Optimizing Northrop Grumman's GEM63XL Solid Rocket Booster with novel propellant formulations to gain an increase in maximum thrust

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Rocket motors are essential parts of a rocket to help it carry specific payloads into space and further space exploration. Thus, it is imperative that more powerful motors be designed to improve their efficiency and aid future missions to space. While the GEM63XL solid rocket motor is powerful, it can still be further improved while keeping the same motor dimensions. These improvements can be attained by chemically analyzing new propellant formulations, modifying grain geometry structure, and adjusting propellant distribution throughout the motor. The improvements that these changes bring can be measured by using a theoretical approach involving code-based simulations and rocket equations with assumed conditions to measure their boosts in overall thrust. This will deliver four novel propellant mixes, a new propellant cross-section, a new propellant distribution, and new nozzle geometry. From these improvements, we can expect to see up to a 98% improvement in the motor's maximum thrust.

1. Nomenclature

F_T = Thrust

m = Mass flow rate

v_{esc} = Exit velocity

A_e = Exit area

P_e = Exit pressure

P_a = Atmospheric/ambient pressure

P_c = Chamber pressure

I_{sp} = Specific Impulse

g = Gravitational constant

r = Radius of the grain's cross-section

l = Length of the booster

ε = Number of points on the star

β = Decay rate

t = Current time of burn

a = Propellant constant

n = Propellant constant

ρ = Density of the propellant

A_b = Burn area

A_t = Throat area

$C*$ = Characteristic velocity
r_b = Burn rate
h = Current altitude
F_D= Drag Force
v = Velocity
ρ_a = Density of air
C_D = Coefficient of drag
A = Cross-sectional area

2. Introduction

Rockets are a means of transport used to carry certain objects to space. Rockets often consist of a payload and a rocket motor. All rocket motors work by creating a chemical reaction that requires fuel and oxygen and ejecting the reaction byproducts at immense speeds. As a result, by Newton's Third Law, they generate thrust, a force that opposes gravity. Motors often include an injector to bring the reactants into the combustion chamber, where, after reacting, the reaction byproducts are ejected out of the motor through the nozzle.

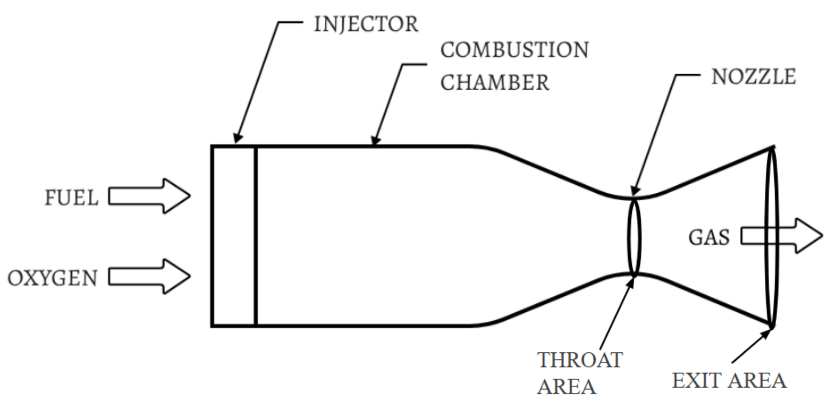

Figure 1: Diagram of an average rocket motor with notable parts labeled [1]

Solid rocket motors are one type of rocket motor, with the other types being liquid, electric, nuclear, etc. Solid rocket motors are categorized by their propellant, as their propellant is mostly composed of powders and packed into a solid form, whereas liquid rocket motors have liquid oxygen and fuel for combustion. In addition, solid rocket motors are also commonly used as boosters since they can provide more powerful thrust compared to other rocket motors in exchange for not being able to control the flow of the motor. While current space agencies such as SpaceX and NASA use liquid engines, many rockets used to launch satellites and other objects to orbit use solid rocket motors, such as Japan's H-IIA rocket and China's Kuaizhou. In addition, rockets such as NASA's Space Launch System use solid rocket motors as boosters to gain enough initial thrust. Thus, research in obtaining more powerful and efficient propellants is invaluable to the aerospace industry.

With many different applications for solid rocket motors, their performance can be altered in two separate ways. The first way is by changing what the rocket uses as fuel and how the fuel is arranged in the rocket motor. This is done by changing the propellant chemical formulation, modifying the propellant cross-section, or changing the way the propellant is distributed along the rocket motor [2].

Many solid rocket motors rely on a propellant primarily made of an oxidizer, fuel, and burn rate modifier to generate thrust. The propellant formulations in this study follow the de Saint-Venant formulation; thus, the burn rate, or how quickly the propellant is consumed, will follow an exponential function of the chamber pressure, which is the pressure in the combustion chamber. There are other types of formulations that follow a different burn rate curve based on pressure that yield mesa and plateau curves; however, the propellants in this study do not fit those formulations. The burn rate is a component in determining the thrust of the motor. Changing the composition of the propellant affects the overall thrust and behavior of the motor.

