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Effect of Nozzle Throat Diameters on Kerosene-NOS Liquid Rocket Performance

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This paper presents experimental results for the performance effects of different converging-diverging graphite nozzle throat diameters on an in-house developed kerosenenitrous oxide liquid rocket test stand. The project aims to enhance the performance and efficiency of small-scale liquid rocket engines by experimentally investigating the effects of nozzle throat diameter on thrust and specific impulse. By confirming the correlation between nozzle geometry and the experimental data, it provides valuable insight for improving propulsion systems and components used in experimental rocketry such as sounding rockets. This study will evaluate two different nozzle throat diameters under varying propellant pressures and mass flow rates. The liquid rocket test stand consists of an external aluminum casing with a combustion chamber measuring 20" in length with an outer diameter of 76 mm and an internal diameter of 1.66". The nozzle throat diameter tested will be 58/64" and 60/64", each with a fixed exit diameter of 1.82". Experimental results were collected over a range of total mass flow rates using data acquisition systems and analyzed using graphs and trend lines. The results indicate that as the throat diameter increases, the thrust output and specific impulse increase, although the results are inconclusive due to leaks and a back flame during testing, possibly skewing the results. The ablative wear was analyzed based on the nozzle throat size and mass flow rate. The knowledge gained from this study can be used to prevent future accidents for small-scale liquid rocket engine test stands and verify if the trends seen will be applicable to different nozzle materials and find the optimum nozzle throat diameter.

I. Nomenclature

A_e	= Exit area of the nozzle [ft^2]
I_{sp}	= Specific Impulse [s]
$\dot{m}$	= Mass flow rate of propellant [lbm/s]
P_0	= Ambient pressure [psi]
P_e	= Pressure at the exit of the nozzle [psi]
P_t	= Total Pressure [psi]
T	= Thrust of the liquid rocket engine [lbf]
T_t	= Total temperature [R]
V_e	= Velocity at the exit of the nozzle [ft/s]
$\dot{w}$	= Weight flow of the propellant [lbf/s]
MFP	= Mass flow parameter [lbm-$\sqrt{R}$/lbf-s]
Q	= Volumetric Flow Rate [ft^3/s]
C_d	= Discharge Coefficient []
ΔP	= Pressure Drop between Fuel Regular and Measured Pressure [psi]

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ρ	=	Density of Jet-A [lbm/ft³]
$\dot{m}_{a\_l}$	=	Mass Ablation Rate [lbm/s]
L	=	Linear Ablation Rate [in/s]

II. Introduction

The development of small-scale liquid rocket engine test stands has gained significant attention in recent years due to their application in academic research, technology, and material testing, as well as potential application to largescale liquid rocket engines used by companies such as SpaceX and Blue Origin. The performance of these engines is key for suborbital and orbital space travel. Performance metrics include the thrust and specific impulse of the liquid rocket engine that can be affected by many aspects of the engine. Nozzle geometry is an essential factor in performance as it directly affects the exhaust gas velocity, and as such, poor nozzle geometry could be a determinant of various performance metrics. Thus, small-scale liquid rocket engines offer a cost-effective method for experimental propulsion testing, enabling various testing parameters such as different propellants, combustion methods, or system optimization. This could have a significant impact on the field as new technology emerges; testing on a large scale could be expensive, and a reliable small-scale test stand could be applicable on a larger scale. A small-scale liquid rocket engine test stand provides the opportunity for amateur rocketeers and universities to learn/showcase relationships between certain components/variables on the performance for educational purposes.

This study presents the evaluation of various graphite nozzle throat diameters on the performance of a kerosenenitrous oxide (NOS) liquid rocket test stand. These graphite nozzles were tested at various propellant flow rates to see the impact on performance. By analyzing key performance metrics such as specific impulse and thrust, possible correlations can be discovered between nozzle geometry, propellant flow rates, and oxidizer-to-fuel ratios to thrust and specific impulse of the system. This study showcases the ability to test components on the small-scale level, which can facilitate collaboration with industry partners. The liquid rocket test stand consists of an external aluminum combustion chamber length of 20 inches, a diameter of 76 millimeters, and an internal diameter of 1.66 inches. The nozzle throat diameters tested were 58/64-inch and 60/64-inch with a fuel pressure varying from 300 – 340 psi. The propellant used for this test stand includes Jet-A as the fuel and NOS as the oxidizer. While these propellants are not widely used in the industry, they greatly simplify the complexity of the test stand due to their storage ability, affordability, and handling. To handle the high temperatures inside the combustion chamber, an ablative liner is manufactured in-house to absorb the heat. These choices allow for a cost-effective way of testing liquid rocket components while maintaining the validity of the results. Additionally, the versatility of the test stand allows for modifications leading to future iterations that reflect more closely to realistic liquid rocket engines.

