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Methods of increasing volumetric specific impulse in HTPB based hybrid rocket motors

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ABSTRACT

This study covers the investigation and preliminary testing of fuel formulations for hybrid rocket motors utilizing nitrous oxide as the oxidizer, with the primary objective of enhancing volumetric specific impulse (Isp * ρ). Conventional HTPB based fuels, while reliable and widely used, often face limitations in energy density and regression rates. To address these challenges, a series of fuel blends incorporating energetic additives, metal powders, and halogen donors were formulated and partially tested. Experimental and computational analyses were conducted to evaluate combustion performance, exhaust gas properties, and propellant density for these formulations. Results demonstrated significant improvements in volumetric specific impulse, with selected formulations achieving up to a 10% increase in delivered volumetric impulse compared to the chosen theoretical baseline. The study also explores the tradeoffs between performance gains, ease of manufacturing and safety.

Keywords: HTPB hybrid rocket motors, volumetric specific impulse, combustion efficiency, propellant formulation.

INTRODUCTION

Hybrid rocket propulsion systems have gained increasing attention in recent years due to their unique combination of safety, simplicity, and relatively low cost [1]. Positioned between solid and liquid propulsion technologies, hybrid rockets offer a compelling alternative for various aerospace applications, particularly in the domain of low-cost, suborbital launch systems and green propulsion. Among the most commonly used oxidizers in small to medium scale hybrid propulsion systems is nitrous oxide, valued for its self-pressurizing nature and ease of storage and handling. On the fuel side, hydroxyl terminated polybutadiene remains a popular choice due to its mechanical properties, processability, and historical performance data. However, despite its advantages, conventional HTPB based fuels exhibit limitations in energy density, regression rates and fairly high optimal oxidizer to fuel ratios, all of which restrict the achievable performance compared to traditional solid rocket motors of similar class. Especially, the volumetric specific impulse is a key parameter for systems constrained mostly by volume, as smaller rockets will experience less aeroelastic phenomena such as flutter or aerodynamic bending, both of which are fatal to the [2].

In response to these limitations, our research has turned to advanced fuel formulations that incorporate energetic additives, metal powders, halogen donors and metal oxides to improve combustion characteristics and overall propulsion performance. These changes are intended to increase the heat of reaction, modify the overall exhaust composition and enhance the density of the fuel, thereby directly contributing to higher specific and volumetric specific impulse values. However, the integration of such additives introduces new challenges related to manufacturability, safety and combustion behavior, necessitating a careful balance between performance enhancement and operational feasibility.

This study focuses on the development and preliminary testing of two HTPB based fuel formulations designed specifically for use with nitrous oxide as the oxidizer. Through a combination of computational modeling and further basic experimental testing, two main formulations were evaluated for their combustion performance and physical properties such as propellant density and pourability. The objective was to identify promising candidates that could offer meaningful improvements in both standard and volumetric specific impulse while maintaining manageable levels of complexity.

STUDY METHODOLOGY

Thermochemical simulations

Nitrous oxide was chosen as the oxidizer for this study due to its widespread use in small-scale suborbital launch systems and amateur rockets, particularly in Europe, where it has proven to be an effective and reliable choice for hybrid rocket [3]. Its favorable self-pressurizing properties make it particularly suitable for compact propulsion systems [4], offering significant advantages in terms of simplicity and cost-effectiveness. Additionally, nitrous oxide's decent specific impulse and ease of storage under moderate pressure further contribute to its selection as the oxidizer of choice for suborbital missions and experimental applications.

To identify initial fuel formulations for enhancing both specific impulse and volumetric specific impulse in hybrid rocket motors using nitrous oxide, a series of thermochemical simulations were conducted. The primary objective was to establish two baseline fuel compositions for experimental evaluation, leveraging theoretical performance data to guide mixture ratios. The research focused mainly on improving the volume specific impulse, which is crucial in volume-limited applications such as small suborbital rockets where the main limitation is the possible size of the vehicle. A potential slight reduction in specific impulse was consciously accepted, as in this type of systems the higher energy density allows for a reduction in the total size of the structure and thus a final reduction in the required mass of the entire system due to smaller aerodynamic stress. Future experiments are planned with other additive ratios that may allow to get increased performance in both metrics without significantly sacrificing one at the expense of the other. Thermochemical calculations were performed using a combination of NASA CEA [5] and AFAL programs [6]. These tools allow detailed modeling of combustion thermodynamics under equilibrium conditions. Two initial formulations named "Hybrid Propellant Experiment A/B" were defined based on AFAL performance optimization analysis and practical considerations such as cost and ease of processing (Table 1). Formulation A (HyPEx-A): A mixture of HTPB binder, aluminum powder and parlon. Aluminum serves to increase the heat of combustion while parlon contributes chlorine to improve energy release from aluminium and affect flame chemistry in similar way to solid rocket [7]. Formulation B (HyPEx-B): Based on Formulation A with the addition of bismuth trioxide which is commonly used in solid rocket motors to increase volumetric specific impulse [8]. Bi_2O_3 acts as a condensed-phase oxidizer [9] while also increasing fuel density. This combination promises more stable combustion with even lower O/F ratios (Figure 3). Each formulation was evaluated over a range of chamber pressures and O/F ratios corresponding to nominal hybrid rocket motor working conditions. Key outputs included flame temperature, theoretical specific impulse and total propellant density including oxidizer. Volumetric specific impulse was obtained from standard Isp using equation:

