Solid Propellant-Based Alternative Propulsion System for Small Satellites

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Currently, small-satellite propulsion is limited to either gas, liquid, or electric-based systems; however, each one of these has its drawbacks. An alternative solid propellant system would offer a simpler, less complex, and more energy-dense solution. However, solid propellant has typically not been used due to the lack of thrust control. This paper will investigate a proposed technical solution to this issue of throttling solid propellant for small spacecraft usage. The system is known as the Solid Propellant Adaptive and Responsive Combustion Control (SPARCC). SPARCC would consist of pellets of propellant that ignite and pressurize a chamber, then the pressure would methodically be released using an electromechanical valve to achieve control. Critical parts and interfaces of SPARCC have been identified, characterized, and designed. Although the applied use case for this system would be onboard small spacecraft, this specific investigation focuses on proving the propulsion system would generally work. This is to say, typical considerations for satellite systems (mass, electrical power consumption, etc.) have not been strictly abided by. Future work includes optimizing the system for these characteristics.

I. Nomenclature

Vol_{chmb}	=	Volume of the Combustion / Pressure Chamber
Thr	=	Thrust
A	=	Area
MW_{prod}	=	Molecular Weight of the Combustion Products
γ	=	Ratio of Specific Heats
R_{univ}	=	Universal Gas Constant
T_{total}	=	Total Temperature
P_{amb}	=	Ambient Pressure
$\dot{m}$	=	Mass Flow Rate
a	=	Local Speed of Sound
M	=	Mach Number
V	=	Velocity
n	=	Number of Moles
m	=	Mass
ρ_{grain}	=	Density of a Propellant Grain
t	=	Thickness
S	=	Factor of Safety
σ	=	Yield Strength
r	=	Radius
P	=	Pressure

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II. Introduction

Propulsion systems exist on spacecraft for various purposes; for example, establishing orbits, correcting, correction burns, perturbation correction, or docking maneuvers [4]. The selection of an appropriate propulsion system is often driven by the mission the spacecraft is carrying out. Most small spacecraft do not have propulsion systems due to size, mass, power, and operational constraints [2]. As a result, small spacecraft are very limited in the type of missions they can conduct. As more complex missions arise, the need for small spacecraft that can maneuver themselves is going to rise; for example, see Refs. [1, 3]. Currently, propulsion systems that can provide enough thrust to quickly maneuver a spacecraft are gas or liquid based propulsion systems [6]. Solid propellant-based propulsion systems do currently exist but are primarily single use or operate on milli-newton levels of thrust, primarily utilized for attitude control [6]. Solid propellant has thrust and specific impulse values similar to liquid propellant systems, but has the advantage of being lower complexity, thus lower cost [6]. However, the greatest disadvantage of solid propellant is its lack of controllability, once lit it typically cannot be stopped. This paper will explore a solution to this issue of solid propellant controllability, and its possible application to small spacecraft missions. This proposed solution has been named the Solid Propellant Adaptive and Responsive Combustion Control (SPARCC) system.

This project is currently being worked on by a senior design capstone team at the Florida Institute of Technology and is a continuation of a 2021-2022 senior design capstone team. The work being presented in this paper is the efforts of the current team, and any design decisions, component choices, or general work done by the previous team will be mentioned as such. Since this is a unique propulsion system and due to limited resources, the scope of the project was set to design a proof-of-concept level propulsion system. As a result, the system is not constrained to typical considerations for a satellite system (such as mass, volume, power, etc.). Once the team has a better idea for the system characteristics and functionality, mitigation of these previously mentioned budgets would be done as future work.

III. General System Functionality

Instead of trying to have full control of the combustion of solid propellant, the SPARCC system completely burns small pieces of solid propellant in a sealed tank. Once this tank has been pressurized by a single piece of propellant, the gas can be released on command using an electromechanical value. As a result, the amount of delta V the propulsion system can produce is proportional to the amount of propellant that can be stored onboard. To achieve this, a combustion/pressure chamber capable of being resealed by a mechanical system, solid propellant, and sensors that actively monitor the conditions of the system are required.

