tune into different frequencies.

5.4 Payload

To satisfy our stakeholder requirements (SR.5), the spaceshot vehicle will have a payload, which will not be essential to the successful flight of the vehicle. The payload will include a camera, but if there is more available mass and volume, we hope to include additional items. To support additional payload mass, as well as general overruns in component design, the vehicle is being sized with an apogee of 150km, well above our true target.

The camera system will, at a minimum, consist of a single camera looking radially out of the second stage. This imposes a requirement on the vehicle to de-spin if it is spin stabilized, so that good imagery can be captured (MEC.1). The detailed design of the camera bay is beyond the scope of this review, but we expect the hole cut in the rocket to remain uncovered, as opposed to being blocked by a transparent window. The specific model of camera to be used will be based primarily on reliability and flight heritage; based on a brief study of comparable-performance amateur rockets, GoPro cameras seem to be the leading candidate.

If the payload subsystem is allocated more mass and volume than a single camera requires, we plan to add additional components to the payload. Ideas under consideration include

- A camera inside the recovery bay, watching the deployment of the parachute
- A thermal camera inside the nosecone, to characterize the thermal loading
- A collection of COTS avionics boards, so we can later publish their performance on such an extreme flight
- A biological experiment, as minimal as a Petri dish, to explore the effects of a zero-g environment
- A LEGO Minifigure of the Star Wars character Mace Windu, which has flown on all previous HA flights

The specific components of the payload subsystem will be determined by PDR, once the actual mass and volume constraints are solidified.

5.5 Durability

While actively collecting, calculating, and transmitting data, the rocket will undergo very high accelerations and pressures. The avionics systems need to be able to survive these intensities throughout the flight with a successful recovery (AVI.5). To ensure this survivability, the system will be prepared to withstand certain forces, vibrations, and thermal environments (AVI.5.1, AVI.5.2, and AVI.5.3).

For the design of the SRAD board, we do not expect to have major problems with structural loading, since soldered and especially surface mounted (SMT) components should be able to handle forces well beyond what the rocket is expected to experience. However, we are considering potting the SRAD board to ensure any larger components like capacitors do not break off. We need to further investigate the viability and benefit of doing this. One potential issue with potting the board is that heat produced by the board may be trapped, leading to it overheating. However, we expect the power draw of our board to be low, so this issue is unlikely to be a problem. Another concern is that the barometer may get blocked from reading the ambient pressure. This issue requires more investigation into the specifics of potting.

For testing performance under acceleration, we are planning a test in which we will attach the board to the wheel of a vehicle, and drive it to simulate a 200g load on the board (the maximum our planned linear accelerometer would be able to measure). This acceleration is likely much higher than any sustained forces during the flight, but there should be little to no cost to additionally verifying that the board can work up to those accelerations.

The biggest problem we anticipate is vibration causing breakage of connectors and wires going to and from the board. Our current plan of mitigating this is soldering as many connections as possible before assembly, and using latching connectors for any final connections. In addition, for the final flight we will try to use SMT components as much as possible, including for the storage media. We currently do not have a good way of testing the impact of this problem or the efficacy of our solutions, but we will continue to look into leveraging resources at Purdue and beyond to determine if we can perform useful vibration testing.

The final consideration related to avionics durability is thermal resilience. Many of the components we will be using have a maximum temperature rating as low as 85 degrees Celsius. Early thermal estimates show internal air temperatures in the avionics bay exceeding that value. It is not possible for us to take any action on this front until the structural design of the vehicle and the thermal analyses become more developed.

6 Mechanisms

6.1 Recovery System

As discussed earlier in this report, we are currently unsure whether the first stage of the spaceshot vehicle will be expended. For this reason, we are designing systems to safely recover both stages (MEC.2), with the understanding that the first stage recovery system may be removed in the future.

For the second stage of the rocket, descent speeds are not to exceed 50 feet per second above 1000 feet in altitude, and must not exceed 20 feet per second below 1000 feet to keep the more sensitive structures of the vehicle intact (MEC.2.2). Under the assumption that the booster stage also needs to be recovered, it has a similar set of requirements, though the allowable velocities will be some amount higher since it is expected that the first stage will be more durable and more able to withstand a hard landing (MEC.2.1). Depending on the booster recovery design selected, this velocity will likely range between 20 and 40 feet per second.

The recovery systems for each stage will deploy from a single airframe separation in each of their respective airframes (MEC.2.1.2 and MEC.2.2.2). Prior to separation, each of the airframe sections will be held together through the use of nylon shear pins. The booster recovery system will deploy from a break in the airframe between the motor and the staging interface. The sustainer recovery system will deploy from a break above avionics and below the nosecone. Due to the high altitude of both recovery deployments, a simple separation mechanism will be used to break apart the airframe (MEC.2.1.2.2 and MEC.2.2.2.2). The specifics for this device are found later in Section 6.2.

6.1.1 First Stage Recovery

Two minimal recovery designs for the first stage are being considered, in case it is determined that the first stage must not be expended. Line diagrams for all the recovery schemes discussed in this report are provided in Appendix C, for further reference. The first design consists exclusively of a separation mechanism, one line of shock cord approximately twice the length of the booster, and a mid-sized parachute. This option is intended to be extremely compact and lightweight to allow for minimal design work by the structures and avionics teams. As only a single chute is used for this design, descent speeds will be much higher than the ideal 15 to 20 feet per second below 1000 feet. Descent speeds will likely range between 30 and 40 feet per second for the entirety of the descent to prevent excessive drift from the stage's apogee. The landing impact for the booster will be more forceful than what a typical recovery system would result in, but we currently expect the booster to be tough enough to withstand the conditions without severe damage. Since no valuable payload is onboard, this rough landing is deemed acceptable.

The second parachute option we are considering is a "Sombrero". This device is a mini-parachute attached to the canopy cords, which is able to freely slide up and down, allowing the main chute to unfurl roughly depending on its position. Early estimates indicate this system could use only five times the rocket length in shock cord compared to the nine times used by a conventional drogue-main setup. Acquiring one may be difficult, as the "Sombrero" is a patented device created by Butler Parachutes. They do offer custom orders for parachute systems, but pricing is unknown and requires further inquiry. Another option would be to manufacture a similar system in-house, but that leads to risk of the device being of poor quality and failing during use. This parachute system offers a great way to reduce mass and volume, but it is difficult to source or manufacture.

While the two recovery systems are similar, they each pose their own advantages and disadvantages. The single chute alone is by far the simplest, most reliable, lightest, and cheapest method, but it has the booster landing at a high velocity of 30 to 40 feet per second. This will undoubtedly damage the booster to some extent, but it is likely to remain in one piece. Also, it is not likely to bury itself in the launch range, allowing for it to be easily located and retrieved. The sombrero avoids a hard impact, but it is significantly more difficult to manufacture, is far more expensive, and adds more mass to the rocket. Further research will be conducted into both systems to choose an ideal option.

6.1.2 Second Stage Recovery

There are also two options for second stage recovery systems. Line diagrams for these options are provided in Appendix C. The first option is the most massive as it requires 8-9 times the length of the sustainer in shock cord. However, currently it is deemed the most reliable method, and the team has the most experience with this system as similar recovery designs have been used for two of our previous flights. This method uses a traditional drogue-to-main parachute scheme. At apogee, the separation mechanism separates the single sustainer airframe break and deploys the drogue parachute, also releasing the main parachute bundled in a parachute bag. A single line of shock cord will run from the drogue parachute to the rocket body where it is attached by use of a tender descender. A tender descender is a simple device that holds two quick links together until a black powder charge that was previously contained within the device detonates in response to a signal from the avionics computers. This line connecting the drogue to the tender descender will carry all of the chute tension above 1000 feet in altitude. During descent, the drogue will slow the rocket down to acceptable speeds around 50 feet per second before the system reaches 1000 feet in altitude. Shock packs, various devices that reduce the high forces of parachute deployment or snatch force, will be implemented throughout the system to limit stress on mechanical mounting structures during descent.