While changing the composition of the propellant can affect thrust, another component of determining the thrust of the motor is the burn area, or the surface area of propellant in the motor. When looking at a propellant cross-section of a rocket motor, the cross-section can be a variety of designs, as seen in Figure 2. Each of these cross-sections has its own unique burn area equation, thus altering the thrust. [3][4][5][6]

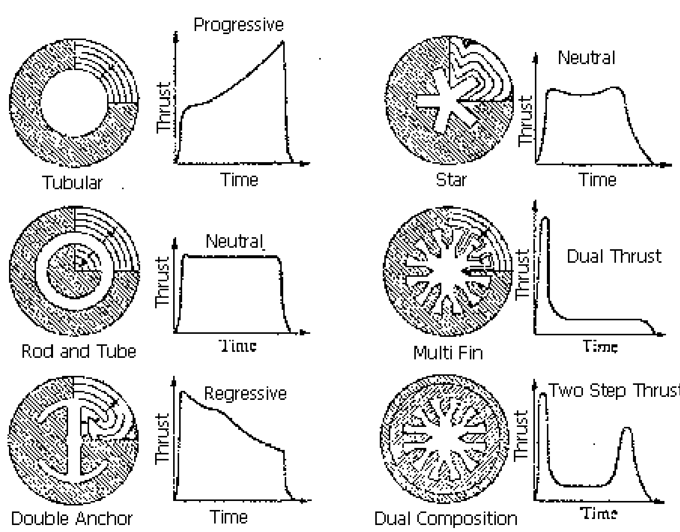

Figure 2: Different grain cross-section geometries and their thrust vs time performance [7]

Another way to modify the burn area is to change the way the propellant is distributed along the rocket motor. For example, by having more propellant concentrated near the back of the nozzle, the thrust generated peaks at a certain time after ignition, which is especially useful when designing missions to outer space or other planets.

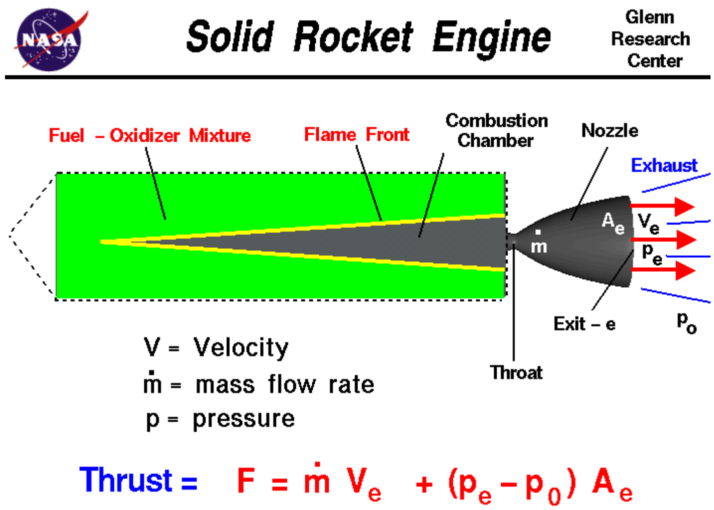

Figure 3: Solid Rocket Engine Diagram with propellant linearly distributed along the motor [8]

The second way to alter a solid rocket motor's performance is by changing its nozzle geometry. Specifically, changing the throat area and exit area in Figure 1 can affect how a rocket motor performs. A nozzle serves to accelerate the mass from subsonic speeds at the throat area to supersonic speeds, thus generating high amounts of thrust.

Changing a nozzle's geometry alters the thrust because thrust is dependent on mass flow rate, or the rate at which reaction byproducts are ejected out of the rocket. There are two types of mass flow rate: the mass flow rate into the nozzle and the mass flow rate out of the nozzle. When the two mass flow rates are equal, maximum efficiency is reached. Thus, having a nozzle geometry that allows the propellant to reach maximum efficiency is essential to avoid limiting the rocket's thrust output.

Another factor to consider when using a nozzle is the isentropic flow of the mass as it flows out of the nozzle. Isentropic flow is fluid flow that has small and gradual changes in its variables [9]. Using the isentropic flow equations, certain ratios essential for thrust can be calculated.

The fluid flow through the nozzle can generate substantial amounts of heat, which can cause the inner walls of the nozzle to erode, especially near the throat area. This is known as nozzle regression, and it can play a minor role in affecting the overall performance of the rocket.

While the improvements may seem significant, this paper used many theoretical equations to solve for thrust. As a result, many assumptions were necessary because it was highly difficult, if not impossible, to accurately model the propellants under real-life conditions. These findings are primarily preliminary theoretical findings due to the assumptions, which will be elaborated in the Results and Discussion of Results section, instead of definitive performance predictions. Thus, further testing while factoring in or dealing with these assumptions is needed in order to accurately gauge the performance of the propellants and their geometry design.

Ultimately, this study will deliver four different propellants and their performance, a propellant cross-section geometry area, a propellant distribution equation, a nozzle that generally works with the propellants, and the effects of nozzle regression on the performance of the rocket.

3. Methods

To get an adequate approximation of propellant performance without performing ballistic tests, this study used a Python code [10] and various rocket equations to calculate a thrust vs time graph of the motor. In addition, the study will analyze the rocket's performance by delivering a thrust vs time under an ideal performance, where maximum efficiency of mass flow is assumed to be reached, and a thrust vs time and height vs time graph in an applied performance, where the motor will act as a rocket booster with a payload and nozzle.