To better illustrate the structure and mechanisms of SPARCC, a digital model of this simplified system can be seen below in Figure 1. On the top of this figure is the model without any labels. Below is the same model, with major components labeled including hardware utilized to operate the system. The diagram is color coded to identify which subsystems were responsible for the different components of SPARCC: Propellant (Green), Mechanisms (blue), Structures (brown), and Control Systems (yellow).

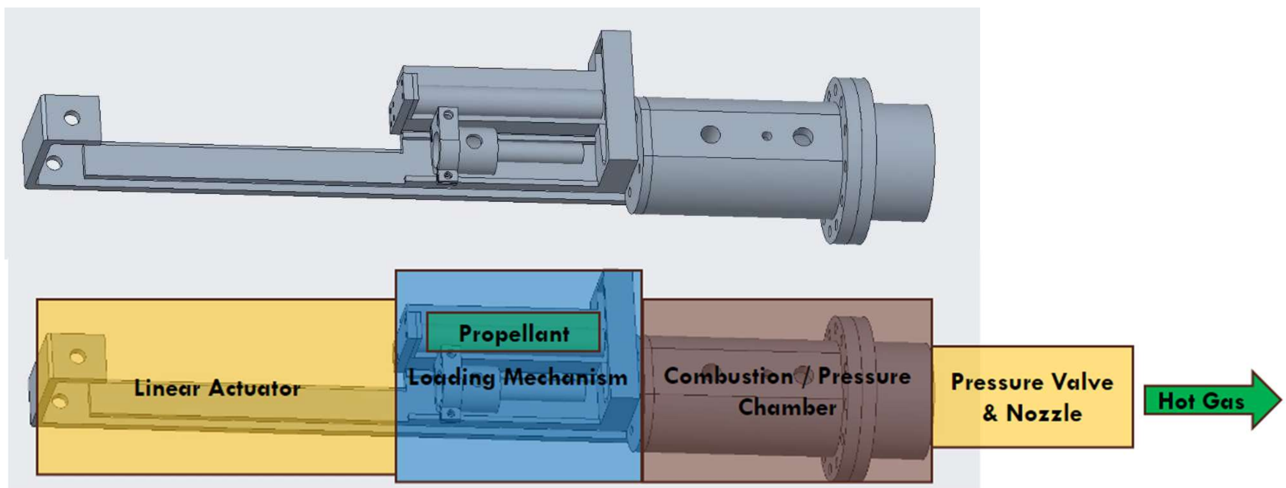

Figure 1: System Layout and Division

The propellant subsystem is responsible for propellant grain manufacturing and design. Mechanisms is responsible for ensuring multiple grains can be stored onboard and transferred to the combustion chamber without compromising the pressure within the chamber or the other propellant stored in the system. The structures subsystem designed the pressure vessel along with its interfaces to ensure the system was able to withstand the stresses of pressurization. Controls selected and integrated all hardware and sensors for the system.

Below is the general concept of operations for SPARCC (Figure 2).