Running parallel to the tender descender to drogue shock cord during the entirety of the descent above 1000 feet will be a secondary set of shock cords which connect the main chute to both the drogue chute and the rocket body. This shock cord will use a different mounting location than the tender descender. Above 1000 feet, this system will be in slack, and the main parachute will be contained within its bag. At 1000 feet, the tender descender holding the force of the drogue chute will separate, transferring the path of tension through the shock cord connected to the main chute. This tension will rapidly remove the main chute from its bag as the rocket body falls away from the drogue parachute. As the path of tension will transfer from the rocket, to the main, and then to the drogue, the upward force of the drogue will help to quickly inflate the main parachute. The deployed main parachute will then bring the rocket descent speed down to a gentle 15 to 20 feet per second.

For the sustainer stage recovery, the second recovery option that is being researched is reefing parachutes. This system consists of a main chute that can be variably deployed by temporarily fixing the shroud line length via a reefing ring. Essentially, this system allows the single parachute to act both as the drogue and the main chute. This system would greatly decrease the amount of shock cord required to roughly four times the length of the rocket and entirely eliminate the need for a drogue chute, saving valuable mass and volume. Also, if only one chute is used, it would reduce the snatch force the rocket would experience, compared to deploying a chute in the standard fashion. However, as a team, we lack experience using this system and creating or sourcing a reefed chute has an unknown price tag or level of reliability.

After researching several reefing options, the following are being considered:

1. Single reefing ring: simplest option, but not too predictable especially at high altitudes.
2. Double reefing ring configuration: Using friction, we can delay the opening process of the parachute. We still need to further study this option.
3. Active reefing process: Using a triggered event from avionics to loosen or drop the reefing ring, allowing the chute to expand from a drogue to a main at a specified altitude. This option requires significantly more research.

As it contains 8 to 9 times the length of the rocket in shock cord, two chutes and a tender descender, the first recovery method mentioned, is by far the heaviest and least space-efficient option. However, this is the option the team has the most experience with, is more reliable, and is easier to manufacture and assemble. The reefing method will take up approximately half of the volume of the first option, immediately making it a strong contender. Various methods of reefing will continue to be explored and tested, but at this time the reliability and feasibility of this much more compact method is still in question.

6.2 Separation Mechanism

Due to the high altitude required of the recovery deployments for both stages of this vehicle, traditional black powder airframe separations are not viable. These simple systems require atmospheric pressure to both fill the airframe with compressed gas and provide a medium for heat transfer to ignite nearly all of the black powder used in the initial charge. In the low pressure environments in which this rocket will require an airframe separation, another solution is needed (MEC.2.1.2.2 and MEC.2.2.2.2). As recovery systems using tender descenders will not require their use above 1000 feet in altitude, the operation of tender descenders will not be impacted by these same concerns.

Many rockets flying at these high altitudes use a compressed CO_2 separation system for airframe separation. This technique was researched, but CO_2 canisters and the various systems used to release the gas and separate the airframe are rather bulky, heavy, and relatively complicated which could lead to various points of failure. Since two separation systems could be required for the rocket, this method was deemed unsatisfactory. Instead, a simple black powder piston-like device was designed. A preliminary CAD model of this device is shown in Figure 7. This device consists of a mounting plate, a compartment of O-ring sealed atmosphere within a piston like chamber, a pushing plate, and a firebolt.

A firebolt is simply a bolt that has had a hole drilled through it to allow for the passage of e-match electronic ignitors through the bolt. These e-matches are then sealed within the bolt through the use of epoxy and a small sealed container of about 0.5 grams to 1 gram of black powder is placed around the ignitor tip of the e-matches. The exact amount of black powder will be determined through on the ground testing of the system long before launch. The threads of the bolt can be wrapped with pipe sealant and placed within the device to form a proper seal, leaving the black powder on the

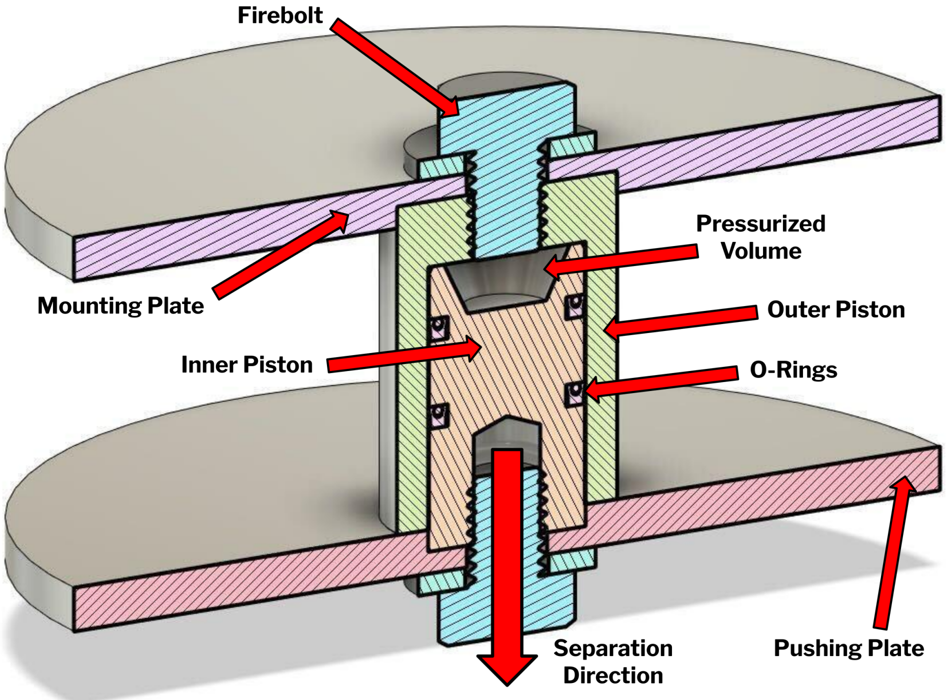

Figure 7: CAD model of the preliminary piston-based separation device

inside of the separation mechanism and the e-match leads on the other end. These leads will be connected to the avionics computers which will then be able to ignite the black powder at a specified time or altitude. As the black powder is contained within a small pocket of atmospheric pressure at any altitude, proper ignition is possible and the separation mechanism will separate itself and the airframe of the rocket in the process.

To actually achieve separation of the airframe, the black powder ignition will force the inner piston upwards relative to the rocket. This movement will be transferred to the airframe coupling through the use of the pushing plate. Since the coupling is attached to the upper half of the airframe and the mounting plate of the separation mechanism is attached to the lower airframe, this force will push the airframe apart.

Two separation mechanisms will be included in the entire rocket if the first stage is required to be recovered: one in the booster stage and one in the sustainer. In both cases, the separation mechanism will be located directly below the parachute bay and above the avionics bay. This allows for easy and safe wiring of the firebolt to the avionics computers. During final rocket assembly, the connection of the two halves of the piston will be the final step, leaving any accidental ignition of the black powder free to expand safely without causing harm to the assembly team or equipment.

6.3 Inter-Stage Mechanism

An essential part of the rocket's flight is the clen separation of the two stages between their burns. Since a failure to separate would very likely lead to catastrophic mission failure, it was decided that a stage separation mechanism would need to be developedto satisfy MEC.3.

After preliminary development and research, we concluded that the main separation mechanism should be as simple as possible. The optimal time to separate, based on minimzing the loss of velocity due to drag, is when the two stages would naturally drag-separate, i.e. the drag on the first stage "pulls" it off the second stage. However, we are unable to simulate the complex aerodynamics of two free bodies in proximity. Thus, the separation scheme we propose is as follows.

1. The first stage burns out
2. The optimal time of separation occurs. Depending on the aerodynamics of the vehicle, this may occur immediately after first stage burnout. Additionally, the stages may not naturally drag-separate at this point
3. If the stages do not naturally drag-separate, we will ignite the upper-stage motor, to "hot-separate" the stages
4. Either from the hot separation, or after a short coast after drag-separating, the second stage burn begins

We considered using a mechanism to retain the stages together until the ignition of the upper stage. Shear pins were the leading candidate for this. However, there are potential stability concerns with hot-separating with shear pins. Specifically, if one shear pin breaks even a fraction of a second before the others, a strong moment will be put on the vehicle, disturbing it from its trajectory while the second stage motor is firing. This is unacceptable, so we decided the two stages would simply be nested into each other geometrically, with only a small amount of friction to overcome to stage. Using this separation scheme allows the rocket to be reduced in mass, as no extra mechanism is required in order for the stages to separate. This also saves on cost, as no extra materials are needed. Additionally, it would still keep the second stage stable during separation, allowing the second stage to continue on course as predicted.

