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

2.1 Internal Stakeholder Requirements

2.2 External Stakeholder Requirements

2.3 Functional Requirements

2.4 System Requirements

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.

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

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 seed for the following generation. This process was repeated for the designated amount of gener-

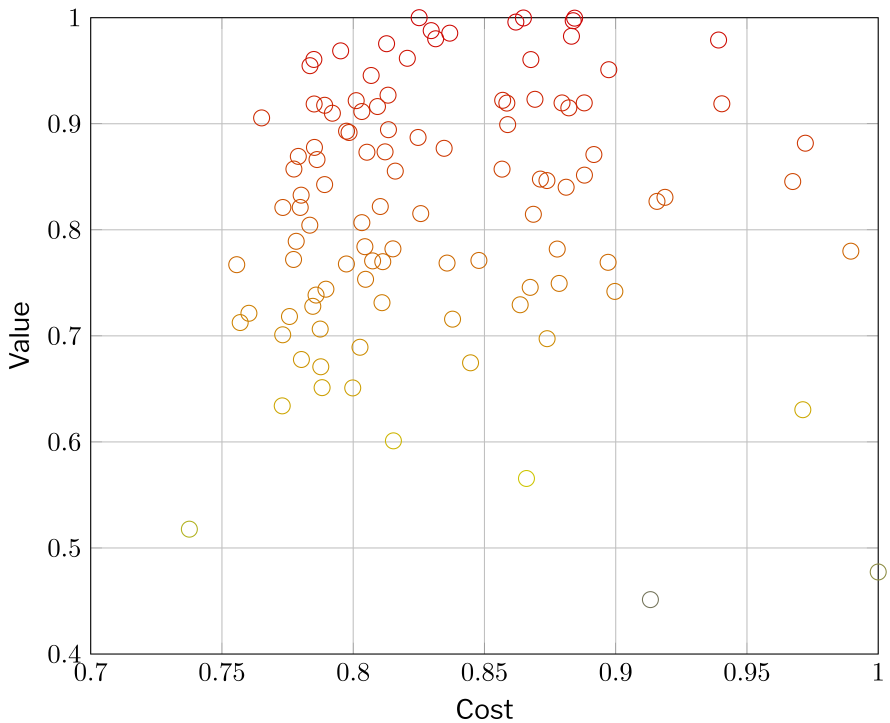

Figure 1: Pareto analysis results scatter plot

ations 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" [3] 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.

