Liquid Bipropellant Rocket Design

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In today's world, rockets serve a diverse range of purposes, and liquid-propelled rockets are a mainstay. Florida Tech BiProp is designing a liquid bipropellant rocket to compete in the FAR-OUT Competition on June 5-11, 2024, in the Mojave Desert. This document will cover the approach, technical design, and testing of the team rocket to achieve this goal. BiProp is a Capstone Project that builds off lessons learned from a 2022-2023 capstone project. The team is challenged with launching and recovering a liquid bipropellant rocket that must be launched to an altitude band of 5,000ft to 15,000ft and carry a payload above 1kg. The team has selected a target altitude of 6,000 ft, carrying a payload of 1 kg to the target altitude. It will be achieved by building a propulsion system that uses a propellant combination of Nitrous Oxide and Ethane, which provides a thrust-to-weight of 10:1 and a burn time of around 2 seconds. To accomplish this, the team has been divided into four technical subsystems: Grounds, Structures, Budget/Safety, and Avionics/Recovery. The team aims to complete a variety of tests before the design showcase in April of 2024: hydrostatic pressure tests of both tanks, an avionics test, and a static test fire. These tests are designed to validate the design, manufacturing, and construction of the final product to allow for revisions as necessary before the competition.

I. Nomenclature

A^*	=	Throat Area
A_e	=	Exit Area
V_c	=	Chamber volume
$\dot{m}$	=	Mass Flow rate
I_{sp}	=	Specific Impulse
P	=	Pressure
P_0	=	Stagnation Pressure
T_0	=	Stagnation Temperature
γ	=	Specific heat constant (1.2163)
$\bar{R}$	=	Universal Gas Constant
h	=	Heat Transfer Coefficient
v	=	Fuel Velocity
t	=	Time to Heat
q	=	Heat Transfer
D_r	=	Diameter of Drogue Parachute
D_m	=	Diameter of Main Parachute
A_m	=	Main Parachute Area
F_s	=	Parachute Snatch Force
H_l	=	Piping Head Loss
P_{loss}	=	Piping Pressure Loss
K	=	Loss Factor
P_{proof}	=	Proof Pressure

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II. Introduction

Many of the most commonly used rockets today utilize a liquid bipropellant design, often with some combination of liquid oxygen and various fuel options. Liquid propellants are an excellent choice for rocket design due to their high specific impulse and have been used in rocketry as far back as the 1920s when Robert Goddard first experimented with a liquid oxygen and gasoline-powered rocket design. Due to their prevalence in the modern market, it is essential that students develop a strong understanding of the principles and practice of liquid bipropellant rocket design. This project was conceived as an effort to provide that practical experience and to improve upon previous attempts at developing a liquid bipropellant rocket at the Florida Institute of Technology. Should the design prove successful, it will be the first rocket of its kind to be designed, tested, launched, and recovered in the school's history.

Due to the large scale, projected altitude, and propulsion used, it will not be possible to launch this rocket in the local area. It was determined by the team that the launch capability and developmental support needed to increase the project's chance of success would be best provided by the Friends of Amateur Rocketry, who hold a yearly rocket competition for student teams in the Mojave Desert of CA, called FAR-OUT (Friends of Amateur Rocketry – Oxidizers Uninhibited Tournament). Consequently, the main driver of the design choices for this liquid bipropellant rocket design are the rules and requirements provided by the FAR-OUT competition team. If the design succeeds beyond static testing, it will be flown at the competition during the week of June 5-11, 2024. The team has chosen to enter the 5,000 – 15,000 ft. category with an intended altitude of 6,000 ft.

The final rocket design (Fig. 1) uses Nitrous Oxide for the oxidizer and Ethane for the fuel. The rocket body consists of a Von Karman nosecone made of filament-wound fiberglass and an aluminum tip, a 1kg payload of epoxied lead shot, an aluminum oxidizer and fuel tank with bulkheads in double sheer with double O-rings and an AN-8 compression fitting and eyebolt ends, an impinging jet mild steel injector, a mild steel combustion chamber that can support a combustion temperature of 1300 K and 400 psi pressure, a conical nozzle made of mild steel, a 3-D printed avionics bay with altimeter and battery, four aluminum trapezoidal fins, and a recovery system that utilizes a 10-ft main and 5-foot drogue parachute. These design choices and systems will be explained in greater detail in section IV.

