Design and Development of A Hybrid Rocket Engine Test Stand to Study Exhaust Flow Gas Dynamics and Chemical Kinetics

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Hybrid rocket engines have several advantages over purely liquid or solid rocket propulsion systems. However, due to the fuel grain regression rates and the decreased fuel efficiency in larger rockets, hybrid rockets have a long way to go before being implemented for payload carrying applications. While certain aspects of hybrid rocket propulsion have been studied extensively, such as comparing low and high fuel regression rate materials, the effects of gas dynamics and resulting shockwaves in the nozzle exhaust have yet to be fully explored. We aim to study the specific impulse and overall efficiency of a gaseous oxygen (GOX) and paraffin wax propulsion system while focusing particular attention on the gas dynamics of the rocket exhaust plume. The overall objective of this work is to employ high-speed Schlieren imaging and chemiluminescence to examine the gas dynamics and chemical kinetics in the supersonic exhaust flow of the rocket. The present work details the design and development of the hybrid rocket motor test stand, as well as presents preliminary measurements from schlieren imaging and chemiluminescence acquired for flows relevant to exhaust flow from a hybrid rocket motor.

I. Nomenclature

A_e	=	nozzle exit area
A^*	=	nozzle throat area
a	=	regression rate empirical constant
k	=	specific heat ratio
M_e	=	exit mach number
m	=	mach number
$\dot{m}$	=	total mass flow rate through nozzle
P_e	=	pressure at nozzle exit
P_o	=	chamber pressure
R	=	gas constant
T_o	=	Chamber temperature

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II. Introduction

A. Purpose

Hybrid rocket combustion, rocket exhaust plumes, supersonic shockwaves, and the chemical kinetics of combustion are extensively studied topics on their own, but the combination of the topics is not. The aim of this paper is to build a hybrid rocket test stand and employ high speed Schlieren and chemiluminescence imaging to analyze the behavior of the resulting exhaust plume. Louisiana State University's Energy and Propulsion Laboratory (LSU's EPL) has focused its attention on supersonic jet flow [8] and combustion of paraffin wax [4] in the past, but the current research aims to combine the imaging and analysis techniques for the analysis of hybrid rocket propulsion. Finally, the effect of multi-phase properties of paraffin [11] on post-chamber combustion inside the exhaust plume will be examined.

B. Background

A. History and Overview

Hybrid rocket propulsion systems utilize both fluid and solid elements in the combustion process, providing an appealing alternative to conventional liquid and solid rocket systems. Experimentation with hybrid combustion systems began in the early 1930s, with the earliest significant effort towards a flightworthy hybrid rocket being initiated in the mid 1940s by the Pacific Rocket Society, utilizing fuels such as douglas-fir wood, a waz infused with carbon black, and a rubber-based fuel. The group achieved their first successful flight in 1951, the rocket reaching a height of about 30,000 feet with a version of their rubber-based fuel [1]. Since that time, hybrid rocket technology has advanced significantly, with the adoption of synthetic fuels such as paraffin waxes and the synthetic rubbers based on polybutadiene providing a more reliable platform to build the technological concept on. Contemporary research into hybrid rocket powertrains now seeks to optimize the technology to the point where it can be more widely adopted in place of the current solid and liquid rocket systems [1].

B. Advantages

Compared with their solid and liquid rocket counterparts, hybrid rocket systems offer numerous benefits which the two former options lack. Although liquid rockets can offer a high performance, high efficiency package, they are incredibly complex and expensive with all the required plumbing and cooling for the fuel and oxidizer. Solid rocket platforms avoid these drawbacks by premixing the oxidizer in a solid, dense state, resulting in lower costs than their liquid equivalents. However, solid rockets suffer drawbacks of their own, being highly susceptible to explosions due to the premixed complexion of the propellant and lacking the capability to control or terminate thrust once ignited [1].

