Spring 2019

Improving the Delivered Specific Impulse of Composite Rocket Propellant through Alteration of Chemical Composition: Methodology and Parameters for Characterization of Propellant and Validation of Simulation Software Common to the Amateur Rocketry Community

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4/26/2019

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Senior Honors Project Course No. 4200 497-002

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Abstract

In this study, two solid composite rocket propellants were designed utilizing ProPEP, a rocket propellant formulation software common in the amateur and hobby rocketry communities. The two propellants were designed to optimize specific impulse relative to a literature propellant designed by 1020 Research Labs. The literature propellant was also tested in order to validate the design of experiment as well as the mixing and testing procedures. All three propellants, which includes the literature propellant RCS-P, and the two novel propellants AKR-P1 and AKR-P2 were characterized with static tests. The results of the static tests provide data on propellant performance and characterization parameters to be used in the design of scalable rocket motors. AKR-P2 delivered a specific impulse of 219 seconds, a 20% improvement compared to the base case literature propellant RCS-P. AKR-P2 also delivered up to 22% more thrust than the other test propellant AKR-P1.

Executive Summary

One of the propellants designed, AKR-P2 was found to be the most efficient and improved propellant relative to the 1020 Research Labs base case (RCS-P). This propellant was mixed with an additional 1.8 wt% ammonium perchlorate, 16.9 wt% aluminum, and 1 wt% red iron oxide compared to the literature propellant. AKR-P2 delivered a specific impulse of 219 seconds, a 20% improvement compared to the base case literature propellant RCS-P. AKR-P2 also delivered up to 22% more thrust than the other test propellant AKR-P1. Fitting the test data to a power law model in the form of Saint Robert's Law resulted in a burn rate coefficient (a) of 0.0282 for AKR-P2 and a pressure exponent (n) of 0.3564. The other test propellant, AKR-P1, was formulated with an additional 2.3 wt% ammonium perchlorate, and 17 wt% aluminum. The added material in each propellant replaced the secondary oxidizer, strontium nitrate used in the base case propellant as this ingredient exists primarily to color the combustion flame purple. AKR-P1 delivered a specific impulse of 202 seconds and can be characterized by a burn rate coefficient (a) of 0.0360 and a pressure exponent (n) equal to 0.3005. The literature propellant, RCS-P, was also prepared and found to deliver an average specific impulse of 182 seconds and can be characterized with a burn rate coefficient of 0.0051 and a pressure exponent of 0.6719. 1020 Research Labs reports that RCS-P delivers a specific impulse of 185 seconds with a coefficient of 0.0215 and an exponent of 0.3866. The performance of the RCS-P tested in this study closely matches that reported in literature. The burn rate characterization parameters are quite different likely because two of the four motors constructed with the RCS-P propellant resulted in failed tests. One of the tests did not record pressure and the other over pressurized the system. As a result, only two data points were able to be used in the determination of the burn rate coefficient and pressure exponent. ProPEP predicted that the three propellants, RCS-P, AKR-P1 and AKR-P2 would deliver specific impulses of 176 s, 195 s and 194 s respectively. RCS-P, AKR-P1 and AKR-P2 actually delivered 182 s, 202 s and 219 s respectively, proving that ProPEP regularly under predicts propellant performance. The largest discrepancy in specific

impulse between the measured value and the predicted value was observed for ARK-P2 at 25 seconds. This result suggests that ProPEP may be inadequate in predicting the performance of composite propellants with high burn rate modifier loadings.

Introduction

Rocket propellants are chemical mixtures, designed to provide ample thrust leading to high-performing and precise rocket ascents. While generating experimental data from mixed propellants is crucial to verifying a propellant's characteristics, developing models to predict a mixtures behavior is quite advantageous and expedites the development process. Modeling software offers characterization methods and tools to predict propellant behavior based on empirical data. Propellant burn rate characteristic parameters will be determined in this project to provide predictions for a range of propellant mixtures. The model validation will be completed through static test stand measurements of each investigated propellant. The parameters required to determine the burn characteristics of each test propellant will be measured. Once experimental analyses are completed, data analysis techniques will assess the predictive accuracy of the modelling programs. The verified method of developing base models for propellants will establish reliable and predictable model data for propellant mixtures ultimately decreasing the testing requirements when determining propellant behavior. As long standing members and leaders on the Akronauts rocket engineering design team at The University of Akron, it is known that development and construction of a novel and efficient propulsion system is critical to maintaining a competitive advantage.

The objective of this project is to formulate, design, mix, and fly an optimized propellant at the Spaceport America Cup. This Intercollegiate Rocket Engineering Competition (IREC) is hosted each year in New Mexico by the Experimental Sounding Rocketry Association (ESRA). The optimization of this propellant is to be performed empirically using static test data and assisted by software common in the composite rocket propellant industry. The test data will also serve as a source of validation for associated software predictions. Through a well-researched and well-designed propulsion system, the Akronauts rocket engineering design team will be eligible for a higher score at IREC, and thus a higher ranking among the more than fifty participating collegiate teams from more than 6 countries ("What", 2018). High performance at the Spaceport America Cup could result in increased exposure for The University of Akron and could improve the sponsorship opportunities available to the Akronauts rocket engineering design team.

Background

Solid Propellants

Solid propellants are generally regarded as easier and safer to combust compared to liquid propellants. Propellant consistency and reliability is greater for solid composite propellants compared to liquid propellants (Sobczak, 1996). Solid propellants offer an opportunity for grain geometry design optimization to maximize surface area of combustion compared to liquid propellant. Solid propellants require high temperatures for ignition, posing a potential safety benefit. Solid propellants are more stable than liquid propellants which establishes them as a favorable phase for amateur rocketry (Braeunig, 2012). Solid propellants reach dynamic equilibrium quickly and produce consistent results.

Thrust Curves

Thrust curves are curves produced from performing static tests to determine the force generated from a propellant mixture over the course of a complete burn. Thrust is graphed as a function of time as exemplified in Figure 1. Thrust curves generated from static testing allow for the determination of mass flow rate, burn time, total impulse, characteristic velocity, and specific impulse. Integrating the area under a thrust curve gives the total impulse exerted over the course of the test (Nakka, 2000). Since the mass of the propellant and total time of the burn are known, the specific impulse can be determined. Balancing the duration of the burn as well as the force generated from the burn is required to improve total impulse. A motor with very high thrust and a short burn time will likely produce a lower total impulse than a motor designed to provide the required amount of thrust over an extended burn duration.

Three major types of burn profiles can be observed from thrust curves when performing static tests. Progressive burning is defined by an increase in the reacting surface area during the interval of combustion. This behavior can be observed in a thrust curve as the thrust increasing over time after the initial jump. This type of burn is characteristic of bates grains, the geometry used in this experiment. Highly progressive burning fuels are undesirable as the increased reaction rate results in a dramatic pressure increase. A thicker walled and thus heavier motor casing must then be used to contain this peak pressure. This design is inefficient as the casing thickness is too great for any portion of the burn that is not the peak pressure. The next profile of burn is neutral. Neutral burning is defined by a burn area and reaction rate that remains roughly constant throughout the duration of combustion. This profile is observed as a constant thrust between ignition and burnout. Neutral burning is the most desired and is considered to be the most reliable and efficient profile. The third major type of burn profile is regressive burning which is defined as a decrease in burn area and reaction rate over the duration of combustion. This profile can be observed as a negatively sloping thrust during the burn time. Regressive profiles are characteristic of ending burning propellant geometries. Erosive burning is independent of the three major profiles and can be observed as a quick decrease in thrust just after the ignition spike. This type of burn is likely caused by unreacted propellant released through the nozzle throat before the reaction conditions, particularly temperature and pressure are high enough to combust this material. Very slight erosive conditions can be observed in Figure 2. It can be very difficult to predict the performance of highly erosive propellants so this condition should be avoided (Kosanke 2012).

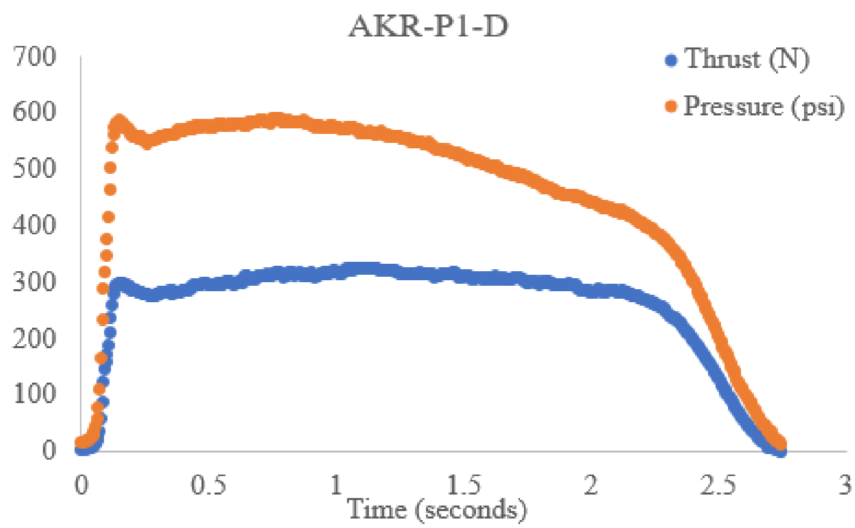

Figure 1 - Thrust and pressure graphed against time for propellant AKR-P1 motor D.

Grain Geometry

Although rocket motors are almost always cylindrical in terms of their outer geometry, the cross section can reveal a variety of designs. Grains are the geometric shapes into which a propellant is casted at the termination of the propellant mixing process. Grain geometry is exceedingly significant concerning the thrust behavior over time and the available burn area of a propellant. Typically, a version of an annulus is created with a specific core geometry ranging from stars and circles to tubes as displayed in Figure 2 (Nakka, 2001). The variety of grain geometries results from experimental efforts to manipulate the thrust profile of a propellant. The burn area is sought to be optimized throughout the motor to produce a neutral thrust profile for any propellant formulation. NASA conducted studies ultimately determining that the optimized grain geometry is a 10-point star core, producing a flat, consistent thrust curve (Johannsson, 2012). For amateur rocketry purposes, circular grains with an annular core are most popular and yield consistent data despite the slightly progressive nature. Achieving complex grain geometries can be quite difficult in amateur rocketry due to limited access to specialized tools and equipment needed to manufacture detailed grains. For this reason annular grains stacked in a motor referred to as bates grains are most commonly used. A bates grain geometry was applied to all of the test motors in this study.

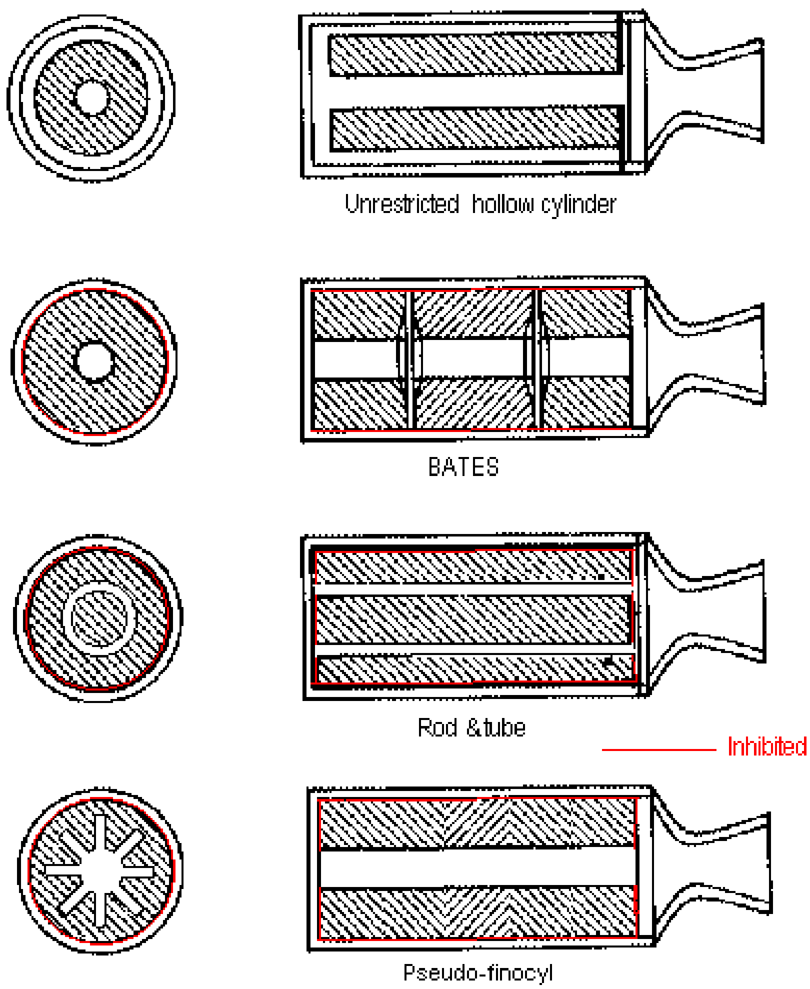

Figure 2 - Examples of typical grain geometries for solid rocket propellants.