The objective of this study is to evaluate the relationship between nozzle throat diameter and liquid rocket performance. This entails varying oxidizer-to-fuel ratios and propellant mass flow rates for each nozzle to investigate the effects on specific impulse and thrust. Furthermore, the efficacy of this test stand must be proven in order for the results to be accurate. In theory, both the thrust and specific impulse are going to decrease as the nozzle throat diameter increases due to exhaust gas velocity decreasing. As the propellant mass flow rate increases, the thrust should increase while the specific impulse will decrease. If these trends hold true, the small-scale liquid rocket test stand will be validated, and the results from this study can be applied to a boarder scale. The ablation rate of custom-made ablative liner is evaluated to assist with future tests as it can used to determine the maximum duration of a test. The ablative liner formula could also be adapted/varied to identify which provides the best ablation rate while absorbing the heat generated. Future iterations of the test stand can be implemented to evaluate various components and study their effects on performance, a feature that will be appealing to industry partners.

III. Background

A. Theory

Modern liquid rocket engines primarily use a bipropellant system characterized by the implementation of two separate liquids as propellants: one liquid serves as an oxidizer, and the other liquid serves as the fuel. Compared to other propulsion systems, the bipropellant system results in higher efficiency and performance due to the oxidizer and fuel mixture, resulting in a greater energy density of the flow. Monopropellant systems incorporate only a single propellant as the fuel, and this dramatically reduces the complexity of a liquid rocket system. However, there are downsides due to a single fuel source, such as the inability to stop ignition as the fuel is actively reacting. In this bipropellant system, the oxidizer and fuel are kept in separate tanks until the point of combustion, reducing the risk of accidental ignition or explosion and contributing to an overall safer system. In this system, the propellants are mixed and atomized after exiting an annular injector into the combustion chamber to be ignited for operation. For this study, Jet-A is used as the fuel, and NOS is used as the oxidizer. Jet-A is typically used as airplane fuel and in large-scale liquid rockets; other fuels, such as RP-1, are used due to their reliability and higher energy density for ignition. However, it is a great option due to the cost, ease of storage, and safety on the small-scale level. NOS is used due to its unique property of self-pressurization at about 700psi, meaning the oxidizer pressure is roughly kept the same throughout all tests. This reduces the complexity of this system significantly, as only the fuel pressure will be different.

Thrust will be one of the main performance parameters that will be evaluated in this study, as well as how it is affected by the oxidizer-fuel (OF) ratio and nozzle throat size. Thrust is the reaction force resulting from imparting momentum to a mass, which in this system is the ejection of propellant at high velocities. Thrust is used as one of the primary parameters to evaluate the performance of a rocket as it exhibits the rocket's acceleration capabilities, which are vital in completing missions. Acceleration is essential for overcoming Earth's gravity; thus, having a large amount of thrust enhances the capabilities and performance of a rocket. The equation for thrust accounting for the pressure difference between the nozzle exit and ambient pressure is shown in Eq. (1) below, where T is the thrust, $\dot{m}$ is the mass flow rate of the propellant, V_e is the exit velocity of the flow, P_e and P_0 are the exit and ambient pressure, respectively, and A_e is the exit area of the nozzle.

$$T = \dot{m} V_e + (P_e - P_0) A_e \tag{1}$$

Specific impulse (Isp) is another an important parameter that will be measured in this study. The specific impulse is the measure of thrust per unit propellant weight flow rate. In other words, it is a measure of the efficiency of a rocket. Equation (2) for Isp is shown below where $\dot{w}$ is the weight flow rate of the propellant.

$$I_{sp} = \frac{T}{\dot{m} * g_0} = \frac{T}{\dot{w}} \tag{2}$$

These are the key performance parameters used to identify the performance of a rocket and analyze the capability and efficiency of this system. To calculate these parameters, the mass flow rates of the oxidizer and fuel are required. The oxidizer mass flow rate is measured using the mass flow parameter (MFP). The equation for MFP can be seen in Eq. (3) below, where A is the oxidizer area of the injector, T_t and P_t are the temperature and pressure at the oxidizer injector, $\dot{m}$ is the oxidizer mass flow rate. MFP can also be defined as the second part of Eq. (3) where γ is the ratio of specific heat at constant pressure to specific heat at constant volume (equal to 1.4 for air), R is the gas constant, and M is the Mach number.

$$MFP = \frac{\dot{m}\sqrt{T_t}}{A\,P_t} = \sqrt{\frac{\gamma g_c}{R}}\, M\left(1 + \frac{\gamma-1}{2}M^2\right)^{-\frac{\gamma+1}{2(\gamma-1)}} \tag{3}$$