To calculate a thrust vs. time graph for an ideal performance, some measurements and constants are needed, namely: rocket motor dimensions, the propellant distribution equation, propellant mathematical properties, and certain physical constants. Thrust can be calculated by

$$F_T = m * v_{esc} + A_e(P_e - P_a) \quad \text{(Eq. 1)}$$

Since the ideal case assumes the propellant is at maximum efficiency, it also assumes that the $A_e(P_e - P_a)$ term is zero and doesn't exist. The exit velocity is calculated with

$$v_{esc} = I_{sp} * g \quad \text{(Eq. 2)}$$

Specific impulse is a measure of how efficiently the motor uses the propellant to generate thrust; thus, a higher number means a more efficient propellant [8]. For the ideal case, the mass flow rate is calculated with

$$m_{in} = A_b \rho r_b \quad \text{(Eq. 3)}$$

The density of the propellant can be calculated based on its chemical composition. The burn area of the propellant is dependent on the cross-sectional geometry. In this study, all propellants will utilize a star-shaped cross-section. Therefore, the burn area can be calculated with

$$A_b = 2 * \pi * r * l * (1 + \varepsilon * e^{-\beta * t}) \quad \text{(Eq. 4)}$$

The burn rate, which is how fast the propellant's interior surface recedes, can be calculated with

$$r_b = a * P_c^{\ n} \quad \text{(Eq. 5)}$$

a and n denote mathematical constants and properties of the propellant, which are given for each propellant in the Results section. The chamber pressure can be calculated with

$$P_c = \left(\frac{a * \rho * A_b * C^*}{A_t}\right)^{\frac{1}{1-n}} \quad \text{(Eq. 6)}$$

The characteristic velocity is assumed to be 1500 m/s, as many other solid rocket motor propellants are around that value.

Since the burn rate affects the burn area, chamber pressure always changes, which affects the burn rate itself. Thus, Euler's method of approximation must be used to calculate burn rate and chamber pressure. The code uses time steps of 0.1 s to calculate the initial chamber pressure and burn rate given the initial burn area, uses those values for the thrust calculation, then calculates the chamber pressure and burn rate values for the next time step. This process repeats until all the propellant is used up and the burn area becomes 0.

To calculate a thrust vs time and height vs time for the motor's applied performance, some additional values need to be found in addition to the ones for the motor's ideal performance. This includes the nozzle geometry, payload and propellant mass, atmospheric pressure from altitude, and the drag coefficient.

The mass flow rate (Eq. 3) for the thrust (Eq. 1) is the mass flow rate in the nozzle. However, for the applied case, the mass flow rate must be calculated with

$$m_{out} = \frac{P_c A_t}{C^*} \quad \text{(Eq. 7)}$$

If Eq. 7 and Eq. 3 are set equal to each other, or when the mass flow rate in and out of the nozzle are equal, the necessary throat area for maximum efficiency can be calculated. The necessary throat area calculated is reported in the Results section. In addition, the thrust calculated for the applied performance uses all terms of Equation 1, compared to the ideal performance, where the latter term was considered to be 0.

Generally, the payload mass is always assumed to be around 1/10 of the propellant mass. The propellant mass was calculated by creating a 3D model of the rocket motor, taking its volume, and multiplying it by the density of the propellant.

To calculate the atmospheric pressure term for the thrust (Eq. 1), the formula [11] used was

$$P_a = 100 \ * \ \left(\frac{44331.514 - h}{11880.516}\right)^{\frac{1}{0.1902632}} \quad \text{(Eq. 8)}$$

An isentropic flow calculator from the pygasflow Python library was used to calculate the exit pressure of the rocket [12]. The code assumed an area ratio, or the ratio of exit area to throat area, of 10 to calculate the exit pressure at the end of the nozzle, since many rocket nozzles have an area ratio of 10.

To calculate the height the rocket attained, the same Euler's method of approximation using time steps of 0.1 s was used. By using Newton's Second Law and summing up thrust, gravity, and drag, the code was able to calculate acceleration and determine velocity and height from kinematics. Drag was factored in with the drag equation

$$F_D = \frac{1}{2}\rho_a v^2 C_D A \quad \text{(Eq. 9)}$$

The coefficient of drag was assumed to be 0.2, and the cross-sectional area could be calculated by taking the cross-sectional area of the booster.

The Python matplotlib and numpy libraries were used to calculate the thrust vs time and height vs time and plot the graphs for ideal and applied performance.

This paper also analyzed how a nozzle regressed during flight by taking nozzle erosion rate data from NASA [13] and changing the throat area at each time step to simulate how the nozzle would change. The nozzle material in this paper is assumed to be phenolic graphite.

The study did not do an experimental test of the propellant performance; thus, the performance of the propellants may be overestimated due to the theoretical nature of the calculations and assumptions made.

4. Results

This study analyzed four different propellant mixtures with their ideal performance and applied performance. The propellants will use the same motor dimensions, or the width and height, as Northrop Grumman's GEM63XL, which are a length of 21 m and a radius of 0.809 m [14]. The propellants will also be applied on a mission to Low Earth Orbit (LEO) carrying a payload of 5,500 kg. The results will also include a percentage comparison of the specific propellant's ideal performance compared to the GEM63XL's performance, as given by Northrop Grumman's specification sheet, as well as data from the applied performance.

Some assumptions of the study include a characteristic velocity (C*) of 1500 m/s, a nozzle area ratio (ratio of exit area to throat area) of 10, a gravitational constant of 9.807 m/s^2, air density a constant 1.225 kg/m^3, and a rocket coefficient of drag of 0.2. When calculating atmospheric pressure from altitude, the formula assumes a base pressure of 101,325 Pa, 288.15 K at an altitude of 0, and 0% relative humidity. To calculate the a and n values for each propellant, experimental equations [15] were solved to get a rough approximation. For nozzle erosion, the paper assumes the nozzle is made of phenolic graphite and has an erosion rate of 0.0062 in/s [13].

