MAE 150R : Current Topics in Solid Rocket Motor Research

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A review of current research topics in solid rocket motor systems was conducted. A selection of recent papers are discussed as they pertain to their topics of study. More detailed analyses are conducted of several papers whose work touches on basic topics in rocket propulsion. Discussed topics include optimization techniques, dual-thrust systems, propellant additives, burning rate measurement techniques, and nozzle erosion. Our analysis of optimization techniques suggests that while hybrid methods can have certain advantages in computational time, the extreme operating conditions and high costs associated with rocketry make genetic algorithms a more appropriate choice for optimization efforts in this field. Finally we were able to demonstrate the importance of accurate burn rate measurements and nozzle erosion prediction techniques.

I. Introduction

Solid rocket motors (SRMs) represent the oldest form of rocketry and are still in common use today for a variety of purposes. While less efficient than liquid bi-propellant engines, SRMs can provide higher thrust at a lower cost. For this reason, they are commonly used to 'boost' initial launch stages in order to maximize payload capacity. By increasing the initial thrust of the rocket, the boosters are able to minimize gravity losses by helping the rocket to clear the relatively high gravity region at the Earth's surface more quickly. SRMs have also found extensive use in military application for tactical warhead delivery systems due to their ability to be stored for long periods of time.

Solid rocket motors burn a solid grain composed of both the fuel and oxidizer needed for combustion. A commonly used propellant in modern SRMs is ammonium perchlorate composite propellant (APCP) which contains ammonium perchlorate as the oxidizer and a binding agent such as hydroxyl-terminated polybutadiene (HTPB) which also acts as a fuel. Composite propellants such as APCP have found wide use in modern rocketry due to their high performance, elasticity (reduces fracturing), and curing process. HTPB is a liquid resin, so it can be mixed with the aluminum perchlorate and cast in a desired grain geometry with the curative, allowing for easy and repeatable grain production. Aluminum is often added to these propellants to increase performance and stabilize combustion.

Grains are commonly cast as cylinders with a number of 'bores' running through their length where the combustion takes place. Since the thrust produced is directly proportional to the burn area of the grain, many different grain geometries have been developed to provide various thrust profiles. In most rocket applications, it is important to have higher thrust during the launch stage, making a regressive thrust profile highly desirable. Cruciform and double anchor geometries were developed for this purpose, providing relatively smooth regression rates.

Finally, like most rockets, solid rocket motors convert the thermal energy produced during combustion into kinetic energy by directing the combustion products through a converging-diverging nozzle. Unlike bi-propellant systems, SRMs cannot use regenerative cooling, so they rely entirely on ablative cooling to keep their nozzles from overheating. Graphite and carbon-carbon composites are commonly used ablative materials in nozzle construction. While they provide insulation between the combustion products and the nozzle's metallic housing, the harsh conditions in which they operate lead to mechanical and chemical erosion during operation.

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In the following section we will discuss recent studies in nozzle erosion, burning rate measurement, propellant additives, dual-thrust systems, optimization techniques, and combustion stability. All of the papers presented were published in the last couple of years and represent a significant cross-section of the current research being done in solid rocket motor design. Basic analyses of some of the topics are provided in the subsequent section for a more in depth perspective on the research.

II. Discussion

A. Internal Stability Analysis Techniques

Large segmented SRMs such as the Titan IV rocket exhibit pressure bursts due to internal pressure oscillations very near the frequencies associated with axial acoustics. Originally thought to be the result only of acoustic instabilities, numerical models have shown that these are significantly influenced by a phenomenon known as "parietal vortex shedding". Injection induced internal flow dynamics give rise to hydrodynamic instabilities in the mean flow profile which couple with the acoustic modes of the chamber. Standard modal approaches to solve for the frequencies of these instabilities rely on a Taylor-Culick linearization approximation to solve the incompressible Navier-Stokes equations.