Specific Impulse

The specific impulse of a rocket motor is one of the most important values to determine the overall propellant performance and is commonly used as an indication of efficiency. Equation 1 defines the specific impulse (I_{sp}) expressed in seconds as a ratio of the total impulse produced from the rocket motor (I_t) and the mass of the material (m) multiplied by the gravitational acceleration (g) (Braeunig, 2012). Specific impulse is often described as the motor’s efficiency, which is sought to be improved in the study. In essence, the specific impulse measures the amount of thrust produced over a given time per the amount of propellant consumed. ProPEP modeling software generates a theoretical specific impulse value for any prospective propellant mixture and was used to determine which test propellant batches to mix and test. Data generated using ProPEP is detailed in Data and Results section. Inefficiency in rocket motors results from a variety of mechanical energy losses in force. Such losses can arise from incomplete chemical combustion and nozzle pressure drop (although necessary for generating a large C* value). Improving specific impulse of a mixture from a chemical standpoint chiefly involves ensuring complete combustion. Maximization of the specific impulse offers the opportunity for full scale rockets to save weight and money by using less propellant. Weight not added by propellant can then be made available for control systems, payload materials, recovery materials, or to operate a lighter overall rocket. Increased payload weight for full scale commercial rocket industries have direct relationships to a project's financial viability.

$I_{sp}=I_t/mg$ (1)

Nozzles

Nozzle sizing of rocket motors must be completed to determine the system's operating pressure. Nozzles establish an immense pressure drop between the chamber pressure and ambient pressure through which the gaseous materials produced in combustion reactions exit the motor at supersonic velocities. Equation 2 displays the Knudsen Number equation which expresses a ratio between the combustion area (A_b) of the grain to the area of the nozzle throat (A_t). Although nozzles are almost always designed with circular exit orifices, the diameters of the orifices are often adjusted. Varying the Knudsen number for propellant characterization is achieved for a single grain size through changing the nozzle size. Manipulating the Knudsen number allows for control over the pressure in the chamber. Safety should be considered when executing nozzle sizing as well as propellant mixture design. Motor casings are rated to specific chamber pressures, and consequently the Knudsen number is altered in order to ensure mechanical integrity of the motor. Figure 3 is a diagram of a basic combustion chamber and nozzle (Braeunig, 2012).

The design of the rocket nozzle has large effects on the thrust generated from a propellant. The nozzle throat is indicated as A_t while the chamber is defined as P_c. Equation 3 displays the equation used to determine the chamber pressure as a function of constant B, Knudsen number, and the pressure exponent from Saint Robert's Law (Nakka, 2000). Since the pressure exponent constant and constant B cannot be changed unless significant alterations to chamber pressures are made, decreasing the Knudsen number by increasing the nozzle size can allow for lower, safer chamber pressures based on the casing's material of construction. The Knudsen number is also useful in the scale up of rocket motors. If a smaller motor is tested using a particular K_n and the operating pressure of that motor is measured, a larger motor can be designed using the same K_n. This larger motor, which is likely geometrically different and utilizes a different size nozzle, will operate at the same chamber pressure as that of a smaller motor if the K_n is maintained the same. The tests completed in the study use different nozzle areas in order to generate a range of operating chamber pressures, Knudsen numbers, and thrust curves for characterization purposes.

$K_n=A_b/A_t$ (2)

$P=B(K_n)^{1/(1-n)}$ (3)

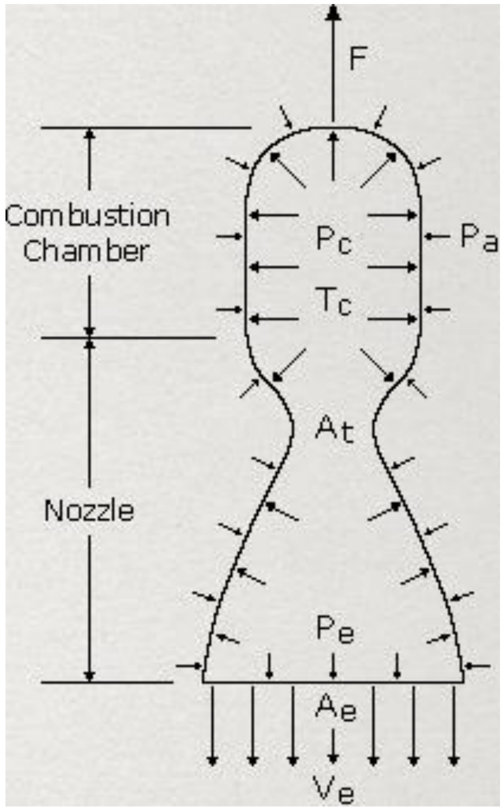

Figure 3 - Diagram of a typical rocket motor pressure chamber, nozzle, and exit area.

St. Roberts Law

Equation 4 displays Saint Robert's Law, also known as the burn rate equation. Burn temperature only exerts a negligible effect upon a propellant's characteristic parameters and is consequently not included in the equation. The burn rate 'R_b' is expressed in distance per time and can be modeled for a specific propellant formulation once two parameters are determined. The first parameter is the burn rate coefficient (a) and the second is the pressure exponent (n). The burn rate coefficient is a unit less value which can be found for a specific chamber pressure range (Sobczak, 1996). The process of determining the 'a' and 'n' constants from Saint Robert's Law is described as propellant characterization. ProPep rocketry program utilizes experimental thrust curves from static test stand experiments for different nozzle sizes in order to calculate the burn rate coefficient and pressure exponent through power law regression. This program fits test data to Saint Robert's Law using the method of least squares. Typical burn rate coefficients for systems where the burn rate units are expressed in inch/s and chamber pressure is expressed in psig are near 0.0387 (Braeunig, 2012). Depending on the propellant, 'a' and 'n' values can be appropriate for wider or smaller ranges of chamber pressure. A burn rate of 0.15 inch/s at 1atm chamber pressure for an average potassium nitrate formulation will yield a burn rate of 0.60 inch/s when the chamber pressure is 1000 atm (Nakka, 2003). The pressure exponent 'n' is also gleaned from experimental data. As the pressure exponent increases, the burn rate becomes increasingly responsive to any changes in the parameter value. Pressure exponents for typical propellants range from 0.3-0.6. An accurate and repeatable characterization for the Saint

Robert’s Law parameters is integral to depicting a motor’s performance. Determining the burn rate aids in establishing reliable motor sizing, propellant characterization, and performance modeling. Experimental methods employed to determine the parameters for each mixture are explained in the Experimental Methods section.

$R_b = aP_c^n$ (4)

Ingredients

Ammonium perchlorate (AP) is a popular and proven oxidizer commonly used in solid rocket propellant. Aside from rocket propellant uses, ammonium perchlorate is utilized for its explosive characteristics within the mining and firework industries. One safety benefit with using ammonium perchlorate is that the strong oxidative potential of the chemical remains stable below 65.6°C. Explosion dangers are prevented from ensuring that no exposure to possible contaminants occurs. Ammonium perchlorate has also been analyzed extensively by amateurs, engineers, chemists, and NASA. Aluminum acts as the fuel for such mixtures, resulting in a heterogeneous propellant where oxidizer and fuel exist in separate chemical structures. Solid rocket propellant mixtures are composed of chemicals other than oxidizer and fuel in order to execute a variety of functions (Sobczak, 1996).

Ammonium perchlorate propellants are typically binded by HTPB (hydroxyl-terminated polybutadiene). Isocyanate acts as the curative for propellants using HTPB where the terminated hydroxyl functional groups execute polymer crosslinking. During mixing, HTPB acts as the main medium through which the solids in the propellant are intermixed with one another. Using a binder whose viscosity can be lowered to aid in mixing is essential for propellants with high solids fractions. Binder systems are necessary for solid rocket propellants as they establish physical strength for the mixture once the mixing process is completed (Sobczak, 1996). Propellants must hold the mechanical strength required to protect grains when straining forces are applied. Case bonding is the process in which the motor casing and propellant grains are bonded to one another typically using a polymeric binder.

Plasticizers are introduced at larger solid loadings in solid rocket propellant formulas to lower the overall viscosity of the liquids in the propellant. Lowering the viscosity allows for maximization of the amount of solids added to the mixture. Isodecyl pelargonate is employed as the plasticizer for the test propellants compared to dioctyl adipate and dioctyl azelate due to the material’s low health dangers and improved mixing performance (Sobczak, 1996).

Strontium nitrate is a popular secondary oxidizer used in ammonium perchlorate based propellants. Strontium nitrate is included in many amateur rocketry AP propellants for aesthetic reasons as the material’s reaction in the motor produces a bright purple exhaust flame exiting the nozzle. Opportunity for performance optimization is considered in the study’s test propellants by removing strontium nitrate.

Burn rate modifiers of various kinds are included in AP solid propellant recipes in order to alter the burn rate. Both inhibitors and promoters of the burn rate speed exist; where metal oxides are used to increase the rate, and salts are used to decrease the rate (Sobczak, 1996). Iron oxide is investigated in the study test propellants as a result of its ability to encourage decomposition of ammonium perchlorate. Additional thrust is anticipated with the addition of iron oxide as a result of the increase in gas production. Copper chromite is included in the test propellant mixtures as well as a burn rate promoter; however, previous studies have produced mixed results regarding the oxide's effectiveness in increasing the burn rate.

Tepanol is used as a preservative for AP propellant mixtures to extend the shelf life by establishing stronger bonds between the HTPB binder and the AP particles. The entire grain's strength is improved considerably by the presence of Tepanol (Sobczak, 1996). Tepanol also improves the propellant mixing process by adding a small amount of another liquid to the mixture.

Experimental Methods

Chemicals

In this study, a variety of chemicals were used in order to produce each propellant used in this experiment. Each of the chemicals employed in the construction of the propellant mixture has a unique contribution to the overall performance of the propellant mixture. All data concerning the chemicals themselves such as particle size were provided by the manufacturers and were not verified as part of this study. The ammonium perchlorate oxidizer used is standard grade, rotary rounded, and 200 microns in size ("Bulk", n.d.). The aluminum powder, obtained from Alpha Chemicals ("Alpha", n.d.) is 99.5% pure aluminum, features a 50% pass particle size of 30 microns (500 mesh), is uncoated, and is produced through atomization. The copper chromite catalyst used is a proprietary blend of copper and chromium oxides ("Copper", n.d.). The red iron oxide used exclusively in propellant AKR-P2 features an average particle size of 30 microns and a loose packed density of 55 lb/ft^3 ("Red", n.d.). Strontium nitrate was obtained through FireFox Enterprises and used in propellant RCS-P ("Chemicals", n.d.). The low molecular weight hydroxyl terminated polybutadiene resin (HTPB) used has a molecular weight of 1300 g/mol, a polydispersity index of 2, and a viscosity of 1500 centipoise at 30°C ("Low", n.d.). The plasticizer isodecyl pelargonate (IDP) used in each of the propellants was also obtained from RCS Rocket Motor Components ("Isodecyl", n.d.). Tepanol is a dark yellow and very viscous liquid obtained from RocketsRUs. Modified MDI isocyanate curative is a prepolymerized diphenylmethane diisocyanate that has a viscosity of 450 centipoise at 25°C ("Modified", n.d.). The very small amount of castor oil used in the two experimental propellants was obtained from Sky Organics ("Organic", n.d.) and was used to help maintain a consistent solids fraction as well as very slight improvement in predicted performance. The composition of each propellant is detailed as a percentage in Table 1 and the actual amounts of each chemical mixed into each batch are detailed in Appendix A2.

Table 1 - composition of control propellant as represented in literature as well as the composition of each propellant manufactured for this study.