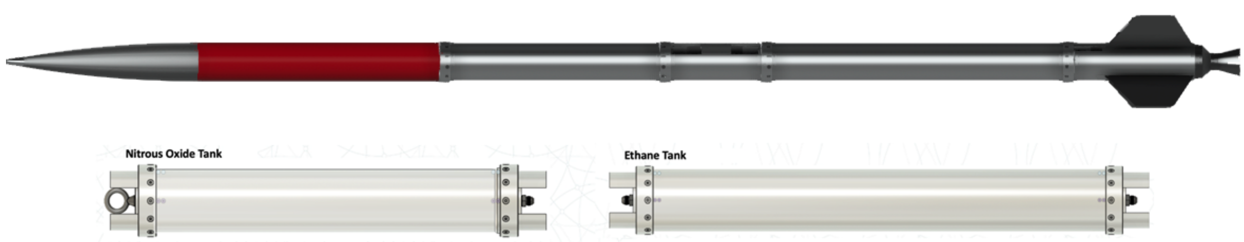

Fig. 1 Full rocket with nitrous oxide tank *(bottom left)* and ethane tank *(bottom right)*

III. System Objectives

A. Deliverables

The deliverables for the project were broken down into two main categories: documentation and analysis and hardware and software. The first category of documentation and analysis consists of all paperwork needed to compete in the FAR-OUT competition and paperwork required for the senior design project for Florida Tech. These include the System Requirements Review (SRR), Preliminary Design Review (PDR), and Critical Design Review (CDR) documents for analyzing the design, progress reports with technical personnel from FAR-OUT, and documents for budgeting. Additional paperwork will be developed for the project based on safety requirements and testing procedures. Analysis for the design of the rocket was conducted in Fusion360, OpenRocket, and Ansys. Primary analysis was conducted for the fuel and oxidizer tanks of the rocket.

The rocket's hardware and software consist of physical components, along with Python and MATLAB scripts for additional system analysis. The physical hardware developed for this project included the structures of the rocket, fuel, and oxidizer tanks, recovery bays, electronics bays, chamber, nozzle, injector, and fins. Software developed for the project included a Python script to analyze fuel and oxidizer tank lengths for an optimal burn. The script provided a burn time and specific impulse based on a target thrust value. The MATLAB script developed was used to determine the optimal size needed for both the drogue and main parachutes of the rocket, as well as to determine the expected loads during recovery.

B. System Requirements

To ensure success of the overall design of this project, several key system requirements were identified, based on rules and guidelines from the FAR-OUT competition as well as needs of the rocket design itself. Each of these requirements will be verified by analysis, test, inspection, or demonstration. The main requirements included a minimum thrust-to-weight ratio of 10:1. The rocket will also be required to reach an altitude of between 5,000 and 15,000 as established by the competition. Additionally, the rocket must be passively stabilized such that its pitch open ration is between 0.05 and 0.3, and its flight path does not deviate from vertial by more than 20 degrees during boost. The rocket will also include a liquid bi-propellant propulsion system and a two-stage recovery system. Per the rules of the competition, it must also carry a mass of at least 1.0 kg and have a GPS tracking system. Lastly, the team and the design must meet all safety requirements set forth by the competition officials and Florida Tech.

The systems architecture for this project is broken down into three main areas: Structures, Recovery Systems and Payload. The payload is detailed above. The recovery system includes all devices needed for successful recovery of the rocket including parachutes, GPS tracking, and deployment. Structures is the largest of the three and is broken up into additional subsystems, including Main Structure, Propellant Feed System and Propulsion (Fig. 2)

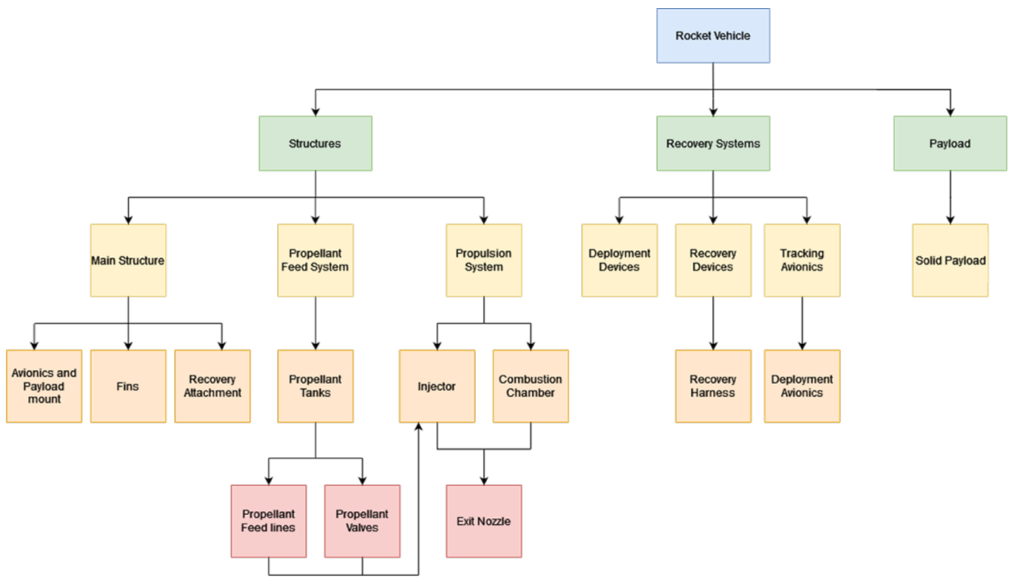

Fig. 2 BiProp systems architecture flow chart.

