Hydrogen Peroxide Liquid-Fuel Motor Design and Testing Report

by

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Executive Summary

The driving goal and question to be answered of this experimental based report is to examine the progress made in designing, developing, testing, and reporting of a student-built liquid-rocket motor using materials commonly acquired via online and local retail venders. The development of this motor required the cross-disciplinary understanding and application of thermodynamics, fluid dynamics, kinematics, material sciences, chemistry, and several areas of mathematics into one cohesive design. Work and research through the manufacturing and testing of this motor has demonstrated an improved understanding of fuel optimization, both chemical and physical in nature; advance in my intuition of fluid interactions and requirements for "smooth" combustion; and finally, has continued my understanding of the design and manufacturing abstract process to prepare for a career in engineering. The chosen combustion pressure is in concept 1.25 megapascals which in theory should deliver 200 newtons of thrust for the selected design parameters. The challenges and limits of my welding expertise and understanding of material sciences were undoubtedly tested in designing a skeleton that survived the trial of combustion. Temperatures and cooling measures are also addressed and will potentially be implemented once this proof of concept is verified.

The order of presentation in this report will then follow a chronological progression of the abstract design, followed by the necessary steps taken to produce the final product. Commentary and calculations will be present in both the text and the appendices and will be explained through the design process. Testing figures and recommendations for improvement will then conclude the report, along with any additional references and needed commentary.

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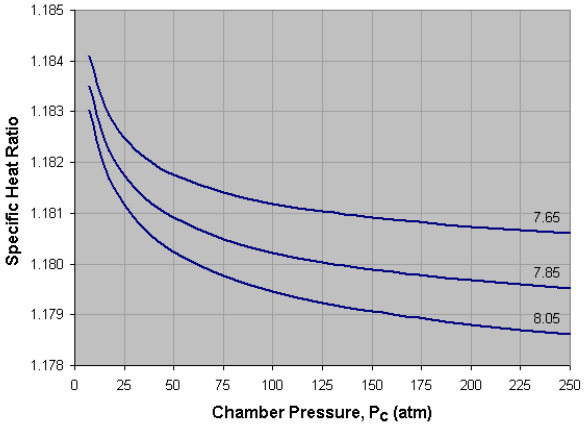

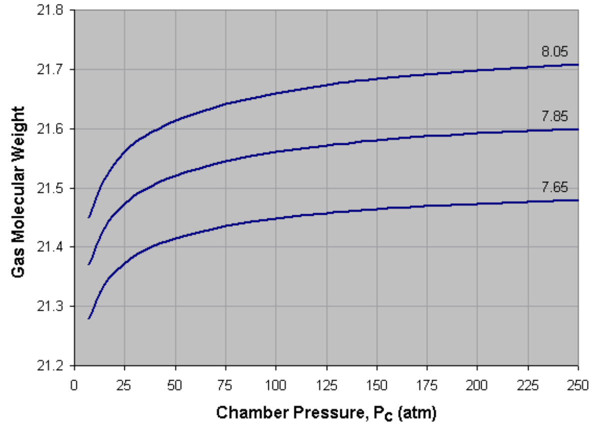

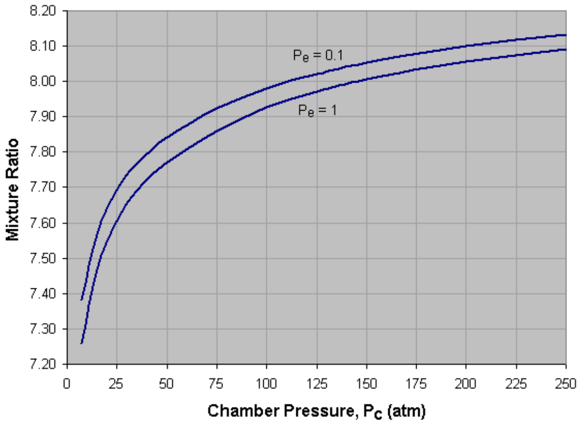

List of Symbols

Isp - Impulse in seconds

T - Thrust in newtons

w_t - Total mass flow rate in grams per second

w_o - Mass flow rate of the oxidizer in grams per second

w_f - Mass flow rate of the fuel in grams per second

P_t – Gas pressure at nozzle throat

T_t – Temperature at nozzle throat

P_c – Chamber pressure

T_c – Chamber temperature

R – Universal gas constant

M – Molecular weight of the gas

γ – Specific heat ratio

A_T – Area of throat cross section

A_e – Area of nozzle exit

P_a – Ambient atmospheric pressure (1 atm)