Propellant Compositions	Literature RCS-P	RCS-P	AKR-P1	AKR-P2
Ammonium Perchlorate	63.3%	63.3%	65.6%	65.1%
Aluminum	0.5%	0.5%	17.5%	17.4%
Copper Chromite	0.2%	0.2%	0.2%	0.2%
Red Iron Oxide	-	-	-	1.0%
Strontium Nitrate	19.5%	19.5%	-	-
HTPB	12.1%	12.1%	12.1%	12.0%
IDP	1.5%	1.5%	1.5%	1.5%
Tepanol	0.5%	0.5%	0.5%	0.5%
MDI Isocyanate	2.3%	2.4%	2.4%	2.2%
Castor Oil	-	-	0.1%	0.1%

Overview of Mixing Procedure

In order to prepare each of the propellants, a mixing procedure such as the example shown in Appendix A3 was followed. In this procedure all of the liquid components are first mixed together using the paddle attachment of a KitchenAid Professional 6 quart stand mixer. Next solid components are added one by one with 5 minutes of mixing in between each addition. Once all ingredients are added with the exception of the curative, the mixture is stirred for 60 minutes. During this time the casting tubes, casting caps, and coring rods are all prepared to receive propellant. Once the mixture is homogeneous, curative is added and mixed for 15 more minutes. The propellant is placed under vacuum for 5 minutes to degas and then packed into molds to cure. A more detailed procedure with relevant safety precautions is detailed in Appendix A2.

Test Sample Grains

Test propellant grains were cast into annular geometries. The approximate dimensions of each grain are 3 inches in length, 1.81 inches in outer diameter, and 0.625 inches inner core diameter. Maintaining consistent propellant dimensions and geometry is essential to ensuring that each variation in composition is compared appropriately. Small differences in dimension are accounted for through pairing of two grains of slightly different masses in order to achieve a certain total weight for each trial. In order to maintain similar total masses for each set of 2 grains, combinations for each set were established to ensure that the total mass of each motor was as near as possible to the grain's mass average.

Four burns were completed for each propellant formulation. Based on the chamber pressure at a base nozzle size of 0.375", the nozzle size for the following test burns was manipulated to generate a range of chamber pressures for the four burns. The Data and Results section details the nozzle sizes and exit diameters for each motor test in Figure X.

The static test stand used for each burn test is depicted in Figure 4. Two electrodes are attached to a copper wire connected to an ignition charge placed into the bottom of the pressure chamber. Ignition was executed remotely. The rocket motor is placed such that the thrust is directed into the ground, maintaining a stationary test as the nozzle is placed into the top of the motor. A pressure transducer is fixed to the pressure chamber in order to monitor the pressure throughout the test. A load cell is also fixed to the bottom of the system to monitor the thrust generated throughout the test. Measurements were recorded through ThrustCurve (TC) Logger software every 0.005 seconds. Exporting TC Logger data into ProPEP along with grain composition and geometry information allowed ProPEP to calculate the burn rate and pressure exponent factors.

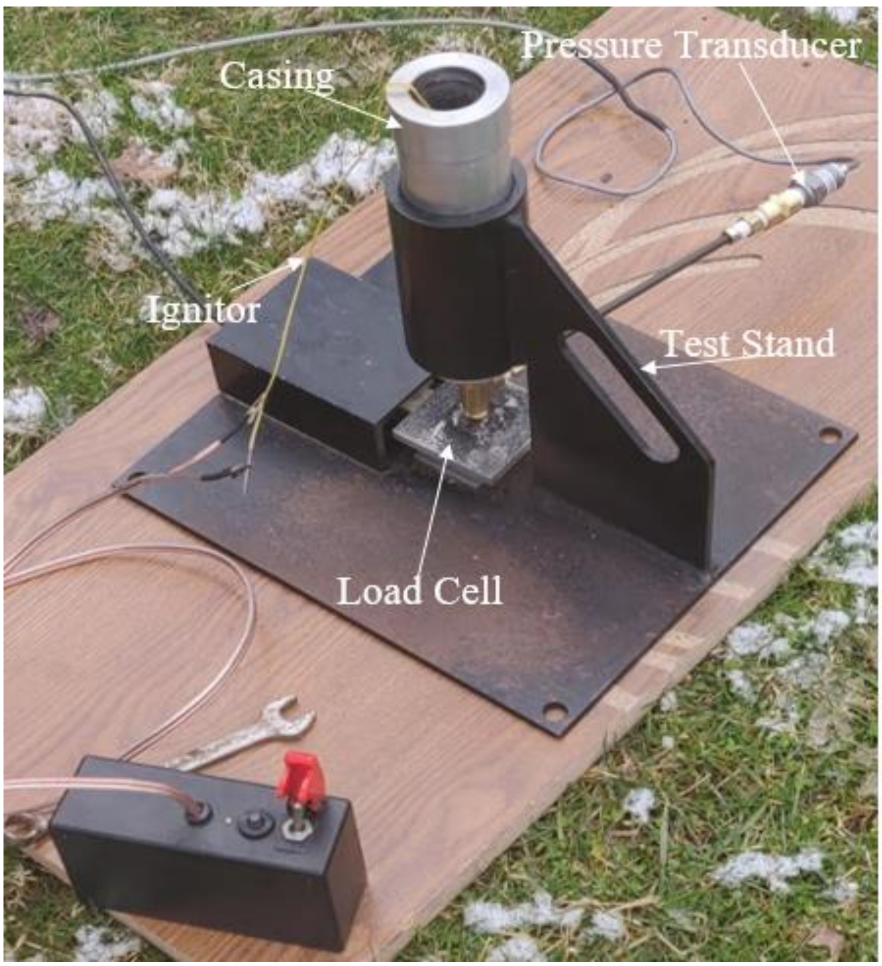

Figure 4 - Static test stand used for each burn test with equipment components indicated

Results and Discussion

In order to determine what parameters to vary in the static tests, ProPEP simulations were executed where the RCS Purple generic propellant formulation was altered in order to improve the specific impulse of the motor. RCS Purple was still mixed and tested even though literature values already exist for the burn rate coefficient and pressure exponent. RCS Purple was included in the project analysis in order to verify that the characterization technique is accurate and provide a baseline for the two experimental propellants. After completing research regarding the base RCS Purple's chemicals, it was determined that strontium nitrate should be removed from the formulation for propellant improvement purposes since strontium nitrate is included mainly to establish a purple color to the rocket's exhaust.

A study was performed in which each component in the original RCS-Purple formula was systematically increased. Each formulation was normalized to be 100 grams and ran in ProPEP. After multiple iterations were completed for each component in the propellant recipe, the specific impulse of each formula was graphed against each formulation with the increased component. This type of graph was assembled for each component in the RCS-Purple recipe and an example is shown in Figure 5 where the concentration of aluminum was systematically increased in the recipe. Based on this study, it was determined that the fuel, aluminum, and the oxidizer, ammonium perchlorate, had the most significant impact on the resulting specific impulse of the formula. It was then concluded that in order to improve the performance of the base RCS-Purple propellant, the amount of aluminum and AP in the recipe should be increased.

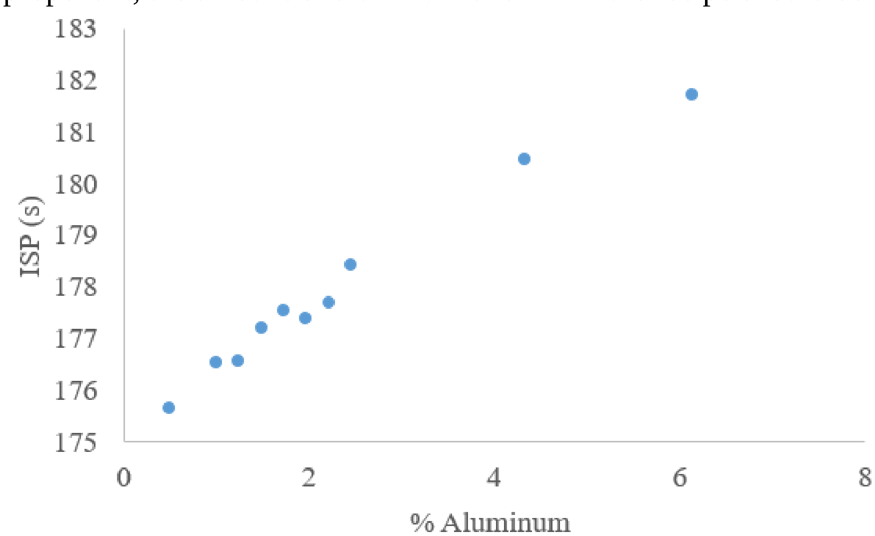

Figure 5 - Specific impulse graphed against formulas with varying concentrations of aluminum

With a goal of improving the specific impulse, increasing the amount of ammonium perchlorate oxidizer and aluminum fuel in place of the 19.5% strontium nitrate in the original recipe was investigated. The simulation sensitivity study that yielded the best specific impulse at

197.85s according to ProPEP occurred when the aluminum loading increased to 17.56g, while the ammonium perchlorate loading was increased to 65.74g as shown in Table 2. Figure 6 clearly displays the local maximum of specific impulse for the ratio of additional aluminum and ammonium perchlorate at ⅛ AP and ⅞ Al. This ratio of ingredients was applied to the base case RCS-Purple recipe to replace all strontium nitrate in the formula. The novel propellant generated from this substitution of additional fuel and oxidizer was mixed and tested as AKR-P1.

Red iron oxide is a burn rate modifier known to catalyze and accelerate combustion. The modifier was included in formulation AKR-P2 in order to determine if the addition of the burn rate accelerator red iron oxide would further catalyze the reaction and cause more aluminum to react over AKR-P1. Burn rate modifiers are valuable additives for rocket propellants since they can cause noticeable improvements to specific impulse and thrust while only being added at low loadings (0.1%-1%). The detailed loadings of each chemical added to each propellant mixture is shown in Table 1. In order to evaluate the potential improvement added by a burn rate catalyst, the second test propellant AKR-P2 was designed to be the same as AKR-P1 but with an added 1 wt% red iron oxide into the propellant formula, (Table 1).

Table 2 - Simulated formulations where the 19.5% strontium nitrate in the RCS Purple mixture was replaced with varying ratios of aluminum and ammonium perchlorate.

	RCS-Purple	**All AP**	**1/2AP 1/2 Al**	**3/4 AP 1/4 Al**	**1/4 AP 3/4 Al**	**1/8 AP 7/8 Al**	**All Al**
R45HTLO (HTPB)	12.11	12.11	12.11	12.11	12.11	12.11	12.11
IDP	1.51	1.51	1.51	1.51	1.51	1.51	1.51
Tepanol	0.50	0.50	0.50	0.50	0.50	0.50	0.50
Castor Oil	0.12	0.12	0.12	0.12	0.12	0.12	0.12
Aluminum	0.50	0.50	10.25	5.38	15.13	17.56	20.00
Copper Chromite	0.20	0.20	0.20	0.20	0.20	0.20	0.20
Strontium Nitrate	19.50	0.00	0.00	0.00	0.00	0.00	0.00
Ammonium Perchlorate	63.30	82.80	73.05	77.93	68.18	65.74	63.30
E744	2.26	2.26	2.26	2.26	2.26	2.26	2.26
Sum	100.0	100.0	100.0	100.0	100.0	100.0	100.0
ISP (s)	175.66	189.39	195.36	192.68	196.91	197.85	195.54

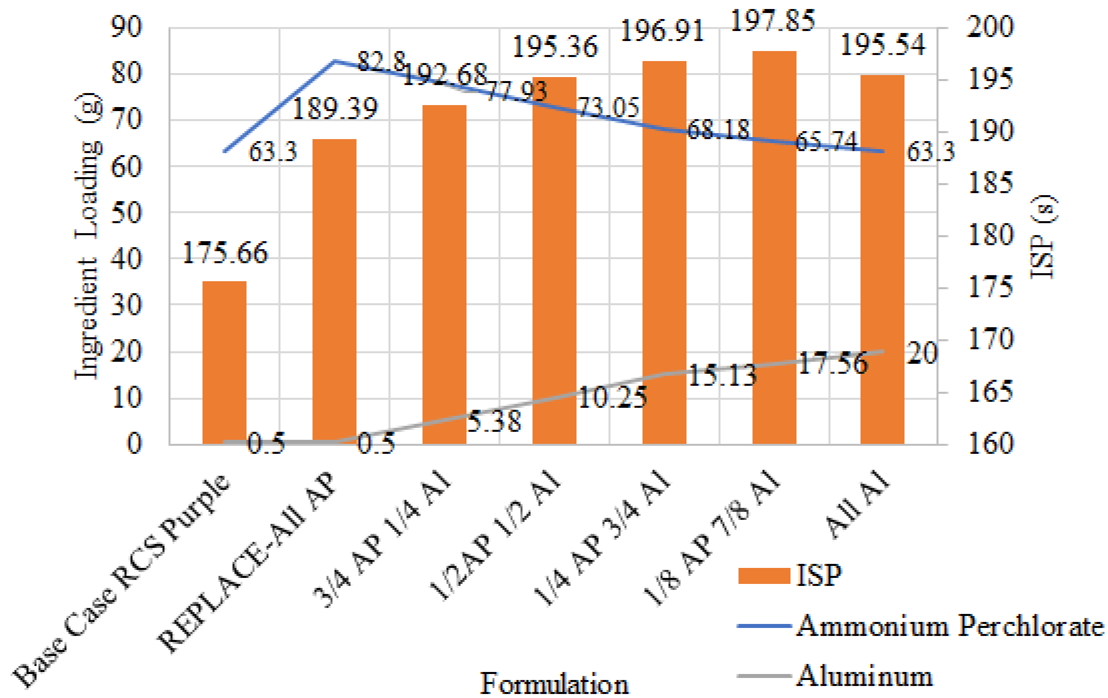

Figure 6 - Change in specific impulse is graphed for each of the different formulations altering the loadings of aluminum and ammonium perchlorate

Ten grains were cast from each propellant in an effort to generate five test motors to characterize each propellant. One grain from each batch cured in a malformed geometry and consequently could not be used in this study. As every grain must be paired in order to be tested, this resulted in one extra grain of each propellant type. Because only 8 grains were available to be practically used, 4 motors of each propellant type were assembled. The measured weight of each grain is reported below in Table 3 after each had fully cured.