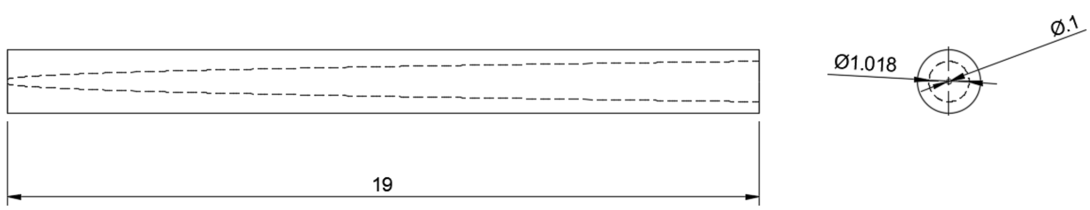

Figure 4: An engineering drawing of the horizontal cross-section of the booster, showing the propellant distribution as well as its initial and final radii values of 0.05 m and 0.509 m, respectively.

The propellant performances were all calculated using a star-shaped grain cross-section with two points. The propellant distribution followed a square root curve of the function $r = 0.1053\sqrt{l} + 0.05$ to achieve the desired thrust vs. time peaks that matched closely to a mission going to LEO. The initial and final values of the propellant radii can be seen in Figure 4. These radii were chosen because they provided the minimum chamber pressure necessary for combustion and mass ejection while keeping the largest amount of fuel. The motor in Figure 4 has a length of 19 to factor in the nozzle, which usually has a length of 2 m.

The propellant weight was calculated using the CAD model (Figure 4) and multiplying it by the density of the propellant for the applied performance. Since most of the propellant masses were around 55,000 kg, the payload was taken to be 5,500 kg. In addition, the throat area for the nozzle in the applied performance section, as obtained from setting the mass flow equations (Eq. 3 and Eq. 7) equal to each other, is 1 m^2.

Northrop Grumman's GEM63XL Performance [14]:

Name	Quantity
Propellant Formulation	QDL-4, HTPB, 19% Aluminum
Propellant Weight (Mg)	47.853
Burn Time (s)	87.3
Max Thrust (MN)	2.061
Specific impulse of propellant (s)	280.3
Density (g/cm^3)	N/A
a value	N/A
n value	N/A
Total Impulse (MNs)	131.534
Average Thrust during Burn Time (MN)	1.503

Figure 5: Table of notable data values for Northrop Grumman's GEM63XL's performance

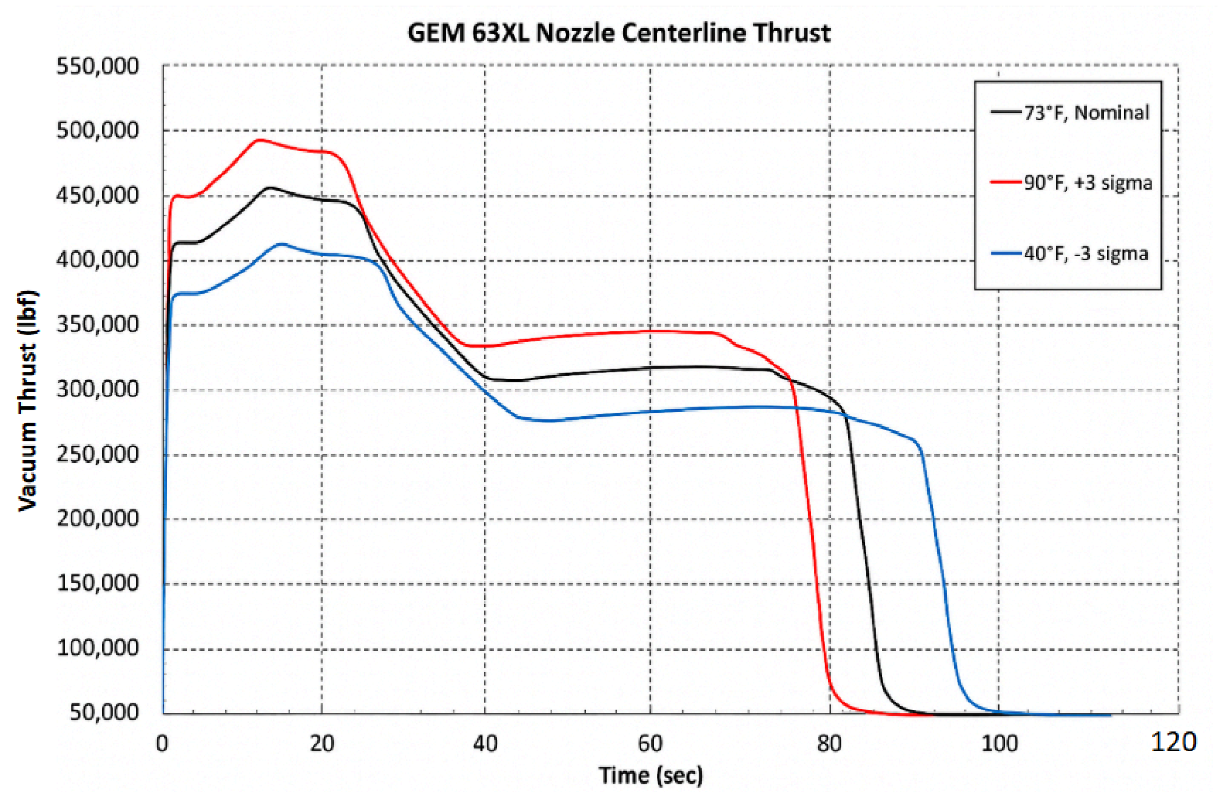

Figure 6: Northrop Grumman's GEM63XL Thrust vs Time graph with different ambient temperatures at ignition [14]