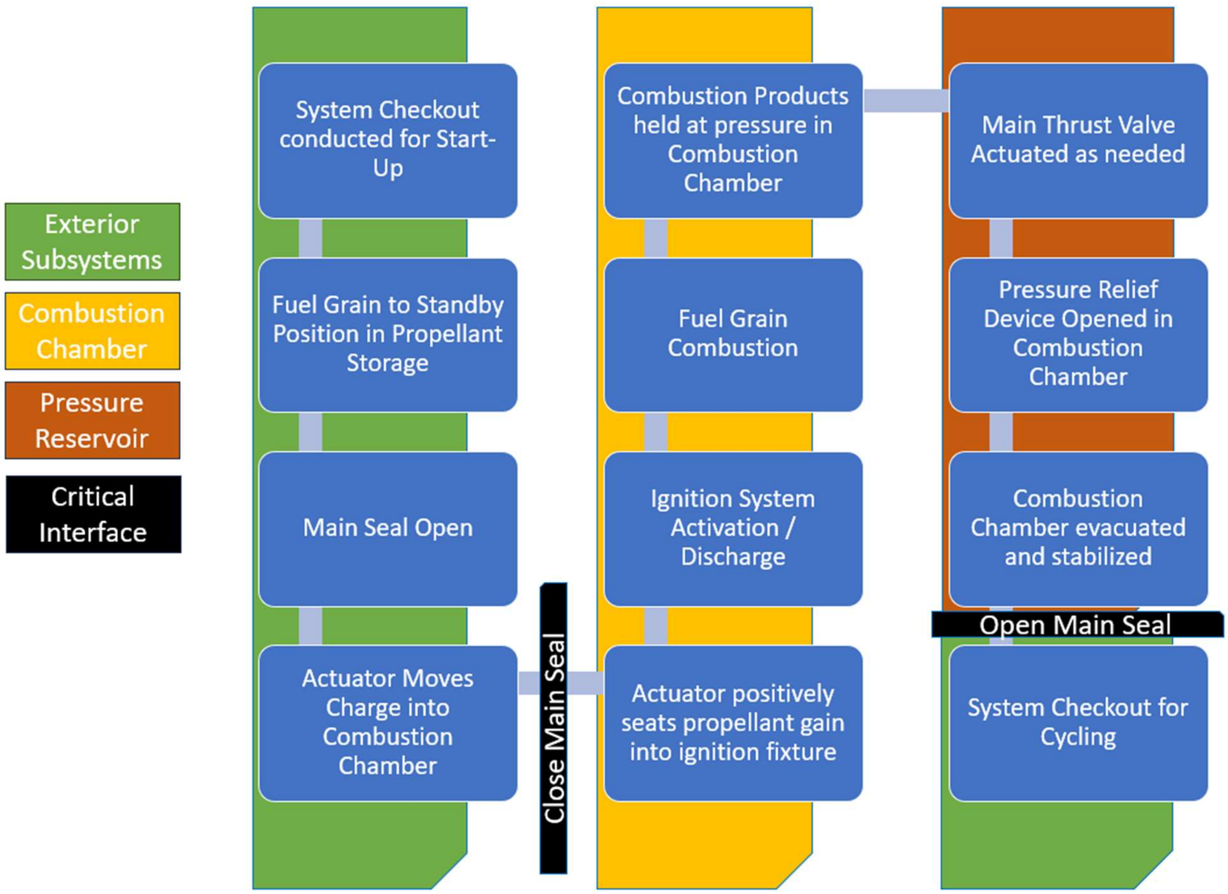

Figure 2: Concept of Operations

The major phases of the operation are color coded in the figure above to better emphasize the key events. The system starts by transferring a propellant grain from storage into ignition position (green). Once this has occurred, the system ignites the propellant grain, and the tank is pressurized (yellow). Finally commanded thrust is provided by actuating the valve (red). Once pressure has depleted, the tank is opened, and the cycle can repeat.

IV. Loading Mechanism Design

For reference, Figure 3 and table 1 below depicts and labels the specific parts of the loading mechanism mentioned in this section.

Table 1: Loading Mechanism Component Labeling

Number	Component Name	Component Material
1	Linear Actuator	Purchased Component, Mostly Aluminum
2	Wing Ring	Aluminum
3	Piston Cap	Steel
4	Wings	Aluminum
5	Back Plate	PLA (3D Print)
6	Magazine	PLA (3D Print)
7	Bracket	PLA (3D Print)
8	Combustion Chamber	Steel

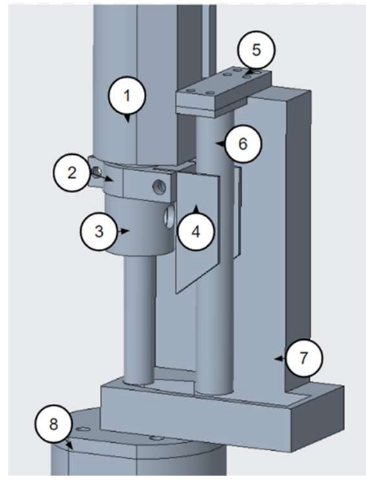

Figure 3: Loading Mechanism Model

The loading mechanism was designed to transport stored propellant grains to an ignition source inside the combustion chamber. The ignition of each propellant grain will be carried out by a single glow plug located within the combustion chamber. Although a custom ignition system could have been developed to reduce power and weight, it was opted to go with a prebuilt solution to reduce the scope of this project.