With the significant increase in computing resources available to researches in the last century, Chedevergne et al.[1] were able to verify these approximations with direct numerical simulations (DNS) of the internal flow dynamics of SRM combustion chambers. Chedevergne et al. were able to reproduce modes predicted by Taylor-Culick stability analysis through direct numerical simulation of the unsteady motion caused by an isolated fluctuation in the combustion chamber. DNS simulations were also able to verify temporal damping of the eigenmodes and axial amplification of the eigenfunctions predicted by such stability analysis. In addition, secondary eigenmodes were observed in the DNS simulations resulting from the coupling of the acoustic modes with nearby eigenmodes associated with hydrodynamic instability. Chedevergne et al. suggest that these nonlinear effects may be responsible for the temporally unstable modes observed in longer rocket combustion chambers.

Boyer et al.[2] recently showed that these approximations are highly sensitive to numerical disturbances at the top end of the combustion chamber. This combined with axial amplification of the eigenfunctions for longer chambers makes the resulting modes highly dependent on axial discretization. As the axial discretization is refined, the eigenmodes fail to converge to discrete values. Boyer et al. suggest that these modes can therefore be reinterpreted as pseudo-modes that can be used to verify the stability equations and analyze the global stability of larger SRMs. By introducing a small defect in the grain however, Boyer et al. were able to show that the same modal analysis could be used to predict the discrete modes that result from grain segmentation or the development of a crack. Since the defect occurs downstream from the injection point, it does not suffer from the same sensitivity to injection point disturbances, resulting in eigenmodes that are fully independent of axial discretization.

B. Standard Optimization Techniques

Traditionally, vehicle design and optimization has been based on a trial-and-error approach, relying heavily on the experience of engineers to develop optimal designs and then testing their performance with slow computational models. Combined optimization techniques have been developed using simplified computer models for their decreased computation times, however, the complex dynamics ignored by these simple models reduces the reliability of their results. Recent advances in computing resources have made the development of more advanced optimization routines more advantageous.

Yumusak[3] developed a routine to optimize combustion chamber and nozzle design by modeling viscous, axis-symmetric flow and applying a gradient-based numerical optimization method. This approach requires modeling the dynamics under a variety of conditions in order to determine the local slopes of performance characteristics with regard to design variables. Once these slopes are determined locally, an adaptive stepping routine is employed to locate the overall maximum in performance criterion. Yumusak's routine successfully optimized the nozzle to the standard bell shape, but with additional parameters, such a tool could be used to predict the optimum shape including manufacturing costs. The geometry of the combustion chamber was also optimized, and given a specific thrust requirement, the overall geometry of the combustion chamber and nozzle could be tailored to the needs of a given mission. While the basic design principles of SRMs are already well understood, the development of more advanced optimization tools such as this one will help to make the design process more efficient and improve the efficiency and reliability of SRMs in the future.

C. Developments in Dual-Thrust Systems

Dual-thrust systems were developed for vehicles that require an initial high "boost thrust" to achieve a high velocity in a short period of time followed by a lower "sustain thrust" designed to maintain a constant vehicle velocity for the remainder of the burn. In order to achieve dual-phase thrust in a solid rocket motor, special grain geometries must be developed. A commonly used geometry for this purpose is the wagon-wheel grain profile which uses "spokes" of grain to provide two distinct burn phases. Initially, the spokes provide a high burn area resulting in relatively high levels of thrust, however, once the spokes have burned away, the effective burn area drops suddenly, resulting in a lower average thrust throughout the second phase of the burn. While design and optimization techniques for single-phase SRMs are relatively well developed, dual-thrust SRMs have received comparatively little attention.

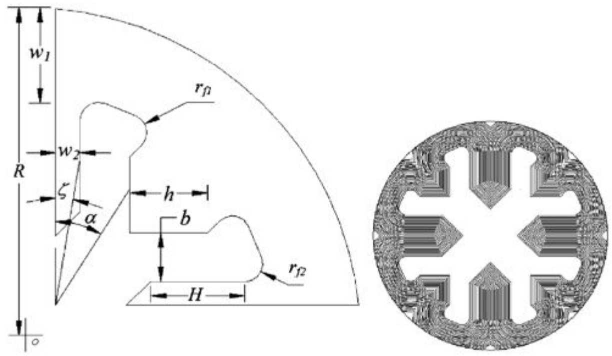

Figure 1. Wagon wheel grain geometry parameters and surface regression.[4]

