5-2008

Restart Capabilities of Hybrid Rocket Motor Utilizing Gaseous Propane and Oxygen Injection System

Joseph Gracy
University of Arkansas, Fayetteville

Recommended Citation

Gracy, Joseph, "Restart Capabilities of Hybrid Rocket Motor Utilizing Gaseous Propane and Oxygen Injection System" (2008). *Mechanical Engineering Undergraduate Honors Theses*. 10.
http://scholarworks.uark.edu/meeguht/10

Restart Capabilities of Hybrid Rocket Motor Utilizing Gaseous Propane and Oxygen Injection System

A thesis submitted in partial fulfillment
of the requirements of the Honors Program, for the degree of
Bachelor of Science in Mechanical Engineering
and as requested of the
Arkansas State Undergraduate Research Fellowship Grant

by

Joseph D. Gracy, Mechanical Engineering

Thesis Advisor- Dr. Larry Roe

May 2008
University of Arkansas

Acknowledgements

Special thanks to the following individuals for all of their dedication and support in assisting me with testing and fabrication, and keeping me full of pizza and laughs after long nights in the lab.

Robert Gracy
Aaron Schoolcraft
Roger Schoolcraft
P. Blair Shelton
Steven Reynolds

Table of Contents

Abstract

Hybrid rockets have become an increasingly popular application in professional and amateur rocketry for their outstanding performance and reliability. An issue pressing the marketability and functionality of these rockets is the ability to restart with an exclusive system after primary ignition. Research and development of a system that can be used reliably in either application to achieve restart under various conditions has been made recently using dual injection of GOX and C_3H_8 using a 200kV ignition system while implementing a polymethylmethacrylate (PMMA) formable polymer as primary fuel. The system used a manual valve arrangement for control. The design features most of the necessary performance adjustment components and both additional and intrinsic safety mechanisms. Analysis of test data indicate improvement in ignition lag by increasing operating pressure, minimizing plumbing system, and increasing electric igniter durations.

List of Tables and Figures

Introduction

Imagine what it would be like to start a car in the morning on the way to work and not be able to slow down, much less stop, until the gas tank went empty yet attained excellent gas mileage. This scenario would be analogous to having a solid rocket motor replace the engine in a car: no throttle, no method of arresting the reaction, but superb performance. Obviously this situation is not ideal for use on the road. What if it were possible to store two tanks, one of gasoline and one of air, and have the advantages of being able to stop and even to throttle the mechanism but with only average gas mileage. This particular setup would be like having a liquid propellant rocket system and would allow control, however it would require a very precise and sophisticated plumbing network. With no ado, imagine combining the advantages of both of these systems. Gas mileage would be well above average, the throttle could be controlled, and at any time the car could be stopped with plenty of fuel in the tank. The aforementioned setup almost completely describes what would be equivalent to a hybrid rocket. The advantages are clear, but attaining these advantages proposes a challenge. The challenge that is of primary interest is the ability to restart the rocket after shutdown.

Although the primary topic examined herein is designated as the restart capabilities of hybrid rockets, it is crucial to understand the components of the entire system and how they complement each other. When analyzing these components close examination of similar components used in other rocket systems, such as liquid bipropellant and solid rocket systems, must be made to distinguish among the systems. By analyzing these systems and their components it will become clearer how applications in various military and commercial industries demand unique and economically feasible solutions for various mission requirements.

Objective

The objective of the research herein is to display the capabilities of a hybrid rocket motor to restart multiple times with a unique system. Although inspired from theoretical and industrial applications, the system component layout in and of itself will be completely unique. Reliability analysis and necessary operating conditions will be determined from experimental observations and ideal theoretical processes. Once these analyses are made, performance characteristics of similar models will be made.

Research Designs Disclaimer

One must first consider the nature of the design process before discussing current and proposed systems that attain the common objective of effectively and reliably initiating and terminating hybrid rocket burn sequences. As with any design it can be argued that any particular design may be far superior at meeting a certain design goal for one application while failing considerably to meet requirements a separate mission may require. Therefore, it is impossible to claim any of the following designs as an either superior or inferior design since a mission has not been specified. Instead, the overall functionality and recommendation of scenarios of which each system may or may not be inclined to succeed will be determined for each system based on industrial standards and general practice.

Research Designs

Rocket motors are usually segregated into three categories: liquid monopropellant or bipropellant, solid propellant, and hybrid rocket motors. Liquid monopropellant rockets utilize a fuel which spontaneously decomposes under certain conditions which are created in the combustion chamber. Liquid bipropellant rockets mix two reactants necessary for combustion precisely when the reaction is initiated, thus storing them separately until thrust is desired. Since the two reactants are stored in either gaseous or liquid form, the mixing and regulation network can be quite complicated requiring added weight of structure and plumbing components. Solid rockets mix the two components in a precisely mixed grain that performs well but is incapable of arresting (Brown 2002). This leads to the design of the hybrid rocket, a system that employs a solid fuel-only grain with a single plumbing system to inject the oxidizer when desired. Separating the fuel and oxidizer provides much safer handling while only having to regulate the flow of the oxidizer. With a separate fuel and oxidizer, the system can be initiated, regulated, and arrested at any time.

Incorporating both a solid fuel and either liquid or gaseous oxidizer allows for the wide range of flexibility for single and multiple ignition systems- the simplest of these systems of course being the single ignition system. Although these systems are proven and reliable and can easily launch a rocket, they do not allow for restart capabilities. For example, one such hybrid rocket igniter system involves a portion of solid rocket propellant mixture on the primary fuel that is ignited with an electrical igniter. The electrical igniter initiates the burning of the solid propellant pre-heater grain that preheats the gaseous oxidizer to initiate the reaction of the primary fuel and oxidizer ("Hybrid" 2007) as shown in Figure 1 on the next page. The sequence is a waterfall reaction that exhausts the supply of the pre-heater grain. Since amateur rocketry usually demands the oxidizer and fuel supplies be burned until completion the system fits the mission requirements.

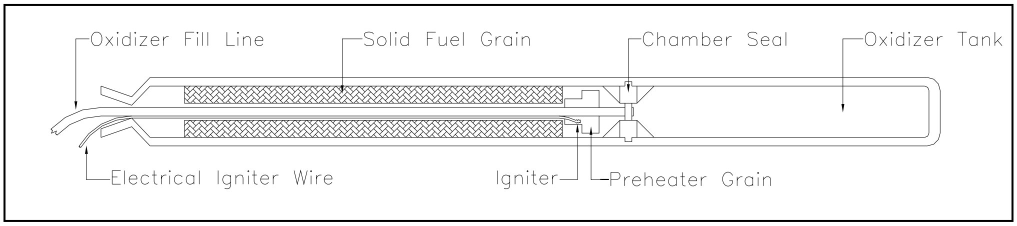


Figure 1, Single Start Solid Propellant

The single start solid propellant system utilizes axial symmetry of main components to maintain predictable flight conditions. The oxidizer tank is charged from the fill line traveling through the center of the annular solid fuel grain. Igniter wire makes contact in the pre-heater grain and mates with the oxidizer fuel line to exit out the nozzle as shown in Figure 1. Note that the structural integrity of the side walls is maintained by invading only from the nozzle opening. Uniformity of the chambers allows for lighter materials to be used. If fill and electrical lines were to invade from the side of the chamber, the corresponding stress concentration would pose a greater threat of mechanical failure. Thus, the single start solid propellant design utilizes lightweight materials while satisfying mechanical rigidity and a mission specific requirement of exhausting all the oxidizer for maximum altitude.

When it comes to systems that have the capability of reliably starting and restarting a hybrid rocket the injection of a hypergolic fluid is the most common industrial solution. A hypergolic fluid is one which spontaneously ignites when combined with an appropriate oxidizer under a given set of conditions. Being such, one can easily reason that the spontaneous nature of the chemical reaction would make hypergolic restart systems very reliable when correctly engineered.

The hypergolic system usually requires an intricate network of plumbing to ignite the primary fuel with the oxidizer. A minimum of two tanks must be present- one containing the hypergolic starting fluid and the other an oxidizer. Since unforced flows must always pass from a region of high pressure to one of low pressure, the two tanks must be kept at uniform pressure to allow proportional flow. One solution could be a set of carefully monitored pumps controlled by a unique electrical system. Having only the additional mass of the pumps and a feedback electrical system incorporating two pressure transducers, this system negotiates the pressure differential by a fairly complicated method. In addition, a greater number of critical components poses a greater risk of failure of one component and, therefore, the entire system. A more common design incorporates a single pressurized inert gas tank in addition to the oxidizer and hypergolic fluid tanks as suggested by Bradford et al. (1996) as shown in Figure 2 below. A network system of a set of parallel lines from the gas tank to the oxidizer and hypergolic fuel tanks distributes a common pressure to the tanks. According to Campbell (1964) the oxidizer and hypergolic fluid tanks must also contain each respective fluid in a sealed bellows to prevent mixing with the inert gas while adapting to compensate for diminishing reservoir quantities from multiple starts and varying burn times.

