Launch Vehicle and Supporting Ground System for a Self-Pressurizing Liquid Rocket Engine

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The Space Hardware Club at The University of Alabama in Huntsville is developing its first liquid propelled vehicle capable of carrying a 3U payload for the 2018 Spaceport America Cup Intercollegiate Rocket Engineering Competition. The rocket system, including the vehicle and supporting ground system, is being developed and manufactured by the student team. Due to the comparatively low intra-organizational knowledge of liquid propulsion systems versus solid propulsion systems, the team has planned for numerous design iterations of the liquid propulsion system and has designed accompanying systems to be adaptable to interface changes. The rocket must be developed, manufactured, tested, and flown within a year to participate in the 2018 competition; consequently, a simple design architecture is employed in almost every facet of the overall rocket system. The liquid propulsion system uses self-pressurizing propellants, Ethane and Nitrous Oxide, to eliminate a third pressurant tank and to counteract the engine performance penalty of a large ullage volume blowdown cycle. Additionally, the airframe is designed to use existing composite manufacturing techniques and tooling and the recovery system is COTS to reduce development time and to increase assumed reliability compared to an in-house solution. The supporting ground system is a significant investment or personnel and financial resources of the project due to stringent safety considerations and launch-rail structure. This paper discusses major architecture and integration decisions made during development, along with planned validation testing to ensure a safe final flight.

I. Introduction

WITH almost 5 years of experience with solid propellant launch vehicles, The Space Hardware Club (SHC) at the University of Alabama in Huntsville (UAH) is expanding capabilities by developing its own liquid rocket engine and accompanying flight vehicle and support systems for competition in the Spaceport America Cup, held from June 19th to 23rd, near Las Cruces, New Mexico. In its inaugural year, over 1,100 students and faculty from more than 70 institutions competed in the competition, and the competition was sponsored by multiple aerospace companies, such as Virgin Galactic, Blue Origin, ULA, SpaceX, and others. 2018 is the second year of the Spaceport America Cup, and has over 200 teams from across the world registered to compete. The team will compete in the 30,000 ft student researched and designed (SRAD) Hybrid/Liquid category. 9 other teams are registered to compete in the same category.

One of the primary vehicle design challenges of switching from commercial-off-the-shelf (COTS) propellant to a SRAD liquid propellant system is the integration of multiple subsystems within the vehicle and the integration of the vehicle to a ground system. Over the past 5 years, SHC has found ways to expedite and optimize the production of many rocket components. As the club grows, so will the capabilities to progress to new, technologically advanced rocketry projects such as this liquid propelled vehicle. However, existing tooling and club heritage from previous projects is a driving force behind many design decisions.

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II.Vehicle

The vehicle, from engine to nose cone, was designed with student manufacturing capabilities in mind. A CAD model showing the overall vehicle can be seen in Figure 1.