Propellant Data and Comparisons:

Name	#1	% DIFF	#2	% DIFF	#3	% DIFF	#4	% DIFF
Propellant Formulation	AP (70%) Al (10%) HTPB (17%) PbSt (3%)	N/A	AP (65%) Al (15%) HTPB (15%) PbSt (5%)	N/A	AP (70%) Al (10%) HTPB (17%) FeO (3%)	N/A	AP (68%) Al (15%) HTPB (12%) FeO (5%)	N/A
Propellant Weight (Mg)	54.523	+13.94%	54.831	+14.58%	53.599	+12.01%	55.447	+15.87%
Burn Time (s)	133.7	+53.15%	123.5	+41.47%	138.3	+58.42%	112.5	+28.87%
Max Thrust (MN)	3.309	+60.58%	3.529	+71.24%	3.149	+52.80%	4.086	+98.31%
Specific impulse of propellant (s)	250	-10.81%	245	-12.59%	250	-10.81%	255	-9.03%
Density (g/cm^3)	1.77	N/A	1.78	N/A	1.74	N/A	1.80	N/A
a value	3.6	N/A	4.0	N/A	3.2	N/A	3.9	N/A
n value	0.32	N/A	0.30	N/A	0.38	N/A	0.38	N/A
Total Impulse (MNs)	348.444	+164.91%	343.340	+161.03%	342.488	+160.38%	361.560	+174.88%
Average Thrust during Burn Time (MN)	2.606	+73.41%	2.781	+85.03%	2.476	+64.74%	3.214	+113.84%
Applied Performance Max Height	231.395	N/A	217.804	N/A	233.624	N/A	216.925	N/A

(km)								
Applied Performance Max Height with Nozzle Regression (km)	225.594	N/A	212.824	N/A	227.488	N/A	212.606	N/A
Effect of Nozzle Regression on Max Height	-2.51%	N/A	-2.29%	N/A	-2.63%	N/A	-1.99%	N/A

Figure 7: Table of all four propellants, their performance, and % comparison with GEM63XL's metrics

Ideal Performance (Thrust vs Time):

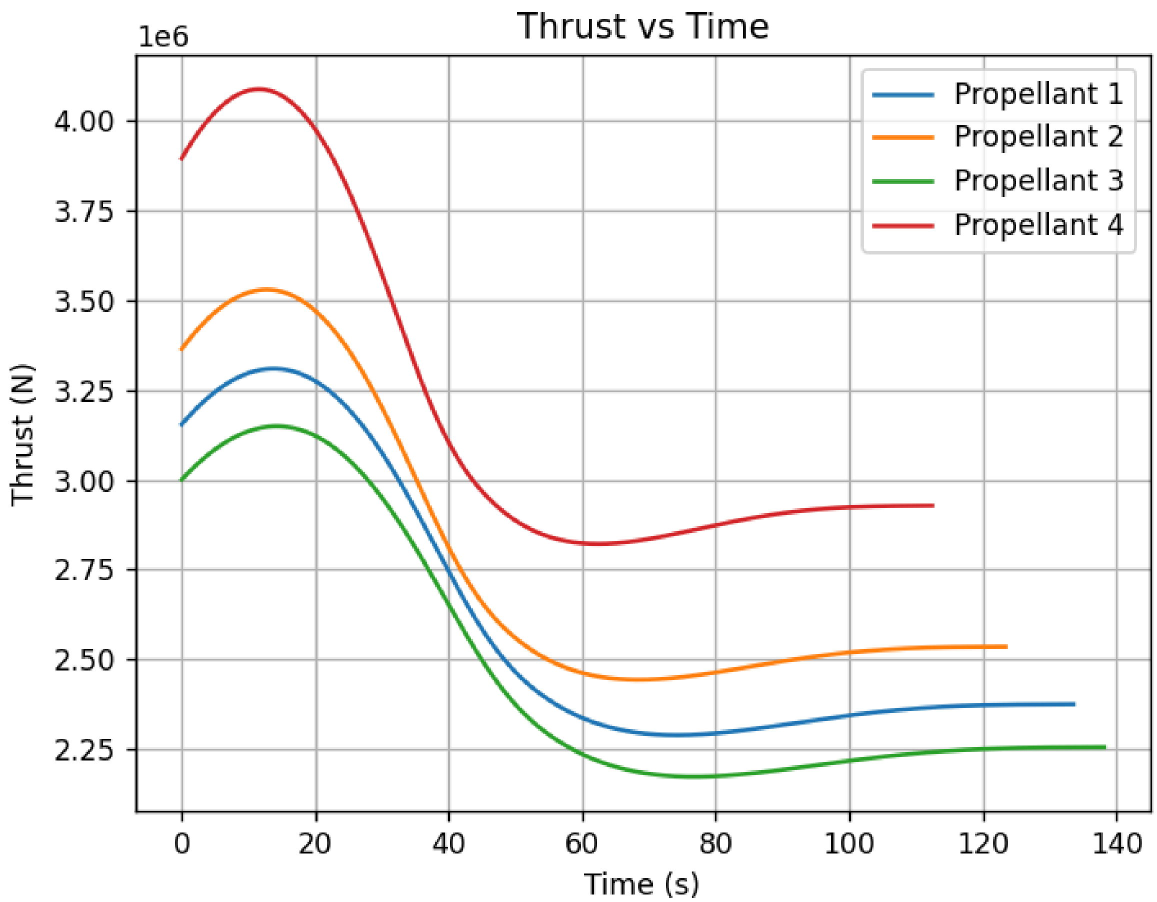

Figure 8: All four propellants and their ideal performance's thrust vs time curve