IV. Design Decisions

A. Injector, Combustion and Nozzle

1. Injector

An impinging jet injector is going to be used in the propulsion system. This design choice was carefully made after doing a trade study on different injector types. The main reason for this choice was the manufacturability of the part. The injector size is based on the total mass flow rate and Oxidizer/Fuel (O/F) ratio. The specified thrust-to-weight ratio required by the competition is 10:1 off the launchpad, whereas the current engine will be designed for 1500 lbf, giving the rocket approximately 13:1 at liftoff. The injector will be composed of three individual mild steel circular plates as it is a material that can withstand the expected pressure and temperature loads (approximately 500 PSI and 1328 K, respectively). These plates will have a diameter of 4 inches. The thickness of the plates will be 0.75 inches, which was optimized based on weight requirement and static stress simulation analysis. The injector will be radially bolted onto the chamber. There are O-rings on the 3rd injector plate in order to make the system leakproof.

All plates will be bolted together and attached to the chamber. The oxidizer and fuel enter the injector (Fig. 3) from the tanks; they are then distributed through the set of orifices that orient them in a collision angle. This will then cause individual propellant streams to impinge down in the chamber and combust. The number and size of the orifices are based on the required mass flow rate and ultimate thrust requirements. The Open Rocket simulator software was employed to conduct iterative simulations for rocket parameters and sizing. To achieve the desired altitude and launch speeds, a propellant mass flow rate of 8.29 lbs./s is required. Using Rocket Propulsion Analysis (RPA) software, the individual propellant mass flow rates were calculated at 3.569 lbs./s for nitrous oxide and 1.711 lbs./s for ethane.

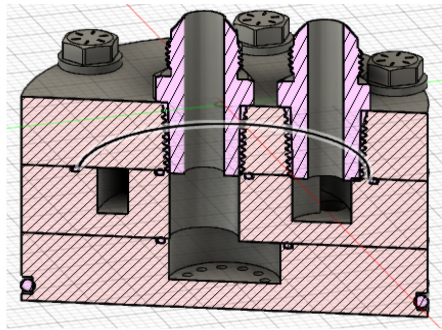

Fig. 3 CAD model of final injector design, section view.

2. Chamber and Nozzle

The combustion chamber had two main requirements, to generate a thrust of 1500 lbf and to not melt the engine material. Thus, we decided that the chamber should be made out of a mild steel tube with a thickness of 0.25 inches and a length of 6 inches. With the choice of nitrous oxide as oxidizer and ethanol as fuel, we performed chemical simulations to see what kind of state the combustion chamber would experience during operation. The chemical analysis was done with CEARUN courtesy of NASA. The input parameters of the program involved the operating pressure (400 PSI), what O/F ratio we wanted (This is highly dependent on the combustion temperature and desired I_{sp}), and finally, what exit pressure we would like from the nozzle. It was possible to run the program with multiple settings, so we examined O/F ratios from 0.1 to 10.1 and graphed the results. Final graphs are shown in Fig. 4 and 5.

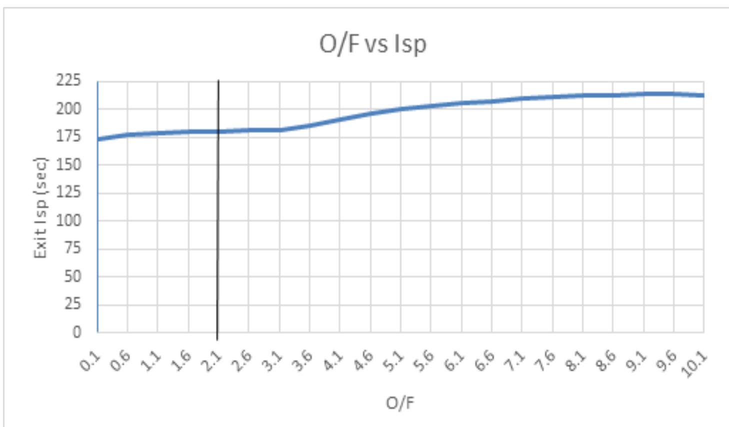

Fig. 4 Oxidizer/Fuel ratio vs exit I_{sp} (sec).

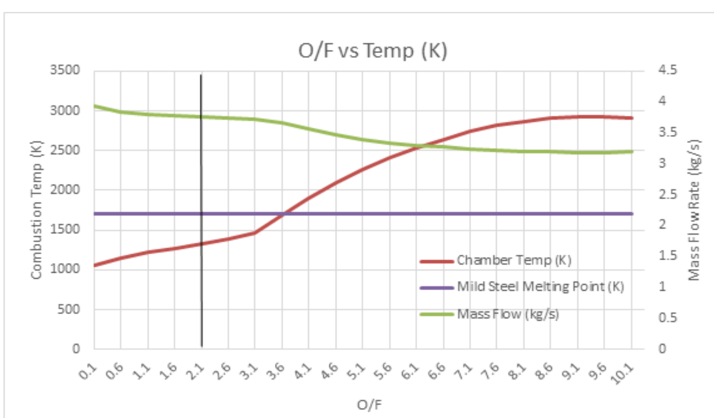

Fig. 5 Oxidizer/Fuel ratio vs. combustion temperature (K).