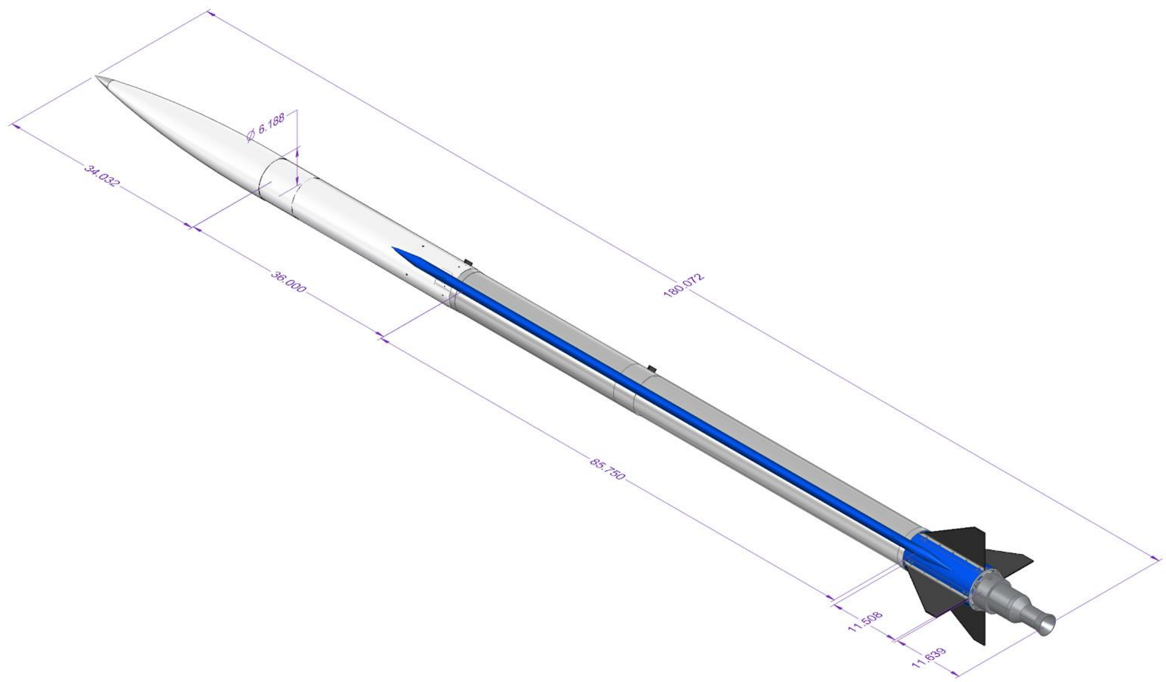

Fig. 1 Dimensioned CAD of overall rocket (inches)

Most components were manufactured in-house by team members at the UAH Prototyping Facility. Additionally, the design was optimized to use pre-existing tooling from solid vehicle development. The rocket is 6 inches in diameter and approximately 15 feet long, with 4 fins and 2 composite fluid shrouds. It is powered by a self-pressurizing blowdown Nitrous Oxide-Ethane engine. An image can be seen in Figure 1. More information on the engine and fluid systems development can be found in Development of a Self-Pressurizing Ethane-Nitrous Oxide Propulsion System for an Amateur Rocket by Marcus Shelton and James Biaglow [1]. The largest portion of the outer airframe is the aluminum tanks, with a carbon fiber tube and fiberglass nose cone above it. A CAD model showing material selection can be seen in Figure 2.

Fig. 2 CAD model displaying material selection

A. Aerodynamic Surface Design

As a passively stabilized flight vehicle, maintaining aerodynamic stability throughout flight is a top level system requirement. While the avionics and simulation team created software to actively calculate coefficient of drag in-flight for trajectory prediction, it would be time consuming to adapt for theoretical calculation for theoretical center of pressure. Instead, multiple models were created in commercial software to model different aerodynamic aspects. The center of gravity was found through assembling components in CAD and assigning them proper mass or density, depending on the component. This center of mass was input into Open Rocket, and the fins were designed to keep the center of pressure between 2 and 4 calibers away with both the wet CG and dry CG through all regimes of flight. The body for simulating the coefficient of drag with wet CG can be seen in Figure 3.

Fig. 3 Coefficient of drag shown in Open Rocket simulation

B. Fluids and Engine Integration

Fig. 4 Aluminum tanks holding fuel and oxidizer

The largest part of the vehicle are the aluminum tanks holding fuel and oxidizer. They share a common bulkhead, with fuel above oxidizer. A layout can be seen in Figure 4. There are 2 main areas of valves: relief and check valves located above the tanks, and engine valves located below the tanks. The two composite shrouds house the line from the oxidizer tank to the relief valves and the line from fuel tank to the engine valves, respectively. The upper section of valves is housed in a carbon fiber body tube, with a supporting bulkhead that also separates the fluid system from the recovery system. The lower section of valves connects to the engine and is protected by fin brackets and fiberglass panels.

The load path of the vehicle was designed with serviceability in mind. Because there are important points of connection and assembly between the engine and the tanks, any thrust structure had to be laterally removable. Thrust is transferred through a thrust plate, then the 1/8th inch angle aluminum fin brackets. This design traded most favorably for manufacturing time and and mass savings, in addition to its serviceability. Each bracket integrates into the thrust plate and lower tank end cap with 4 #10 low profile socket head cap screws, as can be seen in Figure 5. (screws not shown). The fins are attached with #6 countersunk screws. In each par, one bracket is threaded and the other is threaded, to save mass and maintain a low aerodynamic profile. The thrust plate is connected to the engine cap by 4 ½-20 bolts. The brackets and thrust plate are then covered by thin fiberglass panels for aerodynamics. The engine is left uncovered, as the effect on the aerodynamics is outweighed by the mass savings and cooling benefits.