Applied Performance (Thrust vs Time + Height vs Time):

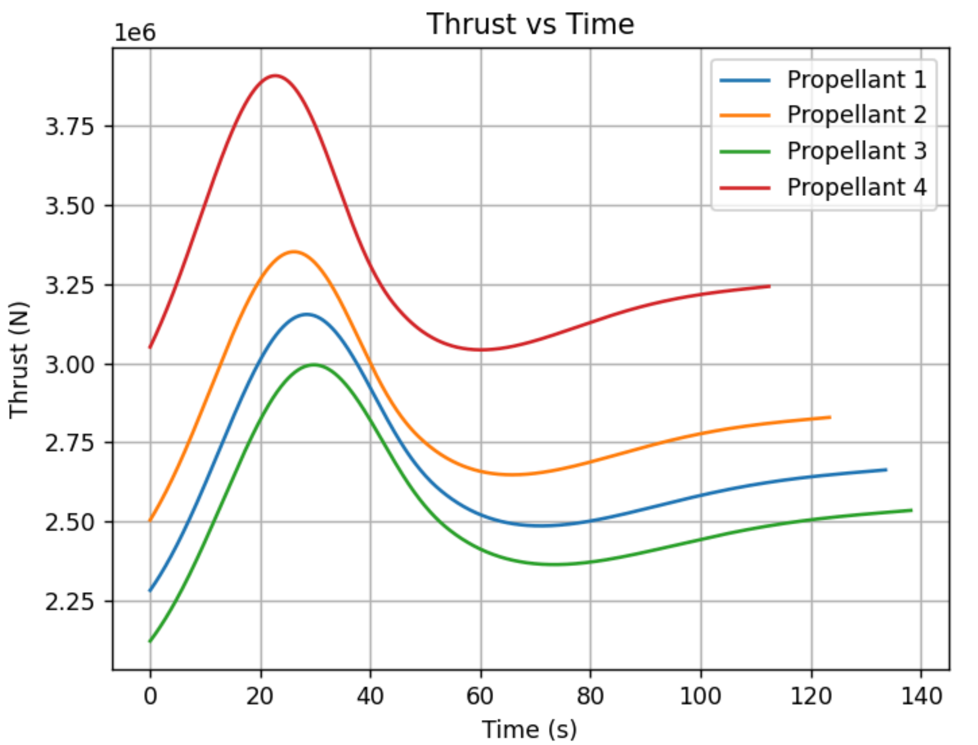

Figure 9: All four propellants and their applied performance's thrust vs time curve

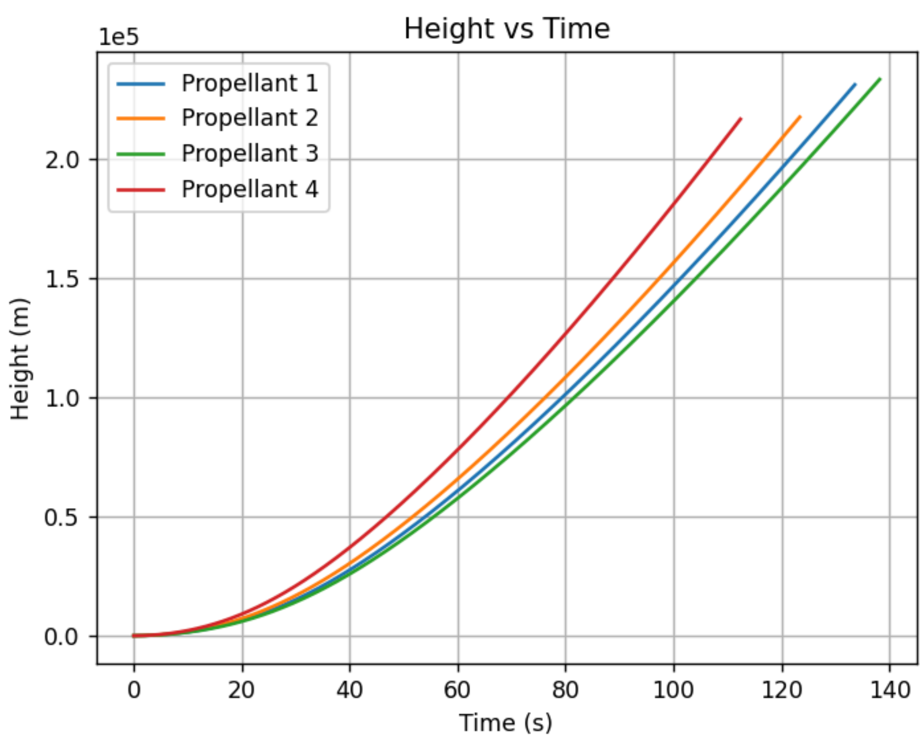

Figure 10: All four propellants and their applied performance's height vs time curve

Applied Performance with Nozzle Regression (Thrust vs Time + Height vs Time):