After examining the possible range of I_{sp} s from different O/F ratios, we saw there is a sweet spot at O/F = 2.1 where the chamber has a margin of ~350K from its melting point, but still produces around 180 seconds of I_{sp}. CEARUN also g a nozzle Ae/A* of 3.855 which was for an ideal 15-degree cone nozzle. The minimum required thickness of the combustion chamber wall was determined using the thin-walled pressure vessel calculation. That value was then taken multiplying it by the safety factor to get our final thickness of 0.25 inches. The rocket thrust equation was used to get the mass flow rate of the vehicle, Eq. (1).

$$F_{Thrust} = \dot{m}V_{exit} + (P_{exit} - P_{inlet})A_{exit} \tag{1}$$

Rearranging to get $\dot{m}$ from the desired thrust of 1500 lbf and known exit velocity of 5817.12 feet/s, we find the total mass flow rate of the engine is 8.296 lb/s. The final piece of the puzzle was finding the throat area given these calculated conditions. This value was found using Eq. (3).

$$A^* = \frac{\dot{m}}{P_0} \cdot \sqrt{\frac{T_0\bar{R}}{\gamma}} \cdot \left(1 + \gamma\frac{-1}{2}\right)^{\frac{\gamma+1}{2(\gamma-1)}} \tag{3}$$

A conical-shaped nozzle was selected due to its manufacturing simplicity compared to a bell nozzle. Performance calculations proved that there was not much efficiency lost due to the shape of the nozzle for the speed the vehicle was going to be at; therefore, we selected the conical nozzle. Now that we had the design parameters, we made the CAD model of the combustion chamber and nozzle. In order to simplify the manufacturing process, the nozzle will be made and then welded to the chamber. The diagram below shows the cross-section of the full assembly of the engine (Fig. 6).

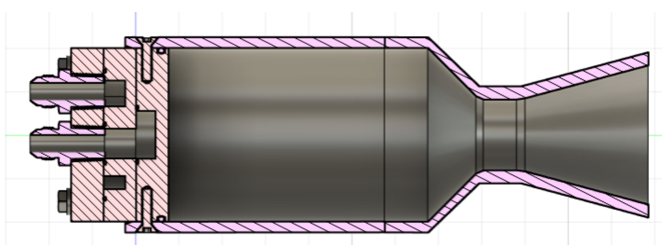

Fig. 6 CAD Design of Chamber and Nozzle.

B. Tanks and Bulkheads

The tanks are made of aluminium due to its ability to withstead pressure while being light. The volume of the tanks was gotten for volume calcuations of the propellants needed for the mission. The volumes were then used to get the heights of the nitrous oxide (oxidizer) tank and ethanol (fuel) tanks. To get the wall thickness we used the hoop stress formula shown in Eq. (4) as the weakest part of cylinder is the wall under hoop stress. We chose to have an external radius of 2.5. The self-pressurization of our propellants finds the pressure; nitrous oxide has a pressure of 720 psi, and ethane has a pressure of 540 psi. This gave us a minimum wall thickness of 0.1028 inches. After accounting for the safety factor, we decided to have a thickness of 0.15 inches.

$$thickness = \frac{P \cdot r}{\sigma} \tag{4}$$

We selected to used ¼-20 by 1 inch counter bolts as they are common, strong and cheap. After selecting the bolt we did shear stress calculations using the double shear formula shown in Eq. (5), due to the geometry of the tank bulhead. The internal diameter of the tanks is 4.5 inches.

$$P = F/A => F = PA$$

$$\sigma = \frac{4*F}{2\pi d^2} \tag{5}$$

The bearing stress was then calculated for the inner wall using Eq. (6). Accounting for the safety factor we decided to have 12 bolts radially ataching the bulkheads to the tanks. Each bulk head was designed to have two O–rings inoder to make the system leakproof. Below is the final assembly of the tank with the bulkhead (Fig. 7).

$$B_s = \frac{F}{t * d} \tag{6}$$

$t \text{ is thickness of bulkhead wall and } d \text{ is diameter of bolt}$

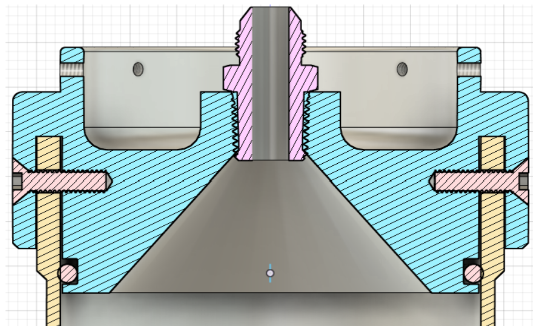

Fig. 7 Cross-section of final assembly of bulkhead and tank.