A concern of ours with this scheme is that the uncertainty of the natural drag separation leads to imprecision in our simulation. However, we plan to mitigate this by running simulations where the stages drag-separate immediately after burnout, and simulations where the stages remain together until the second stage ignition.

6.4 De-Spin Mechanism

While many rockets with similar missions to ours spin-stabilize, we are not yet able to determine if we will need to. While that uncertainty exists, we have been examining the feasibility of de-spin systems. To satisfy other needs of the mission, namely recovery and payload concerns, the vehicle must be rolling at a rate no higher than 60 revolutions per minute when it reaches apogee (MEC.1). At this point, our analysis has assumed a very large range of possible initial spin rates, between 50 and 1000 revolutions per minute. Of course, as the stability analyses develop, this range will be tightened. After researching the common options for despin, the mechanisms that the team considered were yo-yo despin, cold gas thrusters, hot gas thrusters, and a reaction wheel system.

Yo-yo despin is achieved by wrapping two cables, each attached to a mass, around the rocket. A release mechanism allows the cords to unwind at the desired time, allowing the masses to spread out to the full length of the rope which increases the rocket's moment of inertia. When the cords are fully extended, they release from the vehicle, carrying a significant amount of angular momentum.

Additional optimizations can be done, such as using cables with elasticity, but our preliminary analysis was done with the most basic model. Yo-yo despin is relatively simple, and requires only a small amount of mass and volume.

Cold and hot gas thrusters operate on similar principles. Two small nozzles are aimed tangentially to the airframe, and exhaust gasses are ejected, slowing the rotation of the vehicle. Cold gas thrusters are fuelled by a tank of compressed gas, typically nitrogen or carbon dioxide, while hot gas thrusters use small solid rocket motors to generate the exhaust gas. For both cases, we considered only passive systems with fixed amounts of impulse; a more complex system might, for example, close the valve between the tank and the nozzles when the rotation rate slows sufficiently.

Finally, a reaction wheel would also be capable of reducing the spin of the vehicle by using a motor to spin a flywheel inside the stage. The angular momentum of the stage as a whole would be transferred to the flywheel alone. However, unlike the other three designs considered, this system is actively controlled, requiring a microcontroller to drive the motor. Also unlike the other three, this system does not actually discard the angular momentum of the vehicle, so the flywheel must continue to spin until landing.

Analyses were performed for each of these candidates. The results of these analyses were condensed into the decision matrix shown in Table 5. Each system was ranked on eight metrics, each on a scale from 1, the worst, to 10, the best. These metrics are generally simple to understand, such as cost and mass, but some have more nuance. The explanations below clarify our interpretation of each metric.

- **Testability:** Ease with which a test rig can be built with similar dynamics to the flight despin mechanism. A higher number indicates that we believe that the design would be easier to test in a meaningful way.
- **Reliability:** How infrequently the system fails in the flight environment.
- **Precision:** How frequently the despin mechanism is able to reduce the spin within the acceptable range. A higher number indicates the system is more consistently capable of slowing the vehicle to zero spin, regardless of initial spin, while a lower-ranked system would leave the vehicle spinning more often.
- **Simulability:** The accuracy of a simulation of the system. A higher number indicates a system that can be modeled with very high accuracy.

Based on this evaluation, yo-yo despin is the most optimal method. It is not the most highly rated method in every category, but is the best overall. Assuming that spin-stabilization is required, the team's de-spin method of choice to ensure successful parachute deployment will be the yo-yo method.

While parameters of the rocket are currently in flux this early in the design process, using the current estimates available, performance for the yo-yo despin mechanism can be calculated. We conducted a literature search for the governing equations for yo-yo despin. In Appendix E, equations from three sources are shown to be equivalent. It is currently estimated that two 170 gram masses will be used at the end of tethers of negligible mass. The tethers are assumed to be between 22 cm and 27 cm long. These estimates are based on the following assumptions, drawn from our sources for despin equations.

Metric	**Weight**	Yo-yo	Cold Gas	Hot Gas	Reaction Wheel
Mass	5	6	2	9	1
Volume	5	8	1	9	3
Reliability	8	8	6	5	8
Cost	5	9	2	8	3
Testability	8	10	6	8	10
Precision	7	8	1	1	10
Manufacturability	8	9	3	7	5
Simulability	7	8	2	2	9
	Totals:	443	166	311	352

Table 5: Decision matrix for the de-spin mechanism

- Cables unwind at a constant rate
- Cables are massless
- Despin masses are point masses
- System is perfectly conservative
- Gravitational effects are negligible
- Cables are released when radial (see Figure 8)
- Rocket rotation is perfectly on-axis

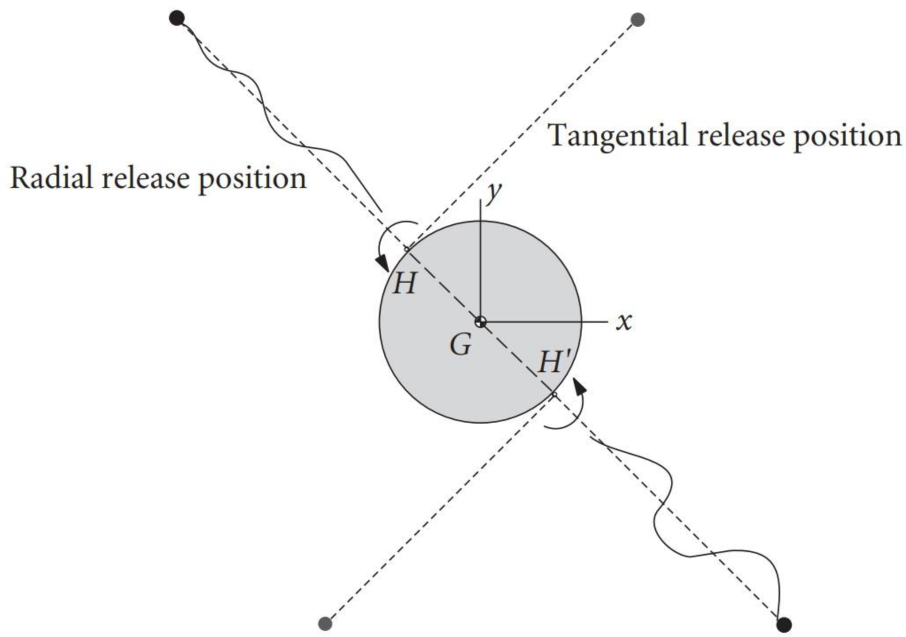

Figure 8: Comparison of radial and tangential release for a yo-yo despin system (from [2])

7 Structures

7.1 Introduction

7.2 First Stage (Booster)

7.3 Second Stage (Sustainer)

8 Next Steps

8.1 Highest Risks

The highest risk items within the recovery systems consist of potential designs for both the sombrero (first stage) and reefing (second stage) parachute schemes. Neither of these parachute designs can be adequately tested during ground or subscale trials, meaning they will only experience the conditions presented by spaceshot on the final launch day. This makes it difficult to determine proper descent speeds across the significant distance from which the stages will be falling. This in turn makes the prediction of landing locations for both stages incredibly difficult. Also, since neither of these designs are widely used, there is a nonzero chance that both systems will have to be manufactured in-house. This could lead to significant challenges as extremely precise manufacturing is required involving parachute lines, something we have no prior experience with. These concerns could lead to prolonged manufacturing and testing times if not handled properly.

Primary risks associated with the separation system come mainly in the form of prolonged development time. The first prototype is currently being manufactured and will need to pass a number of tests before the design can be continued. Mainly, the mechanism must be able to hold atmospheric pressure in a near vacuum similar to flight conditions and burn at least 80% of black powder detonated in vacuum separation tests. Failure for the system to achieve these goals would likely lead to the need for major redesigns and significantly prolonged development time.