Moving propellant grains into the combustion chamber naturally proposes some constraints, namely the seal between the piston cap (labeled a 3) and the combustion chamber (labeled as 8). For this design, a ring system similar to that seen in car engines, was selected. Two rings will be housed near the bottom of the piston cap, and when inserted into the combustion chamber, will push against the sides of the combustion chamber wall to create a seal. As for propellant grain storage, the design is a spring-loaded magazine that will be attached to the main body of the loading mechanism, labeled as 6. This magazine will be mounted to a back plate, labeled as 5. This semi modular design was chosen for its ease of assembly and reloading during testing.

The linear actuator (labeled as 1) takes two positions during a loading cycle; extended into the combustion chamber and retracted out of the combustion chamber. These positions are depicted below in Figure 4.

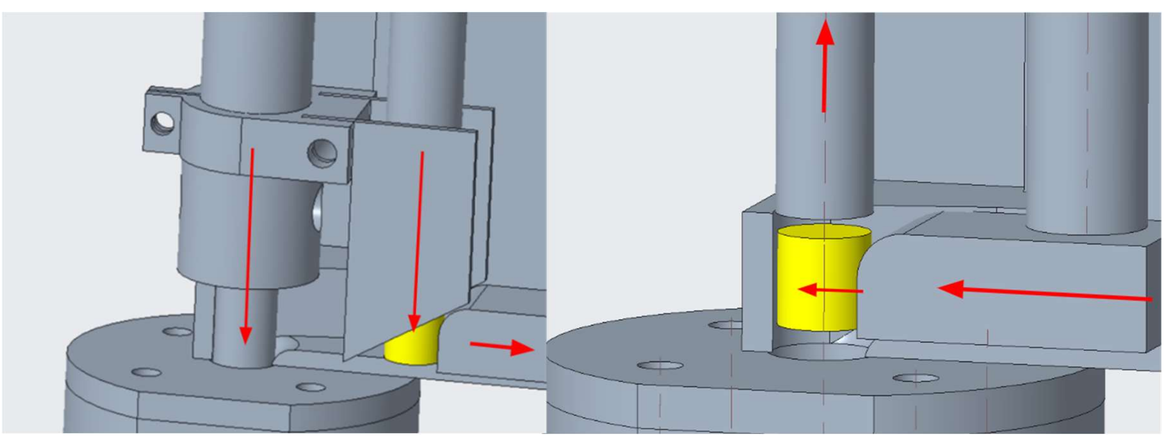

Figure 4: Extended (Left) and Retracted (Right) Positions

When the linear actuator is in its extended position, there are a pair of wings (labeled as 4) that are connected to the actuator above the piston cap that push a spring-loaded block away from the bottom of the magazine. This action allows a single propellant grain to get deposited out of the magazine. Once the piston retracts back from the combustion chamber, these wings no longer restrict the spring-loaded block. This allows the block to push a propellant grain into the path of the linear actuator. The linear actuator then extends into the combustion chamber, creating a seal with the piston rings, and remains there until combustion is complete and the generated pressure is fully released. Once the pressure has been spent, the loading cycle will begin anew.

Complex geometry which is not expected to receive any significant loads or heat will to be 3D printed out of PLA. Although a system designed for a satellite should not have any parts made of PLA on board, this compromise was made for the scope of the project. Parts which receive significant loads, but do not interact with any significant temperature, are aluminum for ease of manufacturing. Components which interact with a significant temperature and hold significant loads, are made of steel. The material choice for each component can be seen in Table 1.

V. Propellant Choice and Sizing

The propellant for this system is a mixture of Hydroxyl-terminated Polybutadiene (HTPB) as fuel and Ammonium Perchlorate (NH_4ClO_4) as the oxidizer. The propellant also contains a curative of Isodecyl Pelargonate and a plasticizer of modified MDI Isocyanate. This was a decision made by the previous team working on this project, and this previous team had also chosen an oxidizer to fuel ratio (O/F) of 4 for each propellant grain. Although they moved forward with this O/F ratio, they had reported issues with residue and propellant stiffness. The goal of SPARCC is to store propellant onboard, and to be used multiple times, so a malleable propellant grain which leaves significant residue once burned was not acceptable. After some initial testing, the current team had determined to get these optimal characteristics, the same propellant can be used, but at an O/F of 12 instead of 4.