To calculate the mass flow rate of the fuel, the general volumetric flow rate equation and the volumetric flow rate equation for an orifice are combined. These equations are shown in Eq. (4) where C_d is the discharge coefficient, A is the fuel area of the injector, ΔP is the difference in pressure between the pressure regulator and the pressure measured during the test, and ρ is the density of the fuel.

$$Q = \frac{\dot{m}}{\rho} = C_d A \sqrt{\frac{2 * \Delta P}{\rho}} \rightarrow \dot{m} = C_d A \sqrt{2 * \Delta p * \rho} \tag{4}$$

During ignition, this system will produce immense heat that should not come in contact with the aluminum chamber. A custom-made ablative liner is used as a heat-absorbent material. Over the duration of the test, this ablative liner will lose mass, and the thickness near the nozzle will change. This can be measured by using Eq. (5) and Eq. (6) shown below where $\dot{m}_{a\_l}$ is the mass ablation rate and L is the linear ablation rate.

$$\dot{m}_{a\_l} = \frac{(m_f - m_i)}{\Delta time} \tag{5}$$

$$L = \frac{(t_i - t_f)}{\Delta time} \tag{6}$$

B. Previous Work

Huzel [6-7] book delves deep into liquid rockets, including injector and combustion chamber design, along with fuel and oxidizer choices and their impacts on engine performance. It is vital to use this book to help validate liquid rocket engine design choice to ensure maximum performance out of a small-scale test stand. He discusses various fuels and oxidizer combinations and shows how Jet-A and NOS may not be the best for performance but do significantly reduce complexity and cost.

Richter [8] studies how various compositions of ablative materials eroded based on their densities. He investigated both ablatives made of micro balloons and ceramic microspheres, the prior of the ablative materials used a similar composition to the one used in this study. His data shows that the micro balloons provide better ablation resistance and that the higher the density of the micro balloons, the more ablation resistance it gained.

Natali et al. [9] study the effects of micro balloon densities and how they affect ablative regression by adjusting their mixture to vary the percentage of micro balloons used in the mixture. After formulating various compositions for their ablatives ranging from low to high concentrations of micro balloons, they put them under extreme heat via torch and found that the balance of micro balloons is key since increasing the density leads to higher use of an increased level of viscosity reducer, which increases the burn rate of the ablative. Inherently increasing the concentration of micro balloons decreases the ablation rate but provides diminishing returns until the viscosity reducer causes it to ablate more.

Rao's [10] paper explores how nozzle geometry and types affect thrust. He delves into how picking the proper nozzle can help optimize the thrust output for a given mission. Nozzles can be used in other ways besides forward thrust, including thrust reversal or vectoring through a variety of methods depending on the nozzle type and system. He finds that the propellant chemical properties greatly impact what nozzle to choose due to incomplete combustion and varying specific heat properties. The chamber efficiency and conditions also have an influence on nozzle optimization.

This liquid rocket engine test stand has been developed and verified to measure performance difference in previous work done by Christison [11] and Biliske and Patel [12]. Design choices such as choosing Jet-A and NOS are discussed further in these papers as well as other design choices. Christison does an excellent job at discussing this test stand specifically such as discussing other injector geometries and deciding why an annular injector would be best for this test stand. Biliske and Patel were able to measure the performance difference in cold flow tests, meaning the test stand can function and capture data properly. Christison highlights and analyses hot-fire data ensuring this test stand is fully capable of collecting accurate data.

Liquid rocket engines have been extensively researched but developing a relationship between nozzle throat diameter and engine performance on a small scale where boundary layer effects have a greater impact compared to a larger scale. The rate of ablation on such a small scale has also had a lack of research and development. Most of the previous ablative research has been conducted with the ablative being isolated and not used in a liquid rocket engine. Understanding how this scale affects nozzle performance and ablative burn rate using NOS and Jet-A is vital for the future development of small-scale liquid rocket engines.

IV. Methodology

This study was conducted at Oklahoma State University Richmond Hill Research Complex at the north loading docks. This location is far off the main campus, and appropriate safety measures were in place to conduct hot-fires and in case any accidents occurred. This area has ample space for spectators and a far standoff distance where no one would be injured. The configuration of the test stand is shown below in Figure (1). Christison [11] outlines a proper full assembly procedure of this test stand that is used in this study.