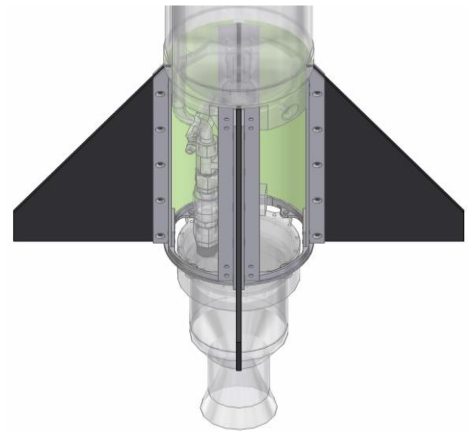

Fig. 5 Thrust Assembly and fin attachment

C. Flight Profile and Recovery

An overview of the flight profile can be seen in Figure 6. The rocket launches at a competition specified 84° angle. Using active onboard altitude prediction, it then flies in powered ascent until predicted apogee is 30,000 feet. It then cuts engine and begins venting excess fuel and oxidizer until it reaches apogee. At apogee, the vehicle deploys its drogue parachute and the payload pod. The payload pod is inspired by the CubeSat P-Pod deployment system from the ELaNa program, but with design alterations to take advantage of gravity. Once the frame is out of the nose cone, a door on the bottom is triggered to open, releasing the payload. The rocket then descends on a 30 inch drogue before deploying a 120 inch main parachute at 1,500 feet AGL.

Fig. 6 Launch and Recovery Concept of operations

D. Aerothermal Simulations

The launch vehicle undergoes more significant aerothermal heating than previous club rockets due to its long flight time. The rocket is expected to reach a maximum velocity of Mach 1.44 at an altitude of 3029 m, or 9938 ft. This puts it in the compressible, supersonic regime of flight. The aerodynamic heating caused by compressibility effects could potentially cause issues if not properly accounted-for, and it is desired to minimize the airframe's coefficient of drag.

For validation purposes, some basic parameters were calculated by hand. At the tip of the nose of the rocket is a stagnation point, and the streamline through it also passes through the normal portion of a shock wave. Thus, the pressure and temperature at the tip of the nose cone should be equal to the total temperature and pressure behind a normal shock.

Given that $M_1 = 1.44$, $P_1 = 70920\,Pa$, $T_1 = 300K$, and assuming that for adiabatic flow through a normal shock $T_{o1} = T_{o2}$,

$$T_{o2} = T_{o1} = \frac{T_{o1}}{T_1} T_1$$

From Table A-1[1], for $M_1 = 1.44$: $\frac{T_{o1}}{T_1} = 1.415$ and $\frac{P_{o1}}{P1} = 3.368$

$$T_{o2} = 1.415(300K) = 424.5K$$

From Table A-2[1], for $M_1 = 1.44$: $\frac{P_{o2}}{P_{o1}} = 0.9476$

$$P_{o2} = \frac{P_{o2}}{P_{o1}} \frac{P_{o1}}{P_1} P_1 = (0.9476)(3.368)(70920Pa) = 226342Pa$$

The nose cone was modeled using a 3-dimensional density-based solver in ANSYS Fluent at conditions spanning the portion of the flight where compressible effects occur: M > 0.3. The ambient air temperature was assumed to be 300K, or 540°R. The air was assumed to be an ideal gas, and viscosity was calculated using the sutherland equation. Viscous effects were modeled using the Spalart-Allmaras model due to its fast convergence and accuracy in high-Re applications; the k-omega model gave non-physical pressures near the boundary layer. To improve computational efficiency and maximize mesh resolution, only 1/12 of the axisymmetric nose cone was modeled.