C. Avionics and Recovery

The avionics and recovery sytems were picked based off the FAR-OUT requirements. The avoinics package consists of two AIM XTRA 4.0 altimeters with built in GPS modules and antennae. Live data can be transmitted from the altimeters with the use of a ground station. This ground station will allow the altimeter pyrotechnic charges to be manually ignited if needed during descent. The complete assembly of the avionics bay can be seen below. It includes the two blue batteries for each altimeter and other components of the avionics bay to keep it together (Fig. 8).

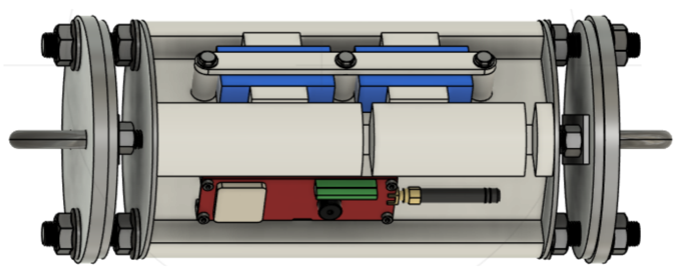

Fig. 8 Complete avionics bay design.

The design of the avionics bay was based on the design concept that the payload for the rocket would be housed within the inner section of the avionics bay. This was chosen to save space in the rocket due to the fact that the requirement for the payload was only that it be at least one kg in mass. The two ends of the avionics bay are u-bolts, which are $3/8^{th}$ inch to help with the recovery loads experienced by the rocket.

The descent speeds under a drogue chute and main parachute were based on the competition requirements of less than 150 ft/sec under drogue chute, and less than 30 ft/sec for the main parachute. With these design requirements in mind, the drogue parachute for the design was determined to be 5ft in diameter to achieve a descent velocity of 70 ft/sec, and the main parachute was sized at 10ft in diameter with a descent velocity of 25 ft/sec.

D. Fluid Systems

The rocket's fluid systems (Fig. 9) are vital for ensuring a controlled supply of fuel and oxidizer to the injector, guaranteeing the desired mass flow and thrust during launch. Ground systems store fluids under pressure until launch, where pressure forces them to the manifold, maintaining separation until distribution to the injector. Operating on principles of fluid mechanics and chemistry, the system is verified through hand calculations and hydrostatic testing.

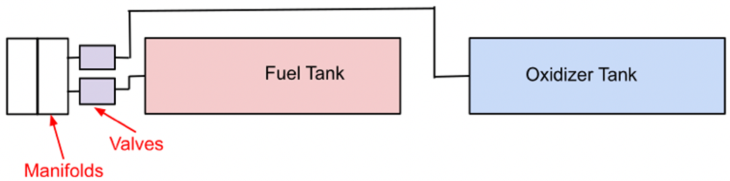

Fig. 9 Basic Fluid System Layout.

The final layout includes fuel and oxidizer lines. Using specific calculations based on Bernoulli's equation, Eq. (7), with velocities of 83.92 m/s for the ethane and 79.34 m/s for the nitrous oxide, the size chosen was a ½" outer diameter for both lines. The results of a trade study on materials as well as considerations of the tube size, desired strength and factor of safety of 2, led to the final choice of Aluminum 6061 for the tubing. These design choices will ensure the needed mass flow of 1.21 kg/s for ethane and 2.55kg/s for nitrous oxide and that we meet our safety and system requirements. Equation (8) shows the calculation used to establish the minimum tubing diameters of 0.146" for the oxidizer and 0.322" for the fuel line. The larger size of ½" was ultimately chosen for both lines because boundary layer implications were not considered in these calculations.

Sample calculation for tubing mass flow, with oxidizer example:

$$P_1 + \frac{1}{2}\rho v_1^2 + \rho g h_1 = P_2 + \frac{1}{2}\rho v_2^2 + \rho g h_2 \rightarrow v = \sqrt{\left(\frac{2\Delta P}{\rho}\right)}$$

$$v = \sqrt{\left(\frac{2(5102120.4Pa - 2757902.92)}{744.8\frac{kg}{m^3}}\right)} = 79.34 \text{ m/s} \tag{7}$$

Sample calculation for tubing minimum diameter, with oxidizer example:

$$\dot{m} = \frac{m}{t} = \rho v A = \rho v \cdot \frac{\pi d^2}{4} \rightarrow d = \sqrt{\left(\frac{4\dot{m}}{\pi \rho v}\right)}$$

$$d = \sqrt{\left(\frac{4 \cdot 2.54981\frac{kg}{s}}{\pi \cdot 744.8\frac{kg}{s} \cdot 79.34\frac{m}{s}}\right)} = 0.0074\text{m} = 0.2918\text{in} \tag{8}$$

Divide this value by 2 since there will be 2 oxidizer pipes. d = 0.146"

To meet mass flow requirements of the rocket, a Pyrotechnic Half Cat Valve design was chosen (Fig. 10), with a compact design the easily fits within the rocket tube and allows unrestricted fluid flow when ignited. The valve, under 4 inches long and 2 inches in diameter, is designed to control the flow of fuel and oxidizer effectively. It comprises a cylindrical housing with opposing tapped NPT 1/2" entry and exit holes, a 7/8" passage lengthwise, narrowing to a 1/2" passage at top, a slider with O-rings to block fluid flow, and a potassium-nitrate sucrose structure that burns away upon ignition. The E-match, connected to the engine controller, initiates the valve's opening before the engine ignites. The slider design aims to minimize stress on the solid propellant structure, and the overall design is based on successful past models, modified to meet current flow rate requirements.