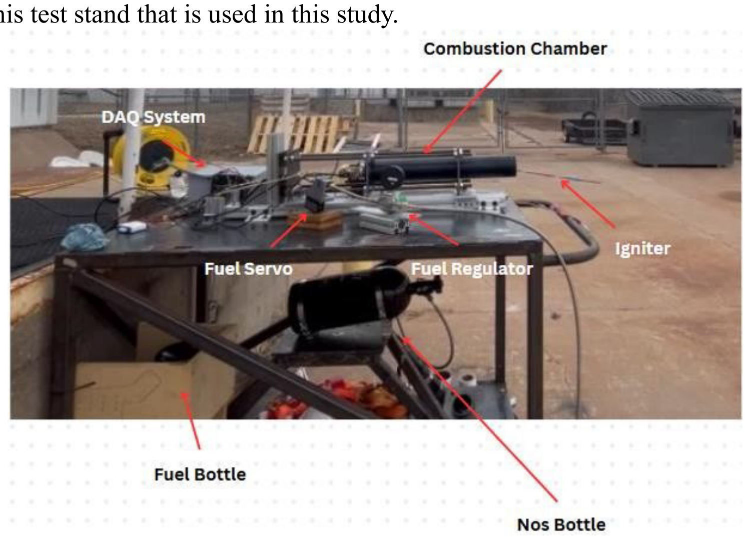

Figure (1): Fully Assembled Test Stand

A. Data Acquisition System

The DAQ system includes a plastic box that houses an Arduino Uno board that captures all pressure data through an SD card module, sends signals to the fuel and oxidizer servos to turn their respective propellant valves, and starts the ignition. The oxidizer and fuel pressure data are read through two different pressure transducers (in milliamps) that are wired to the Arduino board. The data is captured at a sample rate of 35 Hz, and the SD card can be removed post-testing to extract the data. To accurately measure thrust during a test, a thrust ring on the combustion chamber presses on a metal plate on a slider that then presses on the load cell. The load cell uses a software called LV-1000 by Loadstar where it captures the force pushing on the load cell (in this case thrust), and the data can be exported to Excel for further analysis.

B. Ablative Liner

To absorb the immense heat released inside the combustion chamber, a custom-made ablative liner is used. The method developed by HalfCatRocketry [2] is used to make the ablative liners, where the main goal is to develop a low-cost material that can burn away throughout the duration of the test. The ablative liner is composed of four materials: epoxy resin, hardener, sodium bicarbonate, and phenolic micro balloons. The resin-to-hardener ratio is 2:1 respectively, combining for 63.3% of the total mass, sodium bicarbonate accounts for 31.7% of the total mass, and phenolic micro balloons account for 5% of the total mass. The phenolic micro balloons create a char layer on the surface of the ablative, insulating the inside chamber walls. The sodium bicarbonate acts as a film-cooling agent as it is composed of H20 and CO2, which carries the heat out of the chamber during the burn. The epoxy provides structural stability to the mixture and binds the phenolic micro balloons and sodium bicarbonate together. A cross-section view of the ablative liner can be seen in Figure (2). A more detailed way to make this specific ablative liner can be found in Christison [11] and Biliske and Patel [12].

Figure (2): Cross-Sectional view of Ablative Liner

C. Oxidizer Setup

The oxidizer used is NOS, which requires a special pressurized tank to hold. The bottle pressure depends on temperature, so to carry more NOS in this tank, a greater temperature difference between the oxidizer tank and this NOS bottle would be beneficial. To fill this bottle, an external fill station is used. After the bottle is filled, it is then placed in a holder underneath the test stand. The oxidizer line is attached to the oxidizer ball valve which will be controlled through a servo. Prior to conducting both the cold flow and hot-fire test, ensure the oxidizer valve is closed, and the oxidizer bottle is open. Inside the injector, there is a sonic orifice flow control, creating a choke point, limiting the oxidizer Mach number at the injector face to 1.

D. Fuel Setup

The fuel setup is similar to the oxidizer setup in which the fuel is held in a pressurized tank with a pressure relief burst valve to prevent over pressurization. To fill the fuel bottle with Jet-A, the relief valve is taken off, and a tube is inserted inside the bottle, and Jet-A from a separate fuel reservoir will be fed. This bottle also needs to be pressurized with Nitrogen (chosen because Nitrogen is inert), as Jet-A by itself will not flow through the line. Once the bottle is pressurized, it is placed upside down onto a holder underneath the test stand. This causes the nitrogen to “push” down on the Jet-A, allowing a proper flow rate throughout all tests. To set the pressure during a cold flow or hot fire, the Swagelok pressure regulator is used to regulate the pressure between 0 – 500 psi, which controls the flow rate of fuel and, in turn, the oxidizer-to-fuel ratio. Once the fuel bottle is filled, connect a fuel line between the bottle and the pressure regulator and another line between the pressure regulator and the threaded fuel port. Similar to the oxidizer line, a servo driven ball valve will be used to open the fuel line during cold flow and hot-fire tests. The injector setup with all the propellant lines and pressure transducers attached is shown in Figure (3) below.