Raza and Liang[4] recently developed an optimization technique for the development of three-dimensional wagon-wheel grains given a desired dual-phase burn profile. The wagon-wheel grain geometry can be parameterized by a number of design variables including number and width of "spokes" present (see Figure 1). Raza and Liang developed a hybrid optimization technique that combined a global genetic algorithm with local simulated annealing. The genetic algorithm creates a population of candidate solutions with randomly assigned values of design variables and rates them on performance and compatibility with given constraints. Parameters of successful candidates are recombined and subjected to random "mutations" in order to create a new population of candidate solutions. Repetition of this procedure eventually produces optimized values of the original design variables. Raza and Liang enhanced this process with simulated annealing which uses pre-defined search criterion to further refine the candidate population with each iteration. In addition to showing good optimization with this technique, Raza and Liang were able to analyze the sensitivity of thrust performance to various design parameters. The number of "spokes" proved to be a particularly sensitive parameter as an increased number of "spokes" both increases the effectiveness of the boost phase while simultaneously decreasing the overall specific impulse.

Another form of dual-thrust system developed primarily for military applications is the dual-pulse SRM. This type of SRM uses the first pulse during the launch phase and then a delayed second pulse for high maneuverability and speed during an "attack" phase. In order to maintain compactness and aerodynamic stability of the design, the dual-pulse system is achieved by the addition of a second combustion chamber stacked upon the first and separated by a pulse separation device (PSD) as shown in Figure 2. This allows the second pulse to utilize the same nozzle as the first pulse. In order to determine potential losses due to second phase flow through the first pulse chamber, Javed et al.[5] performed a numerical simulation of the second phase flow dynamics. They found that nearly 60% of the overall pressure drop occurred as a result of flow through the PSD and first pulse chamber, leading to a significant reduction in efficiency. In order to determine the precise trade-offs between this design system and other potential dual-pulse designs, further numerical models would have to be performed, but the research conducted by Javed et al. does provide some insight into the efficiency of the second pulse phase for such a dual-pulse system.

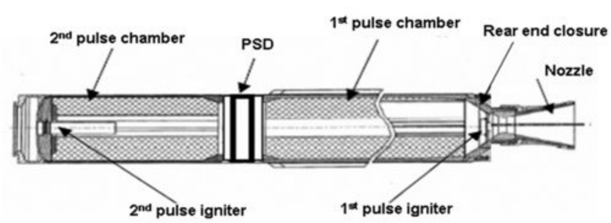

Figure 2. Diagram of a dual-pulse rocket motor.[5]

D. Propellant Additives

Propellant additives, such as aluminum powder, are commonly used to increase the specific impulse of SRMs. Until recently however, it was far too computationally expensive to attempt to model the specific effect these additives have on the dynamics of the combustion chamber. Doisneau et al.[6] have been working on a way to model these effects efficiently and accurately with access to modern computing resources and massively parallel techniques. They developed a model for non-evaporative sprays in a gaseous environment that models the sprays as groups of point particles with exchanges of mass, momentum, and heat governed by statistical properties. The particle spectrum is divided into sections based on relative particle size and the distribution of particle sizes in each section is based on a two-coefficient exponential approximation. Doisneau et al. were able to predict the hydrodynamic and acoustic instabilities of such an SRM with relatively few sections, allowing for relatively fast modeling of these flow dynamics. While there are a number of effects that have yet to be implemented in this model, including aluminum combustion and evaporation, this represents an important step forward in the modeling of SRMs with propellant additives. Doisneau et al. also suggest that the addition of secondary break up modeling in the nozzle would allow for accurate predictions of specific impulse, which could significantly improve the implementation of propellant additives in SRMs.