Table 3 - Weight of each grain cast from each propellant type.

RCS-P		AKR-P1		AKR-P2	
Grain	Weight (g)	Grain	Weight (g)	Grain	Weight (g)
RCS-P-1	177.8	AKR-P1-1	178.8	AKR-P2-1	178.5
RCS-P-2	175.7	AKR-P1-2	175.1	AKR-P2-2	179.1
RCS-P-3	178.8	AKR-P1-3	175.6	AKR-P2-3	176.9
RCS-P-4	174.9	AKR-P1-4	175.8	AKR-P2-4	176.9
RCS-P-5	177.4	AKR-P1-5	177.8	AKR-P2-5	175.3
RCS-P-6	178.8	AKR-P1-6	172.4	AKR-P2-6	176.7
RCS-P-7	174.4	AKR-P1-7	174.2	AKR-P2-7	176.2
RCS-P-8	174.7	AKR-P1-8	175.8	AKR-P2-8	177.6
RCS-P-9	177.0	AKR-P1-9	175.2	AKR-P2-9	173.9

Two grains of the same propellant type were required for each single motor. The advantage of using two grains compared to a single grain is that the dynamic nature of the burn shifting from one grain to another is simulated. The test is thus more indicative of a full scale motor burn due to the use of multiple grains per motor. Two grains from Table 3 were selected from each mixture’s grain set to minimize the standard deviation between the weights of each motor. The average combined weight and standard deviation for each propellant is shown in Table 4. Originally for AKR-P1, grains 1 and 6 were paired to have more consistent motor weights. On the day of testing a large inclusion was discovered on the inner diameter of grain AKR-P1-6 so it was replaced with AKR-P1-2, upsetting the average and resulting in a larger standard deviation.

Table 4 - Overview of how each grain was paired in order to make the most consistent combined weight for each test.

Motor	RCS-P		AKR-P1		AKR-P2	
	Paired Grain	Combined Weight (g)	Paired Grain	Combined Weight (g)	Paired Grain	Combined Weight (g)
A	1&2	353.5	1&2	353.9	1&5	353.8
B	3&8	353.5	3&4	351.4	3&6	353.6
C	5&9	354.4	8&9	351.0	4&10	353.8
D	4&6	353.7	5&7	352.0	7&8	353.8
Average	353.8		352.1		353.8	
St. Dev	0.4		1.1		0.1	

The RCS-Purple propellant mixture was tested first since the individuals executing the testing were all quite experienced with the propellant. If the static test stand setup or grains had any issue, RCS-Purple would be the best indication of such problems. RCS-P motors already have literature characterization parameters, so characterization of the propellant was only necessary to serve as a baseline reference for the experimental propellants (AKR-P1 & P2). The RCS-P-A motor was not included in the thrust curves since the pressure was not logged properly but is included in Figure A11. RCS-P-B was also not included since the thrust curve was highly irregular and displayed immense over pressurization as shown in Figure 7. This over pressurization likely occurred as a result of an inclusion or cavity within the wall of the propellant grain. When the flame front reaches an inclusion in the propellant, the instantaneous burn area is dramatically increased, resulting in a spike in the burn rate of the propellant and thus a spike in chamber pressure. This type of burn is flawed and does not represent the actual burn characteristics of the propellant so cannot be used to determine the burn rate parameters. Figure 8 represents a more characteristic burn profile for RCS-P. This profile is decently progressive as the thrust increases over the combustion interval.

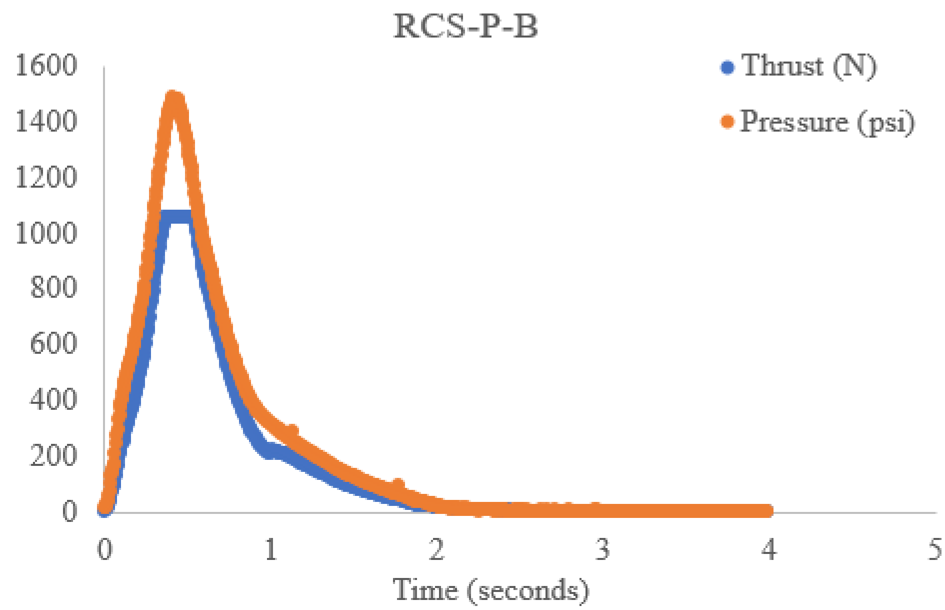

Figure 7 - Thrust and pressure graphed against time for propellant RCS-P motor B. This motor showed a very progressive profile and burned out very quickly. This propellant over pressured and is considered a failed test.

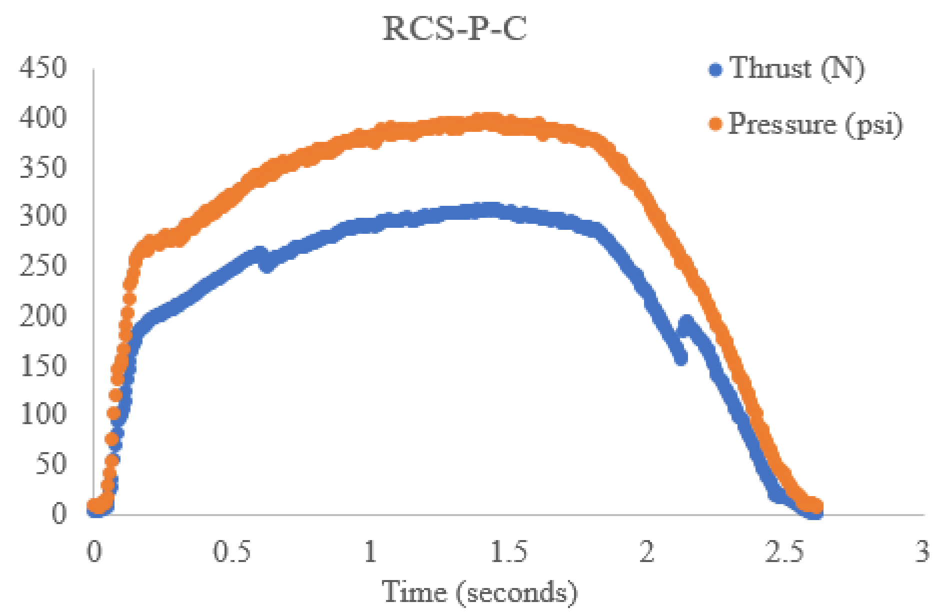

Figure 8 - Thrust and pressure graphed against time for propellant RCS-P motor C. This motor showed a slightly progressive profile

A thrust curve for AKR-P1 is shown in Figure 9 and is representative of the four trials performed with this propellant. This propellant shows very neutral behavior after being slightly erosive. A thrust curve for AKR-P2 is shown in Figure 10 and is characteristic of the trials of this propellant. This curve shows less erosive nature but is slightly more progressive than AKR-P1. Both of the AKR experimental propellants show slightly erosive behavior and this is likely as a result of the very high aluminum content in the propellant formula. It is likely that just after ignition, the vapor stream exiting the nozzle is carrying non-combusted or partially combusted pieces of aluminum, resulting in the erosive profile. While performing static tests, it was noted that AKR propellants produced sparks just after ignition that did not remain throughout the combustion interval. These sparks are likely aluminum particles that did not combust completely at ignition and caused the erosive nature. Once a higher temperature and pressure in the casing was reached, the aluminum was able to combust completely, resulting in no additional sparks. It can also be observed that AKR-P2 is less erosive than AKR-P1. It is possible that the addition of 1% red iron oxide catalyst to this mixture decreased the required reaction conditions necessary to fully combust the aluminum fuel, resulting in a less erosive propellant.

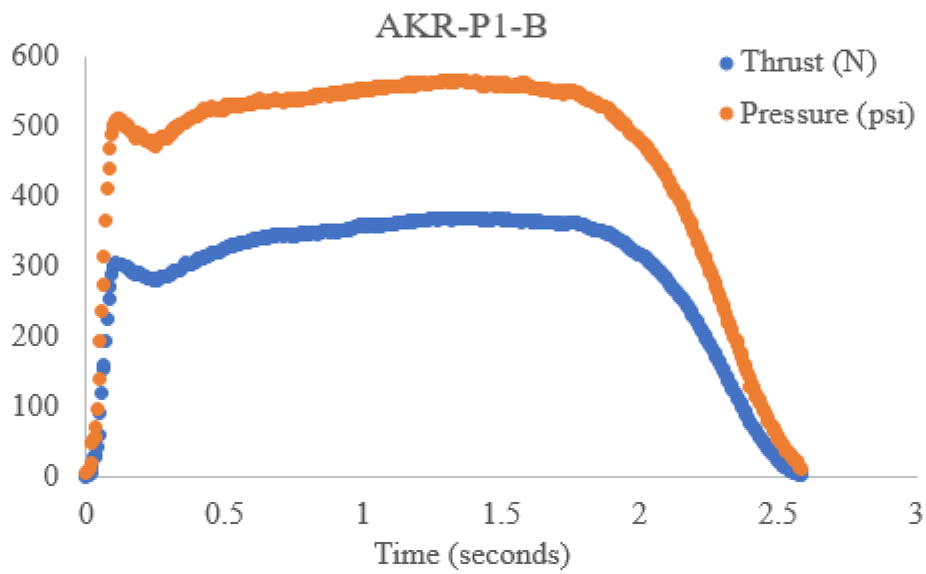

Figure 9 - Thrust and pressure graphed against time for propellant AKR-P1 motor B. This motor showed slight erosive characteristics at the beginning before leveling off into a neutral burn.