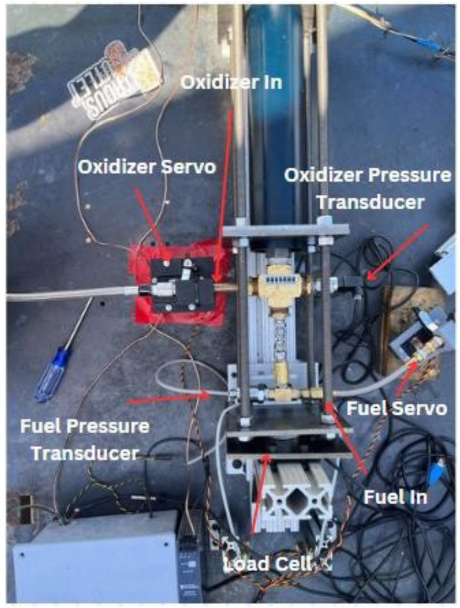

Figure (3): Injector Side of Combustion Chamber

E. Igniter Setup

The igniter consists of an electric match with black powder charges in contact to ensure it meets the activation energy required for flow ignition. This electric match has two ends where a voltage will be sent through one end by the NI PS-15 power supply and the other end by the Arduino. This creates a short circuit which ignites the electric match and black powder charges. This ignition happens 30 milliseconds before the opening of the oxidizer and fuel valves to ensure proper ignition. Testing was done to ensure the igniter functions properly by simply constructing this igniter, laying it on the ground and sending the signal to ignite through the Arduino and verify if an ignition does happen. After verifying, the black powder charge is taped to the electric match around a rod and then inserted into the combustion chamber through the nozzle side. A key safety concern is accidental ignition, so in order to prevent this, the NI PS-15 power supply used to power the ignitor is left unplugged until the test is ready to be conducted.

V. Results and Discussion

Two types of tests must be performed in order to fully characterize this liquid rocket test stand. The first is a cold flow test, meaning ejecting out only fuel or oxidizer without ignition. The second test is a hot fire where both the oxidizer and fuel are ejected inside the combustion chamber, where they mix, atomize, and are ignited. The cold flow helps with determining the mass flow rates of the oxidizer and fuel using Equations (3) and (4). The hot-fire tests evaluate components of the test stand and analyze the performance parameters. As the test stand is modular, different nozzles can be swapped inside the combustion chamber between tests easily. Previous work has been conducted with this test stand to verify the validity of the results. Results from previous cold flow tests from Christison [11] and Biliske and Patel [12] are used in this analysis to determine the mass flow rates of the propellant.

A. Propellant Cold Flow Results

As previously mentioned, the oxidizer flow is metered using a sonic orifice control, which creates a choke point in the flow. This allows the oxidizer mass flow rate to be determined using Equation (3). To verify the pressure was roughly the same throughout the duration of the test, several oxidizer cold flow tests were performed, and the manifold pressure was graphed, which can be seen in Figure (4) respectively.