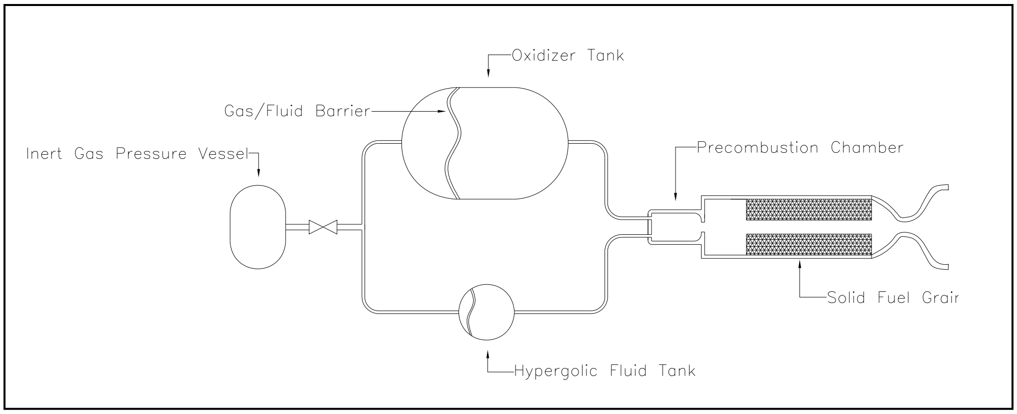


Figure 2, Hypergolic Ignition System Utilizing Single Pressure Source

Once the high pressure mixture of hypergolic fluid and oxidizer is injected into the pre-combustion chamber, the droplets impinge on each other, further reducing the particle sizes until atomized particles begin to react. As localized reactions occur, the exothermic reaction releases energy to vaporize surrounding reactants, thus furthering the intensity of the reactions as described by Sutton et al. (2001). The flame front propagates down the solid grain, liquefying and then vaporizing the solid grain structure. An ultra lean concentration ($\Phi<1$ as described in Appendix C) of oxidizer and hypergolic fuel allows for the initial reaction to proceed to a greater completion with excess oxidizer to react with the vaporizing solid grain. Eventually, the hypergolic fluid injection can be terminated leaving the oxidizer and primary fuel in a self-sustained reaction as is desired.

Though hypergolic fluids provide a reliable method of restarting hybrid rocket motors, they pose special concerns for the system designers. The foremost concern is the nature of the hypergolic fluid itself. Hypergolic fluids spontaneously react with oxygen in an exothermic reaction that produces large quantities of heat and, in a contained vessel, pressure. No spark is necessary to initiate the reaction, only contact at moderate pressures. Therefore, isolation and maintenance of containers of hypergolic fluids must be well kept. In addition to the storage tanks, it is also necessary to ensure no leaks are present in the plumbing system. Any leak would create a potentially hazardous situation. And finally, when the hypergolic fluid is first pressurized in the plumbing system it must also be free from any oxygen. This means that an inert gas such as nitrogen or argon must be used to charge the system and clear it of all the oxygen. All these necessary steps usually deter the average rocket designer from hypergolic fluids and find themselves pursuing a similar, yet safer and more simplistic option.

The safer restart system design similar to the hypergolic system incorporates a gaseous non-hypergolic starting fuel with either a gaseous or liquid oxidizer to initiate the primary fuel and oxidizer reaction. Since the starting fuel is non-hypergolic a spark must be provided to start the reaction with the oxidizer. The integration of a glow plug or a spark plug with an external power device with intruding electrodes usually satisfies this requirement as shown in Figure 3 below.

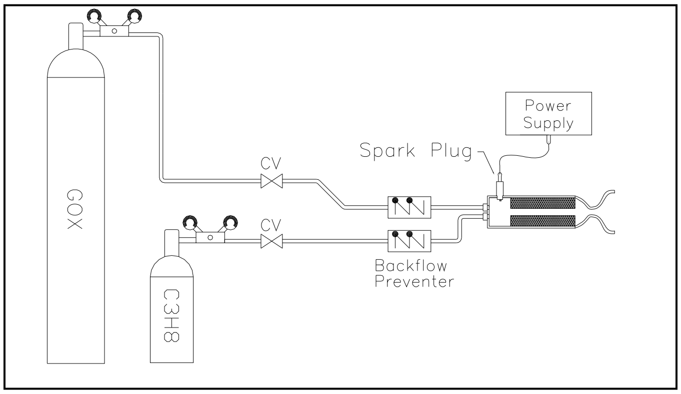


Figure 3, Gaseous Hydrocarbon Injection with Spark Igniter

Plumbing of the starting fuel and oxidizer is more easily separated due to the different storage pressures of the reactants. The storage tanks pressure of liquid oxygen is approximately 2200 psi and the saturation pressure of propane is 124.6 psia (Çengel et al. 2005) at 70 °F. With a nearly twenty fold difference in the pressures, it would be highly impractical to pressurize propane to that of oxygen. The pressure vessel would be of an equal multiple less volume but would require walls of greater thickness to maintain a comparable factor of safety. Also, the risks involved with pressurizing a gas are only increased as pressures increase. Therefore, separate tanks with respective pressures and certifications most properly suit the application.

As with the hypergolic system the entering gases must have equal pressures to allow consistent flow from each tank and prevent backflow through any of the plumbing.

Regulators specifically designed for each gas must be used and throttled down to the design inlet pressure. The regulated pressures of the incoming starter fuel and oxidizer become further topics of discussion later on as reliability and ignition delay are considered.

Proposed Design and General Overview

The restart system designed and tested utilizes a gaseous propane and oxygen injection with an external igniter power supply. The hybrid rocket motor (Appendix E Figure 5, Item #7) consisted of a primary fuel as polymethyl methacrylate (PMMA) and was formed into an annular grain in a galvanized steel 1 ¼” schedule 40 pipe size 10” nipple. A nozzle was formed utilizing a 1 ¼”-3/4” reducer with a ¾”-3/8” bushing. The fuel grain of length 6 inches allowed for both 2 inch pre-combustion and post-combustion chambers.

Male quick-disconnect ¼” NPT hose plugs (#5) were tapped into the center of the end cap and side wall of the chamber. The end cap was then connected with the gaseous oxygen hose with female quick-disconnect hose sockets and likewise the side plug was connected to the propane line. Continuing up the hoses 18 inches next were one-way flow valves (#4) placed in each line. Approximately 3 feet later were ¼” globe valves for the oxygen (#1) and the propane (#2). Remaining hose sections on the order of 20 feet allowed for regulator connections at the tanks at a safe distance (not shown in Figure 5 in order to maintain resolution). The oxygen regulator reduced the pressure the average tank pressure of 2000 psig to a range of 0-150 psig. Similarly, the propane regulator reduced the average tank pressure of 120 psi to a manufacture suggested safe range of 0-15 psig.

The electrical ignition system consisted of a Sabre 200 kV stun gun for the necessary capacitor and inductor components. The circuit components inside the manufacturer’s housing were decoupled and placed in a traditional outdoor socket box. Oversized toggle and rocker switches were mounted on the housing and all electrical connections were soldered and epoxy insulated for strong connections and safe handling. The electrical leads were made from 14 AWG and utilized spade male and female terminals for rocket motor housing connections. All rocket motor cartridges were fitted with corresponding female and male spade terminal connection with an six inch section of 14 AWG wire connectors led to a round connector position at the head of a #6 machine screw and locked with a nut. Oversize holes of 3/8” were drilled along the cylinder circumference approximately 60 degrees from each other in the pre-combustion chamber forward of the grain approximately ½” inch and aft of the propane injection plug. One 7/16-3/16 rubber grommet was held in place by epoxy approximately ½” from each machine screw head. The grommets insulated the metal electrodes from the rocket body. Further electrical and thermal insulation was achieved from high temperature furnace cement coned both inside and outside the motor housing. (Note: Electrical insulation was only achieved from the electrodes after cement was allowed to cure to completion as the electrolytic curing agent conducted electricity. Further electrical insulation was achieved by coating all exposed electrical wiring joints with epoxy.)

System Theory

The theory of the gaseous injection hybrid rocket restart system could be described in intricate detail for the most basic process. However, the following discussion will assume a basic knowledge of general mechanics and emphasize the unique physical process involving fluid dynamics of hybrid rocket combustion.