Hybrid rocket platforms, with solid fuel and separately stored gaseous or liquid oxidizer, combine benefits from both solid and liquid rocket systems to create an attractive alternative. Compared to solid rockets, hybrid systems exhibit the ability to throttle, shut down, and restart fuel combustion as well as being less prone to explosions and having a higher specific impulse. With respect to liquid rockets, hybrids boast greater simplicity and reliability as well as lower costs, with only half the plumbing and higher propellant densities [1,2].

C. Drawbacks

While hybrid propulsion systems offer several advantages over their solid and liquid counterparts, their most significant disadvantage is found in the low regression rate of the solid fuel grain during combustion in comparison to a conventional solid propulsion system [3]. Hybrid combustion systems undergo "boundary layer combustion," where the flame is separated from the solid fuel surface and sustained by fuel vaporized from it, which is itself produced through the transfer of heat through convection and radiation from the flame to the fuel surface. This heat transfer is impeded further by the generation of vapor at the surface, referred to as the "blocking effect [4]."

III. Theory

A. Hybrid Rocket Propulsion

A typical hybrid rocket setup as shown in Fig. 1 is composed of a combustion chamber containing the fuel grain and a separate oxidizer tank connected via an oxidizer injector. An igniter initiates combustion as the oxidizer is injected into the combustion chamber and then sent through a nozzle to produce thrust. The hybrid rocket test stand utilizes a gaseous oxygen (GOx) oxidizer and a 1260A paraffin wax fuel grain. Due to paraffin's multi-phase properties, the fuel regression rate is 3-5 times higher than hydroxyl terminated polybutadiene (HTPB) and other commonly used fuel grains [5].

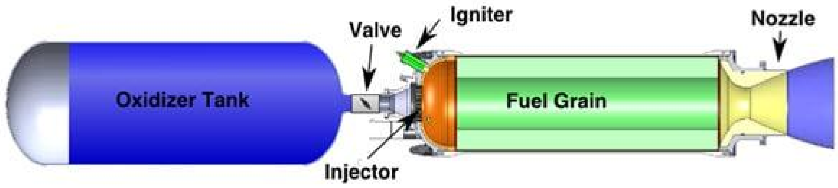


Fig. 1: Schematic of hybrid rocket engine [6]

While larger scale hybrid rocket motors utilize a liquid phase oxidizer, a gas phase oxidizer is utilized in the current research due to safety and handling concerns. Gas dynamics relate the combustion chamber pressure to the mass flow rate (mdot) through the nozzle and the throat area, as shown in Eq. (1).

$$P_o = \frac{\dot{m}}{A^*}\left[\sqrt{\frac{k}{RT_o}}\left(\frac{2}{k+1}\right)^{\frac{k+1}{2(k-1)}}\right]^{-1} \tag{1}$$

An increase in mass flow rate and chamber temperature results in an increase in chamber pressure. The chamber pressure will also increase with the decrease in throat area and specific heat. Maximizing chamber pressure is beneficial when maximum thrust is the objective, but the focus of the current research is to analyze the rocket exhaust plume on a test stand. The mach number of the exit pressure can be found through the relation of the ratio of the chamber pressure to the nozzle exit pressure in Eq. (2)

$$\frac{P_o}{P_e} = \left(1 + \frac{k-1}{2}M_e^2\right)^{\frac{k}{k-1}} \tag{2}$$

Along with the pressure, the area ratio of the nozzle exit area to the throat area (Eq. (3))must be taken into consideration for the design of a nozzle.

$$\frac{A_e}{A^*} = \frac{1}{M_e}\left(\frac{2}{k+1}\left(1 + \frac{k-1}{2}M_e^2\right)\right)^{\frac{k+1}{2(k-1)}} \tag{3}$$

When designing a hybrid rocket combustor, total chamber pressure and temperature are two of the biggest safety factors. NASA's Chemical Equilibrium with Applications (CEA) software, freely available through Glenn research center, was employed to find the optimal oxidizer to fuel ratio (O/F). Given a nozzle with a converging subsonic ratio of 3.52 and a supersonic ratio (A_e/A *) of 5.43, a pressure ratio (P_o/P_e), and a chamber pressure of 1 atm, the optimal O/F ratio is 2.0, with an exit mach number of 2.719.