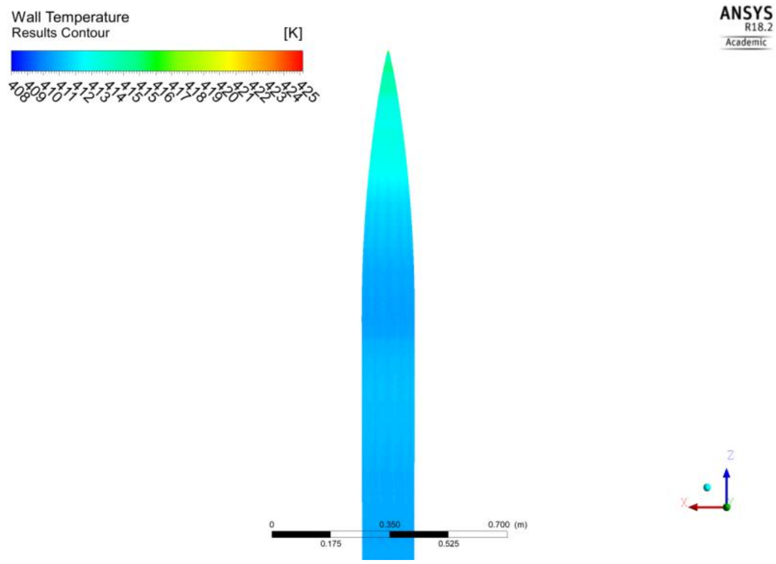

Fig. 7 Nose Cone Adiabatic Temp. Dist. at M1.44

In the model, the total temperature at the nose cone's tip was 424.2K at maximum velocity conditions, a 0.071% difference from the hand calculations. The nose cone tip's total pressure was modeled to be 205320 Pa, a 9.29% difference. By these metrics, the model was reasonably validated. The adiabatic temperature distributions at the nose cone's boundary layer were recorded for the course of the flight where M > 0.3. MATLAB code was written employing a lumped capacitance thermal model, and used to determine the average half-thickness temperature for the nose cone using the ANSYS-generated data points. This analysis could benefit from collecting more data points, and assumptions such as perfect insulation at the nose cone's inner surface cause the results to be overestimated. Nevertheless, the results validate the nose cone's material selection, as the maximum half-thickness temperature is well below the glassing temperature of 100 C.

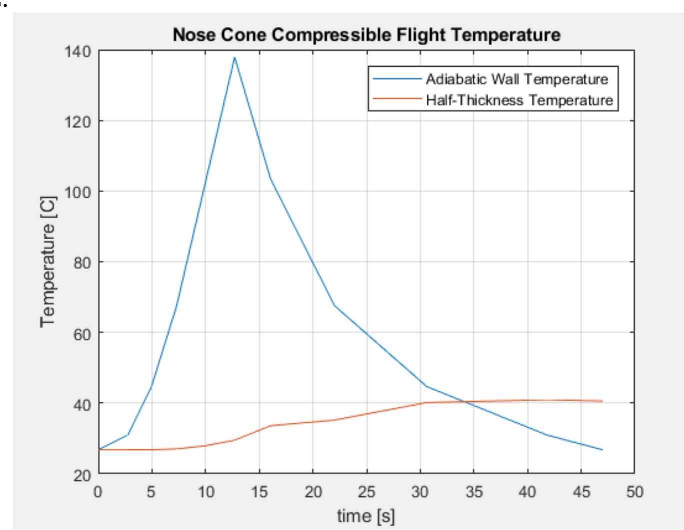

Fig. 8 Nose Cone temperature distribution vs time

III. Ground Support Equipment

A. Systems overview
Launch vehicle ground system's function is to perform fill, purge, and launch operations. Each operation has multiple considerations to manage in order to achieve functionality. These operations are broken up into three auxiliary subsystems, Structures, Controls, and Fluid Systems.

B. Structures
1. Launch Rail

The rocket is attached to the rail by 3 airfoiled 1515 rail buttons, which is standard in high power amatuer rocketry. Given the rocket's low T/W the launch rail has to be of an impressively large size. The current launch rail is 30 ft of triangular truss supported 15-15 rail and 8 ft of unsupported rail yielding a cumulative height of 38 ft. Figure 10 shows this arrangement.

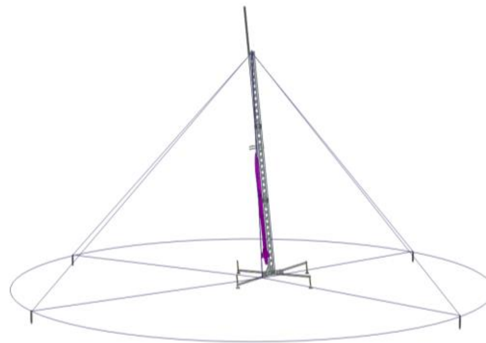

Fig. 10 GSE test stand with integrated rocket

2. Propellant Environment Control

Fig. 11 Exploded View of Propellant Environmental Control System

Both Ethane and Nitrous Oxide thermophysical properties vary considerably with temperature. Nitrous Oxide additionally shows greater instability above its critical temperature (T_{crit}) of 98°F and therefore for safety must be kept below that temperature. In order to keep both propellants thermally controlled without the need for controlled venting Environment Control systems were created for the propellant cradles. Figure 11 shows the arrangement being implemented by the team which includes aluminum cradles allowing for the storage of 4 propellant gas cylinders and 1 nitrogen gas cylinder. The environmental control system consists of a wood paneled, insulated box with a simple window unit air conditioner. This seeks to isolate the propellants from the outside conditions and create conditions that we can control.