The physical shape of each grain was driven by requirements for storage and motion around the system. The design needed to be modular and allow for easy storage, while also allowing for easy transfer into the combustion chamber. Ultimately, a cylindrical design was chosen for these grains. Below, in Figure 5, a manufactured grain can be seen. Each of these grains are 13 mm in height, and 10 mm in diameter.

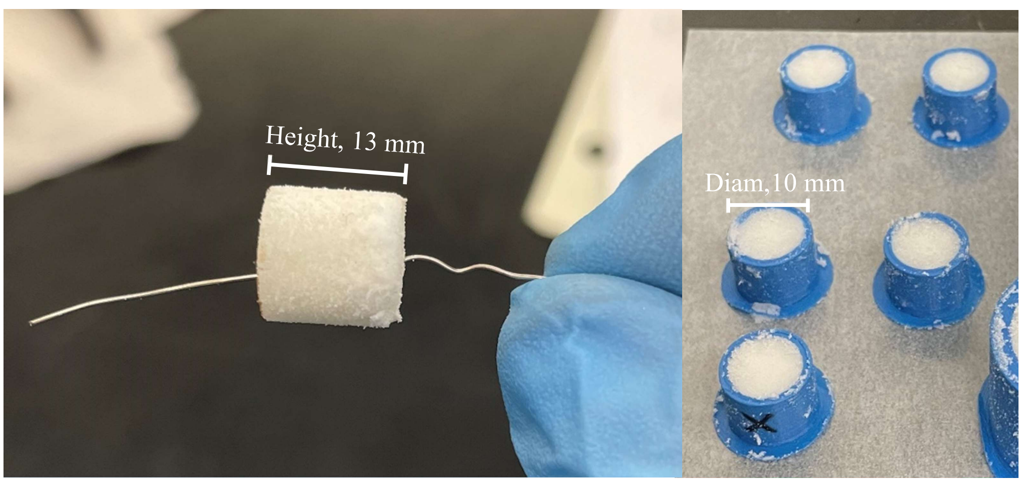

Figure 5: Propellant General Shape and Dimensions

These specific dimensions were determined based on a desired thrust output and chamber volume. As per the State-of-the-Art of Small Spacecraft Technology from NASA, the typical thrust from a cold gas propulsion system on board a satellite can range anywhere from 10 micro-Newtons to 3.6 Newtons [6]. The surveyed solid propellant solutions ranged from 37 to 461 Newtons. Because SPARCC is a sort of hybrid between a gas and solid propellant system, a number in between these values was chosen as the desired thrust, around 20 newtons. It should also be noted that the previous team had manufactured a nozzle to be used for this system. As nozzles are typically expensive and complicated to manufacture, the current team has decided to reuse this nozzle.

Knowing the nozzle geometry and desired instantaneous thrust output, the mass required of a single grain can be calculated. Other important metrics, such as the molecular weight of the combustion products, total temperature, and ratio of specific heats, were either assumed or measured directly during experimentation. Below, in table 2, all values used in the equations found in this section and a brief justification of their values can be found.

Table 2: Values, Symbols, and Explanation of Variables

Name	Symbol	Value	Justification
Volume of the Chamber	Vol_{chmb}	150 cm^3	Desire to have a high chamber pressure, and suggestions from the previous team.
Thrust	*Thr*	20 *Newtons*	Based on a NASA survey of small satellite propulsion systems (State-of-the-Art of Small Spacecraft Technology, 2023).
Nozzle Exit Diam	D_{exit}	5.88 *mm*	Old nozzle provided by previous team.
Nozzle Throat Diam	D_{throat}	3.04 *mm*	
Molecular Weight of the Combustion Products	MW_{prod}	0.0881 *kg/mol*	Measured in the lab during volume tests.
Ratio of Specific Heats	γ	1.28	Assumed - commonly used value for propulsion applications.
Universal Gas Constant	R_{univ}	8.3144 $kg/(mol * K)$	This is the universal gas constant and its units used in all equations in this section.
Total Temperature	T_{total}	1500 *K*	Assumed / Measured – Approximate flame temperature of the propellant.
Ambient Pressure	P_{amb}	0 kPa	The system is designed to operate in space, where there is no ambient pressure.