*L** - Characteristic length

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Mom, without you I could not have even breathed my first breath let alone complete any semblance of an education. You have carried me in the fashion by which God has carried us since before the dawn of natural light and I am forever grateful to you.

Also thank you, albeit you conceded somewhat indignantly so, for letting me convert your garage into a manufacturing facility.

Introduction

The formulas and processes of finding the approximate geometry of the final convergent-divergent motor structure are found primarily through Bremer's work and Wade's given data and tables. Each of these references are cited at the conclusion of this report. The design process follows the selection of certain parameters based on manufacturing or material limitations and then the computation of the calculated variables as a response to the selected ones.

Design Process and Computation

The scope of this project regarding the development and manufacturing of the nozzle, throat, and combustion chamber immediately presents several degrees of freedom on dimensions of the sections of the motor. Following Bremer's work, the impulse, denoted *Isp*; the thrust in newtons, denoted *T*; and the total propellant flow rate in grams per second, denoted w_t are found first.

The desired operation thrust is selected to be 200N as this gives operational conditions (temperatures and pressures) that are achievable using the manufacturing ability currently possessed. Given this thrust of 200N, and 85% of the experimental *Isp* at near-perfect conditions of 271.15 seconds, utilization of

$$Isp = \frac{\mathrm{T}}{w_t}$$

to find desired total flow rate, w_t , gives 75.188 g/s. This value of .85 * experimental *Isp* is taken from the given experimental value for 98% Hydrogen Peroxide burned with kerosene at an oxidizer to fuel ratio of 7.07:1. The 85% efficiency is then the hopeful achieved performance of this engine.

Utilizing the given experimental oxidizer/fuel ratio of 7.07:1, reduction of oxidizer amount to 6.5:1 is then chosen as this allows for lower combustion temperatures as this ratio is fuel-rich and further from stoichiometric combination (over 8:1) , meaning lower thermal energy released with the cost of lower energy efficiency.

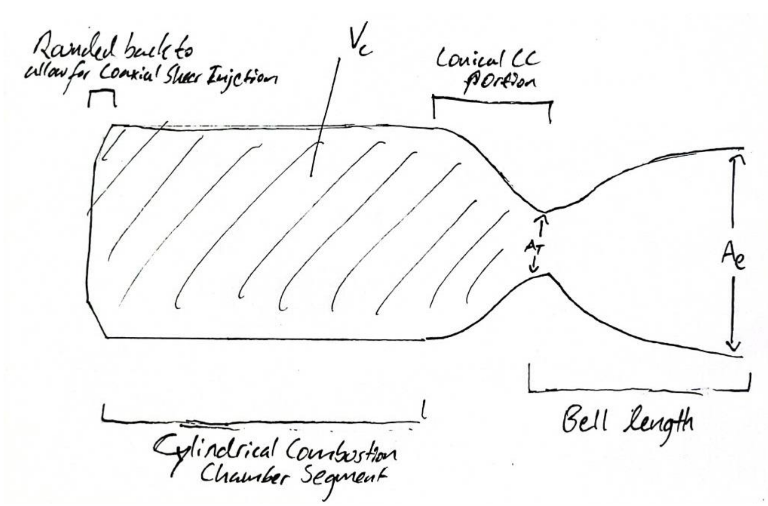
The flow rates of the oxidizer and the fuel, denoted w_o and w_f respectively, can then be found using,

$$w_o = w_t(\frac{R}{R+1})$$

$$w_f = w_t(\frac{1}{R+1})$$

where R is the O/F ratio. Plugging given fields in yields the mass flow rate of our Hydrogen Peroxide to be 65.163 g/s and our kerosene mass flow rate to be 10.025 g/s.