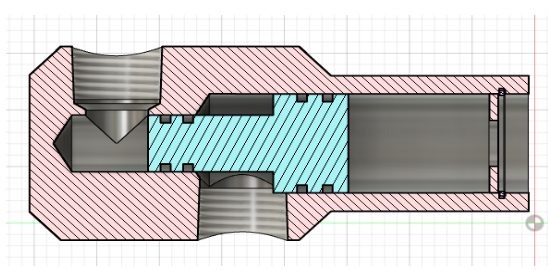

Fig. 10 Valve Assembly Cross Section.

Minor and major losses in the system were calculated using friction factors from a Moody Chart, considering parameters like velocity, diameter, and estimated lengths. For the fuel lines, the head losses are measured at 91.01m, while for the oxidizer line, it amounts to 687.2m (Eq. 9). Minor losses, attributed to specific geometric discontinuities in the tubes, are quantified at 1115.8m for oxidizer and 675m for fuel. These minor losses are disproportionately high due to the relatively short length of the tubes.

Sample calculation for head losses, with oxidizer example:

$$\dot{m} = \rho v A \rightarrow v = \frac{4\dot{m}}{\pi \rho d^2} = \frac{4 \cdot 2.54981\ \frac{kg}{s}}{\pi \cdot 744.8\ \frac{kg}{m^3} \cdot (0.007747m)^2} = 72.63\ \frac{m}{s}$$

$$Re = \frac{\rho v_{avg} d}{\mu} = \frac{744.8 \frac{kg}{m^3} \cdot 72.63 \frac{m}{s} \cdot 0.007747m}{1.54 \cdot 10^{-5}\ Pa \cdot s} = 27187918$$

$$\text{Relative Roughness: } \frac{\varepsilon}{d} = \frac{0.0015 \cdot 10^{-3} m}{0.007747m} = 0.00193$$

$$\rightarrow f = 0.0132 \text{ Friction factor from Moody chart}$$

$$h_{maj} = f \cdot \frac{L}{d} \cdot \frac{v^2}{2g} = 0.0132 \cdot \frac{1.5m}{0.007747m} \cdot \frac{\left(72.63 \frac{m}{s}\right)^2}{2 \cdot 9.81 \frac{m}{s^2}} = 687.17m \tag{9}$$

The estimated pressure losses for the fluid system are specified as 194psi for the oxidizer lines and 30.5psi for the fuel line. This corresponds to a pressure of 540psi in the oxidizer manifold and 510psi in the fuel manifold. Analysis indicates that the pressure required in the manifold exceeds the pressure of the combustion chamber by approximately 100psi (combustion chamber pressure: 400psi), emphasizing that the natural losses in the system may not be sufficient.

In response to these findings, it was determined that introducing a necking region to the tubing could effectively address both the reduction in mass flow rate and meet the required pressure. A valve will be manufactured from aluminum, chosen for its low weight and high ultimate strength, and attached to the pipes using a ½" 37° flared NPT fitting. Before the pipes are mated to the tanks, tests on the valve will be conducted, as well as compressive stress testing of the solid propellant. Once validated, the valve will be integrated into the rocket system, and a hydrostatic test will be performed on the entire system to ensure its reliability under pressure.

In summary, the design of the fluid system combines calculations, material considerations, and valve selection to ensure a precise supply of fuel and oxidizer. The current focus lies on manufacturing, rigorous testing, and seamless integration into the rocket's overall structure.

E. Ground Systems

Ground systems are a crucial component of this project and play an integral role in the overall success of the rocket. From fueling, to ignition and departure from the launch rail, the grounds systems facilitate smooth and efficient operations. As stated, there are two main components of the grounds systems for this project, those being for fuel and oxidizer loading, and launch of the rocket. Loading the rocket with propellant safely and effectively is a challenge, where many designs and plans of attack were considered. The propellant loading system must utilize a quick disconnect so that the umbillical feeding the propellant into the rocket can detatch remotely during launch. The quick disconnect is in line with a fill valve that is remotely actuated by solenoids, allowing the propellant to feed into the rocket while maintaining a safe distance, and in order to minimize anomalies in the flow procedure. An example of the grounds systems architecture can be found in Fig. 11 below.