Ultimately, to determine the size of a propellant grain, the first step is to determine the area ratio of the provided nozzle. This can be found by dividing the area of the exit plane, A_{exit}, by the area of the throat plane, A_{throat}.

$$\frac{A_{exit}}{A_{throat}} = 3.7412$$

Once this ratio has been determined, the following equation can then be iterated upon to solve for the expected exit Mach number, M_{exit}, of this nozzle, assuming the flow is choked.

$$\frac{A_{exit}}{A_{throat}} = \frac{1}{M_{exit}}\left[\frac{2}{\gamma+1}\left(1+\frac{\gamma-1}{2}(M_{exit})^2\right)\right]^{(\gamma+1)/2(\gamma-1)}$$

<u>Iteration Result</u>:

$$M_{exit} = 2.685$$

Next, the speed of sound in the exhaust gas, a, must be determined. This can be done using the following equation.

$$a = \sqrt{\gamma\left(\frac{R_{univ}}{MW_{prod}}\right)T_{total}}$$

Now that M_{exit} and a are known, the velocity of the exhaust gas, or V_{exit}, can be found using the definition of the Mach number.

$$V_{exit} = a * M_{exit}$$

Next, the mass flow rate through the nozzle, $\dot{m}$, Total Pressure (equivalent to Chamber Pressure, P_{chmb}) and exit pressure, P_{exit}, must all be determined. These values were found using the following three equations in a system, assuming the flow to be isentropic.

Rocket Thrust Equation:

$$Thr = \dot{m}V_{exit} + (P_{exit} - P_{amb})A_{exit}$$

Isentropic Pressure Relation:

$$\frac{P_{chmb}}{P_{exit}} = \left(1 + \frac{\gamma - 1}{2}(M_{exit})^2\right)^{\gamma/\gamma-1}$$

Choked Flow Equation:

$$\dot{m} = \left(\frac{A_{exit}P_{chmb}}{\sqrt{T_{total}}}\right)\sqrt{\frac{\gamma}{\left(R_{univ}/MW_{prod}\right)}}\left(\frac{\gamma+1}{2}\right)^{-\left(\gamma+1/2(\gamma-1)\right)}$$

Solutions to the System of Equations:

$$\dot{m} = 0.0163\ kg/s$$
$$P_{exit} = 52.32\ kP$$
$$P_{chmb} = 1269.9\ kPa$$

Now that P_{chmb} is known, the mass of solid propellant needed to pressurize the chamber can be calculated. This calculation can be done by assuming idea gas and using the ideal gas relation.

$$n_{grain} = \frac{P_{chmb}Vol_{chmb}}{R_{univ}T_{total}}$$

Convert from Moles to Mass:

$$m_{grain} = n_{grain}MW_{prod} = 1.341\ g$$

Now, the average density of the manufactured solid propellant, or ρ_{grain}, must be determined. This was done using data gathered in the manufacturing process. The data needed to calculate ρ_{grain} is the average mass and average volume of any specific grain.

$$\rho_{grain} = \frac{Avg.Mass}{Avg.Volume} = 0.0013\ g/mm^3$$

Now that ρ_{grain} and m_{grain} are known, the volume of a single grain, or Vol_{grain}, can be calculated.

$$Vol_{grain} = \frac{m_{grain}}{\rho_{grain}} = 1017.2\ mm^3$$

Now that the volume of a single grain has been determined, assuming the grain is a cylinder with a fixed 10 mm diameter, that would result in a grain height of around 13 mm. The compiled results of all the grain data can be found below, in Table 3. As many assumptions were made through these calculations, future testing will either validate or help restructure the propellant sizing.

Table 3: Final Grain Sizing for the System

Mass	Volume	Diameter (Assuming Cylinder)	Height (Assuming Cylinder)
1.341 g	1017.2 mm^3	10 mm	12.95 mm

VI. Chamber Sizing and Design

The combustion chamber and pressure vessel are, in functionality, the same object. The combustion takes place in the pressure vessel. The design of this pressure vessel can be seen below, in Figure 6.