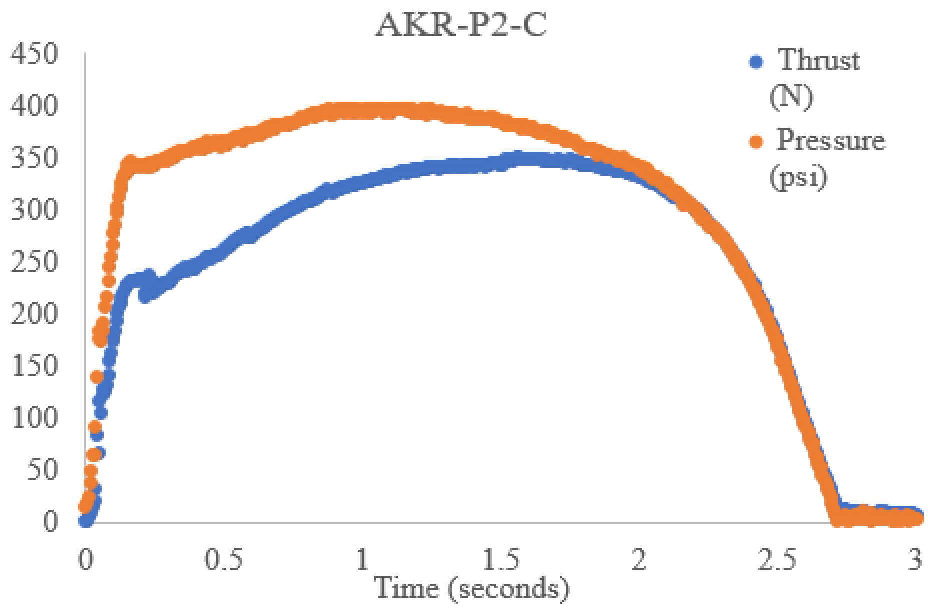

Figure 10 - Thrust and pressure graphed against time for propellant AKR-P2 motor C. This motor showed slight erosive characteristics at the beginning, transitioned to a progressive burn and regressed near the end.

The data expressed for each motor in Table 5 was either measured during testing, calculated by ProPEP, or calculated based on basic rocketry equations provided in the background section. Exporting the thrust curve data from the TCLogger program into ProPEP produced the values for Knudsen number (KN), average chamber pressure, average thrust, and burn time. This program reviews the thrust and pressure data and removes the data collected during ignition spike and burnout in order to produce better averages over the combustion interval. The total impulse, defined as the area under the thrust curve, was calculated through numerical integration. The mass flow was determined by dividing the mass of the propellant by the burn time. The delivered specific impulse was determined by dividing the total impulse by the product of the propellant mass and acceleration of gravity.

Table 5 - Performance data for each motor at varying nozzle sizes.

Motor	Nozzle Throat (in)	Exit Diameter (in)	Burn Time (s)	Total Impulse (Ns)	Average Thrust (N)	Average Chamber Pressure (psi)	Mass Flow (kg/s)	KN	Delivered Specific Impulse (s)
RCS-P-A	0.375	1.28	1.98	641.57	324.03	-	0.179	-	201
RCS-P-B	0.375	1.28	1.69	711.67	421.11	550.98	0.209	-	220
RCS-P-C	0.406	1.31	2.46	579.43	235.54	314.74	0.144	161.02	181
RCS-P-D	0.375	1.28	1.83	674.71	368.69	488.85	0.193	82.08	207
AKR-P1-A	0.375	1.28	3.01	697.21	231.63	286.53	0.118	188.7	196
AKR-P1-B	0.343	1.15	2.49	756.86	303.96	477.31	0.141	225.6	244
AKR-P1-C	0.328	1.35	2.62	621.42	237.18	487.21	0.134	246.7	219
AKR-P1-D	0.312	1.23	2.61	714.25	273.66	478.75	0.135	272.7	216
AKR-P2-A	0.328	1.21	2.9	680.27	234.58	247.66	0.122	246.71	185
AKR-P2-B	0.312	1.23	2.55	845.38	331.52	393.78	0.139	272.66	205
AKR-P2-C	0.375	1.28	2.68	758.69	283.09	330.94	0.132	188.74	167
AKR-P2-D	0.343	1.15	2.26	749.7	331.73	488.87	0.157	225.60	195

Table 6 displays the burn rate characterization parameters (burn rate coefficient and burn rate exponent), experimental average specific impulse, and predicted specific impulse from ProPEP. The coefficients determined for the RCS-P propellant are not comparable to the literature values. The dissimilarity between the literature and experimental parameters is likely due to the lack the data points recorded for the RCS-P mixture. Only two motors (C & D) displayed useful data to use to model the propellant. Fitting a power-law model to determine a pre-exponential and exponential factor to only two data points is less likely to produce reliable results. The plot of this data and associated model can be observed in Figure 12. The parameters determined from the AKR-P1 and AKR-P2 mixtures are comparable to typical burn rate coefficient and burn rate exponent values. Recalling that exponents typically fall between 0.3 and 0.6, both experimental propellants' exponents are reasonable. The burn rate coefficients are similar to the literature value for RCS-P, suggesting that the characterization was executed accurately.

Table 6 - Characterization parameters and comparison of predicted and experimental specific impulse for each propellant.

Propellant	Burn Rate Coefficient [a]	Burn Rate Exponent [n]	Error	Average Specific Impulse (s)	Predicted Specific Impulse (s)
RCS-P literature	0.0215	0.3866	-	185	176
RCS-P	0.0051	0.6719	1.30E-05	182	176
AKR-P1	0.0360	0.3005	5.03E-03	202	195
AKR-P2	0.0282	0.3564	4.22E-03	219	194

Although the literature and experimental burn rate parameters were dissimilar for the RCS-P propellant, the predicted ProPEP specific impulse and experimental specific impulse for RCS-P had a difference of only 11s as displayed in Figure 11. Additionally, the literature specific impulse was only 3 seconds greater than the experimental RCS-P mixture. The harmony between the literature specific impulse and actual specific impulse indicates that the measurement techniques were accurate. As predicted, the experimental AKR-P1 propellant produced a greater specific impulse than RCS-P at 202s. The ProPEP model's prediction was quite similar to the actual specific impulse for AKR-P1. However, ProPEP suggested that AKR-P2, with the addition of 1% red iron oxide, would produce a marginally lower specific impulse than AKR-P1. On the contrary, the red iron oxide mixture produced the most efficient motor, with a specific impulse of 219s, 20.3% larger than the RCS-P value. ProPEP predicted lower specific impulse values all propellants compared to the empirical values determined in this study. ProPEP modeled AKR-P2 most poorly, underestimating the specific impulse by 25s, suggesting that the software may not be adequate for propellants utilizing a significant burn rate modifier loading. ProPEP has been proven to under-predict I_{SP} and was not able to account for the performance increase provided by the addition of red iron oxide.

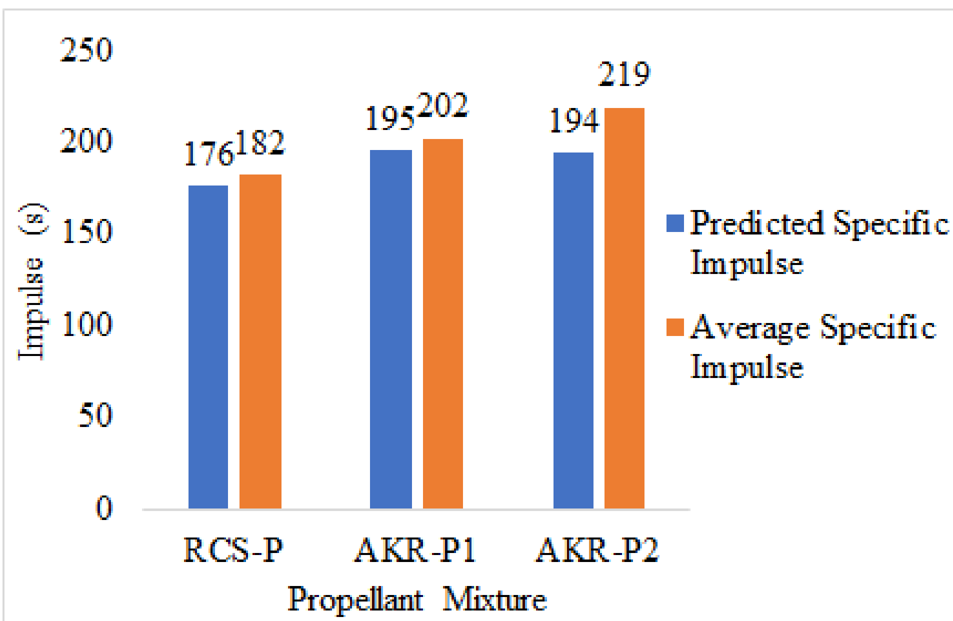

Figure 11 - Comparison of specific impulse between model predictions and experimental data

Figure 12 displays the data points collected from the two useful burn tests completed for RCS-P. As discussed previously, since only two data points were recorded, the resultant Saint Robert's Law parameters were not similar to the values determined through literature. The burn rate coefficient was much lower than a typical value for a solid rocket propellant (~.00215 for RCS-P), while the pressure exponent was slightly higher than a typical value (0.3-0.6). Additional data would be required in order to completely verify the mixing and testing methods for RCS-P propellant. Despite differences in the parameters, the motor's performance in terms of specific impulse was almost identical to literature values.

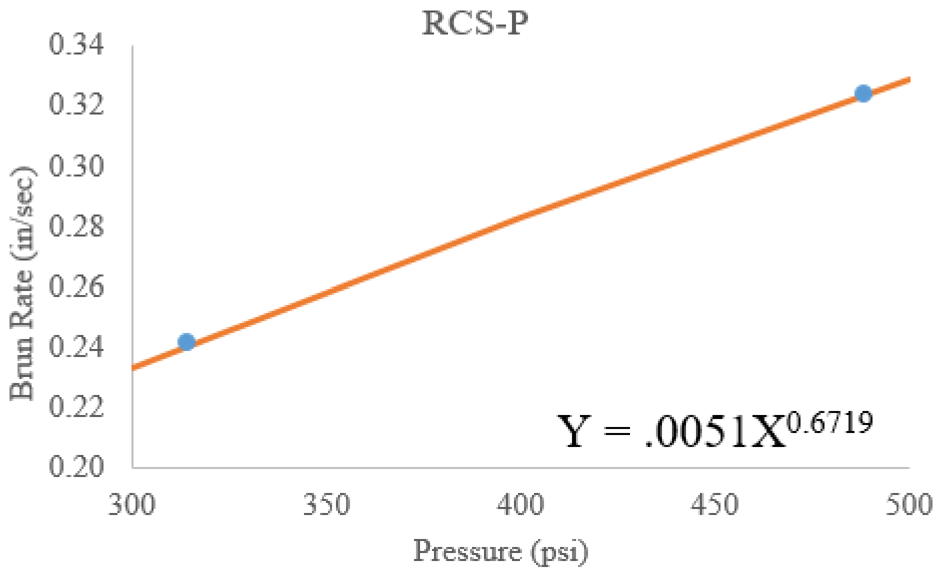

Figure 12 - Burn rate graphed against chamber pressure for propellant RCS-P. This data was fit to a power law in the form of Saint Robert's law as shown in Equation 1. The model equation is shown and graphed.

Figure 13 displays the data points collected from each of the four burn tests completed for AKR-P1. Although the values for the burn rate coefficient and pressure exponent were reasonable values based on comparable literature values, the cluster of three data points was not expected to be observed. The adjusting of the nozzle diameters for each burn test is executed to manipulate the pressure within the chamber, resulting in a faster burn rate. However, the adjustment of the nozzle diameter for three of the tests resulted in a similar pressure and burn rate. The cluster of data points is preferable compared to observing several data points with the same pressure and drastically different burn rates. Such data would result in a large error associated with the Saint Robert's Law parameters. Since the pressure did not change between the three tests, the burn rate remained similar. More drastic changes in the nozzle diameters may need to be completed in order to manipulate the chamber pressure properly.

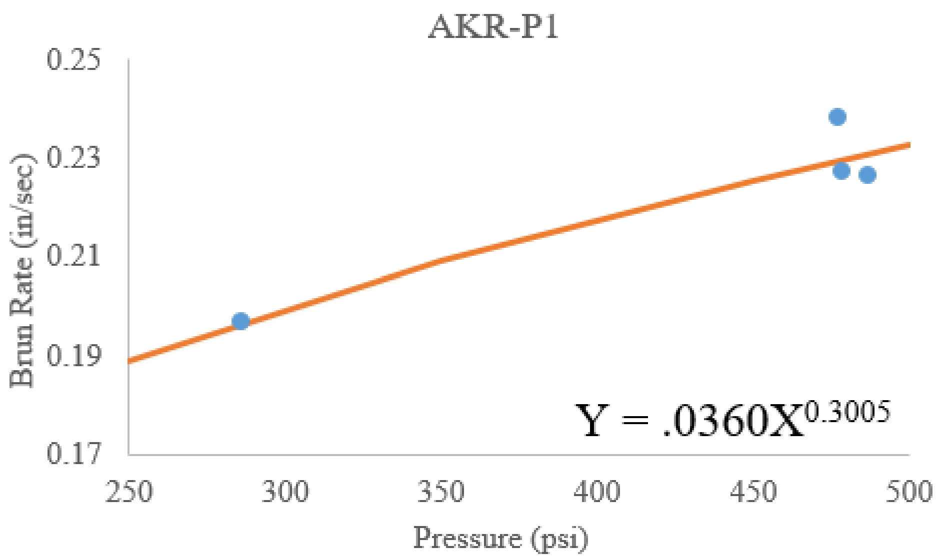

Figure 13 - Burn rate graphed against chamber pressure for propellant AKR-P1. This data was fit to a power law in the form of Saint Robert's law as shown in Equation 1. The model equation is shown and graphed.