$$Isp_{vol} = Isp \cdot \rho \quad (1)$$

where: ρ is fuel density (Figures 1–3).

The inclusion of high density additives like Bi_2O_3 was found to significantly improve the volumetric specific impulse without compromising Isp, while also offering potential benefits for combustion stability due to lower susceptibility to oxidizer mass flow changes. These simulation outcomes directly informed the selection of the two candidate formulations for subsequent manufacturing and hot-fire testing phases.

Table 1. Fuel composition and density

Ingredient	HyPEx-A	HyPEx-B
HTPB binder	59.8%	48.8%
Parlon	8%	6%
Aluminium	32%	30%
Additional additives	0.2%	0.2%
Bi_2O_3	-	15%
Theoretical density	1504 $\frac{kg}{m^3}$	2680 $\frac{kg}{m^3}$

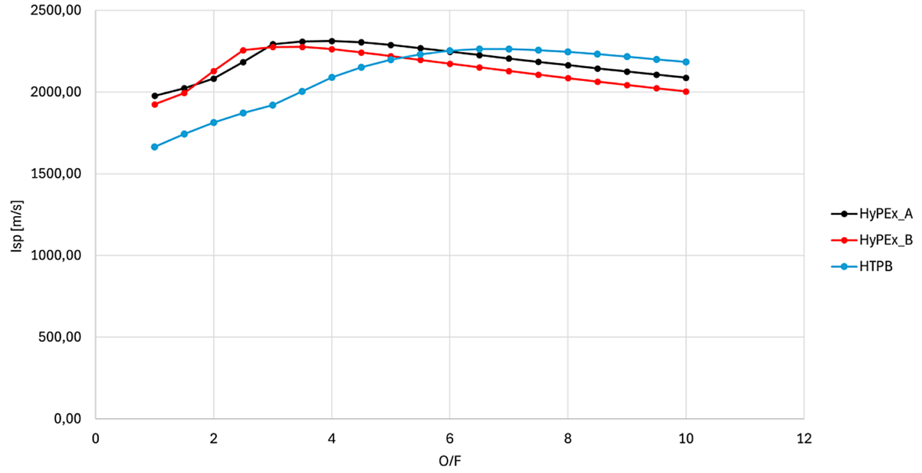

Figure 1. Comparison of specific impulse between formulations

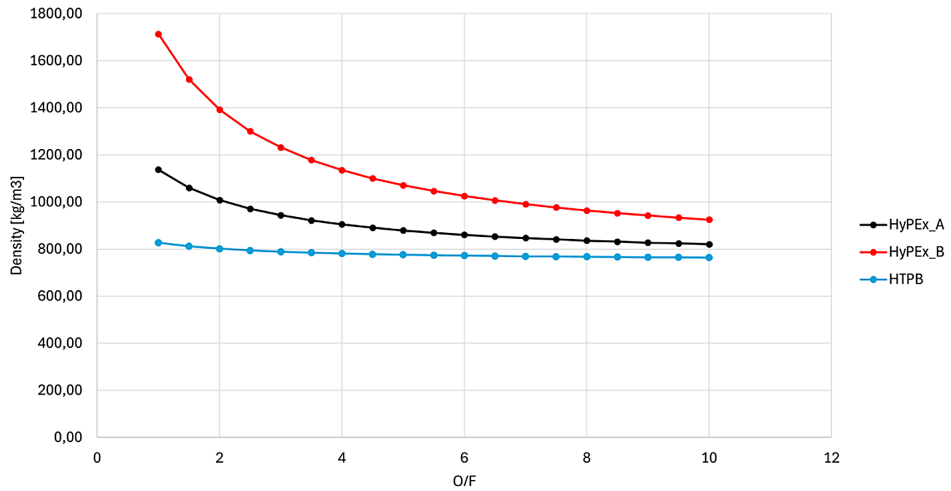

Figure 2. Comparison of propellant (fuel and oxidizer) density between formulations