8.2 Testing Capabilities

The overall purpose of conducting testing is to prove that certain aspects of the system fulfill the outlined requirements. Tests are designed to return data showing that the part being tested is capable of the required functions for the overall rocket to function. During test operations, safety is of course our top priority, and we work with our advisors as well as the facilities we work at to develop and practice safe procedures.

Like any team, we do not have an unlimited capability to test our system. Often times, there are high-risk systems that we are not able to test with high fidelity, which is a difficult combination. This section is intended to summarize the capabilities that our team has for testing. As far as test flights, we are very capable of flights less than roughly 10,000 ft; these flights can be conducted locally. However, high-altitude test flights are very difficult for us, due to the distance to launch sites where such flights could occur. Our long-term plans for such tests are very much undetermined at the moment, but for the sake of practiciality, we hope to minimize the systems that can only be tested on such flights.

Our team is very lucky to work at Zucrow Labs, where we have the ability to manufacture and test motors. Hot-fire testing at Zucrow will be an important step as we approach our spaceshot attempt. We will be able to fully characterize the motors we will fly on, which will be included in our 6DOF to most accurately predict the parameters of the flight. Also, the better we can characterize the motor, the more accurate the state estimation algorithm running on the avionics board will be. While a simplified model of the motor in the avionics software would not likely lead to a mission failure, the better the motor model, the more accurate the in-flight estimation of state, and the more precisely the in-flight events can be triggered. An additional component of the propulsion system is the method by which the second stage motor will be ignited at a high altitude. A leading candidate to solve this problem is a burst disk. An upside of this design is the relative ease of testing it; an unloaded motor with a burst disk could be vacuum tested.

As was discussed earlier in Section 5.1.3, testing of the avionics system will develop over time. The core component of the avionics system, the state estimation algorithm, will receive software in the loop testing. Simulated flight data will be generated by the 6DOF, noise will be injected as the flight data will be passed through sensor models, and the state estimation algorithm will determine the parameters of the flight, which will be compared to the true data directly from the simulation. This will allow us to finely tune the parameters of the state estimation algorithm to perform will with flights like the spaceshot. We are also investigating hardware in the loop testing; however, it may be the case that the amount of time and effort required for this would outweigh the benefits. Finally, for all test flights, the avionics board will fly, at least passively. This will help us characterize the performance of the board and algorithm together, as well as reducing “unknown unknowns”. The other siginificant component of the avionics system we will test is the downlink. Once transmitters, receievers, and antennas are selected, we plan to perform ground testing, bu simply moving the components far apart, as well as flight testing. However, these flights will not be very accurate in simulating the flight, particularly in terms of altitude, so more invesigation is needed.

We are relatively confident in our ability to test the mechanisms of the spaceshot vehicle. A draft of the separation mechanism is already being manufactured, with plans to fly in the spring semester. We are also able to perform higher-fidelity testing of this device on the ground. We are exploring testing the mechanism in a vacuum chamber to verify its performance during the spaceshot flight. The despin mechanism will also be somewhat straightforward to test on the ground with a spinning test rig. The parachutes and associated systems will be tested on lower-altitude test flights, which we are able to perform locally, and without significant overhead (as would be required for a high-altitude flight), though the conditions during these flights will not be able to replicate some aspects of the spaceshot launch. Unfortunately, the inter-stage mechanism will be very difficult to accurately test without a two-stage flight, which by their nature are more complex and risky than single-stage flights. The very simple design for the inter-stage we are pursuing helps to mitigate the lack of associated testing capabilities.

Testing of the aerostructures comes in two types: first, structural testing. We are confident in our abilities to characterize the structural characteristics of the vehicle through structural testing. Much more difficult, however, is testing the aerothermal loading the vehicle will experience. We are exploring testing options in this realm, but the design will likely need large safety factors for thermal loading, because the phenomenon is difficult to test, and to characterize in general.

8.3 Timeline

Next semester, we plan to make headway into the design of the vehicle. In particular, we plan to re-fly one of our existing vehicles (to roughly 5000 ft), primarily to test the stage separation mechanism, though the SRAD avionics board will be passively on board as well. However, we do not expect to hold another design review in the spring. Preparing for a design review stretches the limited resources of our team. We will work to balance the benefits a design review brings, especially in the form of feedback from reviewers, with the costs of holding one. For this reason, we plan to spend the spring semester entirely focused on design and testing, with our Preliminary Design Review tentatively expected to be held next fall. After that, we expect to hold a Critical Design Review, and finally a Flight Readiness Review, though naturally those reviews are much further out.

Appendix A Acronyms and Glossary

Acronyms

Term	Definition
CAD	Computer Aided Design
CDR	Conceptual Design Review
CEF	Characteristic Evaluation Function
CO_2	Carbon Dioxide
COTS	Commerical Off the Shelf
DATCOM	Data Compendium
DOF	Degree of Freedom
EKF	Extended Kalman Filter
FAA	Federal Aviation Administration
FEA	Finite Element Analysis
FM	Frequency Modulation
GPS	Global Positioning System
HA	High Altitude (see PSPHA)
IMU	Inertial Measurement Unit
MEOP	Maximum Expected Operating Pressure
MSL	Mean Sea Level
NASA	National Aeronautics and Space Administration
PDR	Preliminary Design Review
PEPC	Purdue Engineering President's Council
PESC	Purdue Engineering Student Council
PPE	Personal Protection Equipment
PSP	Purdue Space Program
PSPHA	Purdue Space Program High Altitude
PZL	Purdue Zucrow Labs
RF	Radio Frequency
SDR	Software Defined Radio
SMT	Surface Mount Technology
SRAD	Student Research and Development
SRR	Systems Requirements Review

Glossary

Term	Definition
1DOF	1 degree of freedom model for simulating the flight of a rocket.
6DOF	6 degree of freedom model for simulating the flight of a rocket
BurnSim	Simulates motor profile and performance.
COCOM Limit	Limits placed on GPS sold in the US. Shuts off GPS if the speed and height thresholds are met.
Delta V (ΔV)	Change in velocity. Often used as velocity needed to reach height in terms of impulse.
MATLAB	Mathworks computing program with built in functions and user created function.
Missile DATCOM	Program developed by the US Airforce that generates an aerodynamic table based off vehicle parameters.
OpenRocket	An open source rocket modeling programs that allows the user to assemble an amateur rocket and simulate it with an in house 6DOF program.
Siemens NX	CAD program developed by Siemens.
Simulink	Mathworks block diagram enviornment that allows modeling of functions in different blocks and connect them to each other.

Appendix B System Requirement Tables

Some of these requirements are dependent on whether or not we expend the first stage. Requirements that only exist if the first stage is recovered are marked with a $*$, and requirements that only exist if the first stage is expended are marked with a $\dagger$.

B.1 Internal Stakeholder Requirements

These are requirements set internally by Purdue Space Program High Altitude members. These were decided in a team-wide planning meeting early in the vehicle design process.

Req. ID	Requirement	Rationale
SR.1	The rocket shall reach 100 km mean sea level.	To fulfill our mission statement of reaching space, which the 100km mark is widely regarded as the boundary.
SR.2	The rocket shall have two powered stages.	To learn from the complexity of the separation mechanism and develop valuable learning experience, and become the first successful two stage rocket built by a student team.
SR.3	The rocket shall have one or more motors created by students at Purdue Zucrow Labs.	To involve a student design propulsion on a PSP rocket.
SR.4	The upper rocket stage shall be recoverable.	To be able to study the effects of high speed flight on all parts of the rocket on the ground.
SR.5	The rocket shall carry a payload non-essential to rocket performance.	We want to put an object inside the rocket that is meaningful to the team and launch it to space. It should not be a critical part of the vehicle.
SR.6	The rocket development shall follow systems documentation.	This is a requirement meant to address some of the documentation shortcomings of our previous PSP rocket teams. Documentation tends to be lacking, and whenever a core member leaves the team, limited knowledge gets transferred, resulting in having to start certain research from the beginning. This will also standardize the explanation of the function of a system across the teams and pass on our knowledge to future teams and groups.