E. Burning Rate Measurement Techniques

SRM performance is extremely sensitive to propellant burning rate. In order to accurately predict the behavior of any given rocket, it is important to have an accurate prediction of its propellant burning rate. The inherent error involved in batch mixing procedures prevents the reproduction of propellants with sufficiently identical burning rates, requiring each batch to be measured independently. Currently these measurements are conducted via the use of ballistic evaluation motors (BEM). However, multiple BEMs must be tested in order to determine burning rate pressure dependence and significant man-power is required in the preparation of these devices. In addition, a dedicated test facility is required to make these measurements and they are inherently prone to a variety of errors.

Jeenu et al.[7] recently demonstrated the utility of a pulse-echo ultrasonic technique for burning-rate measurements in industry. An ultrasonic transducer is used to make continuous measurements of the location of the propellant surface. By simultaneously measuring chamber pressure, the burning-rate/pressure dependence is able to be determined from a single test. Jeenu et al. demonstrated several other advantages that this method has over BEMs including significantly reduced man-hour requirements and a higher level of accuracy. The standard deviations for measurements with this method were consistently lower, sometimes less than half that of those reported for BEMs. Furthermore, only a small augmentation factor, $\leq 2\%$, was required for prediction of full-scale SRM burning rates. While BEMs do have several advantages over the pulse-echo ultrasonic technique, including the ability to simultaneously measure characteristic velocity and specific impulse, the work by Jeenu et al. represents an important step towards more accurate and more efficient SRM production methods.

F. Nozzle Erosion

As measurement techniques for SRM performance have grown more precise, additional factors that influence this performance have had to be taken in to account. The increase in nozzle surface area due to erosion can have a significant effect on overall performance, but it is only recently that we have had the tools out our disposal to model nozzle-surface recession rate. Thakre et al.[8] developed a model for the erosion of graphite nozzles due to the flow of composite propellants. Non-metalized propellants such as ammonium perchlorate or HTPB cause chemical erosion through heterogeneous reactions with oxygen-containing combustion products, but propellants containing aluminum additives can add a mechanical component due to impingement of

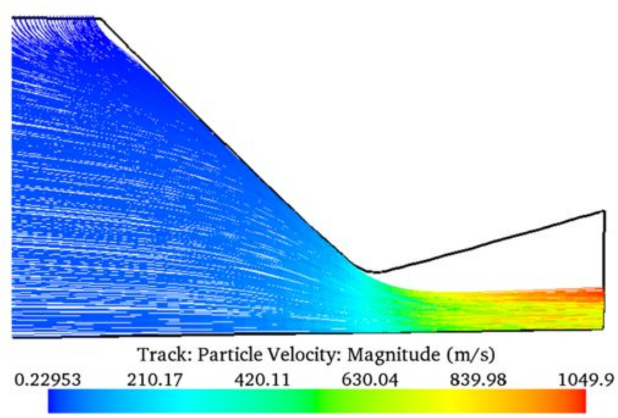

Figure 3. Trajectories of aluminum and aluminum-oxide particles through a rocket nozzle.[8]

condensed-phase aluminum-oxide on the nozzle surface. Thakre et al. model both of the effects through a combined Eulerian-Lagrangian approach that treats the aluminum spray using an imposed Lagrangian framework similar to the work of Doisneau et al.[6] Thakre et al. were able to demonstrate that no mechanical erosion occurs downstream of the throat as particulate matter in this region of the nozzle consistently moves away from the nozzle surface (see Figure 3). Furthermore, the model was able to predict surface erosion rates as they increased up to the point of the throat which corresponded well with experimental data. Inclusion of this data in a model with a dynamically modeled nozzle geometry could potentially lead to a more accurate model of SRM performance.

Bianchi and Nasuti[9] recently conducted a similar analysis of just the chemical erosion process in carbon-carbon nozzles. A linear dependence on chamber pressure was shown for the erosion rate at standard operating pressures of most SRMs. Given this, Bianchi and Nasuti were able to predict the final nozzle shape simply from the mean chamber pressure which correlated well with experimental results. Bianchi and Nasuti went one step further by modeling the flow dynamics as the nozzle surface was eroded. They were able to show that throat erosion was underestimated by as much as 10% when the effects of nozzle erosion were not incorporated into a dynamic nozzle geometry during the simulation. With SRM performance models getting more and more refined, Bianchi and Nasuti have demonstrated the necessity for a model with a dynamically varying nozzle geometry based on the effects of both chemical and mechanical erosion.