B. Rocket exhaust plumes

When dealing with supersonic flow, the presence of shockwaves must be taken into account. A converging-diverging nozzle (CD nozzle) is used to accelerate subsonic rocket exhaust to create thrust. The exhaust travels through the converging section at subsonic speeds (m < 1) until it reaches the throat at the speed of sound (m = 1), where the flow accelerates through the diverging section at supersonic speeds (m >1). If the pressure of the gas exiting the nozzle is equal to the ambient pressure, shockwaves are not present, making an ideal "design condition". When the pressure of the gas exiting the nozzle is less than ambient pressure, the jet is overexpanded, meaning that the flow has been expanded past ambient pressure. On the other hand, when the exit pressure is greater than ambient pressure, the jet is underexpanded. In both underexpanded and overexpanded jets, complex shockwaves and expansion waves are formed and create "mach diamonds".

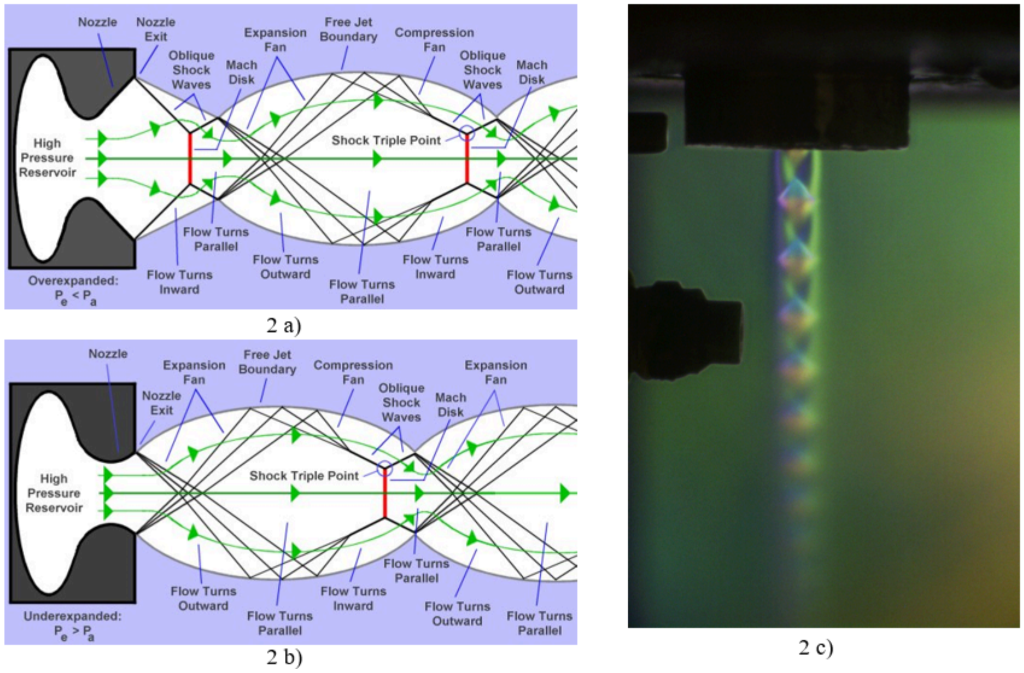


Fig. 2: a) Overexpanded exhaust plume, b) Underexpanded exhaust plume structure [7], c) FCS Imaging of mach diamonds [8]

C. Imaging Methods

While mach diamonds can be seen by the naked eye, more advanced imaging is required to study the shock structures in the jet. The current research utilizes two main techniques the first of which is Traditional Schlieren Imaging. The shock patterns in the jet change the fluid density, which in turn bends light rays. Schlieren imaging takes advantage of those bent light waves to capture images of fluid flow (Fig. 2c)(Fig. 3). A light source emits light through a slit towards a series of concave mirrors set up on either side of the fluid flow. The light passes through the flow and is focused to a point on a razor edge before reaching the recording device.