A hybrid rocket consists of solid fuel with a separate arrangement of gaseous or liquid oxidizer. Unless the oxidizer and fuel combination react at relatively low temperatures and pressures another energy source must be introduced to initiate the reaction. A separate fuel allowed to react with the already present oxidizer poses the most condensed solution. Typical hydrocarbons reactions have been used in many applications, such as gasoline for automobiles. With their plentiful supply, ease of transportation (non-cryogenic and low storage pressures), and relatively low refining and manufacturing costs, hydrocarbons also present themselves useful in rocketry.

Continuing the theme of safety and economic feasibility, a popular gaseous hydrocarbon commonly used for heating, small vehicle fuel, and the occasional recreational cookout demonstrated yet another application. Propane is a simple hydrocarbon consisting of three carbons and eight oxygen atoms singly bonded. As a hydrocarbon propane follows the typical exothermic reaction with oxygen to form carbon dioxide and water.

$$C_3H_8 + 5O_2 \rightarrow 3CO_2 + 4H_2O + Heat \qquad \text{(Eq 1)}$$

Many thermodynamic processes can be applied to the heat of the reaction of propane with oxygen. Common applications include the heating of air to increase its temperature (and pressure if contained at constant volume) to do work as described in the

Otto cycle. For the application of rocketry, the heat itself is the primary interest. In order for the reaction of the primary fuel (PMMA) with the oxidizer to take place the components must be able to interact aggressively at the molecular level. The simplest way to achieve this is to vaporize the fuel (PMMA) and introduce it to the already gaseous oxygen. Since the fuel begins in the solid state it must be heated through two phase changes. The heating of the fuel further increases the kinetic energy of the vaporized particles. Increased kinetic energy means higher particle velocities. When particles collide at a high enough velocity they have enough energy to react with each other and do not simply rebound. Reacting molecules continually heat the surrounding molecules until local thermodynamic equilibrium is reached. Heat radiates away, and the expanding gas mixture exits the chamber at a high velocity. Through conservation of momentum one can calculate the thrust generated by the escaping gas. However, the thrust is not of particular interest since it is an afterthought of the restart system and provides little means of analyzing the restart system itself.

Flow of both propane and oxygen into the pre-combustion chamber must also be precisely regulated to meet the combustion limits of propane. First, the combustion limits of propane with oxygen must be specified. The generally specified combustion limits of propane with air lie between 2.15%-9.60%. However, the oxidizer is injected as oxygen and not air as used in more common applications. A translation using molar masses of air and oxygen 8.61%-31.30% combustion limits of propane and oxygen. A sample calculation of the combustion limit translations can be found in Appendix D Calculations.

Flow rates of the aforementioned reactants determine whether or not the combustion limits may be reached. Initial injection of the oxidizer followed by propane injection discussed later on in allows for combustions limits to be entered from the lower limit. The Bernoulli equation allows for predictions of flow rates by making several generally assumptions. Assumption include that the flow is adiabatic and occurs in a frictionless, constant area duct that has no discharge losses. Also, calculations assume pressure differences are from immediately post-regulator to ambient conditions with static conditions at the inlet. Oxygen mass flow can be calculated to be 0.0174 kg/s from the 10 psig operating pressure with a 1/4 inch hose diameter. A globe valve regulates the propane effective flow diameter to within combustible limits. For the lower combustion limit a diameter of 0.0708 inches a flow rate of 0.00164 kg/s meets the oxygen combustion limit of 8.61%. For the upper combustion limit a diameter of 0.156 inches a flow rate of 0.0079 kg/s meets the oxygen combustion limits of 31.30% as calculated in Appendix D Calculations.

The actual functionality and most of the safety of the system in based on the sequence and timing of the electrical igniter and oxygen and propane valve positions. The usage of an explosive gas such as propane requires careful consideration into relative concentrations and accumulation in confined volumes. Thus, minimizing the concentration of propane in any volume was the primary safety objective. In addition to maintaining the minimum concentration of propane as possible with oxygen for safety purposes, the lean mixture would then allow for the excess oxygen to react with the vaporized fuel (see Appendix C). Achieving these objectives was fairly simple when controlled by the globe valves and maintaining open communication among valve and ignition operators.

Ignition sequence phases also met a secondary objective of minimizing there restart sequence times. A minimum restart sequence time directly relates to a minimization of starter fuel and oxidizer waste. The ignition sequence begins by a “flush” of the oxidizer . The “flush” serves to minimize the concentrations of any vaporized fuel (vapors accumulating from static primary fuel) and to ignite primary fuel if there are any remaining embers from a previous ignition. If the primary fuel begins burning at this point, all other ignition mechanisms are not required and the burn time may be continued until termination is desired. When termination is desired, the oxygen valve must be completely close to arrest any oxidizer supply to the fuel.

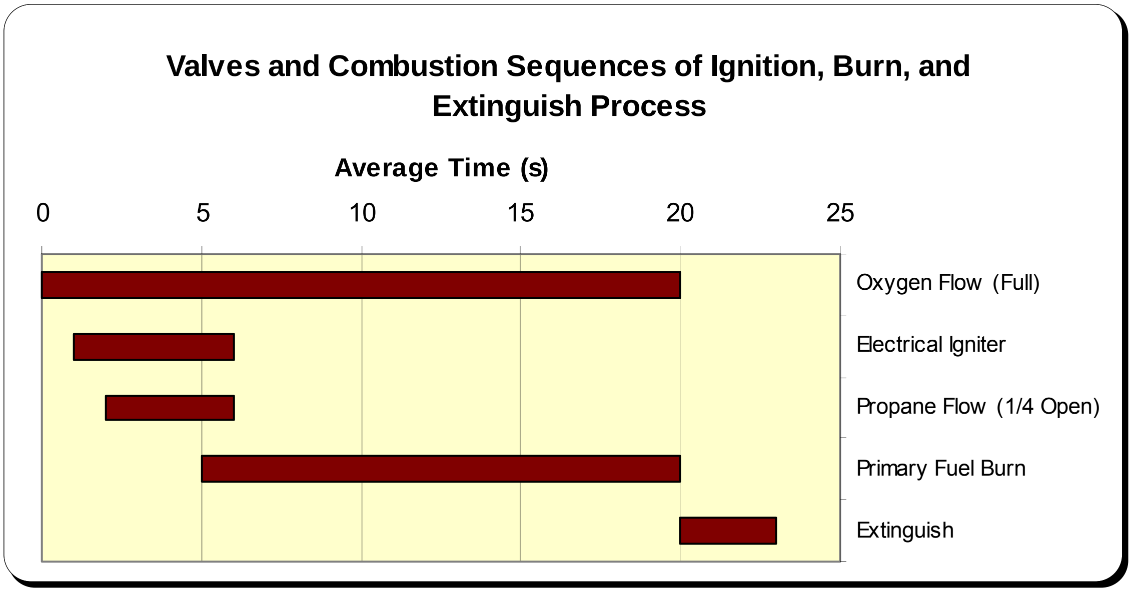


Figure 4, Valves and Combustion Sequence of Ignition, Burn and Extinguish Process

Assuming the primary fuel is cold from either being fresh fuel or from a long enough down time from a previous burn, the next step is to commence electrical igniter pulses in the pre-combustion chamber. Again, if the primary fuel ignites any point before introduction of propane, the remaining start actions may be dismissed until termination is desired. Introduction of a small stream of propane is next to produce the flammable mixture that will soon be ignited by the spark. Figure 4 shows a typical restart sequence as described herein. Times in Figure 4 also correspond to mean phase averages. Delay and improvements in this times are addressed later in the Sequence Averages and Data Analysis section. Once the primary fuel begins to burn the propane valve should be completely closed and electrical ignition should terminate. A slight hangover in either propane or electrical ignition process is unproductive and only wastes valuable starter fuel and electrical energy that are minimized for weight constrictions.

The chamber experiences a side-wall pressure that must first be considered. Motor housings were constructed of 1 ¼" galvanized steel pipe . However, the application of the materials was modified to include high temperature, low pressure gases. Entering pressures from the propane and oxygen line were set to be 10 psig. Since flow occurs from volumes of high pressure to those of a lower pressure, one can infer that the chamber pressure was maintained lower than 10 psig during burn operations since flow was indeed observed. However, primary ignition causes a drastic increase and then decrease in chamber pressure than can be idealized as a singularity. Calculations (Appendix D) reveal that a 116.6 fold increase in the pressure would result in factor of safety of 5, far below was it conceived in the ignition reaction.