III. Analysis

A. Hybrid Optimization Techniques

In Raza and Liang's[4] study of optimization techniques for dual-thrust systems utilizing a wagon-wheel grain cross-section, they compared the results for a simple genetic algorithm (GA), a simulated annealing method (SA), and their combined approach (GASA). In each case the total impulse achieved, I_T, is defined as

$$I_T = m_p \, C_{\mathscr{F}} \, c^*. \tag{1}$$

We can therefore use our standard rocket relations to write the specific impulse as

$$I_{sp} = \frac{c}{g_E} = \frac{C_{\mathscr{F}} \, c^*}{g_E} = \frac{I_T}{g_E \, m_p}. \tag{2}$$

Raza and Liang presented the total impulse and propellant mass achieved by each technique. We can determine their respective specific impulses such that

$$\begin{aligned} \text{GA} &: I_{sp} = 241.42 \text{ s} \\ \text{SA} &: I_{sp} = 238.08 \text{ s} \\ \text{GASA} &: I_{sp} = 235.67 \text{ s} \end{aligned}$$

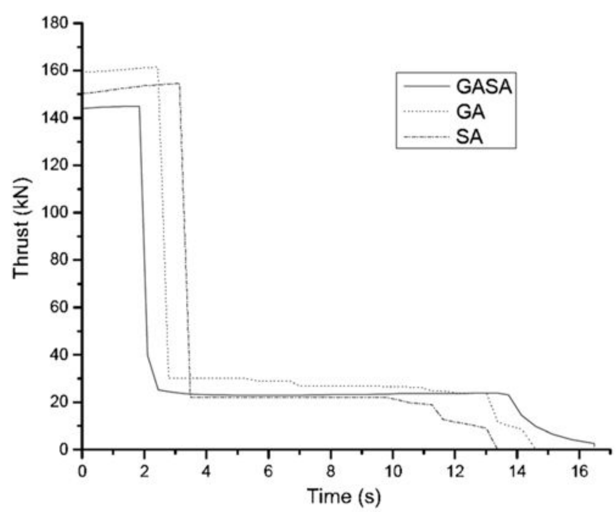

Figure 4. Thrust profiles for dual-thrust system with wagon-wheel grain cross section as determined by three distinct optimization techniques.[4]

Figure 4 shows the thrust profiles for the three techniques. Raza and Liang did not present the computational times for each technique, but they argue that the proposed GASA technique is advantageous for significantly reduced processing time. The GASA approach produces a specific impulse 2.38% lower than the best specific impulse achieved by the genetic algorithm alone. We can get an idea of the consequences of this reduction by analyzing the simplified rocket equation,

$$\Delta v = g_E \, I_{sp} \, \ln\left[\frac{m_0}{m_f}\right]. \tag{3}$$

For a given structural and payload mass, a 2.35% increase in initial rocket mass would be required in order to achieve the same Δv. Given the high cost of launch, it is likely that even this small increase in initial mass would not be worth the savings in computational time achieved by the GASA approach. While such optimization techniques are still quite advantageous, a simple genetic algorithm appears to be the best approach for this application.

B. Burning Rate Measurements

In the study by Jeenu et al.,[7] burning rate measurements are provided for a variety of propellant types. 18% aluminized hydroxyl-terminated polybutadiene (HTPB) containing 65% ammonium perchlorate (AP) and 3% oxamide is a slow-burning propellant used in large booster-stage rockets. The densities of selected components are approximately[10]

$$\begin{aligned} \rho_{HTPB} &\approx 0.93 \times 10^3 \ \mathrm{kg \cdot m^{-3}}, \\ \rho_{Al} &\approx 2.70 \times 10^3 \ \mathrm{kg \cdot m^{-3}}, \\ \rho_{AP} &\approx 1.95 \times 10^3 \ \mathrm{kg \cdot m^{-3}}. \end{aligned}$$