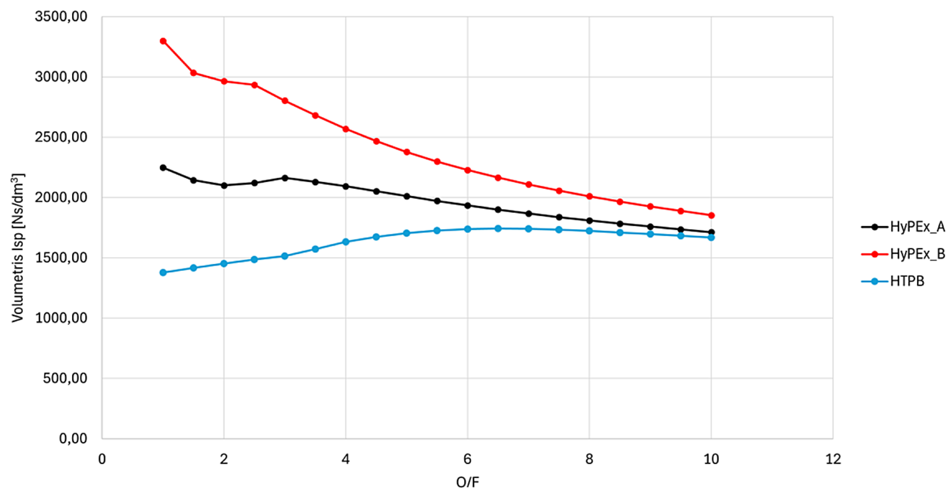

Figure 3. Comparison of volumetric specific impulse between formulations

Motor simulations

In order to conduct hot fire testing of selected fuels, a small scale hybrid motor was designed and manufactured. Critical motor parameters such as injector size and nozzle throat diameter were analyzed and optimized using the Hybrid Rocket Analysis Program developed by Nickel [10] and based on Richard Newland's tank model [11]. It was employed to validate the motor configuration and eventually assess motor performance.

Initial regression modeling was performed using estimated fuel regression rates. Due to high dependence of regression rate on combustion chamber design, the constant O/F method was used. Given selected fuel grain dimensions, such length was chosen so after burnout at optimal O/F web thickness is no less than 3 millimeters to accommodate increased regression zones.

This data was enough to perform numerical simulations which provided estimates of peak pressure and total impulse over burn duration. These results informed nozzle throat sizing and initial injector diameter (Figure 4, Table 2).

Simulation outputs confirmed that both formulations could operate within the chamber pressure and flow limits of the test system.

Table 2. Important motor parameters

Parameter	Value	Unit
Oxidizer tank volume	800	cm^3
Fuel grain outer diameter	38	mm
Fuel grain inner diameter	20	mm
Fuel grain inner length	240	mm
Injector diameter	3.3	mm
Injector C_d	0.56	-
Nozzle throat diameter	8	mm

Technological methodology

To facilitate the transition from theoretical analysis to experimental validation, a dedicated testbed motor was developed based on a modified off-the-shelf design and simulation results. This approach enabled rapid and controlled progress while minimizing development time and associated costs. The selected configuration provided sufficient flexibility to accommodate various fuel variations and was compatible with the available oxidizer delivery and instrumentation systems, making it ideal for iterative performance evaluation (Figure 5).

To improve combustion efficiency at small scale, two mixer plates were added (Figure 6). Their role is to promote mixing of unburnt reactants and thus increase combustion efficiency [12].

Table 3. Important test parameter

Parameter	Value	Unit
Oxidizer mass	0.459	kg
Initial fuel mass	0.321	kg
Oxidizer initial temperature	289	K
Final fuel mass	0.175	kg