System Components Costs and Assembly

The main objective of the test model was to demonstrate the abilities of a system constructed entirely of generally accessible parts and tools. Vendors for a great portion of the assembly included AirGas, Lowe's, and McMaster-Carr. Obtaining the parts from such vendors allowed for low-cost parts that were in high production for various other applications. The system may be created by purchasing the same or similar parts as detailed in Appendix F Table 4, Price List of Rocket and Fabrication Parts and Tools. In addition, a general schematic can be found in Appendix E Figure 5.

Discussion of Data Gathering Techniques and Capabilities

Each restart sequence served to demonstrate specific aspects of the restart system. Each sequence was divided into four primary phases and a connecting sub-phase. The primary phases of the restart sequence were determined to be Electric Starter, Oxygen till Ignition, Burn, and Extinguish. Each phase was determined and timed to precise events that could be timed with relatively high precision from video evidence.

The Electric Starter phase is a measure of the duration of pulses from the starter box. Each pulse was on the order of a few tenths of a second and served to ignite the gaseous mixture in the pre-combustion chamber.

The Oxygen till Ignition phase is a measure of the time from full engagement of the oxygen globe valve until ignition of the propane and oxygen mixture. This procedure was necessary to prevent propane gas accumulation in the combustion chamber and a resulting explosion. Maintaining as lean a mixture as possible by injecting gaseous oxygen first met this safety goal.

The Burn phase is rather self explanatory and measures the time of burn of the primary fuel with the oxidizer. Start times for this sequence were determined as 1/2 second from propane and oxygen combustion initialization to termination of the oxygen flow. The dynamic visible flame difference of primary ignition signified this event and allowed for relatively precise timing as well.

The Extinguish phase is also self explanatory and measures the time from oxygen gas flow termination to a visibly non-existent flame at the nozzle. With a nozzle area reduction of only 76.1% (Appendix D Calculations) a backflow of atmospheric air allowed for a slightly continued, yet unobservable burn.

Times for each phase were determined using digital video of each sequence. Since all the phases mentioned occur at the macroscopic level and are external to the rocket motor chamber, the moderate resolution of the camera and the opaque rocket walls did not compromise the desired time phases. More accurate timing measurement techniques are discussed in the Conclusion. All data in the following sections can be found in Appendix F Table 1, Hybrid Rocket Test Sequences and Corresponding Phase Durations and in Appendix F Figure 6, Hybrid Rocket Sequences and Corresponding Phase Durations.

Attempts to obtain the weight reduction of the rocket motor during burn cycles and the corresponding fuel regression rates were greatly flawed. Due to the system design incorporating two soft hose connections on the front and the side of the rocket motor, the slightest movement in either hose drastically affected the scale readouts. In addition to inconsistent scale readings, the scales had a power saving feature that reset and therefore “re-zeroed” the scales every 30 seconds. Without an absolute reference point it was impossible to draw any conclusive data to analyze the fuel regression rates. Fortunately, the total burn time predictions that would have been achieved through the scale measurements were achieved through the digital video captured during each burn sequence.

Sequence 1

Sequence 1 successfully demonstrated a single start application. With a fully-open oxygen valve for 4.5 seconds followed by 3 seconds of electrical igniter pulses the rocket motor ignited. This particular ignition commenced with a comparatively loud start propane/oxygen explosion. Although the explosion was expected and met the design requirements of flame front propagation down the grain structure, all present at the testing site were somewhat startled. The propane-oxygen mixture was determined to be rich and was later corrected by opening the propane globe valve more slowly.

Other observations of this particular burn included a bright orange flame measuring approximately three feet in length and six inches in diameter. The extraordinary size of the flame was determined to be the combustion of an unintended fabrication residue of Vaseline. When pouring the primary fuel into the motor cartridges, the lubricant was necessary to later pull the annular form away from the fuel grain. More careful observation of the flame front later revealed a separation of flame from the nozzle of approximately two inches. The separation of the flame from the nozzle was possibly due the converging only configuration as opposed to the converging-diverging configuration more commonly used to increase thrust and overall performance in propulsion systems. In addition, the time for the flame to extinguish took only a half second. The quick extinguishment can also be attributed to the excessive flame and the separation it caused from the nozzle. Since the objective of the testing was to conduct as many restarts as possible, the initial starting of the rocket motor was kept to a minimal burn time of 3 seconds in order to minimize unnecessary fuel usage.

Sequence 2

The second burn sequence, or more appropriately the first restart sequence, demonstrated the capabilities of the system to successfully restart a rocket. In the same order of valves as discussed in the System Theory section, the fully-open oxygen valve was followed by a more slowly opened propane valve than the first test, resulting in a 5.5 second delay from oxygen injection to ignition and a 5 second delay with the electrical igniter. The one second increase of oxygen injection time and two second increase of electrical starter time occurred as a result of the more gradual propane injection. Upon propane/oxygen ignition, a noticeable blast front propagated out the nozzle to a distance of about two feet. The starter gas and oxygen then heated the primary fuel and oxygen until their combustion overtook the propane/oxygen reaction. Flame progression consisted of the flame from an initial blast that waned to a flame front of only about four inches from the end of the nozzle. In a matter of about one half of a second the flame progressed to a steady state length of one foot with a diameter on the order of two inches for 13 seconds. The time to extinguish the flame then took 6 seconds.

This monumental burn sequence confirmed the design objective to restart a hybrid rocket motor with a unique system. The purpose of the second burn sequence not only demonstrated the overall ability of the system to restart a hybrid rocket successfully, it also did so after a relatively short cool down period of 30 seconds from the first burn. Downtimes and the effects of varying lengths are addressed later.

Sequence 3

The third burn sequence took place at a nearly identical set of phase times as Sequence 2. However, the electric igniter was pulsed for the entire duration of the oxygen flow. Both the Electric Start and Oxygen till Ignition times were 5 seconds followed by a 19 second total burn time. Again, the initial propane/oxygen flame expanded rapidly out the nozzle to ignite the primary fuel. Also, the flame diminished in a likewise fashion and then increased to a smaller length of approximately 8 inches for 19 seconds. The time to extinguish the flame was one second longer than Sequence 2 at 7 seconds. The 7 second extinguish time being the longest time to extinguish the flame correlates to a shorter downtime from Sequences 2 of only 60 seconds. Later sequences suggest that longer downtimes between sequences decreases the extinguish time as fuel supplies decrease.

Sequence 4

To confirm the similar results of Sequence 2 to Sequence 3, the phase times of Sequence 4 revealed the consistency of the Electrical Start and Oxygen till Ignition times. In fact, both of these phases were slightly decreased to Electrical Start phase time of 4 seconds and Oxygen till Ignition phase time of 4.5 seconds. However, the flame propagation of Sequence 4 had a somewhat different formation from the three previous sequences. First, the propane and oxygen mixture combusted very smoothly together and did not exhibit a long flame blow down. This resulted in a progressively increasing flame as the primary fuel start combustion. Also, the propane was left on for a full second after primary ignition. The combustion mixture became increasingly rich and the flame was observed to be excessively orange as opposed to the previously observed flames with a white center glow. Once the propane was terminated it proceeded to the standard flame size and color. A lower extinguish time of 5 seconds was followed then be a 120 second downtime until Sequence 5.

Sequence 5 and Sequence 6

The phase time of Sequences 5 & 6 demonstrated a very short downtime in what is designated herein as a “hot restart.” Data for Electric Start and Oxygen till Ignition phases for Sequence 5 was unavailable due to a miscommunication among the camera operator and valve operators. However, previous results suggest that the times did not vary from Sequences 2-4. Unlike Sequence 4, the primary ignition resumed the excessive blast front, leading to a waning flame, and then to a fully-developed flame on the order of only 5 inches. Note that the primary fuel consumption influences the flame length by decreasing it as fuel supply also decreases. After a 20 second Burn time the residual flame was extinguished quickly in about 4 seconds, followed by a 6 second downtime until Sequence 6.

Although no discrete window exists to distinguish a normal restart from a hot restart, for the purpose of data analysis it was considered to be “any downtime period in which the typical start sequence is unnecessary for primary fuel and oxygen combustion for the following restart.” In the case of Sequence 6 it can be quantified as 6 seconds of downtime. With respect to the preceding definition, the Electric Start and Propane were unnecessary to initiate primary fuel and oxygen combustion. The reason these two phases were omitted can be found in the System Theory Section. One possible source for the restart of Sequence 6 may have been a lingering flame that was unnoticed inside the combustion chamber. Another plausible source may have been the local high temperatures and presence of vaporized fuel which would then react upon contact with an oxidizer. Regardless of the precise mechanism that caused the restart for Sequence 6, the main value of the restart was conserved energy in the electrical system and starter fuel gas.