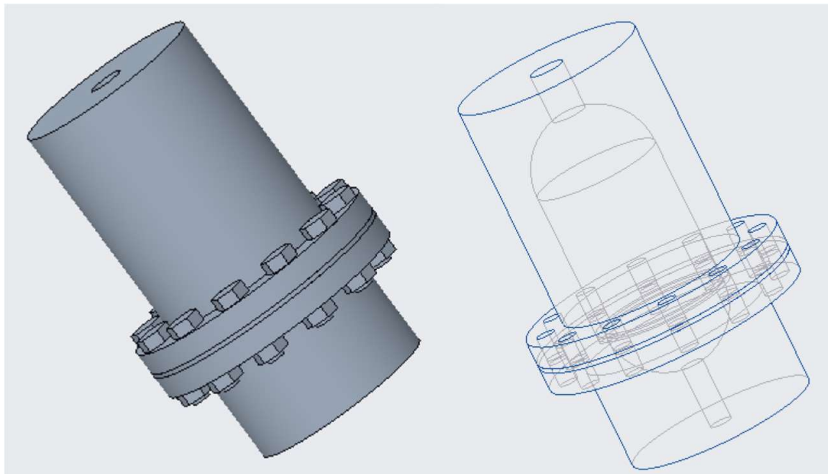

Figure 6: Combustion Chamber Solid Model (Left) and Wireframe (Right)

Through trade studies weighing cost, manufacturability, and strength, it was ultimately decided that the material properties of 1045 steel would be best suited for this application. Knowing the material and required tank volume, calculations can be done to determine an estimate of the required wall thickness of the chamber. In the previous section, it can be seen that $P_{chmb} = 1269.9\ kPa$, or around 12.5 ATM. Before the thickness can be determined, the yield stress of this material at the expected temperature must first be found. The following graph in Figure 7 was used to determine the yield strength of carbon steel at high temperatures. This graph was found in *Marks' Standard Handbook for Mechanical Engineers, 12th Edition* [5].

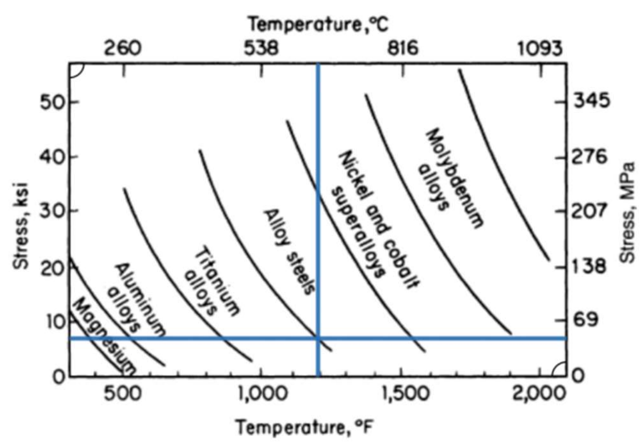

Figure 7: Graph of Yield Stress vs Temperature for a Generic Carbon Steel

According to the graph, the yield stress of carbon steel at 1250 F (667 C) is approximately 7 ksi (48.3 MPa). The found yield stress is then plugged into a simple hoop stress pressure vessel equation. As the graph has notable temperature limitations, a calculation using a yield stress of 24 MPa was also performed. It is also of note, the factor of safety (S) in these calculations was set to 5.

$$t = \frac{S\,P\,r}{\sigma} = \frac{(5)(1317.22kPa)(0.02m)}{48.3MPa} = 2.73mm$$

$$t = \frac{S\,P\,r}{\sigma} = \frac{(5)(1317.22kPa)(0.02m)}{24MPa} = 5.49mm$$

Based on the pressure in the chamber and the approximated yield stress, the wall thickness needs to be at least 5.49mm thick. The walls will be made thicker than this in the final design to provide extra safety and allow for simpler machining. Figure 6 above depict the 3D model of the pressure chamber with a thickness of 1 cm, and a cavity volume of 150 cm^3.