3. Integration considerations

The objective of Ground Support Equipment is to support the launch vehicle during preflight as well as the first moments of flight. Intercollegiate Rocket Engineering Competition (IREC) has requirements regarding launch structures. The launch vehicle (LV) must be supported by at least two rail guides when suspended horizontally. It is also dictated that the LV's weight must be supportable by the bottom rail guide vertically before preparing launch preparations. The launch rail must support a possible 70° launch elevation, along with having a variable azimuth. These launch conditions must be considered regarding launch stability. The launch rail must be able support a rail departure velocity of at least 100 ft/s and not become over or under stable during ascent.

C. Controls
1. Data Collection

Data collected by the Ground Station will be used to determine location of propellants in the fill system, vehicle conditioning, and readiness for launch. This will be achieved using a National Instruments NI USB-6210 connected to a single board computer (SBC) running LabView which will be collecting data and relaying it to the Ground Station.

2. *Valve Actuation*

Valve Actuation will be achieved using a bank of relays to trigger pneumatic solenoids for our S&K Automation 3000 Series pneumatic valves. The control of the relays will again be controlled through Labview on the SBC at the command of the ground station.

3. *Integration considerations*

Ground Systems requires the constant monitoring of vital components. This monitoring involves communicating with various electrical systems within ground systems equipment (GSE) and onboard the LV. Data Collection within GSE encompasses monitoring Pressure and Temperature at various locations along GSE fluid lines. Knowledge of these elements allows our Ground System to make various decisions regarding fill or purge operations. These operations encompass opening various valves to allow for the flow of propellants, pressurant, and compressed gas. In order to have redundant launch controls, GSE is intended to operate as a breakaway avionics system if there is an issue during launch or purge operations. This system is intended to be controllable from a minimum of 2000 feet away. Control will be attained over a 2000 foot optical cable, two router switches, an SBC, and a personal computer (PC). This system method is intended to avoid a loss of control during fill, purge, and launch functions. Ground Control is also intended to have an antenna to provide updated Avionics Data to the Ground Station from the LV. In order to achieve great quality and attainable data these systems will be defined and programed using National Instruments' Labview and Matlab.

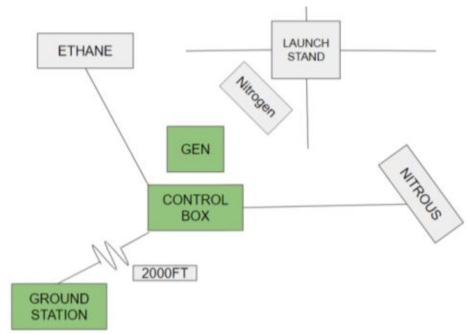

Fig. 12 GSE Launch Field Overview

D. Fluids

1. *Propellants*

Ethane and Nitrous Oxide are the chosen propellants for the LV. Both are high vapor pressure liquids at room temperature. This leads to a pressure requirement throughout the entire fill system but allows for the use of pressure fill operations for loading the LV. The system shown in Figure 13 is identical for both the ethane and nitrous oxide systems with only one caveat of increased cleaning requirements on the nitrous oxide system in accordance with ASTM MNL36 and CGA specifications for oxidizers.

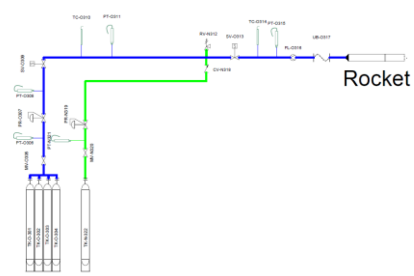

Fig. 13 GSE Nitrous Oxide System Schematic

2. *Pressurant*

Nitrogen has been chosen as the pressurant for the LV. The system shown in Figure 14 provides the high pressure nitrogen to the LV and pressurizes the LV for flight.

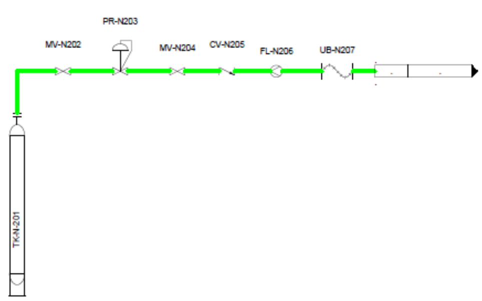

Fig. 14 GSE Pressurant System

3. *Purges*

In both propellant cradles there contains a singular bottle of gaseous nitrogen used for purging and drying the system in order to ensure a clean fluid system while in the field.