The design of this motor assembly will follow the convergent-divergent rocket motor precedent, which consists of a throat that contracts the combustive gaseous products of the combustion chamber, and then expands the cross-sectional width to in turn expand the products into the nozzle. This design allows for the capture of the thermal energy of the combustion reaction to be harnessed into kinetic energy of the gases themselves through the flow process of travelling through the “bottleneck” of the motor assembly.



The next design steps then must involve finding the dimensions of these three sections and their two junctions, resulting in thrust chamber pressure and temperature, nozzle throat and exit cross sectional area, gas pressure and temperature at the nozzle throat, and nozzle exit velocity each being design parameters that define these dimensions.

Following the DESIGN EQUATIONS resource provided by Risacher and the work done by Bremer, these two equations provide a relationship between the throat temperature and pressure, T_t and P_t respectively, and the chamber temperature and pressure, T_c and P_c respectively, using the specific heat ratio of the combustion by-products which is again taken from the experimental data provided by Wade as 1.2. These relationships are solved using a selected chamber operating pressure of 1.25 MPa, which is chosen as it is a pressure that is achievable using the given manufacturing materials and methods.

$$P_t = P_c(1 + \frac{\gamma - 1}{2})^{\frac{-\gamma}{\gamma - 1}}$$

Again, using the experimental specific heat ratio of 1.2 and a chamber pressure of 1.25MPa gives,

$$P_t = (0.5645)(1.25)$$

$$P_t = 0.7056 \text{ MPa}$$

Now using the second equation that involves the temperatures of the system,

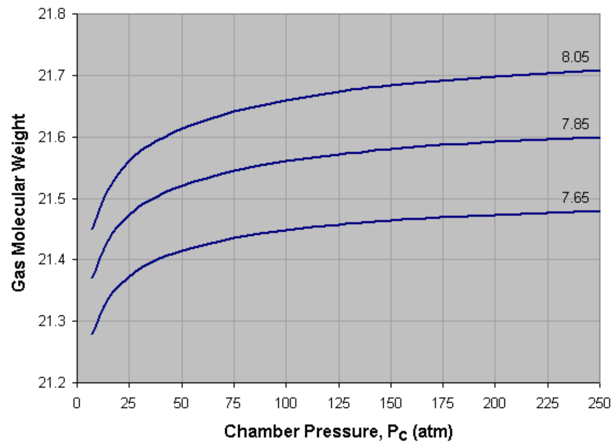
$$T_t = \frac{T_c}{(1 + \frac{\gamma - 1}{2})}$$

Plugging in our specific heat ratio of 1.2, γ, and the experimental temperature of combustion in chamber provided by Wade being 2,975 deg K yields,

$$T_t = \frac{2975}{1.1}$$

$$T_t = 2704.5455 \text{ deg K}$$

Molecular weight of the combustive gases, denoted M, is now found through examination of the chart provided by Braeunig using the selected combustion chamber pressure of 1.25 MPa (12.3365 atm),



Using the fuel mixture ratio of 7.07, an approximate molecular weight of 20 g/mol can be interpolated from the chart.

With these variables now known, along with the universal gas constant, R, and the specific heat ratio, γ, the area of the throat cross-section can now be calculated through the formula provided by Bremer and Risacher,

$$A_T = \frac{w_t}{P_t}\sqrt{\frac{R \cdot T_t}{M \cdot \gamma}}$$

Which gives an A_T of 3.2617 cm^2 for a diameter of throat of 2.0379 cm.

To now define exit nozzle area and thus diameter, we find the exit velocity Mach number of the gaseous products using this formula which relates the chamber pressure, the ambient atmospheric pressure (denoted P_a), and the specific heat ratio,

$$M_e^2 = (\frac{2}{\gamma - 1})\left[(\frac{P_c}{P_a})^{\frac{\gamma-1}{\gamma}} - 1\right]$$

Which when calculated gives an exit velocity of Mach 2.2805.