Sequence 7

The following sequence demonstrated the inherent limitations of the systems to provide enough electrical energy to accommodate a high number of restarts. After a 300 second downtime from extinguishment in Sequence 6, a typical restart sequence with the propane starter gas and oxygen was initiated. However, when the Electrical Igniter was pulsed it was noted that the sound of the spark did not come from inside the rocket motor pre-combustion chamber but rather the power supply unit. With only two standard 9 volt batteries as the energy source and a manufacturer suggested 1% decrease in electrical energy from each pulse, the energy supply was determined to be exhausted beyond the dielectric breakdown of air across the arc gap. An extended downtime till Sequence 8 of 1200 seconds allowed for the installation of new batteries. The chamber motor casing was also allowed to cool during this time. Finally, an aft visual inspection confirmed the new batteries produced the intended spark and testing resumed.

Sequence 8

After six successful burns the fuel supply was of concern. Realization in Sequence 8 revealed it was all but completely consumed. With the confidence of a strong spark in the pre-combustion chamber, all the starting phases were conducted. With a slightly long Electric Start time of 6 seconds and Oxygen till Ignition time of 7 seconds, the propane and oxygen mixture began combustion. Upon ignition the rocket experienced a very hard start with a more pronounced propane/oxygen explosion. Immediately the flame produced was noticed to be much smaller in all aspects with dimensions of approximately 4 ½ inches in length and 1 inch in diameter. As the burn progressed for 23 seconds the flame decreased to a length of about 4 inches while also becoming much whiter in color. All these characteristics indicated a diminished fuel supply and an excessively lean burn mixture. Appropriately enough, the extinguish time was the shortest of all the tests and occurred immediately upon termination of the oxygen flow.

Sequence Averages and Data Analysis

When calculating the mean values of the phase times from all the sequences it was necessary to distinguish among data that did and did not represent the intended objective of restarting a hybrid rocket motor. The three sequences immediately drawn to attention are Sequences 5, 6, & 7. For Sequence 5 the values of Electric Start and Oxygen till Ignition phase times were not able to be obtained from a recording error on the video. The video was observed to start exactly about 1 second before ignition, and thus the remaining phase times for Burn and Extinguish are still valid. Sequence 6 did not require an electric start and was ignited almost immediately after oxygen injection. Even though the downtime of 6 seconds from the previous sequence most obviously was an effector on the start sequence, it still met the objective criteria of restarting a hybrid rocket motor with a functioning restart system. On the other hand, Sequence 7 did not mean the objective even though it had legitimate values recorded for all the phases. The clause prevents Sequence 7 data from being included in the mean value of all data is that electrical energy supply was exhausted. If the values were included they would greatly affect the data given the small population of restarts.

The mean averages of the valid data are as follows for the corresponding phase times and can be also be found in Appendix F Table 2. The Electric Starter average phase time was calculated to be 3.8 seconds with a maximum time of 6 second for Sequence 8 and minimum time of 0 seconds for Sequence 6. The Oxygen till ignition average phase time was 4.5 seconds with a similar maximum time of 7 seconds occurring again during Sequence 8 and a minimum time for Sequence 6. The Burn average phase time was 15.7 seconds with a maximum time of 23 seconds for Sequence 8 and a minimum time of 3 seconds for Sequence 1. The Extinguish average phase time was 3.6 seconds with a maximum time of 7 seconds occurring during Sequence 3 and a minimum time of 0 seconds for Sequence 8.

Downtime analysis was conducted over the entire set of sequences as opposed to data used for previous calculations. The average Downtime phase was 288 seconds. The maximum Downtime was 1200 seconds and occurred between Sequence 7 and Sequence 8 and was attributed to battery replacement and electrical inspections. The minimum Downtime was 6 seconds and occurred between Sequence 5 and Sequence 6 and demonstrated the "hot restart" abilities of the restart system.

Pressure readings for the oxygen and propane gas tanks were also taken intermittently between sequences as shown in Appendix F Figure 7. An initial reading of 2100 psig on the oxygen tank regulator and a final reading 1900 psig along with two other random reading revealed a 1.75 psig tank pressure drop for each second of oxidizer flow. A linear curve fit of the data resulted with a square of the residuals value of 0.993. An initial reading of 118 psig on the propane tank regulator and a final reading 110 psig along with two other readings taken at the same time as the oxygen readings revealed a 0.064 psig tank pressure drop for each second of oxidizer flow. A linear curve fit of the data resulted with a square of the residuals value of 0.873. Propane tank pressure was plotted against total oxidizer burn time instead of propane flow times as a consistent timing unit with the average burn time 15.7 seconds.

Delay in abilities of ignition times from propane injection to initial combustion can be attributed to several lag mechanisms. The first and most apparent solution for increasing starter response time is to provide a continuous spark in the pre-combustion chamber instead of pulses at the rate of 2-3 per second. The associated dramatic increase of ignition availability should directly coincide with an equally shorter ignition time.

A less direct yet still effective ignition delay solution is to decrease the length of hose from the control valve to the injection port. Although low flow gas injections can be idealized as incompressible, a slight compressibility delay is realized with high flow rates and contributes significantly over extended distances. With operating pressures on the order of only 10 psig, the effects of compressibility are minimal.

However, the low pressure operation creates its lag mechanism. At lower pressures a gas mixture does not come in contact with surrounding molecules with near the force as at higher pressures. At a given temperature the ideal gas law reveals a proportional increase in density with pressure. The increased density also means a increased probability of molecular collisions and a reaction to occur at the spark location. In addition, studies conducted at Hampton University for high pressure combustion revealed a “wrinkled” (“Combustion” 2001) flame propagation under pressures greater than 5 atmospheres. Further inspection suggests the “wrinkles” in the flame front allow for greater mixing area and therefore faster and more efficient flame growth. If the plumbing system could withstand such pressures and the pressure was increased as such, noticeable increases in ignition times would result

Safety and Ethical Analysis

Rockets and the systems affiliated with them are required to perform with almost excessive safety considerations. Since a rocket operates from the momentum exchange of high pressure and high temperature gases, any disruption in the process can lead to catastrophic failure. Therefore, all aspects of the design must be carefully evaluated.

When considering the pressure distribution in the rocket chamber the motor casing is the first limiting factor. By the limitations of the propane gas regulator the propane and oxygen were both injected at 10 psig. Idealizing the pipe as a thin-walled pressure vessel allowed for analysis of the tangential, longitudinal, and radial stresses of the side walls. Calculations yielded in Appendix D Calculations that an increase in pressure of 116.3 times the operating pressure would be required to bring the safety factor of the pressure vessel below a suggested value of five. Clearly, the initial explosive blasts of the propane and oxygen mixture would not create this enormous of a pressure singularity increase.

Although, the nozzle was improvised using a 1 ¼” to ¾” pipe reducer and a ¾” to 3/8” bushing, the nozzle area reduction ratio still demanded attention. With consideration to the nominal sizing of standard schedule 40 pipe the nozzle area reduction as calculated in Appendix D Calculations was only 76.1%. Consider a typical area reduction of a rocket nozzle used in Figure 11.13 of Hill et al. (1992). With operating pressures in excess of 500 psia and much more aggressive combustion the nozzle area reduction from chamber area is also on the order of 82.0% as shown in Appendix D Calculations. However, since choked flow occurs at a lower area reduction for isentropic flow with higher pressures and temperatures, the nozzle does not warrant an unacceptable chamber pressure increase.

A two-fold safety mechanism met the risk of backflow from the chamber into the injection hoses. Safety equipment included one one-way flow check valves shortly after the quick-disconnect sockets. The one-way flow valves acted according to their title to prevent the reverse flow of propane or oxygen in their respective hoses. A reverse flow of either fuel or oxidizer would be very hazardous as a potential flame could also flow into the hoses. A flame would be met with a larger supply combustible gas than intended. The excessive combustion would spike the hose pressure while also surpassing the temperature rating and melting the hose, resulting in uncontained gases. One-way valves drastically reduce this risk and provide back pressure in the event of a chamber pressure spike. Even with one-way valves, globe valves upstream allowed for a manual termination in case of any backflow failure.

Electrical shock also posed a risk when using the 200 kV stun gun components. To eliminate this risk every measure was taken to electrically isolate and insulate the stun gun components, wires, and connections. An all-weather outlet box provided sufficient room to separate all the wires and contain the batteries. All electrical connections within the box were soldered and then covered in epoxy for strength and electrical insulation. Plastic covers protected the spade connectors while a thick epoxy coating was applied to rocket motor electrodes. In addition, high heat stove cement and rubber grommets were used to isolate the electrodes from the motor casing to prevent a short circuit and the resulting electrifying of the entire motor casing. Operation of the electrical system was controlled by a primary On/Off toggle switch and a secondary momentary toggle switch to prevent accidental discharges.