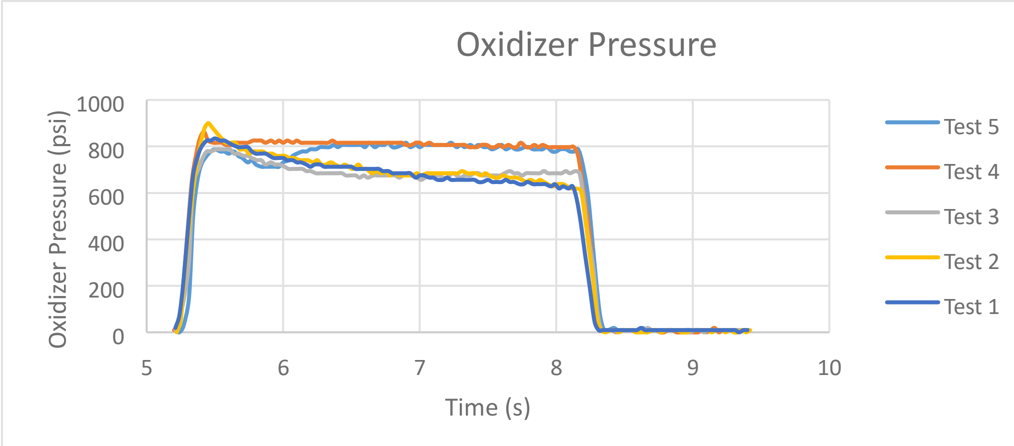

Figure (4): Oxidizer Manifold Pressure vs. Time

Figure (4) highlights how the pressure does vary between the tests. However, this is due to the tests occurring on different days with different ambient pressures. It is important to note that during tests 1 and 2, there was a NOS leak in the fittings, which greatly decreased pressure throughout the test. For tests 3, 4, and 5, after six seconds, the pressure was roughly constant. This justifies assuming an approximately constant pressure and temperature at the NOS injector face during testing, which greatly simplifies Equation (3) as every variable would be constant. Thus, it can be concluded that the oxidizer mass flow rate will be roughly constant throughout the burn duration. Accounting for pipe losses of approximately 0.2 *lbm*, the NOS bottle (at 800 psi) was weighed before and after a three-second cold flow test. There was a change in mass of 2.46 *lbm* and the oxidizer mass flow rate came out to be 0.82 *lbm/s*. This oxidizer flow rate will be assumed for both hot-fire tests. This assumption will likely result in error as the oxidizer pressure could be lower than 800 psi, resulting in a lower oxidizer mass flow rate. However, the oxidizer mass flow should not vary more than ± 0.1 lbm/s. Future cold flow testing is recommended to verify a more precise oxidizer mass flow rate. The fuel cold flow tests were done to measure a fuel mass flow rate by simply collecting and measuring the ejecting fuel mass. Equation (4) can be rearranged to find the discharge coefficient, which can be assumed constant for all tests afterward, leaving the fuel mass flow rate equation to be a function of the pressure difference between the fuel regulator and fuel manifold pressure. Previous cold flow data from Christison [11] and Biliske and Patel [12] were used to determine the discharge coefficient and this is shown in Table (1) below.

Table (1): Fuel Cold Flow Data

	Test 1	**Test 2**	**Average**
Fuel Mass Flow Rate (lbm/s)	0.129	0.09	0.1095
Pressure Difference (psi)	445	285	365
Discharge Coefficient	0.076	0.067	0.0715

The average discharge coefficient is used to determine the fuel mass flow rate for all tests. This is likely to result in errors in mass flow rate as there is not enough testing done. Further testing is recommended to get a more accurate discharge coefficient.

B. Hot-Fire Results

A total of three hot-fire tests were done: two with the 58/64” nozzle at 300 and 340 psi fuel pressures and one with the 60/64” nozzle at 300 psi fuel pressure; a hot-fire is shown in Figure (5). This study aims to examine the effects of nozzle throat diameter on performance, thus, only the two tests at 300 psi will be analyzed. Figure (6) shows the thrust of the two respective tests. Using the measured thrust and calculated mass flow rates, the Isp can be calculated, and this data can be seen in Figure (7).

Figure (5): Hot-Fire at 300 psi Fuel Pressure for 60/64” Nozzle

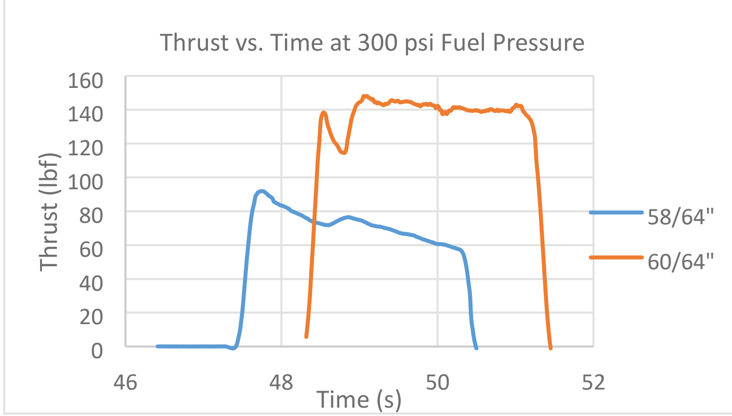

Figure (6): Measured Thrust vs. Time at 300 psi Fuel Pressure for 58/64" and 60/64" nozzles

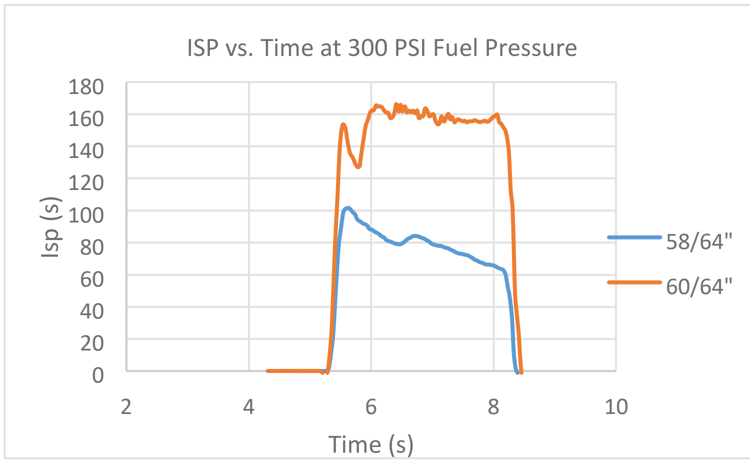

Figure (7): Calculated Isp vs. Time at 300 psi Fuel Pressure for 58/64" and 60/64" Nozzles