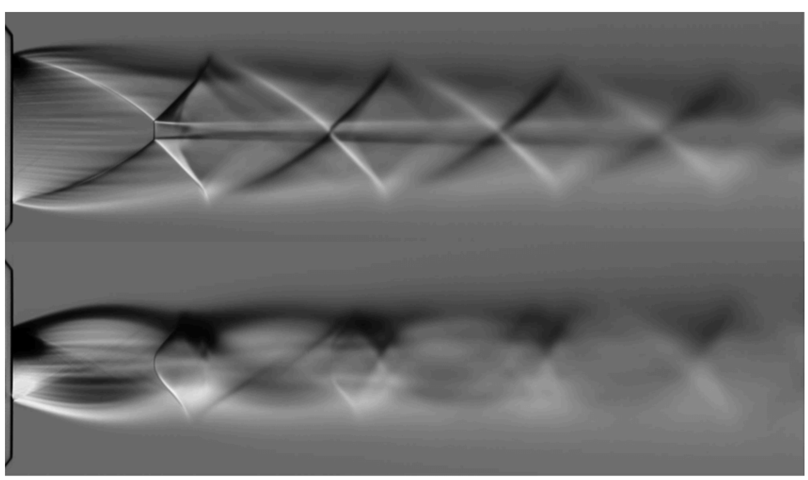


Fig. 3: Traditional Schlieren image of underexpanded elliptical jet [6]

Similar to Traditional Schlieren, Focusing Color Schlieren (FCS) assigns colors to directional changes by using a multi-color light source grid [8]. FCS was employed in previous work at LSU's EPL [8] to analyze liquid jet breakup in a supersonic cold flow jet. The current research discussed in this paper analyzes supersonic hot flow through the use of a hybrid rocket test stand. A Photron SA3 high-speed camera will be used in conjunction with a Traditional Schlieren Imaging setup. A frame rate of up to 2000 fps will be used to better capture the supersonic flow.