4. *Integration Considerations*

Each fluid has unique properties, and these properties can be hazardous if not considered properly. The main danger in our pressurized system is the amount of stored energy within the system. The Nitrogen pressurant experiences the most amount of pressure. The interconnection of Nitrogen with each fluid line leads to designing for the worst case scenario if our Nitrogen system fails. This provides greater factor of safety in our Ethane and Nitrous lines with the ability to maintain our Nitrogen requirements. Each of these systems have been modeled in Applied Flow Technologies Fathom; this software has allowed GSE to better understand the dynamics of our fluid system. This software has allowed us to avoid high line velocities and better determine fill times in varying design situations. To gain additional assurance in this system, calculations have been made to bring better scope to certain components. These components of our design have been relief devices, fluid line boil off rates, and an idealized fill time.

IV.Testing

A. Injector Testing

Injector Testing will be performed on the Workhorse Injector and Flight Injector. These tests will seek to characterize the discharge coefficient of the ethane and nitrous sections in order to properly bound the fluids model for the injector. Workhorse Injector testing will bound the properties of the injector prior to hot-fire and Flight Injector testing will verify design and model performance prior to hot-fire and launch. Figure 15 shows the Cold Flow Stand that will be used to perform this testing with the Workhorse Injector Mounted. Table 1 shows the Test Matrix for the Workhorse Injector tests.

Fig. 15 Cold Flow Stand w/ Workhorse Cold Flow Test Article

Table 1. Workhorse Cold Flow Test Matrix

Block	Goal	Burn Duration	Pressure
0	Activate Facility and Verify Flow	0.5-1.0s	50-200 psig
1	Flow Nitrous Oxide Element	3.0-5.0s	200 psig
2	Flow Ethane Manifold	3.0-5.0s	200 psig
3	Combined Flow	3.0-5.0s	200 psig

B. Recovery Testing

This rocket will use two recovery systems previously unused in SHC projects: The Tinder Rocketry Tender Descender and the Tinder Rocketry Raptor CO2 Ejection System. Because there are no previous organization projects that have validated the systems, testing is required to ensure that the systems operate nominally and that the team is comfortable with setting up the system. The Tender Descender will undergo 2 system tests: a load test and a flight test. The system will be tested to its maximum operating load to ensure that it can handle any erroneous forces of deployment. It will additionally be flown on a lower altitude rocket to ensure that it allows nominal recovery operation. The CO2 system will be tested 3 ways: a pressure chamber test, a ground deployment test, and a flight test. The system will be tested in a pressure chamber to ensure that it operates at the maximum altitude of the flight, as a flight test to 30,000 feet is prohibitively expensive and SHC does not currently have a vehicle capable of reaching that altitude. The ground deployment testing will have the recovery system packed into the final flight volume, and then use the CO2 system to eject the recovery elements on the ground. This is to ensure that the internal pressure created by the CO2 system is enough to overcome the frictional forces of the recovery against the body tube. The flight test will not reach maximum altitude of final flight, as the secondary black powder system would be more prone to failure at higher altitude. However, it will test the deployment of the parachutes in flight to ensure that recovery can operate successfully after deployment.

Depending on decision and purchasing timelines, the parachutes may be flight tested. Because SHC only has solid propelled flight vehicle, there is no vehicle that properly simulates the mass of the liquid propelled rocket. Additionally, the organization has pre-existing infrastructure for ground drag tests.

C. Hot-Fire Engine Testing

Hot-Fire Testing of the Workhorse and Flight Engine will be used to bound the thermal and thermochemical characteristics of the engine. Workhorse Engine tests will be used to guide the design of the Flight Engine and refine the design with experimental data. A Test Matrix for the Workhorse Engine tests is given in Table 2.

Table 2. Workhorse Engine Hot-Fire Test Matrix

Block	Goal	Burn Duration	Pressure
1	Verify start sequence & successful ignition	0.5-1.0s	150-200 psia
2	Increase thrust level	1.0-1.5s	200-500 psia
3	Determine max burn duration	1.5s-X.Xs	300-500 psia
4 (opt)	Ablative and TBC tests	1.5s-X.Xs	300-500 psia

IV. Acknowledgments

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V. References

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