These two graphs have a similar curve, which is mainly because Isp is directly related to thrust. A higher thrust is recorded for the nozzle with a higher throat diameter (60/64") compared to the smaller nozzle (58/64"). Thus, this directly leads to a higher Isp for the larger nozzle size as well. The thrust and specific impulse for the 60/64" nozzle is about 140 *lbf* and 160 s respectively where it's about 70 *lbf* and 80 s respectively for 58/64". This highlights that rocket performance is greater for a larger nozzle. Key features of these graphs include a rise and shutdown time of about 0.15 seconds and 0.13 seconds respectively and show a thrust and Isp decay. This is a result of the fuel pressure regulator failing to keep the fuel mass flow rate at a constant rate. Over the duration of the test, the O/F ratio varies as the fuel mass flow rate varies. The thrust and Isp were graphed against O/F ratio as shown in Figure (8) and (9) below. These figures indicate that the larger size nozzle has a larger O/F ratio which in turn results in greater performance parameters. It should be noted how the thrust and Isp remained around 140 *lbf* and 160 s respectively while the O/F ratio varied from 10 – 16 for the 60/64" nozzle, while the thrust and Isp have a liner trend downward for the 58/64" nozzle. This indicates that performance can be optimized to a certain O/F ratio, where increasing it further does not improve performance.