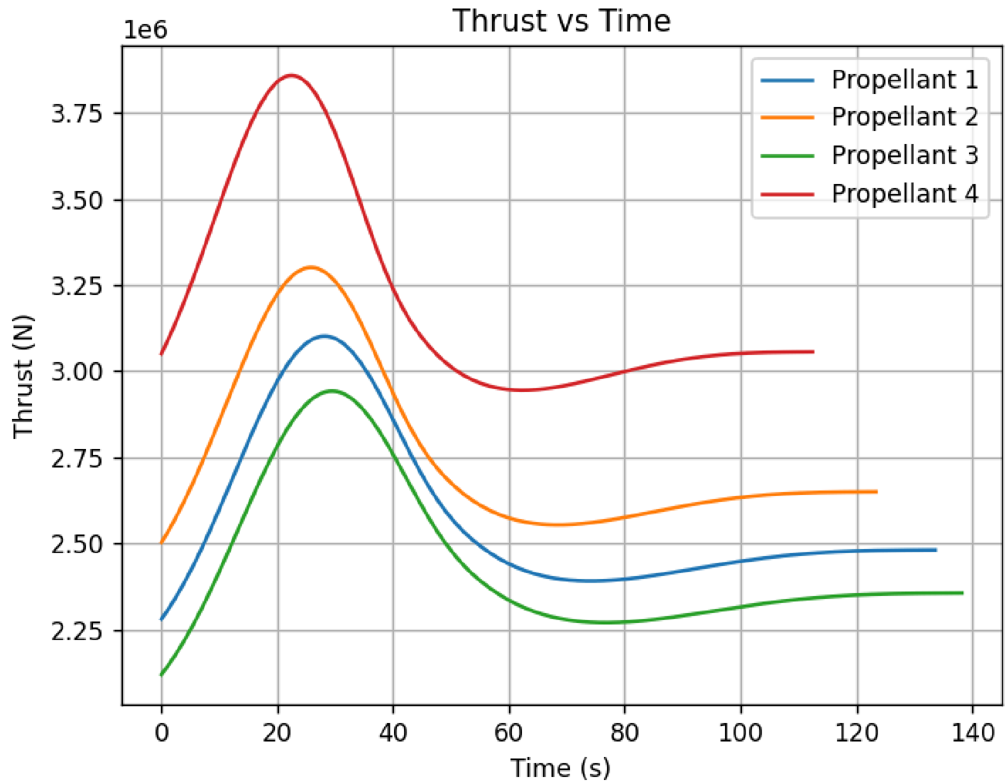

Figure 11: All four propellants and their applied performance's thrust vs time curve with nozzle regression active

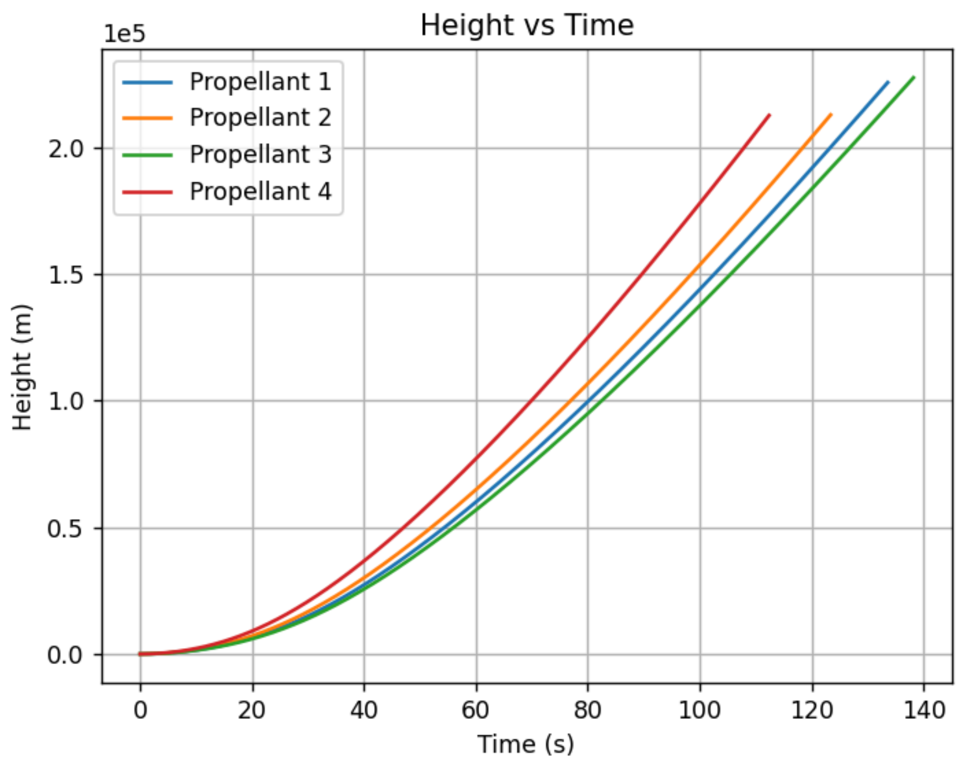

Figure 12: All four propellants and their applied performance’s height vs time curve with nozzle regression active