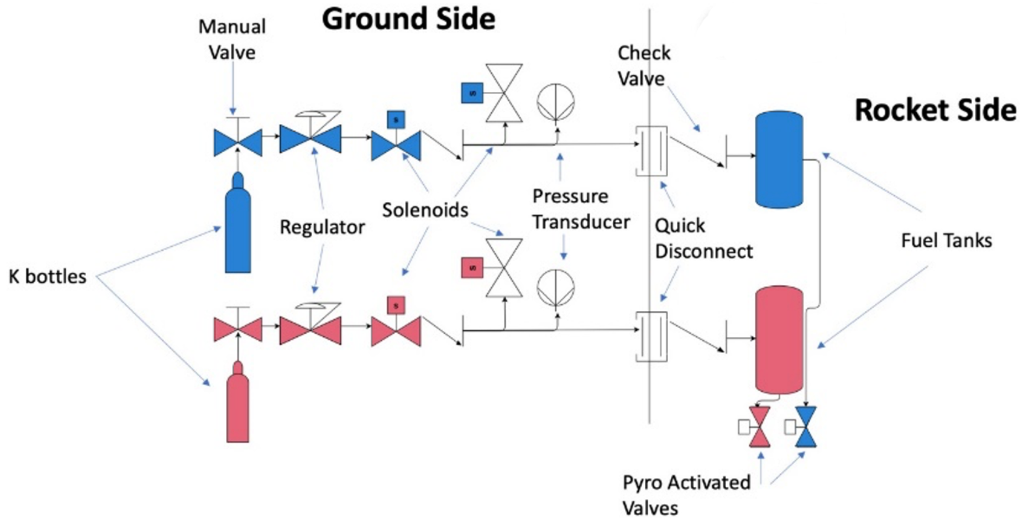

Fig. 11 Ground Systems Architecture

Launch of the rocket is initiated using a remotely controlled hotwire fed into a small low-power model rocket motor, which is placed inside the combustion chamber. This system is very similar to how amateur solid rockets are ignited and is proven to be reliable with a low technical cost. The propellant must begin flowing while the small rocket motor is ignited, which will create a combustion cycle that will nominally continue until the propellant tanks are emptied. Although there are many ways to ignite a rocket of this fashion, this design choice was chosen as it was seen to carry the least risk of failure.

V. Manufacturing and Testing

A. Manufacturing

We plan on manufacturing most of the parts by ourselves at Florida Tech's L3Harris Student Design Center (L3HSDC) and the campus machine shop. The bulkheads for the avionics bay are made of aluminum and were manufactured using the Lathe and mill. The avionics bay will be 3D Printed using PLA. The tanks and bulkheads were also manufactured using the lathe and mill. The injector plates will be manufactured using the waterjet machine and the CNC Lathe. The chamber will be cut to the required length using the bandsaw. The nozzle will be manufactured using the CNC lathe and then welded to the chamber. The valves will also be manufactured using the lathe and mill. Below is an example of the drawing for the nozzle (Fig. 12).

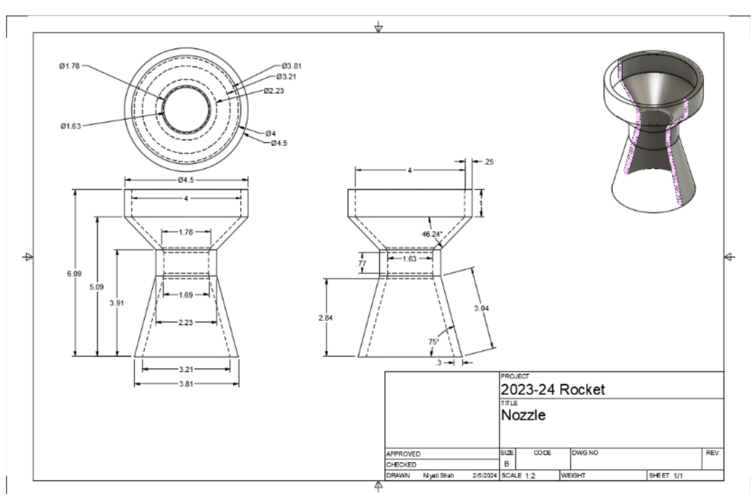

Fig. 12 Nozzle Drawing

B. Testing

Testing is a crucial phase for every component of the rocket. After the manufacturing of a component, it needs to be tested to prove that it will be able to perform its task. Eight main tests were identified to ensure the rocket would work as intended. The main tests are as follows:

Electronics Test:

The purpose of the electronics test is to validate the design choice and configuration for the ground and avionics subsystems. This test will satisfy all ground subsystem requirements and other avionics and structural requirements. Electronics test subjects include controllers, relays, switches, wiring, power storage, payload, and engineering resources such as cameras.

Avionics Flight Test:

The avionics flight test ensures that the package is functioning properly in flight. The avionics package is integrated into a lower-power rocket that is easily recoverable, where flight data will be collected and analyzed to ensure the equipment is working properly.