B.2 External Stakeholder Requirements

These are the primary requirements set by non-PSP organizations that may constrain our design.

B.2.1 Federal Aviation Administration

Req. ID	Requirement	Rationale
EX.1.1	There shall not be a 90 person per square mile population area within a quarter range of vehicle targeted height.	To minimize public danger or property damage in case of rocket veering off course.
EX.2.1	Certificate of Authorization shall be approved by the FAA.	To confirm that rocket operational area will not endanger the public or interfere with air traffic.
EX.3.1	The rocket shall not reach above 150km.	Above 150km, the vehicle would no longer be classified as an amateur rocket and would be subject to a different set of FAA requirements.
EX.4.1	Form 7711-2 shall be approved by the FAA.	To confirm that rocket operational area will not endanger the public or interfere with air traffic.

B.2.2 Purdue Zucrow Laboratories

Req. ID	Requirement	Rationale
EX.2.1	Purdue Zucrow Laboratories shall set high level requirements based on our mission profile.	They can approve mixtures dependent on our mission instead of a strict standard.

B.2.3 Launch Sites

Certain launch sites have additional requirements due to company policy or local regulations. These are blanket requirements that we have extrapolated from reading different launch sites and are reasonable enough to impose as a team wide requirement.

Req. ID	Requirement	Rationale
EX.3.1	The team shall design its own launch rail.	Most launch site operators requested for us to use our own rails due to the SRAD motor possibly damaging their blast plates.

B.3 Functional Requirements

B.3.1 Flight-Critical Requirements

These are the minimum requirements needed for our rocket to fly successfully.

B.3.2 Recovery Requirements

Requirements for a successful recovery.

B.3.3 Non-Flight Critical Requirements

Requirements not necessarily required for the vehicle but fulfills a stakeholder requirement

B.4 Systems Requirements

In this section, the requirement from which any given requirement is derived from is by default its numerical parent; i.e. requirement PRO.1.2.1 is derived from PRO.1.2. Exceptions and special cases will be noted explicitly. Also, at this early stage in the design process, some specific parameters in requirements are still undetermined. They are given as “BLANK”.

B.4.1 Propulsion

B.4.2 Avionics

B.4.3 Mechanisms

B.4.4 Structures

Req. ID	Requirement	Rationale	Traced From
DEF.1.1	Rocket stages shall have fundamental flight articles.	These are the minimum components for a stage of our rocket to be considered a stage.	SR.1
DEF.1.1.1	The stage shall have an airframe.	Core structural part of a rocket that houses sub-systems.	SR.1
DEF.1.1.2	The stage shall have a motor.	Being a two stage powered rocket, all stages will have a motor.	SR.2
DEF.1.1.3†	The stage shall have a recovery system.	To safely recover the stage.	SR.4
DEF.1.1.3.1†	To be able to study the effects of high speed flight on all parts of the rocket on the ground.	The recovery system will be actively controlled for safety.	SR.4
DEF.1.2	The lower stage shall have the required flight articles to be the first stage.	Lower stage may contain components that are not required on other stages.	SR.1
DEF.1.2.1	The lower stage shall have fins.	Passively-stabilized rockets like ours usually require fins to remain stable throughout the flight.	SR.1
DEF.1.3	The upper stage shall have the required flight articles to be the first stage.	Upper stage may contain components that are not required on other stages.	SR.1
DEF.1.3.1	The upper stage shall have fins.	Passively-stabilized rockets like ours usually require fins to remain stable throughout the flight.	SR.1
DEF.1.3.2	The upper stage shall have a nosecone.	Rockets usually require a nose cone to remain stable throughout the flight.	SR.1
DEF.1.3.3*	The upper stage shall have a recovery system.	This stage travels to apogee and would be able to physically confirm height and performance.	SR.4
DEF.1.3.3.1*	Stages with a non-autonomous recovery system shall have an avionics system.	The recovery system will be actively controlled for safety.	SR.4
DEF.1.4	The vehicle shall have a staging mechanism between stages.	This allows the stages to separate.	SR.2
DEF.1.5	The vehicle shall ignite the upper stage motor.	The second stage motor is ignited by the rocket itself as there will be no external mechanism for rocket ignition.	SR.1, SR.2

Req. ID	Requirement	Rationale	Traced From
DEF.2.1	The upper[9] stage shall be recoverable.	The upper stage travels through the entire stage of the flight and records it.	SR.4
DEF.2.1.1	The upper stage touchdown velocity shall be less than 20 ft per second.	The stage must touch down slow enough to prevent significant damage.	SR.4
DEF.2.1.2*	The lower stage touchdown velocity shall be less than 20 feet per second.	The stage must touch down slow enough to prevent significant damage.	SR.4

Req. ID	Requirement	Rationale	Traced From
DEF.3.1	The vehicle shall have a payload.	Satisfies the payload requirement, and gained data is directly useful as visual proof of rocket location.	SR.5
DEF.3.2	The vehicle shall determine its apogee.	To confirm that the rocket has reached the target apogee.	SR.1
DEF.3.3	The vehicle shall identify its location.	For easier post launch recovery.	SR.4
DEF.3.4	The vehicle shall check its state before igniting second stage.	Implied required safety feature for any two stage rocket.	EX.1.1, EX.1.2, EX.1.4

Req. ID	Requirement	Traced From
PRO.1	The rocket shall have an upper stage propulsion system	DEF.1.1.2
PRO.1.1	The rocket shall have an upper stage motor	
PRO.1.1.1	The upper stage motor shall be made from a solid propellant	
PRO.1.1.1.1	The propellent formulation shall be TS - 78	
PRO.1.1.2	The upper stage motor shall be made of BLANK fuel grains	
PRO.1.1.2.1	The first fuel grain will have BLANK geometry	
PRO.1.1.2.2	The second fuel grain will have BLANK geometry	
PRO.1.1.3	The upper stage motor shall have a nozzle	
PRO.1.1.3.1	The nozzle shall have a converging angle of BLANK	
PRO.1.1.3.2	The nozzle shall have a diverging angle of BLANK	
PRO.1.1.3.3	The nozzle shall have a retainer	
PRO.1.1.3.4	The nozzle shall have an ablative casing made of graphite	
PRO.1.1.4	The upper stage motor shall produce a toal Delta V of BLANK	
PRO.1.2	The upper stage motor shall have an igniter	
PRO.1.2.1	The igniter will activate via a pyrotechnic charge	
PRO.1.2.1.2	The charge will accept signal from avionics to activate	DEF.1.5
PRO.1.2.2	The igniter will have an ingition motor activated by the charge	
PRO.1.2.2.1	The igniter formulation shall burn faster than the main motors	
PRO.1.2.3	The igniter shall operate at BLANK pressure	
PRO.1.2.4	The igniter shall be encased in the bulkhead	PRO.1.1, PRO.1.2
PRO 1.2.4.1	The bulkhead will withstand a chamber pressure of BLANK	
PRO.2	The rocket shall have a lower stage propulsion system	DEF.1.1.2
PRO.2.1	The rocket shall have a lower stage motor	
PRO.2.1.1	The lower stage motor shall be made from a solid propellant	
PRO.2.1.1.1	The propellent formulation shall be TS - 78	
PRO.2.1.2	The lower stage motor shall be made of BLANK fuel grains	
PRO.2.1.2.1	The first fuel grain will have BLANK geometry	
PRO.2.1.2.2	The second fuel grain will have BLANK geometry	
PRO.2.1.3	The lower stage motor shall have a nozzle	
PRO.2.1.3.1	The nozzle shall have a converging angle of BLANK	
PRO.2.1.3.2	The nozzle shall have a diverging angle of BLANK	
PRO.2.1.3.3	The nozzle shall have a retainer	
PRO.2.1.3.4	The nozzle shall have an ablaitve casing made of graphite	
PRO.2.1.4	The lower stage motor shall produce a toal Delta V of BLANK	
PRO.2.2	The lower stage motor shall have an igniter	
PRO.2.2.1	The igniter will activate via a pyrotechnic charge	
PRO.2.2.1.2	The charge will accept signal from the control panel to activate	
PRO.2.2.2	The igniter will have an ingition motor activated by the charge	
PRO.2.2.2.1	The igniter formulation shall burn faster than the main motors	
PRO.2.2.3	The igniter shall operate at BLANK pressure	
PRO.2.2.4	The igniter shall be encased in the bulkhead	PRO.2.1, PRO.2.2
PRO 2.2.4.1	The bulkhead will withstand a chamber pressure of BLANK	