Figure 14 displays the data collected from each of the four burn tests completed for AKR-P2. The figure represents the most desirable data set as the pressure and burn rate shifted as the nozzle diameter changed. The power law model fit the data well and produced parameters reasonable based on comparisons to typical literature values. The spread of the data suggests that AKR-P2 is a more consistent-burning propellant as the relationship between the pressure and burn rate fit the power law model while also producing data points relatively evenly along the model line.

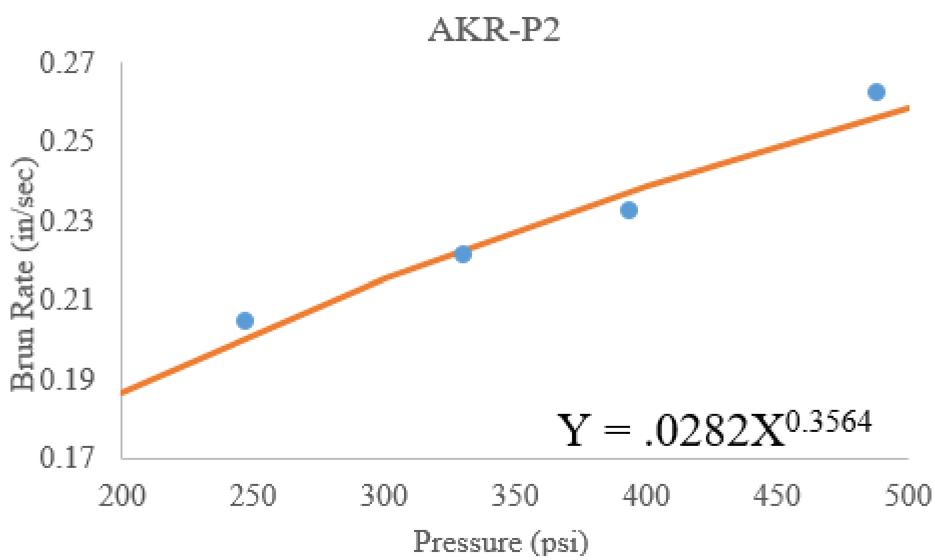

Figure 14 - Burn rate graphed against chamber pressure for propellant AKR-P2. This data was fit to a power law in the form of Saint Robert's law as shown in Equation 1. The model equation is shown and graphed.

Conclusions

The results observed in the project are consistent and comparable to established literature values. The testing methods and modes for characterization were validated by the propellants’ similarity to published data. Although RCS-P propellant mixtures could not produce similar burn rate parameters due to too few data points, the specific impulse difference was a minimal 1.82%. RCS-P thrust curves were relatively progressive in their shape, contrary to a more efficient motor. Despite failing to vary the chamber pressure as expected for three of the four tests, the AKR-P1 propellant yielded acceptable values for the burn rate parameters while displaying a consistent correlation between the burn rate and chamber pressure. AKR-P1 improved the specific impulse compared the average specific impulse of RCS-P by 11% to 202s. AKR-P1 thrust curves were desirable slightly erosive curves, likely due to the the additional unreacted aluminum fuel present at the beginning of the test burn, which was present prior to reaching sufficient pressure and temperature to produce aluminum's decomposition reaction. AKR-P2 propellant testing produced an increase in the specific impulse compared the average specific impulse of RCS-P by 20.3% to 219s. AKR-P2 yielded acceptable burn rate parameter values as well as the most desirable burn data as the pressure and burn rate shifted appropriately as the nozzle size was adjusted. AKR-P2 thrust curves were also slightly erosive as a result of the additional aluminum fuel. The red iron oxide modifier may have reacted additional aluminum content and caused the more neutral thrust curve in AKR-P2 compared to AKR-P1. Red iron oxide’s addition to the formulation resulted in dramatic increases in specific impulse as well as more consistent propellant burns as the burn rate and chamber pressure were easily manipulated by the nozzle size. Adding modifiers like red iron oxide to propellant formulations to improve specific impulse can be incredibly valuable as the loadings required for modifiers are typically at or below 1%.

Increasing the amount of aluminum fuel by nearly 34 times from 0.5g to 17.5g and 17.4g for AKR-P1 and AKR-P2, respectively, improved the specific impulse dramatically. The ammonium perchlorate was also increased to a smaller degree for the two propellants from 63.3g to 65.6g and 65.1g in AKR-P1 and AKR-P2, respectively.

ProPEP propellant modelling software modelled the specific impulse for RCS-P and AKR-P1 well, with errors of only 3.41% and 3.59%, respectively. However, the software’s ability to model a burn rate accelerator’s effect on rocket propellant can be called into question as the AKR-P2 mixture resulted in an error of 12.89%. ProPEP underestimated each propellant’s performance to varying degrees, however, the software was quite useful in order to identify a maximum specific impulse achieved through varying the amount of fuel and oxidizer.

Additional research regarding propellant formulation analysis is recommended in order to verify the data produced from this project. Performing the trials multiple times at each nozzle size for each propellant formulation would allow for statistical analysis techniques to be executed verifying that the data recorded from each test was statistically significant. The red iron oxide was added as one of the final solids to the mixer, a step which was not recommended by the project mentors. The modifier did ultimately mix in rather well, but ensuring that the modifier is added at the correct time would improve mixing quality. Greater care must be taken during propellant casting and filling of casting tubes as three grains were not casted properly. Additional testing would have been possible if all grains were prepared properly. If additional mixtures were to be tested, it is recommended that other modifiers be tested in place of red iron oxide in the AKR-P2 formulation.

Although removing strontium nitrate from a base RCS-P formulation reduces the aesthetic quality of the rocket exhaust, the motor’s efficiency improves considerably. Tremendous value is added to a propellant formulation through the mixture’s ability to produce additional thrust at a reduced weight. Reducing the weight of a rocket is advantageous as additional valuable materials can be added to the payload and rocket structure in lieu of the weight savings. Characterization of solid rocket propellants is integral to executing a proper motor scale up process. The process completed for rocket propellant mixture modelling, mixing, and testing is recommended as appropriate, accurate, and reliable for amateur rocketry.

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Appendices

Appendix A1 - Thrust and Pressure Curves

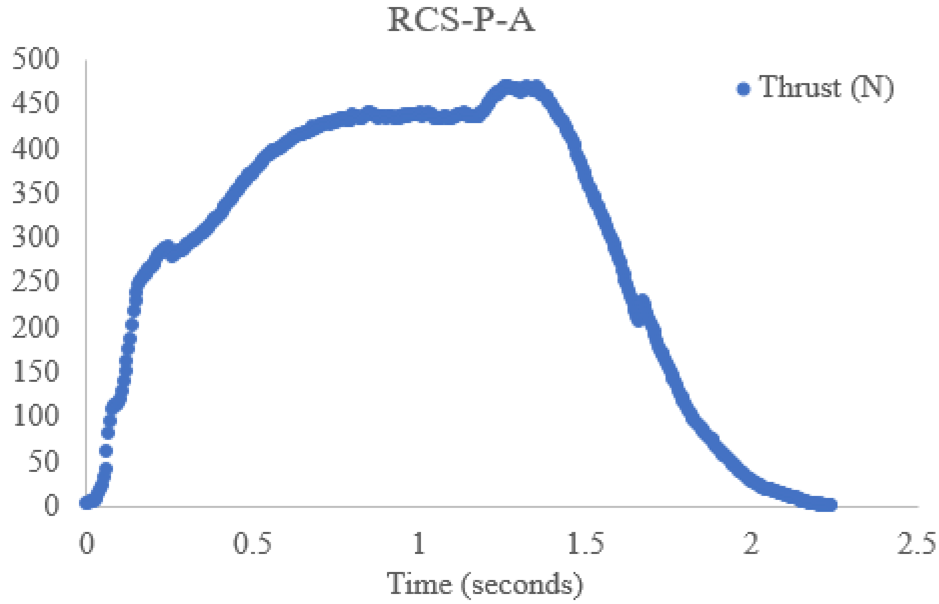

Figure A11 - Thrust graphed against time for propellant RCS-P motor A. Pressure was not recorded for this trial so it could not be used in the determination of empirical parameters.

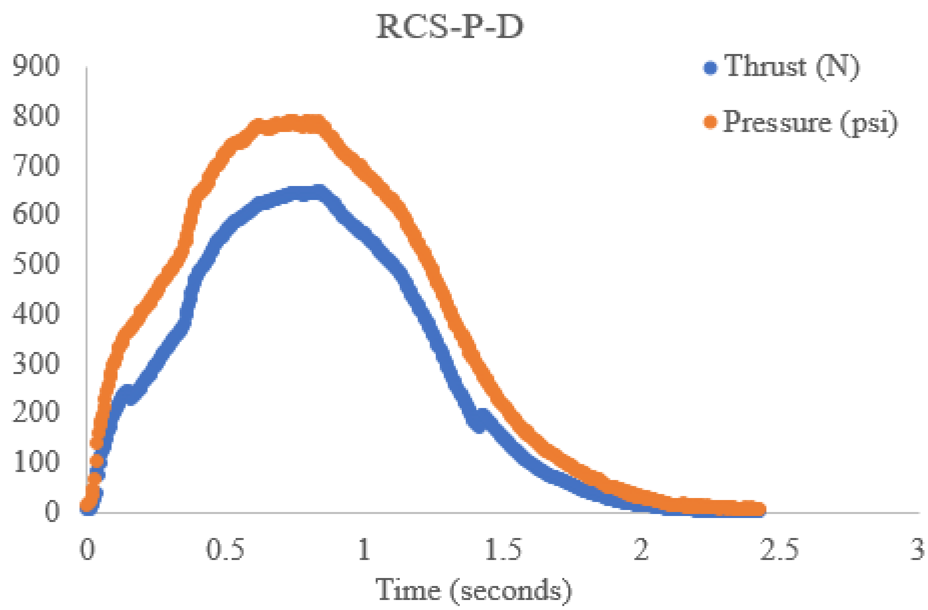

Figure A12 - Thrust and pressure graphed against time for propellant RCS-P motor D.

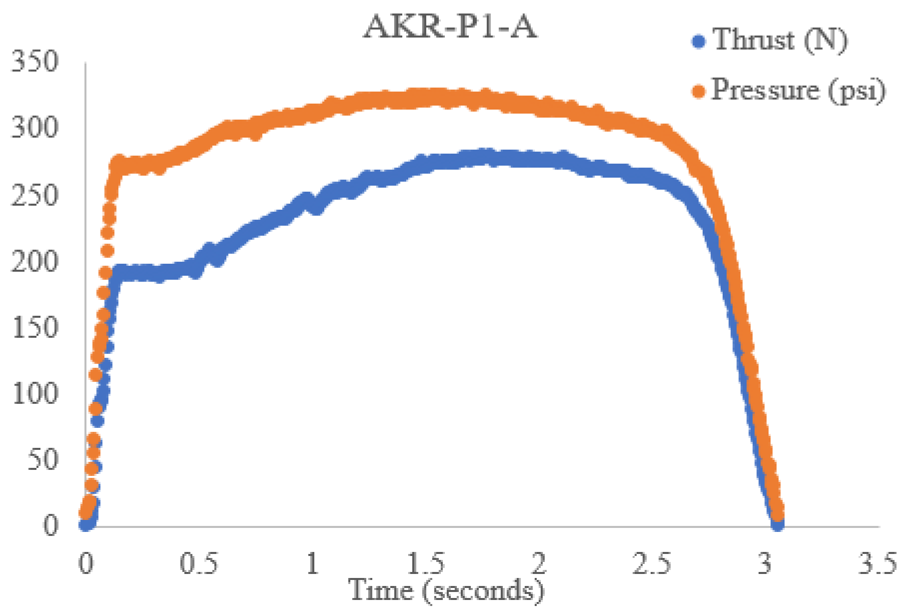

Figure A13 - Thrust and pressure graphed against time for propellant AKR-P1 motor A.