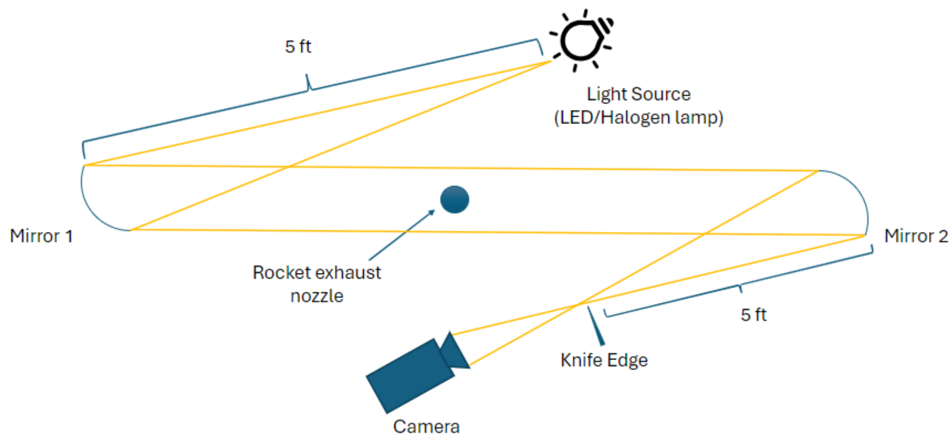


Fig. 4: Traditional Schlieren Imaging setup

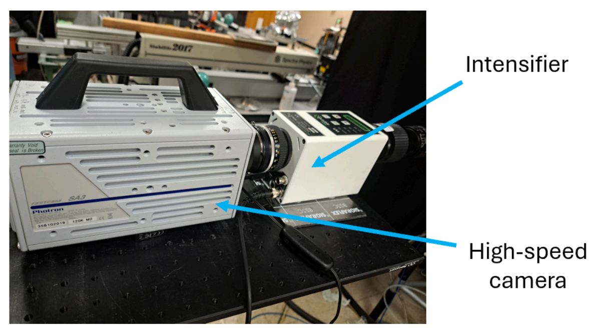


Fig. 5: CI setup consisting of a high-speed camera and intensifier

Schlieren Imaging works exceptionally well for the study of shockwaves in both cold flow and hot flow, but cold flow is missing one key component: combustion. Combustion is a chemical reaction that emits light known as chemiluminescence. When unburnt fuel is present in the exhaust plume, it has the potential to ignite with the atmosphere, creating chemiluminescence. This is known as the "Afterburning effect" [12]. The current hypothesis is that the multi-phase properties of paraffin [11] will have an impact on the Afterburning effect. In order to properly capture the characteristics of a hybrid rocket's exhaust plume, chemiluminescence imaging (CI) is employed (as shown in Fig. 6). A Photron SA3 high-speed camera from the Schlieren Imaging will be used in conjunction with a UVI-1850 intensifier for all CI (Fig. 5). The same imaging setup was used in a similar manner in LSU's EPL by Connor Becnel [4] to study boundary layer combustion of paraffin wax. While Becnel used CI and other imaging to analyze the combustion process, CI will be used in the presented research to analyze the *post*-combustion process inside the rocket exhaust plume.

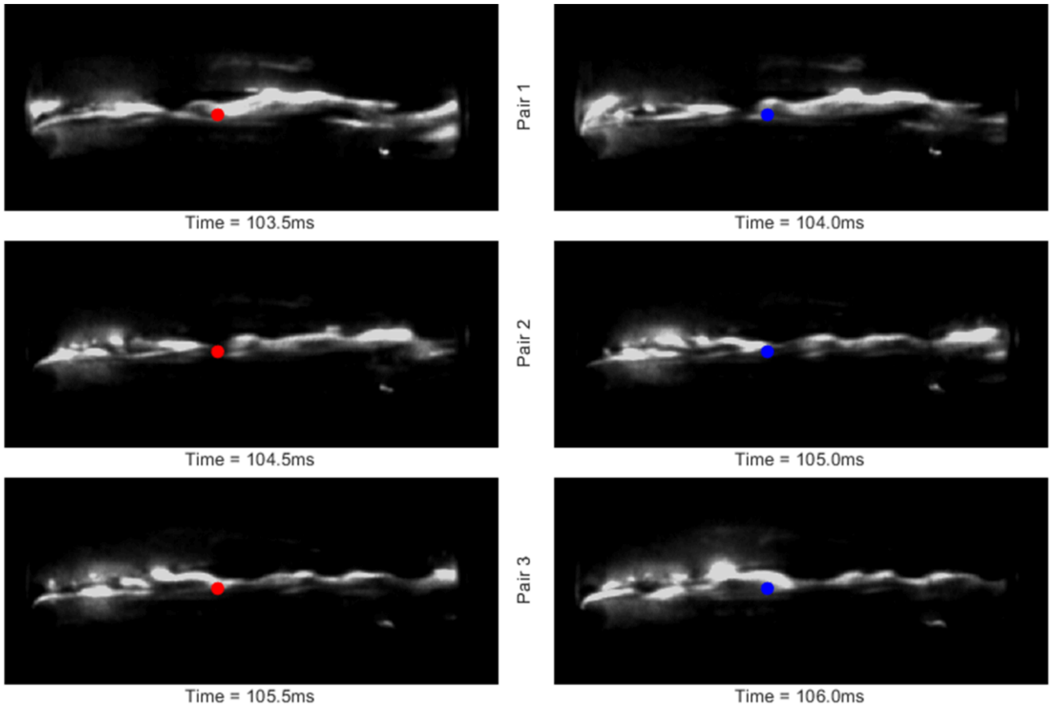


Fig. 6: Chemiluminescence Imaging of paraffin wax boundary layer combustion [4]