Avionics Deployment Test:

The avionics deployment test validates the process of deploying the recovery parachutes during flight. The test involves loading a secured rocket section containing all recovery components in the form they will be in during flight. The pyrotechnic devices at the bottom of the sections will be actuated, which should then blow the nose cone and mid-section off to deploy the parachutes. This test is important in the selection of the pyrotechnic charge used to deploy these parachutes.

Hydro-Static Tank Test:

The hydro-static tank test validates the structural integrity of the fuel and oxidizer tanks used for this rocket. The test involves pressurizing the tank to at least 1.5 the expected operating pressure and holding (while losing less than 1 psi per minute) for five minutes. The tank is pressurized via a solenoid actuated fill valve and relief valve, connecting the tank directly to the k-bottle of nitrogen which is used as the pressurant. The solenoid-actuated valves allow for remote pressurization, which increases safety during the test.

Hydro-Static Injector/Combustion Chamber Test:

The hydro-static injector and combustion chamber tests validate the structural integrity of those subsystems. This test is similar to the hydro-static tank test, where the injector and combustion chamber are pressurized to 1.5 the operating pressure and held at that pressure for five minutes. This test can be done using the same solenoid valves to maximize safety during the test.

Hydro-Static Full System Test:

The hydro-static full system test validates the structural integrity of the entire rocket fluid system. This test also validates the effectiveness of the integration of all the fluid’s subsystems. This test will pressurize the tanks, combustion chamber and injector, and the fluids tubing for the fuel and oxidizer. This test is conducted similarly to the hydro-static tests previously mentioned.

Valve Actuation Test:

The valve actuation test validates the effectiveness of the explosively charged valves used to facilitate the rocket's ignition. This test is done by loading the valve with the explosive charge and detonating it from a safe distance. The valve is then inspected afterwards to ensure it worked as intended.

Static Fire Test:

The static fire test validates the ability of all fluids and structures subsystems to perform as intended once fully integrated. This test involves securing the rocket to a thrust stand containing load cells to measure the force from the rocket's burn. The rocket is then fueled using the ground systems procedure and remotely ignited. The data collected can be analyzed to validate our simulation and performance calculations.

VI. Safety and Risk

For a project dealing with explosive charges, pressurized flammable gases, and heavy machining equipment, it is especially important to always keep safety at the forefront during every step of the process. It is easy to let safety considerations slip when deadlines are at risk of not being met, but using proper safety practices ensures that everyone can walk away at the end of the day in good condition.

One of the most important documents utilized for mitigating risks with dangerous materials are Safety Data Sheets (SDS's). SDS's accompany most materials used in the manufacturing and operation of this rocket, where they outline the properties, proper handling practices and proper safety equipment needed to handle those specific materials. All of these SDS's, as well as other procedures for manufacturing processes need for this project are all kept in a safety binder that is accessible to everyone contributing to the project.

While safety risks are a big concern in this kind of project, they are not the only risks that need to be addressed. Budget and scheduling risks are very prevalent and can stop a project dead in its tracks. It is important to consider that anomalies can cause a disturbance in the flow of the project, where the impact could vary depending on the circumstances. It is important to come up with plans early in the project so that if an event causes a major setback, there is already a set plan to work off while the project keeps moving forward. Budgetary considerations should be made well in advance, ensuring that the project will maintain the proper funding until completion.

VII. Future Work and Considerations

The results of this project have several long-term implications. The knowledge gained through the development and testing of this liquid bipropellant rocket design will add to the body of student-led rocket research that has been conducted at Florida Tech and will also be made available to future participants in the FAR-OUT competition to learn from. Future work may include revisions to the design based on lessons learned through the design process, such as the need to find the proper balance between factor of safety and mass for the tank design and increased use and understanding of simulations to better model what might be expected during development and before launch. Other areas for consideration include alternative fuel and oxidizer combinations, as well as testing and determination of more optimal design configurations.

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References

[1] Sutton, G. P., and Biblarz, O., *Rocket Propulsion Elements*, Hoboken, NJ: Wiley, 2017, p. 206.

[2] Goddard Technical Standard GSFC-STD-0002, 2009, NASA

[3] "ASM Material Data Sheet," *ASM Aerospace Specification Metals, Inc.* Available: https://asm.matweb.com/search/SpecificMaterial.asp?bassnum=ma6061t6

[4] Parks, K., "Aerospace Fittings," *SDSU Rocket Wiki* Available: https://elon.sdsurocketproject.org/wiki/_media/fluids/aerospace_fittings.pdf.

[5] "Parker O-Ring Handbook, ORD 5700," Parker Hannifin Corporation, Cleveland, OH, 2021, https://www.parker.com/content/dam/Parker-com/Literature/O-Ring-Division Literature/ORD-5700.pdf

[6] Bickford, J.H., and Oliver, M., *Introduction to the Design and Behavior of Bolted Joints,* "Non-Gasketed Joints", Marcel Dekker, Inc., Babel, New York, 1995.