Additional secondary safety concerns are discussed in Appendix A, Health and Safety Concerns.

Conclusions

The most authoritative features of the hybrid rocket restart system herein is the dual-plumbing arrangement. One may recall in Research Designs p 7 that a key advantage of hybrid rockets is the simplicity of utilizing a sole oxidizer tank for combustion. So why must the design contradict the very advantages it attempts to employ? The answer is quite simple. If any hybrid rocket is desired to be restarted it must use a separate starter fuel from the primary fuel. Thus, an extra plumbing system is necessary. Although the storage tanks and plumbing equipment used herein were not optimized for any particular mission requirements, consideration for a prescribed number of restarts could drastically reduce necessary storage take capacities.

Improvements in ignition delay could be most easily be achieve by decreasing plumbing hose lengths. Compressible lag effects from decreased length will most greatly be realized at high pressures. High pressures themselves would also assist with combustion ignition for the density and pressure of the gas mixture. In addition, studies reveal a higher speed and efficiency flame propagation with higher pressure combustion.

Extinguish delays could also be improved by analyzing chamber pressure. With lower pressures the chamber to choked area ratio of a nozzle is much greater than high pressure gases. If a proper nozzle were used instead of pipe fittings the area reduction ratio would have exceeded 90%. A smaller throat would decrease backflow of air into the combustion chamber and the resulting continued burns. Another solution mentioned with ignition delays is the decrease in hose length of control valves to the injection ports.

The suction of the flame on the remaining gases in the hoses, primarily the oxygen, also causes undesired extinguishment delays.

Ideally, more precise timing methods would be employed to gather data. The implementation of electrically actuated valves with analog controls for the propane and oxygen hoses could be timed with a feedback system into a data acquisition device. High temperature pressure and temperature sensors placed inside the combustion chamber would reveal the singularity pressure increase associated with the propane/oxygen ignition and could then send a feedback signal to terminate propane flow, thus conserving starter fuel. The electrical igniter would ideally also utilize a larger power supply and be able to operate continuously. A continuous spark would also yield a more consistent and faster ignition of the propane/oxygen mixture. Predetermined burn times could be controlled and pressure decay upon flow termination would more accurately measure the flame extinguish delays. Temperature readings would also be used to determine a minimum "hot restart" time as the heat transfer cooled the rocket to the vapor temperature of the primary fuel.

The success of the system was greatly dependent on the problem solving skills necessary for any original design. All the resources for a particular design were not always necessary and quick decisions had to be made many times to account for unforeseen events. Overall, the support of family and friends made the research herein possible.

References

Bradford, et al., “Hybrid Rocket Combustion Enhancement,” Patent 5,582,001, 10 December 1996.

Brown, C. D., Elements of Spacecraft Design, 1st Ed. AIAA, Reston, VA. 2002.

Budynas, R.G. and J.K. Nisbett, Shigley’s Mechanical Engineering Design, 8th Ed, McGraw-Hill, St. Louis. 2008.

Campbell, Phillip B., “Starter for Rocket Motor,” Lockheed Aircraft Corporation, Patent 3,116,599, 7 January 1964.

Çengel, Y.A. and R.H. Turner, Fundamentals of Thermal-Fluid Sciences, 2nd Ed, McGraw-Hill, St. Louis, 2005.

“Combustion Under Pressure- A New Understanding Revealed.” Microgravity News, Hampton University, NASA Office of Biological and Physical Research, 2001. <http://www.spaceref.com/news/viewsr.html?pid=4051>.

Hill, P. and C. Peterson, Mechanics and Thermodynamics of Propulsion, 2nd Ed, Addison-Wesley, New York, 1992.

“Hybrid Rocket Motors!” Pratt Hobbies, 18 Oct 2007, <http://www.pratthobbies.com/info_pages/hybrids.htm>.

Sutton, G.P., and O. Biblarz, Rocket Propulsion Elements, 7th Ed, John Wiley and Sons, New York, 2001.

Appendices

A. Health and Safety Concerns

Additional health and safety concerns existed and demanded efforts to reduce the risk to the rocket motor operators and bystanders.

Such risks include the expending of high speed small and large particles from the rocket nozzle. Thus, a clear zone was established within a radius of 50 feet aft the plane of the nozzle. In the event any particle was expended, all people present were required to wear safety glasses or equivalent eye protection.

As a result of combustion the rocket motor along with the surrounding test stand became very hot. Therefore, necessary handling of objects was done with thick leather gloves for short periods of time.

The combustion of polymethyl methacrylate, as with most combustion reactions, does not burn to completion. Primary byproducts of polymers are the same as conventional hydrocarbon and include carbon dioxide and water. However, trace elements of carbon monoxide, monatomic elements, and other various combinations of carbon, hydrogen, and oxygen form. Minimum exposure to these trace secondary byproducts is a must to prevent respiratory irritation. Therefore, all burn sequences were conducted outside with light winds to prevent stagnation of fumes.

B. Economic Analysis

Direct economic impact can be made to the individual or group that wishes to purchase the parts necessary to make a similar hybrid rocket and accompanying restart system. Therefore, a price list of all parts used for the motor or restart system or necessary for the fabrication of these items can be found in Table 1, Price List of Rocket and Fabrication Parts and Tools in Appendix F.

Further economic impact can also be seen with the research and development of any sort of technology that could be applied to military applications. The ability to place an object in orbit and change the orbit significantly at a later time draws much attention to missile defense. Additional applications also include commercial satellite attitude controls and replacement of the dispensable RATO (Rocket Assisted Take-Off) rockets on the side of cargo aircraft.

C. General Theoretical References

The equivalence ratio Φ describes the ratio of fuel/oxidizer actually used in a reaction to the stoichiometric ratio.

$$\phi = \frac{\left(\frac{Fuel}{Oxidizer}\right)_{Act}}{\left(\frac{Fuel}{Oxidizer}\right)_{St}}$$

If considering only a change in the oxidizer, an excess of oxidizer would lead the ratio less than unity (Φ<1) and a deficit of oxidizer to greater than unity (Φ>1).

D. Calculations

Combustion limits translation of propane/air to propane/oxygen:

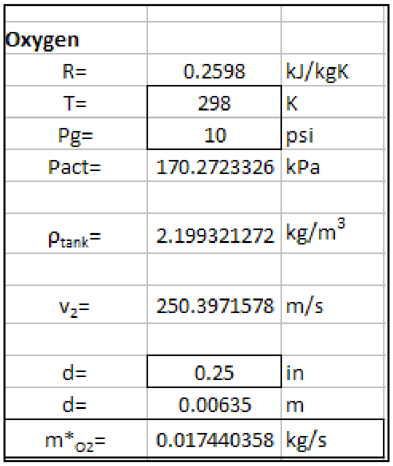
- $$Oxygen\,Limit\% = \frac{Air\,Limit\%}{\left[(1 - Air\,Limit\%)\frac{Molar\,Mass O_2}{Molar\,Mass O_2 + 3.76 N_2} + Air\,Limit\%\right]}$$
- Molar Masses: Air = 137.28 kg/kmol, Oxygen = 32 kg/kmol,

  Propane = 44 kg/kmol

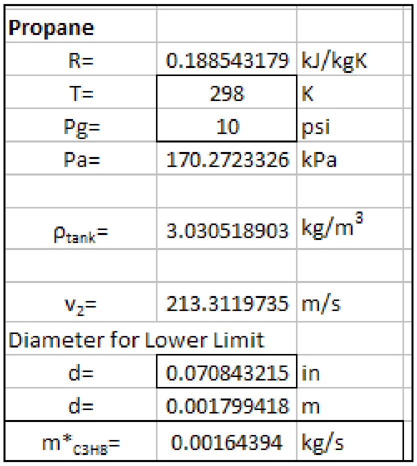
- $$Lower\, C_3H_8/O_2\% = \frac{2.15\%}{\left[(100\% - 2.15\%)\frac{32}{137.28} + 2.15\%\right]} = 8.61\%$$

Flow rates of propane and oxygen using Bernoulli Equation and basic assumption as described in System Theory:

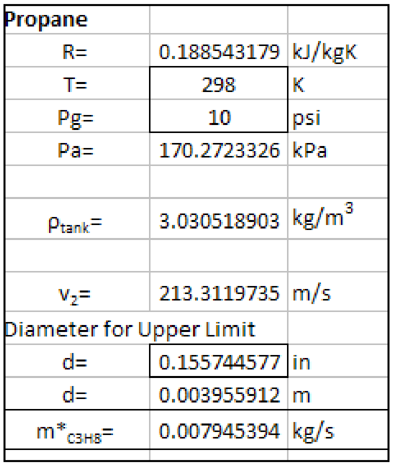
- $$\boxed{\begin{array}{l} \rho = \dfrac{P}{RT} \qquad P_1 + \rho\dfrac{\cancel{v_1^2}}{2} = P_2 + \rho\dfrac{v_2^2}{2} \\ P_1 = P_2 + \rho\dfrac{v_2^2}{2} \qquad v_2 = \sqrt{\dfrac{2(P_1 - P_2)}{\rho}} \\ \dot{m} = \rho A V \end{array}}$$



Oxygen		
R=	0.2598	kJ/kgK
T=	298	K
Pg=	10	psi
Pact=	170.2723326	kPa
ρ_{tank}=	2.199321272	kg/m^3
v_2=	250.3971578	m/s
d=	0.25	in
d=	0.00635	m
m*$_{O2}$=	0.017440358	kg/s



Propane		
R=	0.188543179	kJ/kgK
T=	298	K
Pg=	10	psi
Pa=	170.2723326	kPa
ρ_{tank}=	3.030518903	kg/m^3
v_2=	213.3119735	m/s
Diameter for Lower Limit		
d=	0.070843215	in
d=	0.001799418	m
m*$_{C3H8}$=	0.00164394	kg/s



Propane		
R=	0.188543179	kJ/kgK
T=	298	K
Pg=	10	psi
Pa=	170.2723326	kPa
ρ_{tank}=	3.030518903	kg/m^3
v_2=	213.3119735	m/s
Diameter for Upper Limit		
d=	0.155744577	in
d=	0.003955912	m
m*$_{C3H8}$=	0.007945394	kg/s

Chamber pressure of the motor walls analysis for a thin-walled pressure vessel upon primary ignition. For 1 ¼" Galvanized Steel Pipe Schedule 40 calculations of maximum tolerable singularity pressure increase with stress formulas reference to Budynas et al. (2008):

- For yield, factor of safety goes to unity, n=1, where $n = \frac{\sigma'}{\sigma_y}$
- To be considered thin-walled pressure vessel, d > 10t, d=1.660 in, t=-0.140in, so 1.660 > 1.400 and the assumption is valid
- $\boxed{\sigma_t = \frac{pd_i}{2t} \quad \sigma_l = \frac{pd_i}{4t} \quad \sigma_r = -p_i}$
- Von Mises Stress: $\boxed{\sigma' = \left[\frac{(\sigma_1 - \sigma_2)^2 + (\sigma_2 - \sigma_3)^2 + (\sigma_3 - \sigma_1)^2}{2} \right]^{1/2}}$
- Solve for Pressure with a Yield Strength of 30,000 psi and conservative safety factor n=5
- Results:

Solve for Maximum Pressure				
P=	1163.11	psi	Von mises	
Sy=	30000	kpsi	6000	psi
sig t=	5732.473			
sig l=	2866.236		FOS=	5
-p=	-1163.11			

Given		
t=	0.140	inches
do=	1.660	inches
di=	1.380	inches

Minimum Desired Safety Factor	
n=	5

- With a gas injection pressure of 10 psig, the chamber would be able to withstand a 116.3 multiple increase in chamber pressure.

Nozzle Area Reduction (1 ¼ Sch 40 – 3/8” Bushing):

- D_1 = 1.660 in-2(0.140in) = 1.38 in D_2 = 0.675 in
- $(A_1 - A_2)/A_1 = (D_1^2 - D_2^2)/D_1^2 = 76.1\%$

Nozzle Area Reduction for Figure 11.13 Hill et al. (1992):

- D_1 = 66 mm D_2 = 28 mm
- $(A_1 - A_2)/A_1 = (D_1^2 - D_2^2)/D_1^2 = 82.0\%$

E. General System Arrangement

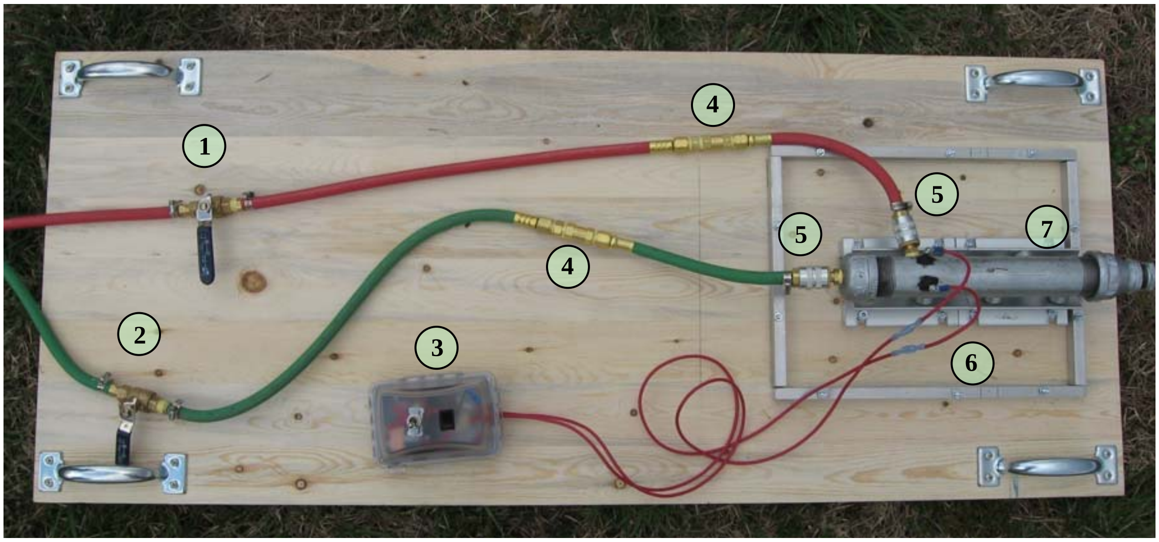


Figure 5, Gaseous O_2 and C_3H_8 Hybrid Rocket Restart System Layout

1-Propane Control Valve
2-Oxygen Control Valve
3-200kV Electrical Starter
4-One-Way Valve
5-Quick Connect
6-Electrodes
7-Rocket Motor

F. Data Tables and Charts

Table 2, Hybrid Rocket Test Sequences and Corresponding Phase Durations

	Phase Time (seconds)					Tank Pressures (psig)	
Sequence	**Electric Starter**	**O2 till Ignition**	**Burn**	**Extinguish**	**Downtime**	**O2**	**C3H8**
1	3	4.5	3	0.5	30	2100	118
2	5	5.5	13	6	60	-	-
3	5	5	19	7	300	-	-
4	4	4.5	20	5	120	2025	113
5	-	-	20	4	6	-	-
6	0	0.5	12	3	300	-	-
7	5	0	0	0	1200	-	-
8	6	7	23	0	-	1950	112
End Conditions	-	-	-	-	-	1900	110

Red highlighted areas are neglected when calculating averages due to invalid or illegitimate values.

Table 3, Average Phase Durations for Valid and Legitimate Sequences

	Phase Time (seconds)				
Sequence	**Electric Starter**	**O2 till Ignition**	**Burn**	**Extinguish**	**Downtime**
Averages	3.8	4.5	15.7	3.6	288
Maximum	6	7	23	7	1200
Minimum	0	0.5	3	0	6