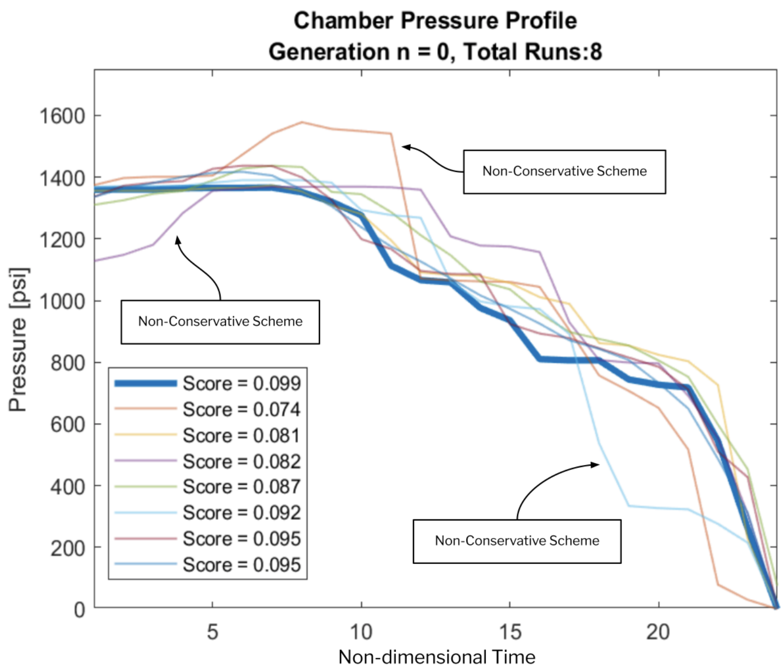

Figure 2: Example genetic algorithm run

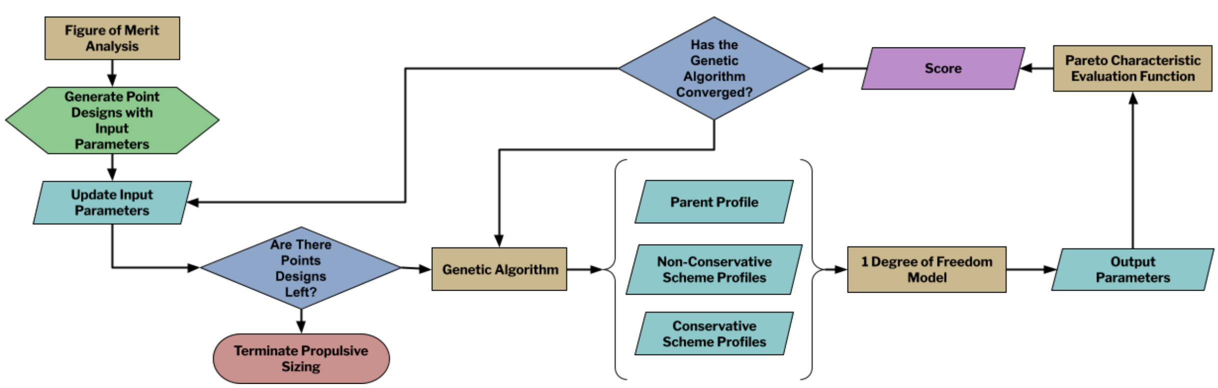

Figure 3: Genetic algorithm selection process flowchart

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 Preliminary Structural and Mass Analysis

3.5 Trajectory Analysis Model and Statistical Methods (6DOF)

3.6 Future Considerations

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 [4] 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 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

5 Avionics

5.1 State Estimation and Apogee Determination

5.1.1 System Architecture

5.1.2 Data Fusion Algorithm

5.1.3 Testing

5.2 In-Flight Events

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

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

6 Mechanisms

6.1 Recovery System

6.2 Separation Mechanism

6.3 Inter-Stage Mechanism

6.4 De-Spin Mechanism

7 Structures

7.1 Introduction

7.2 First Stage (Booster)

7.3 Second Stage (Sustainer)

8 Next Steps

8.1 Highest Risks

8.2 Timeline

Appendix A System Requirements Tables

A.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 100km 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 a motor created by students at Purdue Zucrow Labs.	To involve a student design propulsion on a PSP rocket.
SR.4	The rocket stages 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	To be able to study the effects of high speed flight on all parts of the rocket on the ground.	To be able to assist in high altitude research.
SR.6	The rocket shall follow systems documentation.	To let future PSP and any other interested parties to learn from our mistakes and accomplishments.

A.2 External Stakeholder Requirements

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

A.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.

A.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.

A.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.

A.3 Functional Requirements

A.3.1 Flight-Critical Requirements

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

Req. ID	Requirement	Rationale	Traced To
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 subsystems.	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.[2]	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

A.3.2 Recovery Requirements

Requirements for a successful recovery.

Req. ID	Requirement	Rationale	Traced To
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

A.3.3 Non-Flight Critical Requirements

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

Req. ID	Requirement	Rationale	Traced To
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

A.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.

A.4.1 Propulsion

Req. ID	Requirement	Traced To

Appendix B De-Spin Equations

Three primary sources have been compiled to verify the feasibility of yo-yo despin: a Princeton course [5] (masses released tangentially), an unclassified NASA paper [2] (masses released radially), and an Attitude Dynamics textbooks [1] (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.

B.1 Princeton Course Slides (MAE 342)

Source: [5]

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:

$$l =$$

Bibliography

[1] Howard D. Curtis. *Orbital Mechanics for Engineering Students*. 3rd ed., pp. 576–583. ISBN: 9780080977478.

[2] Donald G. Eide and Chester A. Vaughn. *Equations of Motion and Design Criteria for the Despin of a Vehicle by the Radial Release of Weights and Cables of Finite Mass*. AD0270287. Defense Technical Information Center, Jan. 1, 1962.

[3] Steven D. Heister et al. *Rocket Propulsion*. Cambridge University Press, 2019. ISBN: 9781108422277.

[4] Tijen Seyidoglu and Manfred A. Bohn. *Characterization of Aging Behavior of Butacene®Based Composite Propellants by Loss Factor Curves*. STO-MP-AVT-268.

[5] Robert Stengel. *Space System Design*. MAE 342. Princeton University.