Req. ID	Requirement	Traced From
AVI.1	The avionics shall verify the rocket's apogee.	DEF.3.2
AVI.1.1	The avionics shall use gathered data to estimate altitude.	
AVI.1.2	The avionics shall store the altitude data throughout the flight.	
AVI.1.2.1	The avionics system shall write gathered and calculated data to the system memory.	
AVI.2	The avionics shall activate the recovery system at the proper time.	DEF.1.1.3.1
AVI.2.1	The avionics shall determine the moment of recovery activation.	
AVI.2.1.1	The avionics shall use an algorithm to determine moment of recovery activation.	
AVI.2.2	The avionics shall output a high voltage recovery activation signal	
AVI.3	The avionics shall activate the stage separation and second stage ignition at the proper time.	DEF.1.5, DEF.3.4
AVI.3.1	The avionics shall determine the moment of separation and ignition.	
AVI.3.1.1	The avionics shall use an algorithm to determine moment of separation and ignition.	
AVI.3.2	The avionics shall output high voltage separation and ignition signals.	
AVI.4	The avionics shall locate the rocket after the flight.	DEF.3.3
AVI.4.1	The avionics shall gather location data.	
AVI.4.1.1	The avionics shall gather GPS data.	
AVI.4.2	The avionics shall transmit the rocket location data.	
AVI.4.2.1	The avionics shall have an antenna capable of transmitting the relevant data.	
AVI.5	The avionics systems shall be durable enough to safely fly on the vehicle.	DEF.1.1.3.1, DEF.1.5, DEF.3.1, DEF.3.2, DEF.3.3, DEF.3.4
AVI.5.1	The avionics shall withstand the projected forces during flight.	
AVI.5.2	The avionics shall withstand the projected vibrations during flight.	
AVI.5.3	The avionics shall withstand the projected thermals during flight.	
AVI.6	The avionics shall have a payload	DEF.3.1
AVI.6.1	The avionics shall have an outward recording camera throughout the flight.	
AVI.6.2	The avionics may have additional payload(s).	

Req. ID	Requirement	Traced From
MEC.1	The rocket shall despin to no more than 60 revolutions per minute.	DEF.2.1
MEC.1.1	The rocket shall have a despin mechanism.	
MEC.1.2	The despin mechanism shall deploy at a specific altitude.	
MEC.2	Both stages of the rocket shall be recoverable.	DEF.1.4, DEF.2.1
MEC.2.1	The first stage shall descend with a slower velocity.	
MEC.2.1.1	The first stage of rocket shall have a recovery system.	
MEC.2.1.2	The airframe of the first stage of the rocket shall separate.	
MEC.2.1.2.1	The separation mechanism shall be capable of operation without siginificant ambient pressure.	
MEC.2.2	The second stage shall descend at no more than 20 ft/s below 1000 ft, and no more than 50 ft/s above 1000 ft.	
MEC.2.2.1	The second stage of the rocket shall have a recovery system.	
MEC.2.2.2	The airframe of the second stage of the rocket shall separate.	
MEC.2.2.2.1	The separation mechanism shall be capable of operation without siginificant ambient pressure.	
MEC.3	The two stages of the rocket shall separate at a predicted or commanded time.	DEF.1.4, DEF.2.1
MEC.3.1	The rocket shall have a separation mechanism between the first and second stages.	
MEC.3.1.1	The inter-stage separation mechanism shall not significantly disturb the trajectory of the second stage.	

Req. ID	Requirement	Traced From
STR.1	The rocket shall have fins on the lower stage.	DEF.1.2.1
STR.1.1	The lower fins shall be a BLANK shape.	
STR.1.2	The lower fins shall survive BLANK stage of flight.	
STR.1.2.1	The lower fins shall withstand BLANK temperatures.	
STR.1.2.2	The lower fins shall withstand a compressive load of BLANK.	
STR.1.3	There shall be BLANK fins.	
STR.1.4	The lower fins shall be BLANK inches thick.	
STR.1.5	The lower fins shall have a BLANK cross section.	
STR.1.6	The lower fins shall have a BLANK inch root chord length.	
STR.1.7	The lower fins shall have a BLANK inch tip chord length.	
STR.1.8	The lower fins shall be BLANK inches high.	
STR.1.9	The lower fins shall have BLANK inches of sweep length.	
STR.1.10	The lower fins shall have BLANK degrees of sweep angle.	
STR.1.11	The lower fins shall be BLANK inches from the bottom of the lower airframe.	
STR.2	The rocket shall have a lower airframe.	DEF.1.1.1
STR.2.1	The lower airframe shall have a diameter of BLANK inches.	
STR.2.2	The lower airframe shall survive BLANK stage of flight.	
STR.2.2.1	The lower airframe shall withstand BLANK temperatures.	
STR.2.2.2	The lower airframe shall withstand a compressive force of BLANK.	
STR.2.3	The lower airframe shall be BLANK inches thick.	
STR.3	The rocket shall have an interstage.	DEF.1.4
STR.3.1	The interstage shall have a diameter of BLANK inches.	
STR.3.2	The interstage shall survive BLANK conditions.	
STR.3.2.1	The interstage shall withstand BLANK temperatures.	
STR.3.2.2	The interstage shall withstand a compressive load of BLANK.	
STR.3.3	The interstage shall be BLANK inches high.	
STR.3.4	The interstage shall be BLANK inches thick.	
STR.4	The rocket shall have fins on the upper stage.	DEF.1.3.1
STR.4.1	The upper fins shall be a BLANK shape.	
STR.4.2	The upper fins shall survive BLANK stage of flight.	
STR.4.2.1	The upper fins shall withstand BLANK temperatures.	
STR.4.2.2	The upper fins shall withstand a compressive load of BLANK.	
STR.4.3	There shall be BLANK fins.	
STR.4.4	The upper fins shall be BLANK inches thick.	
STR.4.5	The upper fins shall have a BLANK cross section.	
STR.4.6	The upper fins shall have a BLANK inch root chord length.	
STR.4.7	The upper fins shall have a BLANK inch tip chord length.	
STR.4.8	The upper fins shall be BLANK inches high.	
STR.4.9	The upper fins shall have BLANK inches of sweep length.	
STR.4.10	The upper fins shall have BLANK degrees of sweep angle.	
STR.4.11	The upper fins shall be BLANK inches from the bottom of the lower airframe.	
STR.5	The rocket shall have an upper airframe.	DEF.1.1.1
STR.5.1	The upper airframe shall have a diameter of BLANK inches.	
STR.5.2	The upper airframe shall survive BLANK stage of flight.	
STR.5.2.1	The upper airframe shall withstand BLANK temperatures.	
STR.5.2.2	The upper airframe shall withstand a compressive force of BLANK.	
STR.5.3	The upper airframe shall be BLANK inches thick.	
STR.6	The rocket shall have a nosecone.	DEF.1.3.2