The approximate propellant density, ρ_p, of the grain then is approximately

$$\begin{aligned} \rho_p &\approx \frac{0.14\ \rho_{HTPB} + 0.18\ \rho_{Al} + 0.65\ \rho_{AP}}{0.97}, \\ &\approx 1.94 \times 10^3 \ \mathrm{kg \cdot m^{-3}}. \end{aligned}$$

The propellant burn rate, r_b, is related to the chamber pressure, p_c, by the formula,

$$r_b = a\ p_c^n. \tag{4}$$

Figure 5 shows the burning rate data for this propellant. The chamber pressure increases approximately linearly over the period of the burn such that the majority of the burn takes place over the right-most portion of the plot. The approximate value of the pressure exponent, n, for the majority of the burn period is therefore $n \approx 0.24$. We can determine the value of the burn rate coefficient, a, from Figure 5 such that $a \approx 1.048 \times 10^{-4}$. For a quasi-steady flow approximation, we can represent the chamber pressure as

$$p_c = \left(\frac{a\ \rho_p\ A_b\ c^*}{A_t} \right)^{1/(1-n)} \tag{5}$$

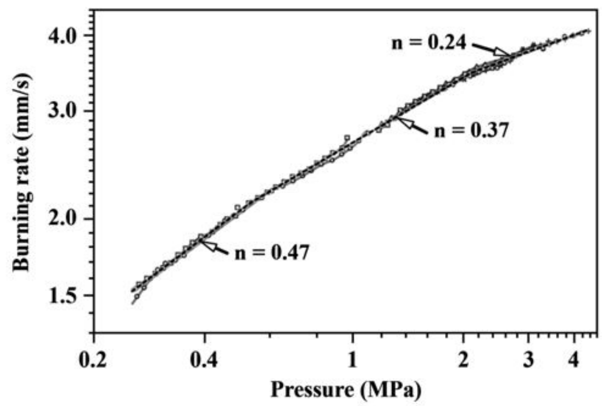

Figure 5. Burning rates computed for small pressure intervals using the ultrasonic measurement technique presented by Jeenu et al.[7]

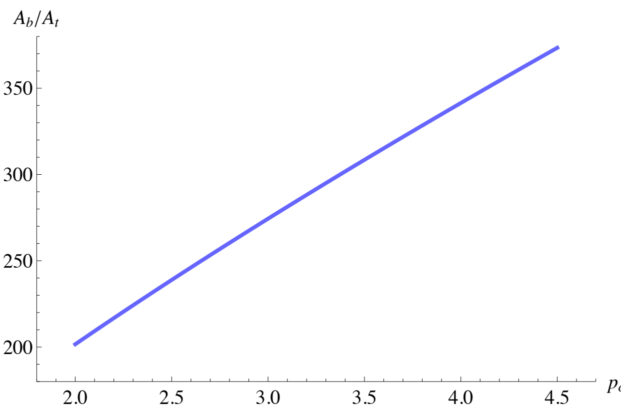

Figure 6. Area ratio as a function of chamber pressure as predicted by a quasi-steady analysis of the ultrasonic data presented by Jeenu et al.

If we assume a characteristic velocity of $c^* = 1500 \ \mathrm{m \cdot s^{-1}}$, we can plot the area ratio as a function of time. Figure 6 shows this relationship over the pertinent range of chamber pressures. Since the throat area remains fixed and the chamber pressure increases linearly over time, it is clear from this plot that the burn area also increases linearly over time for the majority of the burn period of such a thruster.