To find the exit cross-sectional area and thus the expansion ratio of the motor, this formula combines the information of the gas velocity and the details of the combustive products with the throat cross-sectional area,

$$A_e = \left(\frac{A_T}{M_e}\right)\left[\frac{1 + \left(\frac{\gamma - 1}{2}\right)M_e^2}{\frac{\gamma + 1}{2}}\right]^{\left(\frac{\gamma+1}{2\gamma-2}\right)}$$

The resultant exit area is thus 8.4726 cm^2 with a diameter of 3.2845 cm, which leads to an expansion ratio of 2.598.

The final dimension to be then found is the chamber volume. Bremer utilizes a cylindrical combustion chamber design, and thus a polygonal approximation of said cylinder is the path I chose to maximize the strength of the walls. Chamber volume is then determined based on the chosen geometry, the throat cross-sectional area, and the "characteristic length" of the chosen fuel and oxidizer along with their respective injection methods. This last parameter, often denoted L*, is as Bremer states, "often experimentally determined" and thus the chosen value I follow is derived from the experimental work of Khan and Qamar who provide in the attached journal publication this graph that establishes a relationship between the characteristic length and temperature of the combustion chamber (T_c),

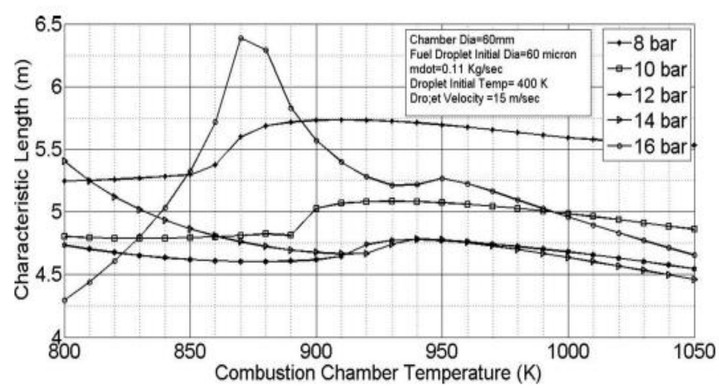


FIG. 13. CHAMBER PRESSURE AND TEMPERATURE EFFECTS ON CHARACTERISTIC LENGTH

If we extrapolate the values to match our combustion chamber temperature of 2975 deg K and our chamber pressure of 1.25 MPa (12.5 bar), a simple regression model calculated yields a value for our characteristic length of 110 cm. The formula provided by both Bremer's work and Khan and Qamar's to relate this characteristic length (L*) to the chamber volume (V_c) and throat area (A_T) is,

$$L* = \frac{V_c}{A_T}$$

When solved for V_c, a combustion chamber volume of 358.787 cm^3 is determined. The cylindrical part of the chamber will have a volume of 304.9690 cm^3, and the converging section of the nozzle-throat assembly will have a volume of 53.8181 cm^3. Then, assuming a 4:1 ratio for the combustion chamber radius to the throat radius and an 85% allocation of volume to the primary cylindrical section, a length of 17.5308 cm for the cylindrical section, and a height of 3.0937 cm for the conical section is derived.

Results and Recommendations

Manufacturing the assembly required a much higher degree of metal knowledge and handling capability than I expected. As a result, fuel testing with the listed fuel and oxidizer will be performed in later studies of this design. The calculated pressures and temperatures involved far exceed what I believe the steel I used is capable of surviving for any significant duration of time and thus safety parameters are what is next in design for testing.

Flow relations presented in the previous section of this report offer an excellent algebraic formulation of how the various sections of the motor - mainly the nozzle, throat, and combustion chamber – will handle combustion. Considering this, I hope to eventually develop either a MATLAB model of the process and product gas flow or find a more calculus-based approach to modeling the flow. The accuracy of this model is far from industrially or scientifically significant, however the entire project was done for marginally less than five hundred dollars, at time of writing, and construction was entirely done in my garage.

I also wish to discuss here briefly that I very heavily relied on existing information and work done by others decades before me with my adding of no measurable progress to the industry. I do feel a keen sense of dissatisfaction with this reality and through eventually fuel testing this design I hope to at least innovate some component of the rocket motor process or find some meaningful performance result to justify my efforts.

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