Appendix C Recovery System Line Diagrams

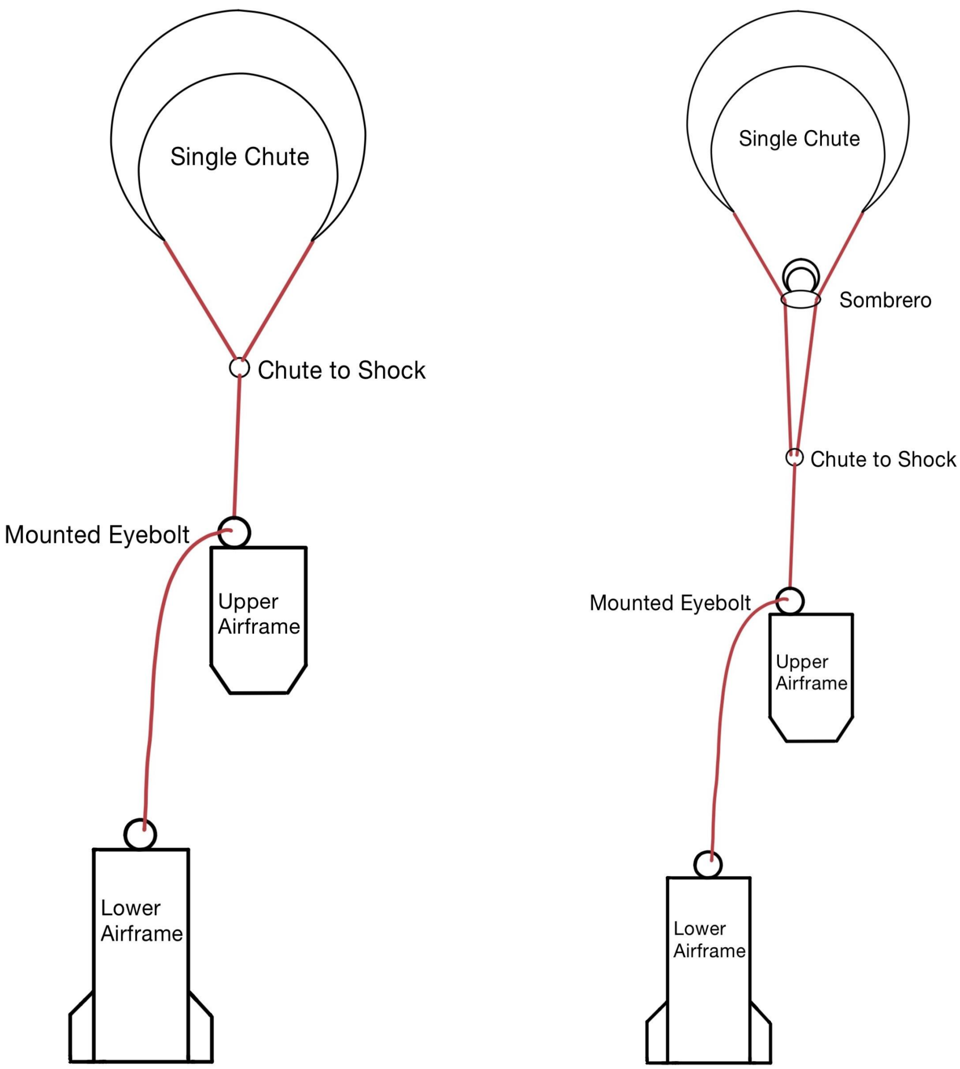

Figure 9: Two recovery schemes for the first stage: on the left, a single parachute, and on the right, the “Sombrero”

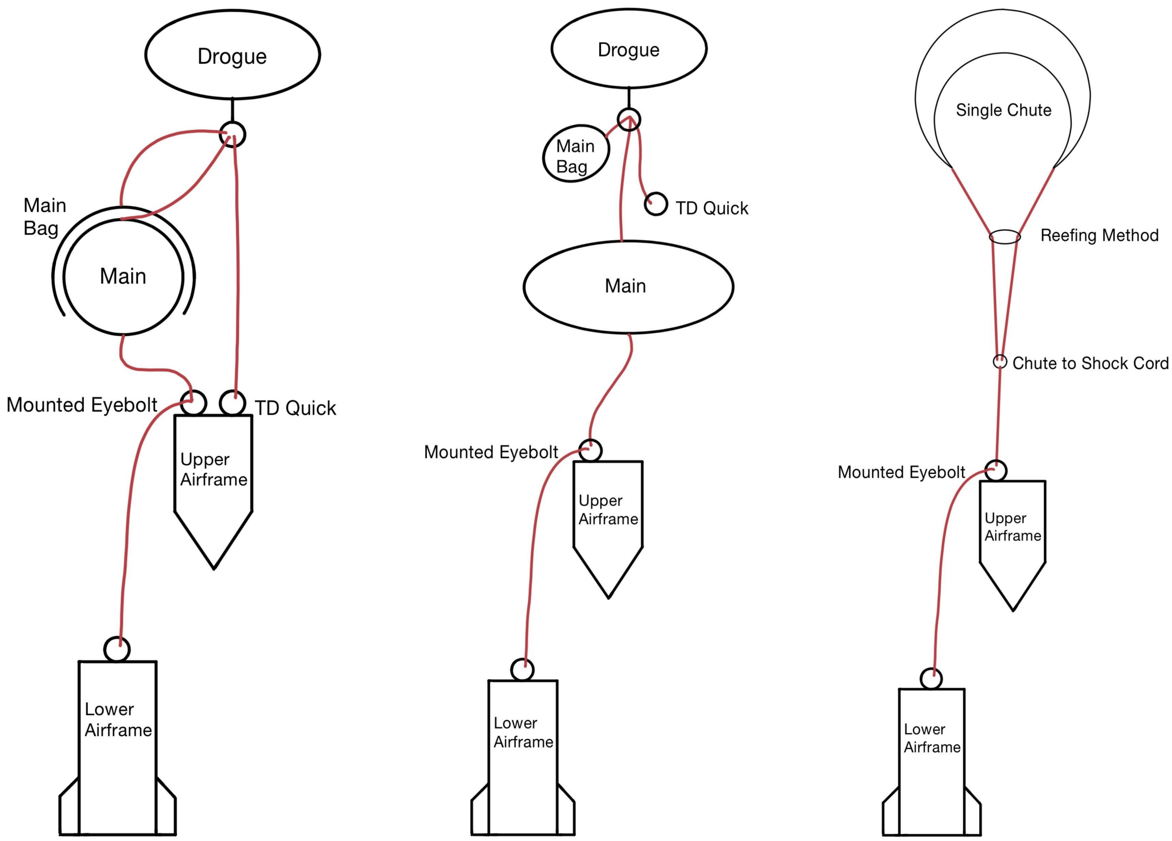

Figure 10: Two recovery schemes for the second stage: the left two images show the dual-deploy with the drogue deployed, and with the main deployed; on the right, the reefing chute design

Appendix D Avionics In-Flight Event Overview

Flight Mode

The computer will track changes through the following phases of flight. Each phase will be represented in a Stateflow model. These states and their transitions may change slightly as the design of the vehicle chages over time.

On the Pad

This phase happens when the computer is powered on and the rocket is vertical on the rail. All sensor calibrations should happen during this phase. During this phase the computer will be checking the accelerometer data to check for liftoff, but not recording the data yet.

First Stage Burn

One liftoff is detected by the accelerometer the computer enters the first stage burn phase. During this phase it is measuring and recording acceleration data at the highest frequency possible. It will also be measuring and recording data from all other sensors, and sending it the state estimation algorithm. Additionally, the first stage lockout countdown begins as soon as the rocket enters this phase. No events (eg. stage separation/ignition/parachutes) can be triggered until the first stage countdown has reached 0.

Combined Coast/Staging

After the primary motor has burned out and the first stage lockout has ended, the computer will enter the staging phase.

Active Separation: If an active staging method is used, the staging event will happen at either a set time delay after the motor burnout or at a set altitude after burnout, or possibly a combination of both for redundancy. The optimal time or altitude for staging will be determined by simulation. It should be noted that we are currently

Hot Separation: If a "hot separation" method is used, the rocket will separate stages by igniting the second stage motor while the two stages are still together. This could be used in combination with/as a backup to drag separation.

Passive (Drag) Separation: If a passive separation method is used, the two stages will not be held together. They will naturally separate after first stage burnout, when the drag force on the bottom stage is greater than drag on the top. This method may include a separation detector so the computer can know when the separation has happened. If this does not naturally occur, the motor second stage motor should still ignite to force separation after a set amount of time, assuming all other ignition conditions are met.