C. Effects of Nozzle Erosion

From the data obtained by Bianchi and Nasuti,[9] the approximate nozzle erosion rate due to chemical processes at the mean combustion temperature for a carbon-carbon nozzle is

$$\dot{r}_{chem} \approx 0.19 \ \mathrm{mm \cdot s^{-1}}.$$

Assuming similar erosion rates between carbon-carbon and graphite nozzles, we can compare this to the data obtained by Thakre et al.[8] for mechanical erosion of graphite nozzles. For a nozzle-material density of $\rho_n = 1.8 \ \mathrm{g \cdot cm^{-3}}$, the maximum mass erosion rate due to mechanical processes may be written as

$$\dot{m}_{mech} \approx 0.06 \ \mathrm{kg \cdot m^{-2} \cdot s^{-1}}.$$

This maximum occurs just behind the nozzle throat however as the erosion rate at the nozzle throat is zero. However, the erosion behind the throat will essentially erode the throat as well. For our purposes, we will assume the effective erosion at the throat is approximately half the maximum erosion rate behind the throat. We can calculate the spatial erosion rate at the throat of such a nozzle then such that

$$\dot{r}_{mech} \approx \frac{\dot{m}_{mech}}{2\ \rho_n} = 0.017\ \mathrm{mm} \cdot \mathrm{s}^{-1}. \tag{6}$$

The effects of mechanical erosion at the nozzle are approximately an order of magnitude less than those of chemical erosion. However, taking both into account, we can determine the approximate change in throat area as a function of the burn time. For an initial throat radius of $r_0 = 10$ mm, the final throat area may be written as

$$A_t = \pi\ (r_0 + \dot{r}\ t_b)^2. \tag{7}$$

where $\dot{r} = \dot{r}_{chem} + \dot{r}_{mech}$ represents the combined erosion rate and t_b is the total burn time. As long as flow remains choked at the throat, the increase in throat area will cause a proportional increase in mass flow rate, such that

$$\dot{m} = \dot{m}_0 \left(1 + \frac{\dot{r}\ t_b}{r_0}\right)^2. \tag{8}$$

Chemical erosion at the throat is much higher than at the exit plane however, and since no mechanical erosion takes place in the expansion portion of the nozzle, the exit-to-throat area ratio will change.

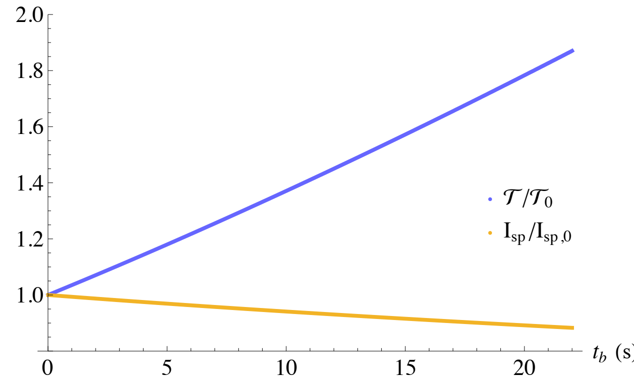

Figure 7. Non-dimensional thrust and specific impulse plotted as a function of time due to nozzle erosion at the throat.

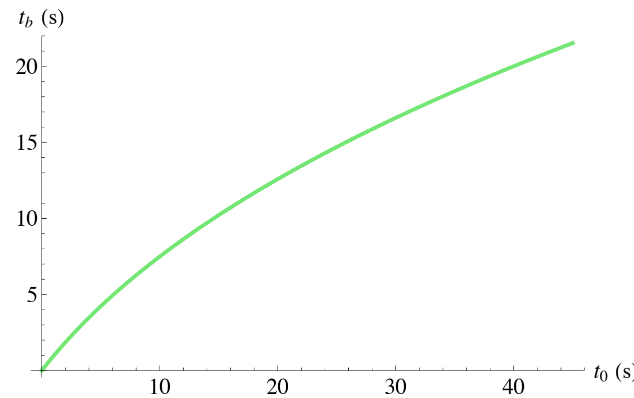

Figure 8. Total burn time as a function of ideal burn time for a nozzle experiencing erosion at the throat.