IV. Methodology/setup

A. Test Stand

The test stand, once completely assembled, will consist of several interconnected subsystems, shown in Fig. 7. The oxidizer and inert gas delivery system will supply compressed gas to the rocket combustion chamber, feeding through either GOx to fuel the combustion or inert nitrogen gas to purge the system of oxygen, removing the possibility of any residual fuel reigniting in the chamber after a given test has concluded. The Stainless Steel 304/304L combustion chamber, shown in Fig. 8a, has an outer diameter of 2.25 in and a thickness of 0.1875 in. Its upstream end, shown in Fig. 8b, will feature a central port for oxidizer or inert gas injection as well as smaller orbiting ports for thermocouple insertion, pressure gauge access, and the pyrotechnic igniter. A 6061-T6 aluminum aft mixing chamber was put in place downstream of the fuel grain to ensure proper mixing. Both the injector interface and aft chamber are sealed with two O-rings, with one surface seal placed between the aft chamber and the mounting plate. Shown in Fig. 8c, its downstream end houses a converging-diverging nozzle, where the exhaust plume will be observed by our CI setup. The combustion chamber is secured on both ends with steel mounting plates, which themselves are to be mounted to fitments which can be secured to a T-slot railing. Data from the stand's diagnostic equipment like the previously mentioned thermocouple, pressure gauge, and CI setup will be sent to our Data Collection System for analysis after the test has concluded. The electrical system supplies power to all electrical components within the test stand system.

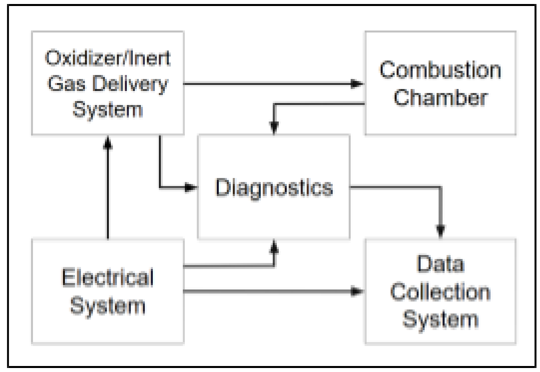
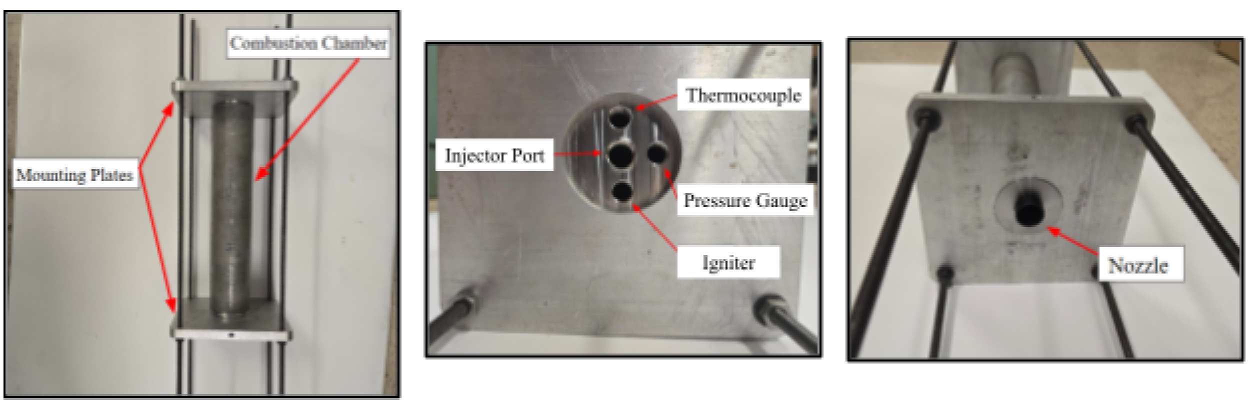


Fig. 7: Test Stand Subsystem Diagram



8 a) 8 b) 8 c)

Fig. 8: a) Hybrid Rocket Combustion Chamber, b) Combustion Chamber Upstream End, c) Combustion Chamber Downstream End

B. Compressed Gas Piping System

Shown in Fig. 9 is the piping arrangement for the test stand's compressed gas delivery system. It begins with two "feeder lines"–one for GOx and another for nitrogen gas–which then converge into a main line running to the combustion chamber. The feeder lines are both fitted with isolation valves, which for safety purposes are to remain closed at all times except when actively testing. Also present on these lines are solenoid valves, which will be