5. Discussion of Results

All of the thrust vs. time graphs from the propellants analyzed and the GEM63XL have a peak at the 15-20 second mark before dropping significantly because of the propellant distribution. While changing the grain geometry affects the overall performance of the motor, it cannot drastically change the shape of the graph. Having a propellant distribution that puts more propellant near the end of the motor and less near the exit would result in a thrust vs. time graph that peaks and then drops significantly before leveling out. Thus, the square root formulation for the propellants in this study is the main contributor to the specific shape of the graph. In addition, during the applied performance graph (Figure 9), thrust slightly increases past the 50-60 second mark because while the thrust is constant, the height is increasing and atmospheric pressure is decreasing, as shown by Eq. 8, thus increasing the pressure difference and increasing the thrust as shown in Eq. 1. When factoring in nozzle regression, the maximum height attained decreases because the motor is not at its maximum efficiency and loses thrust. Additionally, nozzle regression seems to have only a relatively minor impact on performance, since its effect on overall height was less than 10% for all propellants (Figure 7). When comparing the propellants among themselves, the highest performing propellant out of the four is Propellant #4. It has the highest maximum thrust, total impulse, and average thrust compared to the other three propellants. However, it does not attain the maximum height compared to the three because the high speed and thrust generated lead to immense amounts of air resistance that slow the rocket significantly after the thrust drops. In addition, it has a shorter burn time than the other propellants; thus, it is not able to attain the same height as the other propellants before needing to be decoupled. The propellant that achieves the maximum height out of the four propellants is Propellant #3, which is due to its longer burn time and lower thrust, which decreases the amount of air resistance it has to deal with. This also causes Propellant #3 to suffer the largest effect of nozzle regression out of the four propellants because its longer burn time allows for more nozzle erosion, thus reducing the efficiency of the motor. Propellants #2 and #4 are the highest performing propellants. This is primarily due to their composition, which has high amounts of fuel, oxidizer, and burn rate catalysts while having a low amount of binder. As a result, Propellants #2 and #4 have a high-energy and fast burn rate that leads to higher amounts of thrust compared to Propellants #1 and #3. The binders in the propellants are HTPB (hydroxyl-terminated polybutadiene), the oxidizer is AP (ammonium perchlorate), the fuel is aluminum powder, and the burn rate catalysts are PbSt (lead stearate) and FeO (ferric oxide).

When comparing the propellants and their performance to Northrop Grumman's GEM63XL performance, there is an expected improvement in both thrust and burn time due to changes in propellant composition and grain geometries. On average, there is a 60-70% boost in the maximum thrust achieved by these propellants with the grain geometry/distribution given. In addition, the total impulse is increased by around 165% on average. Finally, the average thrust throughout the burn time is near a 70-80% increase for all four propellants on average. Even though a higher performance can often be attained by having a more dense propellant, most of the propellant mass is only 10-20% heavier than the propellant mass of the GEM63XL, something that can be offset given the propellant's high performance compared to the GEM63XL. Since the GEM63XL specification sheet did not give the density of their propellants but rather the mass of the propellants in their rocket motor, the study can only approximate a mass comparison between the propellants. The peak in thrust for all of these propellants was mostly around the 10-20 second mark. When comparing it to the GEM63XL rocket, the time of the peak is around the same. While the propellants could not be tested at various outside temperatures, like the GEM63XL, there would be around a 10% change in thrust and a 20% change in burn time based on the data Northrop Grumman has provided [14]. However, some engineering tradeoffs have been made. For example, some of the more powerful propellants in this study are often very dangerous to manufacture and difficult to transport safely. If these propelants needed to be manufactured off-site and transported to the launch site, it would be very dangerous to transport them. Specifically, Propellant #4 is on the edge of being classified as an unstable and dangerous propellant. Another engineering tradeoff that was made was the increase in mass of the propellants. While it isn't much, more dense propellants are generally more powerful and provide more thrust, but are more massive and add more weight to the overall rocket. Finally, the cost of manufacturing and obtaining the ingredients for these propellants is another tradeoff that may limit the real-life feasibility of the propellants.

Despite the promising results of the study, there are many limitations and assumptions made in the study. All of the assumptions and limitations can be found in the Results section. One assumption made was that the characteristic velocity was 1500 m/s. This assumption was made since many rocket propellants have characteristic velocities lying around the 1500 m/s mark [15]. Since getting the actual value requires experimental testing, this assumption may overestimate the performance of the propellants. Another assumption was that air density was a constant 1.225 kg/m^3. Since air density decreases as altitude increases, the drag force on the rocket would be lower than its calculated value in the code. This would underestimate the propellants' maximum height. Some other assumptions included the atmospheric assumptions to calculate atmospheric pressure from altitude. Since the actual launch site of the rocket could have different launch temperatures and humidities, the actual performance could be lower than the simulated performance. Another limitation that may overestimate how the propellants may perform in experimental settings was the a and n values of the propellants. These values were approximated using a mix of past references and equations [15]; therefore, the actual values need to be verified through experimental testing. To calculate nozzle regression, the study assumed a nozzle material of phenolic graphite. Since GEM63XL's nozzle material might be different, the effect of nozzle regression could change from the calculated values in the code. Additionally, the propellants, grain geometry, and grain distribution analyzed were suited for a potential mission to LEO. That is why the peaks in the thrust curve resided around the 20-second mark. If a mission were to go to another celestial body or geostationary orbit, for example, a different type of propellant would be needed. Thus, future studies that might branch off from this study can analyze and subject the propellants to real simulations that can adequately measure their performance in real conditions and experimentally validate the performance values found in this study.

6. Conclusion

The four propellants analyzed in this study, along with the grain geometry and distribution formulated, suggest potential improvements to Northrop Grumman's GEM63XL booster. However, there are many limitations and assumptions that may impact its actual performance, warranting further experimental validation and testing. Regardless, providing more efficient and powerful propellants is an essential step in making solid rocket motors more powerful and able to lift heavier payloads. The propellants analyzed in this study have been theoretically proven to house possible improvements in solid rocket boosters, potentially improving space transport.

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