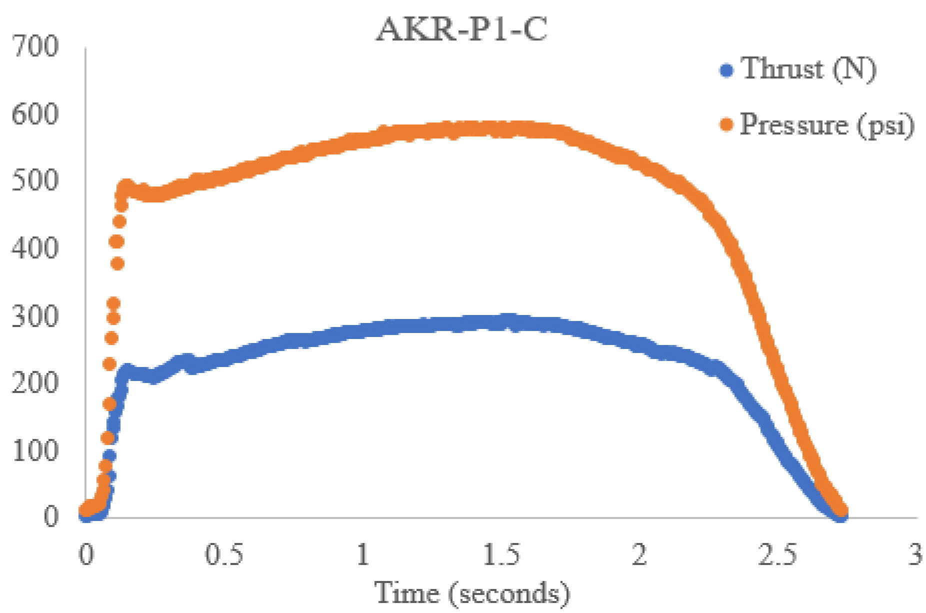

Figure A14 - Thrust and pressure graphed against time for propellant AKR-P1 motor C.

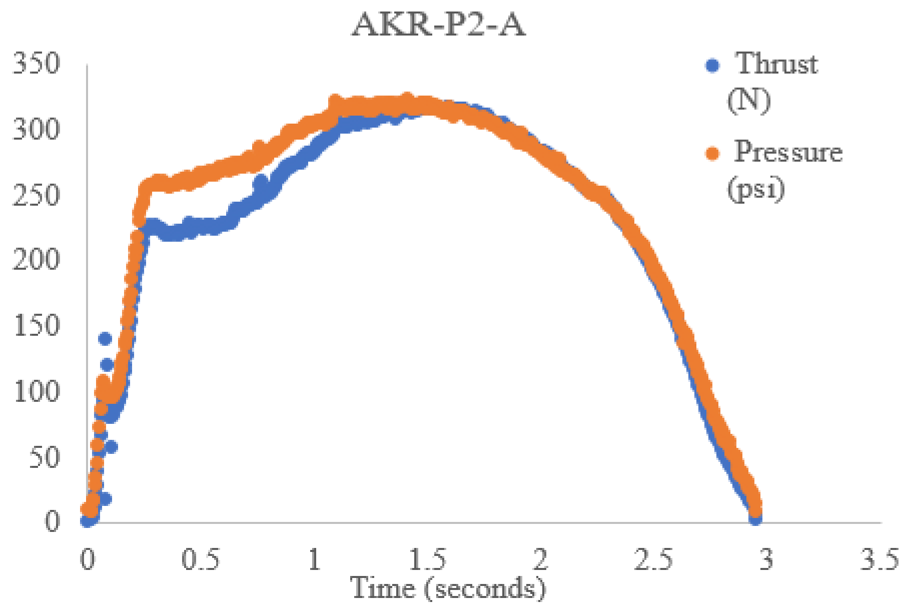

Figure A15 - Thrust and pressure graphed against time for propellant AKR-P2 motor A.

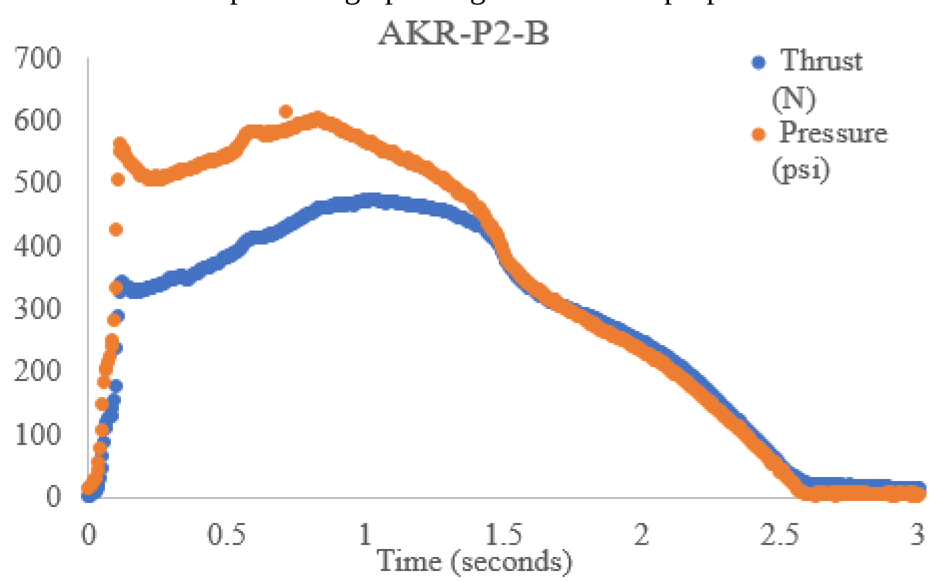

Figure A16 - Thrust and pressure graphed against time for propellant AKR-P2 motor B.

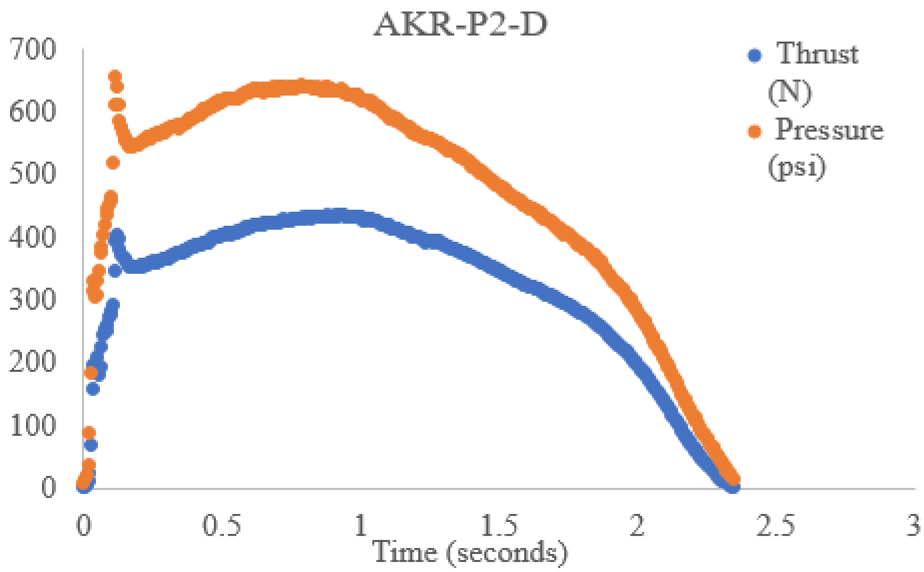

Figure A17 - Thrust and pressure graphed against time for propellant AKR-P2 motor D.

Appendix A2 - Mixing Batch Sheets

Table A21 - Total batch and ingredient mixture weights required for each propellant

Batch Mixtures (g)	RCS-P	AKR-P1	AKR-P2
Ammonium Perchlorate	1267.64	1315.39	1311.16
Aluminum	10.05	351.31	350.27
Copper Chromite	4.02	3.98	4.03
Red Iron Oxide	0	0	20
Strontium Nitrate	390.6	0	0
HTPB	242.56	242.3	241.58
IDP	30.24	30.74	30.36
Tepanol	10.76	9.98	10.46
MDI Isocyanate	47.99	47.6	44.9
Castor Oil	0	2.38	2.38

Table A22 - Mixing sheet for propellant formulation RCS-P

Date	3/5/2019				
Batch No.	RCS-P-M11	Mixer 1 Batch 1			
Propellant designation	RCS-P				
Propellant density (lb/in^3)	0.0607				
GRAINS TO BE CAST					
	Grain #1				
Diameter (in)	1.81				
Length (in)	3				
Core diameter (in)	0.625				
Number of grains	10				
Expected waste, %	1.0%				
Ingredients	Needed	Actual Used		Original batch	
Solids	This Batch (g)	(g)		Weight	Percent
Ammonium Perchlorate (200 micron)	1267.52	1267.64		63.30	63.38%
Aluminum	10.01	10.05		0.50	0.50%
Copper Chromite	4.00	4.02		0.20	0.20%
Strontium Nitrate	390.47	390.6		19.50	19.52%
	0.00				0.00%
	0.00				0.00%
	0.00				0.00%
	0.00				0.00%
Liquids					
R45 HTPB	242.49	242.56		12.11	12.12%
IDP	30.24	30.24		1.51	1.51%
Tepanol	10.01	10.76		0.50	0.50%
Isocyanate	45.25	47.99		2.26	2.26%
	0.00				0.00%
	0.00				0.00%
	0.00				0.00%
	0.00				0.00%
Volume of Grain (in^3)	6.80			99.88	
Weight per Grain (lbs)	0.413				
Total wt. (lbs)	4.13				
Total wt. (g)	1872.27				
Total wt. with waste (g)	2000.00				
Max Grams per batch	2000.00	**That will fit**			
Percent solids	83.6%				
PREPARATION STEPS	DATA AND COMMENTS				
Mix liquid ingredients					
Add metals					
Add curative					
Vacuum process					
Cure					

Table A23 - Mixing sheet for propellant formulation AKR-P1

Date	3/5/2019				
Batch No.	AKR-P1-M21	Mixer 2 Batch 1			
Propellant designation	AKR-P1				
Propellant density (lb/in^3)	0.0624				
GRAINS TO BE CAST					
	Grain #1				
Diameter (in)	1.81				
Length (in)	3				
Core diameter (in)	0.625				
Number of grains	10				
Expected waste, %	4.0%				
Ingredients	Needed	Actual Used		Original batch	
Solids	This Batch (g)	(g)		Weight	Percent
Ammonium Perchlorate	1315.47	315.39		65.74	65.74%
Aluminum	351.38	351.31		17.56	17.56%
Copper Chromite	4.00	3.98		0.20	0.20%
	0.00				0.00%
	0.00				0.00%
	0.00				0.00%
	0.00				0.00%
	0.00				0.00%
Liquids					
R45 HTPB	242.32	242.3		12.11	12.11%
IDP	30.22	30.74		1.51	1.51%
Tepanol	10.01	9.98		0.50	0.50%
Isocyanate	45.22	47.6		2.26	2.26%
Castor Oil	2.40	2.38		0.12	0.12%
	0.00				0.00%
	0.00				0.00%
	0.00				0.00%
Volume of Grain (in^3)	6.80		Sum	100.00	100.0%
Weight per Grain (lbs)	0.424				
Total wt. (lbs)	4.24				
Total wt. (g)	1924.06				
Total wt. with waste (g)	2001.03				
Max Grams per batch	2000.00	**Too much material, Try fewer grains**			
Percent solids	83.5%				

Table A24 - Mixing sheet for propellant formulation AKR-P2

Date	3/5/2019				
Batch No.	AKR-P2-M21	Mixer 3 Batch 1			
Propellant designation	AKR-P2				
Propellant density (lb/in^3)	0.0628				
GRAINS TO BE CAST					
	Grain #1				
Diameter (in)	1.81				
Length (in)	3				
Core diameter (in)	0.625				
Number of grains	10				
Expected waste, %	4.0%				
Ingredients	Needed	Actual Used		Original batch	
Solids	This Batch (g)	(g)		Weight	Percent
Ammonium Perchlorate	1311.15	1311.16		65.74	65.09%
Aluminum	350.22	350.27		17.56	17.39%
Copper Chromite	3.99	4.03		0.20	0.20%
Iron Oxide	19.94	20.00		1.00	0.99%
	0.00				0.00%
	0.00				0.00%
	0.00				0.00%
	0.00				0.00%
Liquids					
R45 HTPB	241.53	241.58		12.11	11.99%
IDP	30.12	30.36		1.51	1.50%
Tepanol	9.97	10.46		0.50	0.50%
MDI E744	45.07	44.9		2.26	2.24%
Castor Oil	2.39	2.38		0.12	0.12%
	0.00				0.00%
	0.00				0.00%
	0.00				0.00%
Volume of Grain (in^3)	6.80			101.00	100.00%
Weight per Grain (lbs)	0.427				
Total wt. (lbs)	4.27				
Total wt. (g)	1936.91				
Total wt. with waste (g)	2014.39				
Max Grams per batch	2000.00	**Too much material, Try fewer grains**			
Percent solids	83.7%				

Appendix A3 - Solid Rocket Propellant Mixing Procedure

MATERIALS NEEDED:

ITEM:	USE:
Mold Release	Helps with release of propellant
Mixer	Mixes the propellant components
Casting Tubes	Cylindrical cardboard tube for casting propellant
Casting Caps	Seals the ends of the casting tubes.
Aluminum Center Rod	core for casting tubes
Nitrile Gloves	Safety gloves required for chemicals
Safety Glasses	Safety glasses required for procedure
Tamping Rod	Used for tamping propellant into casting tubes
Acetone	Used for cleaning propellant from surfaces
Vacuum	Removes excess air from propellant
Plexiglass Cover with Vacuum attachment	Covers the bowl containing propellant

PROPELLANT MIXING PROCEDURE:

***Preliminary Notes*:**

During the creation of this propellant, proper following of the correct safety guidelines is extremely important. At all steps of the process, nitrile gloves, safety glasses, and adequate clothing covering arms and legs must be worn. It is recommended that disposable clothing be worn in case chemicals come into contact with clothing. Caution must be taken during each step of the process to ensure no chemical spills, after pouring of chemicals, the plastic top and cap of each container should be wiped clean with a paper towel and disposed into a proper disposal. Extra remnants of propellant and or towels with propellant should be burned at a safe location away from any flammable sources and away from any structures. This is generally the safest way to dispose of extra propellant instead of throwing it into waste with other potentially flammable items.