Assuming relatively constant exit area, A_e, the exit Mach number, M_e, will be affected such that

$$\frac{A_e}{A_t} = \frac{1}{M_e}\left[\frac{2}{\gamma+1}\left(1 + \frac{\gamma-1}{2}\ M_e^2\right)\right]^{\frac{\gamma+1}{2(\gamma-1)}}, \tag{9}$$

where γ represents the specific heat ratio of the combustion products. As the throat area increases, the efficiency of the nozzle decreases, resulting in a lower exit velocity and lower specific impulse. The thrust produced is proportional to both the mass flow rate and the exit velocity however, resulting in a net increase in thrust. The Zefiro 9 rocket geometry used by Bianchi and Nasuti has an area ratio of 56 and burns an HTPB composite propellant for which we will approximate its specific heat ratio at $\gamma = 1.2$. Figure 7 plots the resulting drop in specific impulse along with the increase in thrust over time for the Zefiro 9 rocket.

While the increased mass flow rate does increase the thrust, it also reduces the overall burn time for a given quantity of propellant, such that

$$t_b = t_0 \left(1 + \frac{\dot{r}\ t_b}{r_0}\right)^{-2}. \tag{10}$$

where t_0 is the ideal burn time of the rocket with no erosion. Figure 8 shows the resulting burn time for a given ideal burn time. As long as the flow remains chocked at the throat, the increased thrust directly counterbalances the decreased burn time resulting in no loss of Δv. However, the drop in specific impulse does affect the potential Δv of the mission.

IV. Conclusions

While many aspects of solid rocket motor design have been well researched over the past century, there are still many active areas of research and development related to SRMs in the field today. Automated optimization methods such as the one developed by Yumusak[3] have the potential to greatly reduce time spent in the design phase and maximize ultimate rocket performance. Our analysis of the study by Raza and Liang[4] suggests that genetic algorithms have the greatest potential for future developments in this area.

An increase in available computational resources has also lead to more efforts in precise performance prediction methods for SRMs. Chedevergne et al.[1] and Boyer et al.[2] were able to model the instabilities that occur in the combustion chamber and predict their normal and secondary eigenmodes. Doisneau et al.[6] developed another model that similarly accounted for combustion instabilities that occur as a result of additives in SRM composite propellants. Such models could be used for more accurate thrust calculations in the future as well as potentially allowing for better optimization with respect to propellant composition ratios and aiding in the development of more stable chamber designs.

Dual-thrust systems are another active area of development. Raza and Liang's[4] study of the wagon-wheel grain geometry provided a method of optimizing dual-thrust systems for a desired mission thrust profile. A study of the efficiency of alternative dual-pulse systems was conducted by Javed et al.,[5] showing reduced efficiency due to flow losses though the pulse separation device (PSD). Continued research in this area could lead to improved PSD design and increased dual-pulse system efficiency.

The study conducted by Jeenu et al.[7] demonstrated the effectiveness of the pulse-echo ultrasonic technique for burn-rate measurement in industry. Adoption of this technique could lead to significant savings in both the cost and time required for the development and production of new SRMs. Our analysis demonstrated the importance of accurate burn-rate measurements. By measuring the burn rate, we can predict the burn area throughout the burn, which is directly proportional to the thrust produced by the rocket. Since grain compositions vary slightly with every batch mix, it is vital to test every batch for effective burn rate in order to accurately predict the thrust that the rocket will produce.

Finally, Thakre et al.[8] and Bianchi and Nasuti[9] developed models for nozzle erosion occurring as a result of mechanical and chemical processes respectively. Thakre et al. were able to show that no mechanical erosion due to propellant additives occurs past the nozzle throat. Our own analysis showed that erosion due to mechanical processes is approximately an order of magnitude less than erosion due to chemical processes. We also showed that erosion can lead to a significant increase in thrust and comparably reduced burn times. The affect on specific impulse was less drastic, but still significant, demonstrating the need to take nozzle erosion into account for accurate prediction of the Δv potential.

Solid rocket motors are used frequently today for small payloads, booster stages for larger payloads, as well as tactical missile applications. The high cost of launching vehicles into orbit makes continued research and development of SRMs extremely important for the aerospace industry. As research in this area continues, we can expect to see better optimization techniques increase the efficiency of the design process, better grain and rocket testing methods improving production processes, and more advanced computer models increasing the accuracy of overall performance prediction, ultimately leading to improved rocket control methods and design.

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