VII. Electrical Components and Controls

There are six main components that control SPARCC. These include one microcontroller to control the system, three devices that physically interact with the system, and two sensors to monitor it's conditions. The microcontroller chosen was the ESP32-Devkit-V1 (ESP32). This is compatible with Arduino, supports Wi-Fi and Bluetooth communications, low-power consumption, and has plenty of pins to run the components. The other three devices that run the system are the Linear Actuator, an L298N Motor Drive Controller, Solenoid Valve, Glow Plug, and relays for the Solenoid and Glow Plug to turn on and off the devices. The ESP32 commands the LA to move the propellant grain into the chamber, tells the GP to heat and ignite the pellet, and tells the SV to open/close when the system needs to produce thrust.

The two sensors onboard are the Pressure Transducer and a Thermocouple. The pressure transducer is responsible for measuring how much pressure is left in the pressure chamber and seeing if there is enough pressure to produce thrust. A K-type thermos couple is responsible for measuring temperature in the wall of the pressure chamber to ensure that the system is within nominal operating conditions. To read the temperature data from the thermocouple, a K-type amplifier is used to measure the voltage change from the thermocouple probe since the change is miniscule. The sensor data feeds into the ESP32 to ensure that it can run the devices safely.

Since there were no considerations for an onboard battery for SPARCC, a lab bench power supply is used to supply power to all electrical components. To deliver power to each device, a power distribution board was used to connect all of the components together. The system is set to run at 12V's, and step-up and step-down converters (also known as buck converters) are used to adapt components to the voltage set by the manufacturer.

VIII. Future Work

As mentioned previously, SPARCC is a proof-of-concept system. Moving forward, previously omitted budgets such as mass, power, and volume, would be reimplemented to further tailor the model as a feasible propulsion system for small spacecraft. SPARCC was also initially conceptualized to have multiple valves connected to a single chamber to provide thrust along all axes. Furthermore, the system operation time needs to be optimized to reduce the minimum time to produce thrust. The propellant storage can also be improved to hold more propellant and to be replaceable to allow for reloading of the propellant bank. Finally, the system currently requires that all of the combustion gases are expelled before it can reload another propellant grain. Ideally, the system should be able to add propellant to a pressurized chamber if more thrust is required.

While not strictly constrained to the future goals, the team did keep them in mind throughout the design process and did come up with some future alternatives that were not pursued due limited resources and time. To reduce the overall mass of the system, the system's linear actuator and steel combustion/pressure chamber could be redesigned since both make up most of the mass. On a more refined model, the linear actuator could be replaced and the combustion/pressure chamber could be redesigned with a lower factor of safety: a minimum of 2 was required but the team opted for a factor of 10 due to safety concerns. Moreover, both above changes would reduce the volume of the system, particularly replacing the linear actuator as it takes up approximately half of the systems length. Further volume savings could be made by replacing the solenoid valve. Finally, changing the onboard electronics for more power efficient components would offer greater power savings.

To reduce the system's operation delay, changes could be made with the propellant and the on-board hardware. Propellant grain geometry could be altered to encourage faster grain burning. Furthermore, iteration of the O/F ratio could further optimize the burn rate while still preserving the desirable traits of the current propellant design. Selecting a different actuator to reduce the travel time of the propellant from storage to the combustion chamber would also provide a time savings. Finally, the ignition method should be changed to be more sustainable to prolong the life of the propulsion system. With the current power supplied and hardware, the glow plug takes a considerable amount of time to preheat, so if a mission requires on-demand thrust with little warning an alternative would provide quicker ignition.

IX. Conclusion

A system which uses solid propellant to pressurize a tank then subsequently releases the pressure to generate thrust, seems to be a viable option as a propulsion system for small satellites. Rough design, Concept of Operations, and preliminary models suggest this. The proposed design is still in the early phases of development and is currently being manufactured and tested at the Florida Institute of Technology, as part of a senior Capstone project. This design has also neglected weight, power, and volume considerations to fit the project within the scope of a senior capstone project lifecycle. To fully test and validate this kind of system, these considerations will need to be accounted for, which has been deemed future work by the current team.

If pursued further, this type of system has the potential to be a promising alternative for the future of small satellite technology. This system was designed to be affordable and simple, which lowers the bar for propulsion systems, and could extend the applicability of small satellites.

Acknowledgments

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