This document contains the specific steps for mixing the propellant, as well as any possible safety hazards during each step of the process. This procedure is designed to produce a 2000 gram batch of RCS Purple Propellant. Adjust the amount of each chemical used based on the batch size and desired propellant.

Procedure:

*Safety - When using the mixer, make sure the lowest speed is always used in order to prevent splash and the creation of any dust

1. Mixing of the liquid components (excluding the curative).

Measure out 242.56g (12.11%) of HTPB directly into mixing bowl. Add 30.24g (1.51%) of IDP, and 10.76g (0.5%) of tepanol. Mix together liquids for 5 minutes on lowest or stir setting.

2. Addition of solid components

*Safety – When dealing with any powdered metals, it is very important to avoid the creation of any dust in the air. Creation of this dust is hazardous to human health, so along with careful procedure, proper ventilation must always be used. Some of the chemicals in this procedure are extremely flammable, therefore the work space must not be near any open flames or possible ignition sources.

Weigh out 10.05g (0.5%) of aluminum powder and carefully add it into the mixing bowl already containing the liquid components. Be very sure not to fluidize any of the dust into the air. Using a disposable spoon, carefully mix in the aluminum powder until all powder is covered in a liquid component. Mix in the bowl on lowest setting for 5 minutes.

Add 4.02g (0.2%) of copper chromite to the mixing bowl following a similar procedure as the aluminum and stir for 5 minutes.

Measure out 390.6g of strontium nitrate (19.5%) and carefully add to the mixing bowl. This chemical is less hazardous than the metals and the particle size is larger so wetting the powder is not necessary. Care should still be taken not to fluidize this powdered material. Mix in the bowl for 5 minutes.

Measure 1267.64g (63.4%) ammonium perchlorate. Add this material 1/3 at a time ~422g and mix for 5 minutes after each addition. Once all components except for the isocyanate curative have been added to the mixture, mix for 60 minutes, scraping the bowl after 45 minutes to ensure a totally homogenous mixture.

3. Preparing to cast propellant

While the propellant mixes, make sure the casting tubes are covered in masking tape if using cardboard or some derivative. Generously apply mold release to the aluminum castings rods, and casting caps. Mix HTPB (18g) and MDI Isocyanate Curative (4g) together. Apply this paste to the insides of the casting tube, making sure to cover all crevices and surface. Mix additional coating at this same ratio if necessary based on the number of castings.

4. Adding curative

Make sure the casting tubes, aluminum rods, and bottom caps are prepared before starting this step, when they are set up you may begin casting.

Once the curative is added, the propellant will begin to cure. If material is not cast into grains fast enough it will solidify into the bowl and will become unworkable. After mixing is completed, measure and add 47.99g Modified Isocyanate Curative (2.26%) to the mixer. Mix for 15 minutes, scraping the bowl after 10 minutes. Vacuum dry the mixture for 5 minutes, making sure to shake the bowl every now and then to release any air bubbles in the propellant.

5. Casting grains

Using hands, roll dough like propellant into cylinders and coil into casting tubes that are already on the core rods. Have a partner tamping the fuel down into the casting tubes as more propellant rolls are formed. Ensure that propellant is sufficiently packed into the casting tube and no gaps exist in the grain. Once the tubes are full and roughly level at the top, add on the top casting caps, and press down on them to extrude excess fuel. After the caps are secured, let the propellant cure for at least 48 hours.

6. Clean-up

Clean up the workstation including bowls, surfaces, and any used utensils. Acetone should be used to ensure adequate cleaning, make sure anything used for the propellant creation is kept separate from any other used lab equipment. All propellant scraps should be accumulated in a box to be safely burned later and not mixed with standard trash.

Appendix A4 - Graphical comparison of propellants at each nozzle size

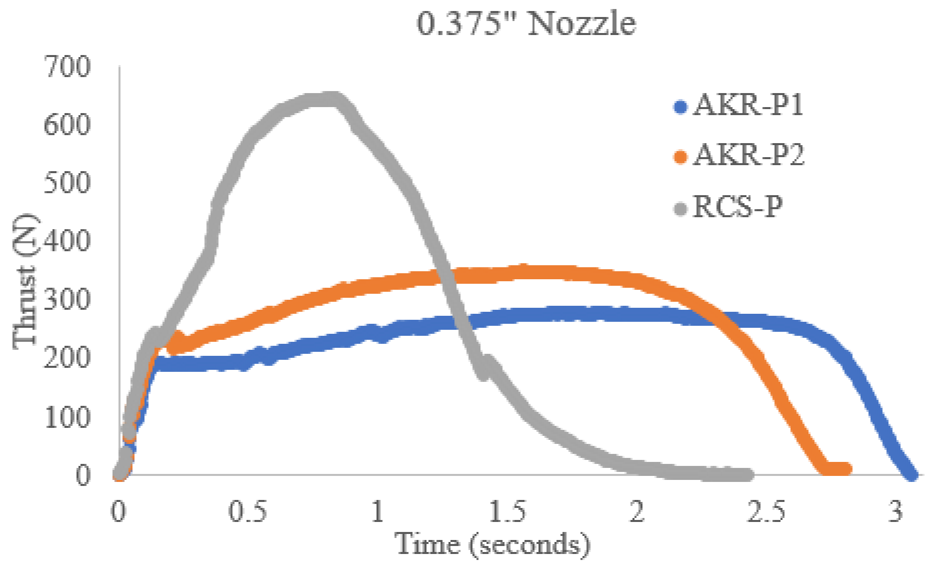

Figure A41 - thrust as a function of time for each propellant utilizing a nozzle with a 0.375 inch diameter throat.

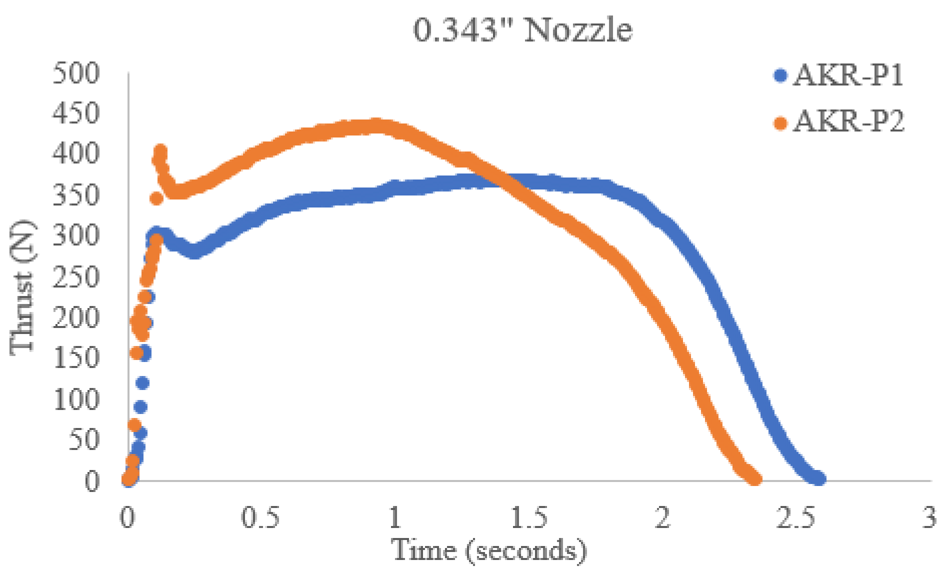

Figure A42 - thrust as a function of time for each propellant utilizing a nozzle with a 0.343 inch diameter throat.

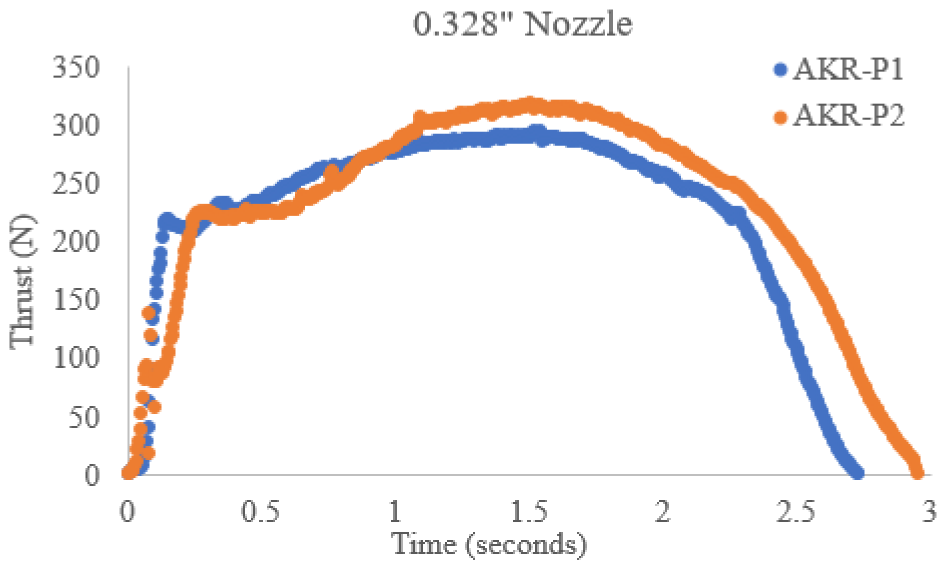

Figure A43 - thrust as a function of time for each propellant utilizing a nozzle with a 0.328 inch diameter throat.

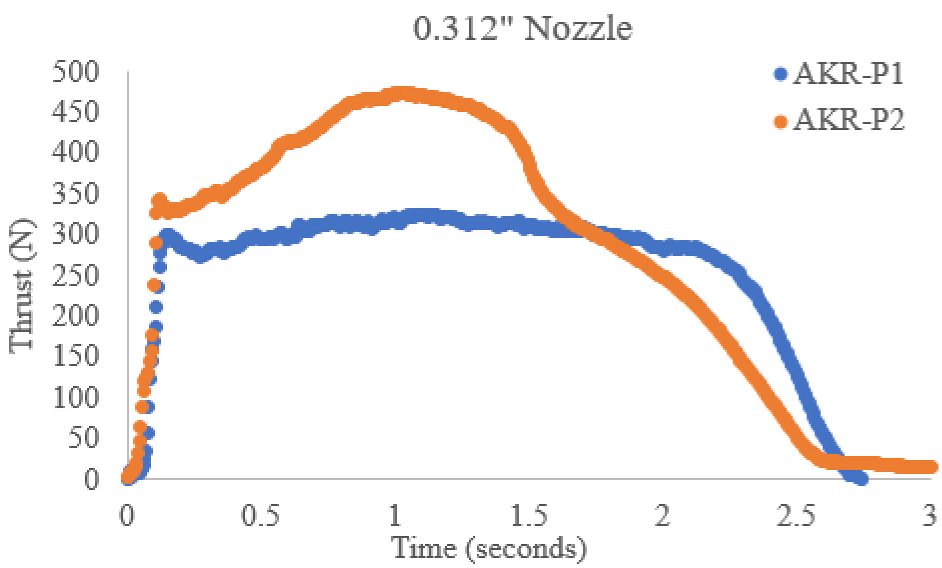

Figure A44 - thrust as a function of time for each propellant utilizing a nozzle with a 0.312 inch diameter throat.