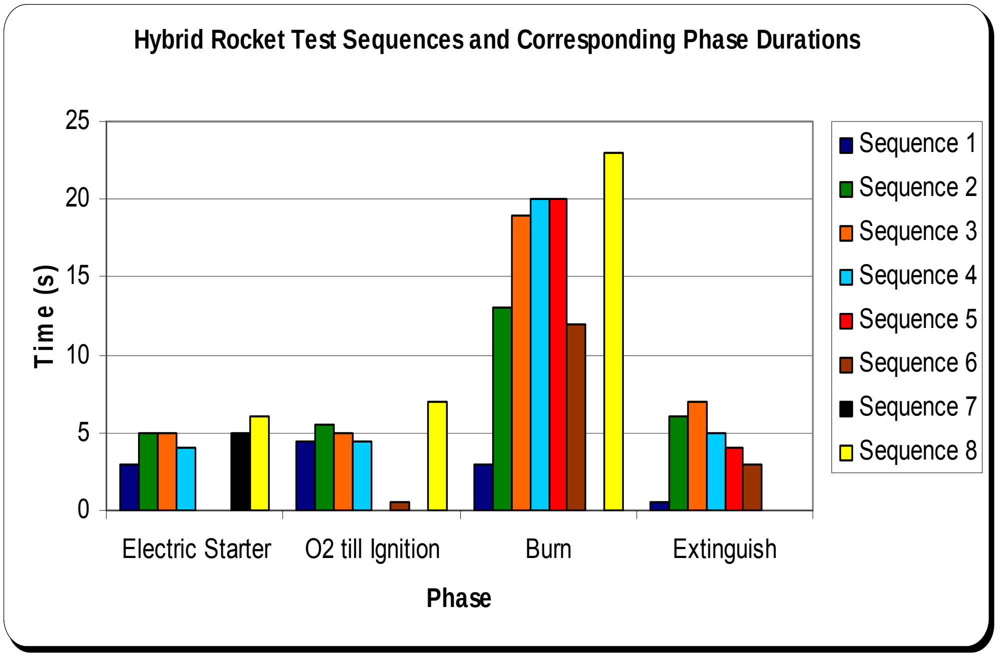


Figure 6, Hybrid Rocket Sequences and Corresponding Phase Durations

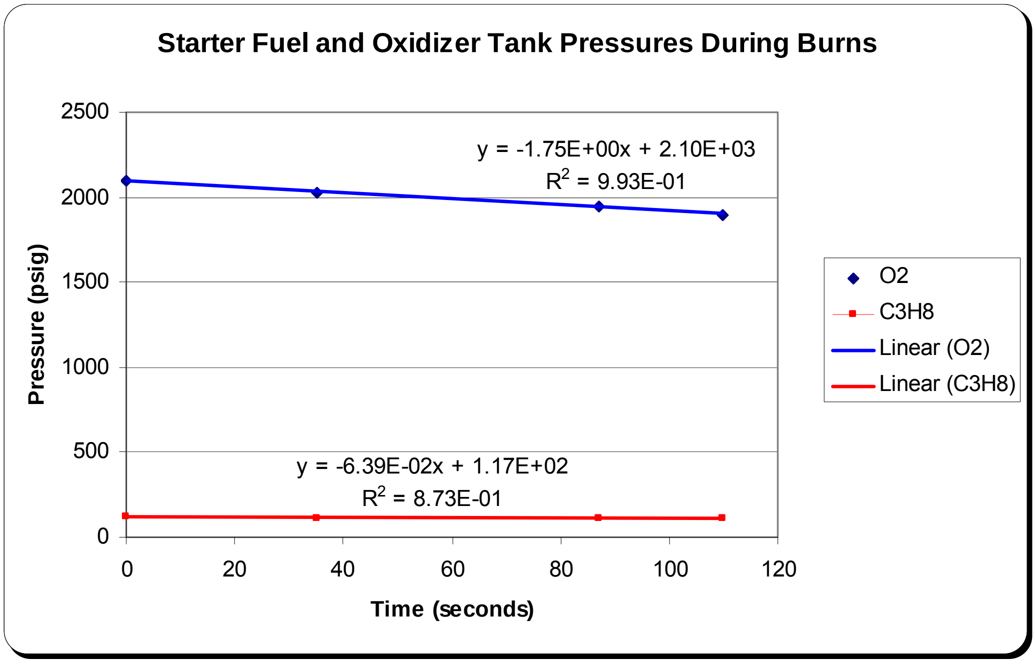


Figure 7, Starter Fuel and Oxidizer Tank Pressure During Burns

Table 4, Price List of Rocket and Fabrication Parts and Tools

Object	Vendor	Price	Quantity	Total	Category
1 1/4" Galvanized Pipe Nipple 10"	Lowe's*	$6.00	3	$18.00	Motor
1 1/4"-3/4" Galvanized Steel Reducer	Lowe's	$2.26	3	$6.78	Motor
1 1/4" Galvanized Steel Endcap	Lowe's*	$2.50	3	$7.50	Motor
3/4"-3/8" Galvanized Steel Bushing	Lowe's	$0.86	1	$0.86	Motor
Brass Air Hose Plug Male 1/4"	McMaster	$1.47	6	$8.82	Motor
7/16X3/16 Grommet	Lowe's	$1.27	4	$5.08	Motor
Rutland Black Furnace Cement	City Lumber	$3.27	1	$3.27	Motor
Two Part Epoxy	Lowe's*	$6.00	1	$6.00	Motor
12"X18" Plate Steel	Lowe's	$4.78	1	$4.78	Motor
Sabre Stun gun (200 kV)	Amazon	$29.95	1	$29.95	Starter
Toggle Switch (ON/OFF)	Lowe's	$4.08	1	$4.08	Starter
Weather Proof Electrical Box 16/1	Lowe's	$8.97	1	$8.97	Starter
14 AWG Wire (XX ft)	Lowe's	$3.00	1	$3.00	Starter
Terminals (Male and Female Spade)	Lowe's*	$2.00	1	$2.00	Starter
#6 Machine Screws - 1 1/2"	Lowe's*	$3.00	1	$3.00	Starter
#6 Nuts	Lowe's*	$0.80	1	$0.80	Starter
9V Batteries	Walmart	$4.50	4	$18.00	Starter
10-24 X 1 1/2 Machine Screw Zn	Lowe's*	$3.50	1	$3.50	Test Stand
#10 Flat Washers (24 pk)	Lowe's*	$1.50	1	$1.50	Test Stand
Digital Bench Scales 2200G	McMaster	$77.52	2	$155.04	Test Stand
Econony V-Block	McMaster	$16.67	2	$33.34	Test Stand
Stud Mount Ball Transfers 5/8" Steel	McMaster	$5.80	6	$34.80	Test Stand
3/4X20X48 Mounting Board	Lowe's	$12.38	1	$12.38	Test Stand

6 1/2" Zn Carrying Handle	Lowe's	$2.97	4	$11.88	Test Stand
Lockable Draw Catch	Lowe's	$4.49	1	$4.49	Test Stand
Brass Sleeve-Lock Hose Coupling 1/4"	McMaster	$5.94	2	$11.88	Plumbing
Barbed Brass Hose Fitting 1/4"	McMaster	$0.76	10	$7.59	Plumbing
Reverse Flow Check Valves	AirGas*	$15.00	1	$15.00	Plumbing
Radnor Hose Coupler 2 pk	AirGas*	$10.00	1	$10.00	Plumbing
25' Air and Gas Welding Hose	AirGas*	$35.00	1	$35.00	Plumbing
#4 1/4"-5/8" Hose Clamps (2 pk)	Lowe's*	$0.50	3	$1.50	Plumbing
Ball Valve 1/4" NPT Female	Lowe's	$6.27	2	$12.54	Plumbing
Radnor Oxygen Regulator	AirGas*	$80.00	1	$80.00	Plumbing
Radnor Propane Regulator	AirGas*	$75.00	1	$75.00	Plumbing
Gas Pipe Thread tape 1/2"X260"	Lowe's*	$2.50	1	$2.50	Plumbing
Kobalt 6-32 UNC Tap	Lowe's	$4.24	2	$8.48	Fab Equip
1/4-20 Ticn Tap	McMaster	$7.96	1	$7.96	Fab Equip
Size 7 Drill Bit for 1/4-20 Tap	McMaster	$3.38	1	$3.38	Fab Equip
1/4 NPT Tap	Lowe's	$9.97	1	$9.97	Fab Equip
Drill Bit for 1/4 NPT Tap	True Value*	$5.00	1	$5.00	Fab Equip
DeWalt 18 TPI Metal Sabre Saw Blades	Lowe's*	$2.50	1	$2.50	Fab Equip
Nicholson 24Tx10" Hacksaw Blade	Lowe's	$1.98	1	$1.98	Fab Equip
Hacksaw 10"	Lowe's	$8.98	1	$8.98	Fab Equip
Rachet Caulk Gun	Walmart*	$3.00	1	$3.00	Fab Equip
Sandpaper 50 Grit Black Zirc	Lowe's	$2.97	1	$2.97	Fab Equip
Sandpaper 80 Grit Black Zirc	Lowe's	$2.97	1	$2.97	Fab Equip
Wiss Compound Action Snips	Lowe's*	$8.00	1	$8.00	Fab Equip
Dowel Rods 1/2" X 36"	Lowe's*	$0.50	1	$0.50	Fab Equip
Dowel Rods 1 1/2" X 36"	Lowe's*	$1.00	1	$1.00	Fab Equip
1/2" Auger Bit	Lowe's*	$7.00	1	$7.00	Fab Equip
1/4" Drill Bit	Lowe's*	$3.00	1	$3.00	Fab Equip
1/4"-1/2" Tap Handle	True Value*	$7.00	1	$7.00	Fab Equip
*Denotes Approximate Value			Total	$722.52	