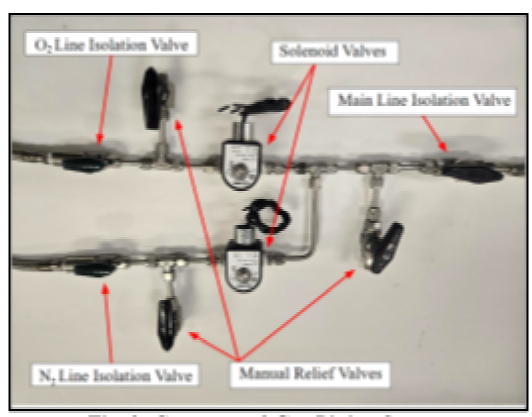


Fig. 9: Compressed Gas Piping System

electronically controlled to allow either the oxygen or nitrogen gas to flow into the main line. The main line also has an isolation valve which, similarly to the two feeder lines, is to remain closed at all times except during active testing. At multiple places in the system, there are manual relief valves for use in case of unwanted pressure buildup. Downstream of the main isolation valve, there is also a check valve which restricts airflow to only travel downstream towards the combustion chamber, preventing any contaminants from potentially reaching the main body of the piping system.

C. Molding of fuel grain

The fuel grain is made from 1260A paraffin wax purchased from Lone Star Candle Supply. While the fuel grain is composed of pure paraffin, additives such as aluminum or carbon black will be explored in the future to test the effects on fuel grain regression rates. The fuel grain is to be molded inside the stainless steel combustion chamber using 3D printed Polylactic Acid (PLA) plugs and a pvc rod to shape the inner diameter. An oil coating will be applied to the plugs and central rod to prevent paraffin bonding to the surface upon cooling. The paraffin will be melted on a hot plate in an aluminum container and poured at a temperature of 180 °F to prevent shrinkage and imperfections. The nominal dimensions are shown in Table 1.

Outer Diameter	Inner Diameter	Length	Weight
1.875 in	1.00 in	8.00 in	240 g

Table 1: Measurements of paraffin fuel grain

V. Discussion

A. Current standing

The combustion chamber is fully assembled, and fuel grain molding is being conducted. The piping diagram for the oxidizer/inert gas delivery system is assembled and only needs to be connected to the combustion chamber and pressurized GOx and nitrogen tanks. The electrical systems for the igniter and GOx/N_2 solenoid valves will be connected to an inverter that is attached to a 24V car battery. The ports in Fig. 2b are fit for a thermocouple to measure temperature, a pressure gauge to measure chamber pressure, an oxidizer injector, and a pyrotechnic igniter.

B. Future goals

The pyrotechnic igniter may be replaced in future testing with a pyrograin ignition method shown in Ref. [9]. In pyrograin ignition, an electric igniter is attached to a small propellant sample near the injector. The working design for pyrograin ignition at LSU's EPL involves a pyrograin sample of aluminum powder and paraffin wax attached to a spark plug via two nichrome wires that deliver the high current required for ignition. This method is commonly used, as pyrograin ignition is affordable and easily reusable.

Our current setup only quantitatively measures chamber temperature and pressure. Future plans include attaching a load cell to the forward mounting plate for thrust and impulse measurements. From the thrust, we can effectively maximize the efficiency of the hybrid rocket.

Once the test stand setup is complete, the following variables can be isolated and empirically tested.

- Chamber/oxidizer pressure
- Fuel grain additives such as Aluminum
- Shape of fuel grain/inner diameter
- O/F ratio

VI. Conclusion

The theory of exhaust plume characteristics were applied to the application of hybrid rocket propulsion. This research plans to use the methodology and instrumentation of previous research and apply it to lesser studied areas of rocket propulsion. The first step in this research is to assemble a fully functioning hybrid rocket test stand for future research and analysis. It is also hypothesized that the high regression rate and multi-phase properties of paraffin will influence the Afterburning effect in the rocket exhaust plume.

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