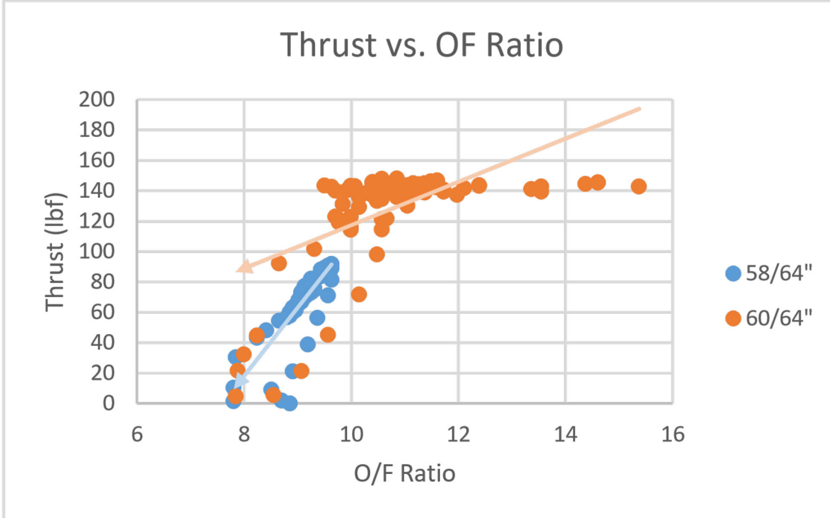

Figure (8): Thrust vs. OF Ratio for 58/64” and 60/64” Nozzles

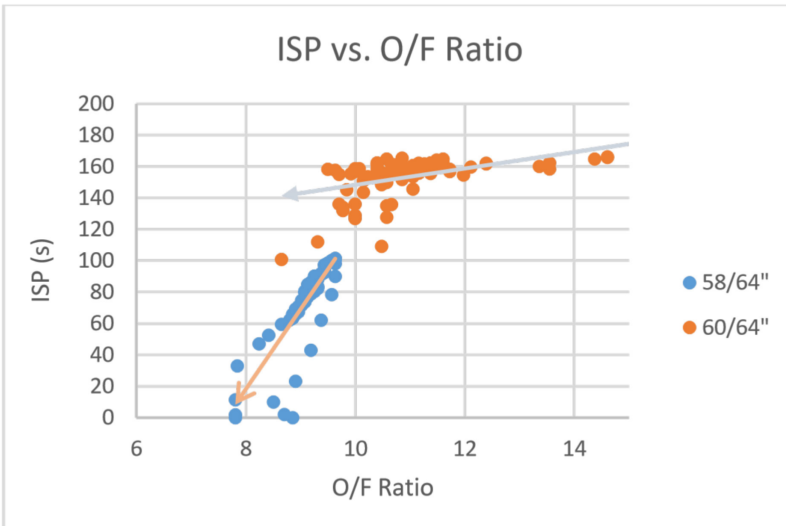

Figure (9): Isp vs. OF Ratio for 58/64” and 60/64” Nozzles

However, it must be noted that both tests had problems of their own. The test with 58/64” nozzle had a propellant leak in the fittings near the injector and the 60/64” nozzle had a back-flame develop inside the chamber. The propellant leak directly leads to a decrease in performance as some propellant went unburnt and reduced the overall mass flow rate. After conducting the test with a 60/64” nozzle, analyzing the phenolic liner showed clear burn marks on the outside face that is in contact with the combustion chamber walls, indicating there was a flame between the phenolic liner and the chamber walls. This greatly reduces the thrust as there are two forces acting on the stand that counteract each other. Thus, it can be concluded that the results are inconclusive. However, the O/F ratio graphs highlight a general trend in performance with respect to the O/F ratio that should be seen in all small-scale liquid rocket engine test stands. Further testing is recommended to identify a proper trend between nozzle size and performance parameters for NOS and Jet-A as the propellant.

In this study, loose fittings and an unsealed combustion chamber led to inconclusive results. Future testing should take into consideration all possible propellant leaks prior to conducting a hot-fire test as this can lead to inaccurate results or potential explosions as the propellant could ignite. Potential leak spots include loose fittings, unsealed combustion chamber due to improper phenolic and ablative liner sizing, or faulty equipment. Back flame is a significant risk as this could escape past the injector, damaging all sensitive electronics and equipment. To prevent this, proper O-rings around the injector and nozzle should be in place along with a properly sized combustion chamber to seal the chamber so the propellant and flame cannot escape.

C. Ablative Wear

The three ablative liners used in this study successfully absorbed heat as there was no visual damage to the aluminum combustion chamber walls. The initial and final dimensions of the liners can be seen in Table (2) below.

Table (2): Ablative Wear Data

Ablative Used	Initial Thickness (in)	Initial Mass (lbm)	Final Thickness (in)	Final Mass (lbm)	$\dot{m}_{a\_l}$ (lbm/s)	L (in/s)
58/64" @ 340 psi	0.474	1.83	0.436	1.744	0.013	0.013
58/64" @ 300 psi	0.474	1.825	0.446	1.748	0.012	0.00933
60/64" @ 300 psi	0.474	1.78	0.424	1.675	0.016	0.017

The common trend is that as the thrust is higher, the ablative wears away more. The 58/64" nozzle tested at a fuel pressure of 340 psi had about 10-15 lbf more thrust compared to when it was tested at 300 psi, thus this trend is seen throughout all three ablative liners. If a hot-fire is conducted with a larger area nozzle or at a higher fuel pressure, the ablation rate should be greater as the temperature inside the combustion chamber is higher. It should be noted the thickness was measured towards the nozzle side as it experienced the greatest heat. This change in thickness over time is only indicative of the one side of the liner as the side where the injector is, experiences little to no charring. This information is useful when conducting future tests as the ablation rate can be calculated for this custom-made ablative liner. This ablation rate can be used to determine the maximum duration a test can be with a certain configuration. However, due to the errors in testing discussed previously, further testing is recommended in order to determine a more accurate ablation rate.

VI. Summary and Recommendations

The key findings in this study include possible ways to prevent a significant error in testing for a small-scale liquid rocket engine test stand. This will lead to more accurate and safe testing to occur, allowing for component testing, and identification of trends between performance parameters and certain changes in the test stand. A fully functional test stand could impact amateur rocket science in a positive way. A modular test stand is a gateway for universities to enter the rocket science field and gain interest from students.

Due to the errors in testing, the results are inconclusive; however, a similar trend as the one shown between the O/F ratio and performance parameters should be identified in other small-scale liquid rocket engine test stands. Evaluation of the ablation rate for the in-house manufactured ablative liners indicated that the higher fuel mass flow rate, as well as a larger nozzle throat diameter, led to a greater ablation rate. This is due to greater performance for those test configurations; thus, the combustion chamber was at a higher temperature, meaning the ablative liner absorbed more heat. This information can be used to determine a different formula for an ablative liner mixture, which could have a greater ablation rate.

Further testing is recommended in all aspects of this study as a result of the errors that occurred during the testing. It is recommended to account for all errors listed in this study when conducting further testing. Further fuel and oxidizer cold flow tests are recommended in order to get a more accurate discharge coefficient as well as a more accurate constant oxidizer flow rate for various pressures. To identify a performance trend between nozzle throat diameter, further testing is recommended as well. These tests should aim to understand the relationship between thrust and specific impulse and the nozzle throat diameter.

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