Second Stage Ignition

Second stage ignition can only occur if the following conditions are met:

- Rocket is above a certain altitude

- Rocket is not tilted more than a certain threshold from vertical
- Stage separation has been triggered (if applicable)
- A set time delay since stage separation has passed (if applicable)

If all these conditions are met, second stage ignition will occur. If they are not met, the computer will skip to the apogee detection phase without igniting the second stage.

Second Motor Burn

The computer enters this phase upon detection of the second stage igniting (from accelerometer data). Upon entering this phase the second stage lockout countdown will start. If ignition is not detected within a certain number of seconds after it should have happened, the computer will skip to the apogee detection phase, without starting the timer.

Coast/Apogee Detection

In this phase, the computer will use data from the state estimation to detect when the rocket has passed apogee.

Drogue Deploy/Descent

After apogee has been passed and any active lockout timers have ended, the drogue chute will deploy.

Main Deploy/Descent

Once data from the state estimation function shows that the rocket is under the main deployment altitude, the main chute will be deployed.

Touchdown

Once the rocket is on the ground stop, it will stop recording data, safe itself, and shut down.

Flowchart of Phases

The state flow diagram is shown in Figure 11. Each rounded box is a flight phase (Stateflow state). The phase will help the State Estimation determine which prediction model to use. Each parallelogram box is an event that needs to be triggered. These could be their own states or just happen upon entering/exiting the previous/next state. The circles show different options that depend on the rocket configuration (stage and separation method). See Table 6 for transition criteria for each arrow. In the table, yet-undetermined parameters are left as “BLANK”.

Lockout Timers

The purpose of the lockout feature is to prevent an early parachute deployment or staging event. This feature will prevent the rocket from igniting any charges until a predetermined amount of time since motor ignition. The time will either be a hardcoded value added when the software is loaded on, or set by the user before launch. Simulations should provide an estimate of how long the lockout timer needs to be. There will be a separate timer for the first and second motor burns. If the

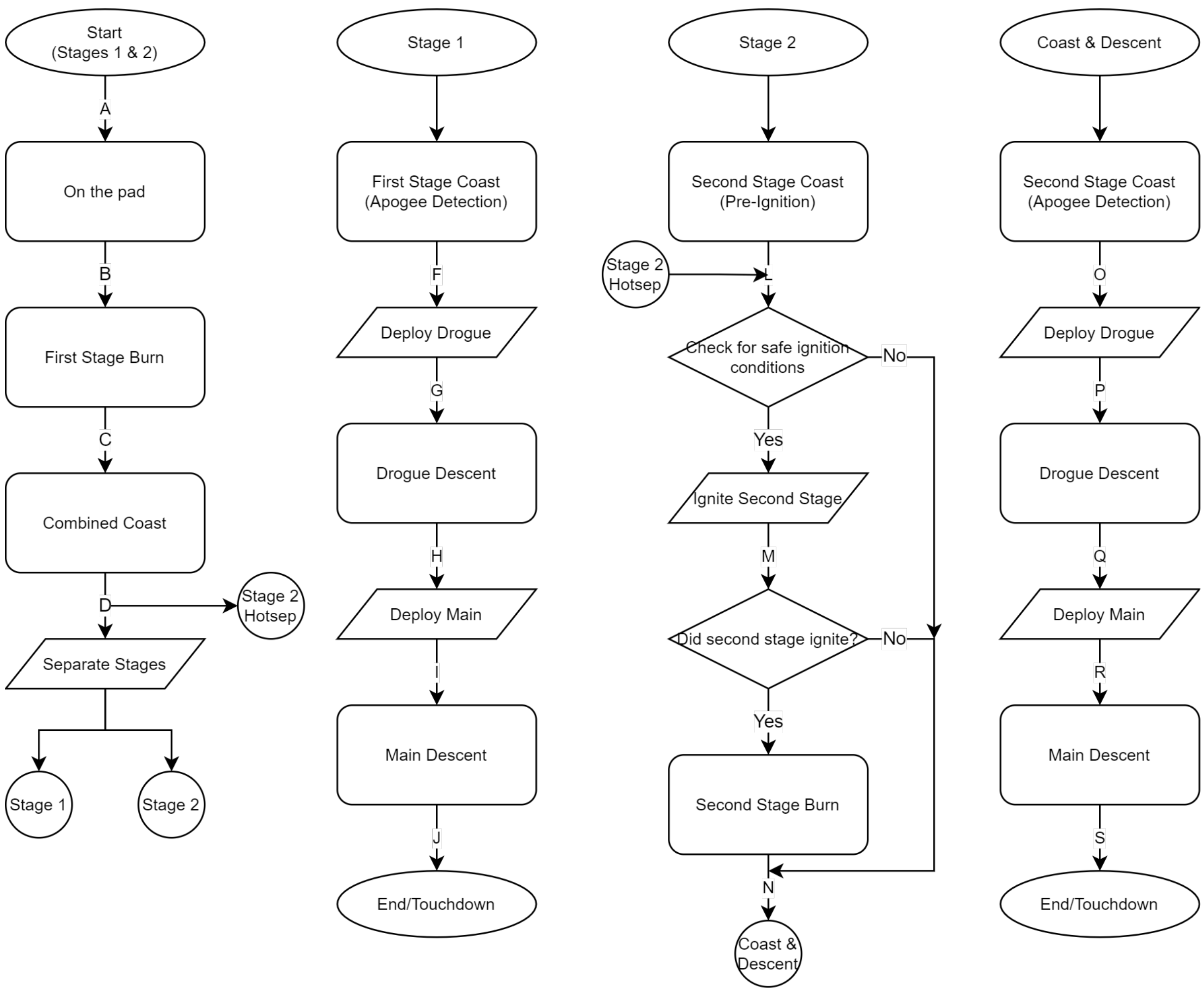

Figure 11: The preliminary avionics flight event logic stage flow diagram

Transition Label	Criteria	Transitions To
A	The rocket is vertical on the launch rail. This means the pitch angle is $90 \pm$ BLANK degrees.	On the Pad
B	The accelerometer detects positive acceleration in the vertical direction greater than BLANK m/s^2 for longer than BLANK milliseconds.	First Stage Burn
C	After first stage burnout: The accelerometer detects acceleration in the vertical direction less than BLANK m/s^2 for longer than BLANK milliseconds.	Combined Coast
D	The first stage lockout timer has reached 0, and the stage separation altitude and/or time delay conditions are met	Separate Stages or Check Ignition Conditions (for stage 2 hot-sep)
M	The accelerometer detects positive acceleration in the vertical direction greater than BLANK. If this is not detected within BLANK seconds after attempting to ignite the second stage, skip and continue to second stage coast.	Second Stage Burn
N	From successful ignition: The accelerometer detects acceleration in the vertical direction less than BLANK	Second Stage Coast (Apogee Detection)
F/O	The rocket has passed apogee: vertical velocity is negative, BLANK seconds since apogee have passed, and the second stage lockout timer has reached 0	Deploy Drogue
G/P	Immediately after drogue deployment is triggered	Drogue Descent
H/Q	The altitude is less than BLANK	Deploy Main
I/R	Immediately after main deployment is triggered	Main Descent
J/S	The rocket velocity is $0 \pm$ BLANK	Touchdown

Table 6: State transition conditions

second motor doesn't light, the second lockout will not happen, to allow recovery events to happen regardless of whether or no ignition is successful.

Appendix E De-Spin Equations

Three primary sources have been compiled to verify the feasibility of yo-yo despin: a Princeton course [6] (masses released tangentially), an unclassified NASA paper [3] (masses released radially), and an Attitude Dynamics textbook [2] (masses released either tangentially or radially). When using the assumptions that: (1) the cord unwinds at a constant rate equal to the vehicle's initial angular velocity and (2) final angular velocity is 0, each source gives the same equation for the corresponding type of release.

E.1 Princeton Course Slides (MAE 342)

If I is the moment of inertia of the satellite excluding the yo-yo masses,

$$\Psi = 1 + \frac{I}{mr^2}$$

If the final angular velocity is zero:

$$l =$$

Appendix F PSP High Altitude Team Members

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Joseph Dunne	Ignacio Esteban	Jacob Fisher	Abigail Frank
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Taylor Vicente	Ryan Williams	Abby Woodbury	

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