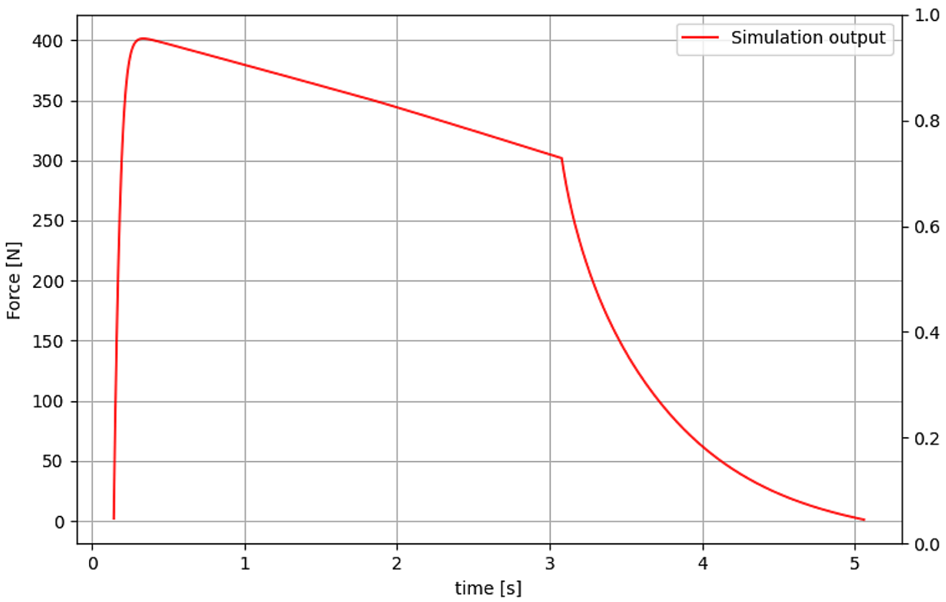

Figure 4. Simulated thrust curve obtained from HRAP

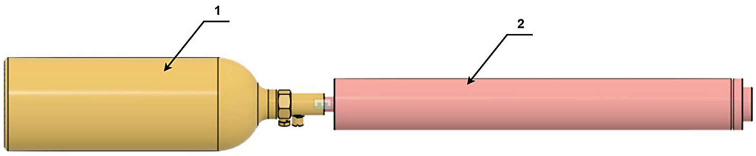

Figure 5. Motor assembly: 1 – oxidizer tank, 2 – combustion chamber

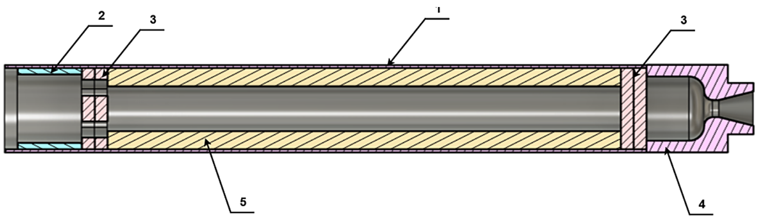

Figure 6. Combustion chamber assembly: 1 – phenolic insulator, 2 – additional insulating material, 3 – mixer plates, 4 – nozzle, 5 – fuel grain

Figure 7. Measured casting ingredients

The fuel formulation was mixed and cast directly into xx-grade phenolic tubes, which functioned both as casting molds and as integrated thermal liners for the combustion chamber, providing insulation during motor operation (Figure 7).

RESULTS AND DISCUSSION

Postprocessing of the experimental data was carried out using the HRAP program in combination with custom-developed Python scripts, enabling direct comparison between simulation results and hot-fire test data. By analyzing the measured thrust curves alongside determined average oxidizer to fuel ratios, it was possible to estimate fuel regression coefficients and adjust simulation parameters for improved accuracy in further tests. This iterative fitting process helped align the simulation outputs with observed motor performance, enhancing the reliability of predictive modeling for subsequent design iterations.

Two static tests were conducted. The first test yielded promising initial results, however due to a valve malfunction, the recorded burn duration was limited to only 0.8 seconds. Consequently, the data from this test were deemed unreliable and excluded from further analysis. The second test was carried out using the motor configuration described in the previous section (Figure 8). Key parameters, including the initial oxidizer mass, fuel mass, and tank temperature, were recorded for following evaluation (Table 4). The test proceeded without anomalies and provided a more complete burn profile. This data will serve as a basis for assessing the motor's performance and validating the design assumptions.

Thermophysical properties of the oxidizer were estimated using temperature–pressure

Table 4. Derived operating conditions

Parameter	Value	Unit
Burn time	4.9	s
Oxidizer pressure	4510	kPa
Oxidizer density	822	$\frac{kg}{m^3}$
Final port diameter (average)	30.9	mm
Regression rate (average)	0.99	$\frac{mm}{s}$
Average mass flow ($\dot{m}$)	0.0918	$\frac{kg}{s}$
Average oxidizer mass flux (G_{ox})	173	$\frac{kg}{m^2 * s}$

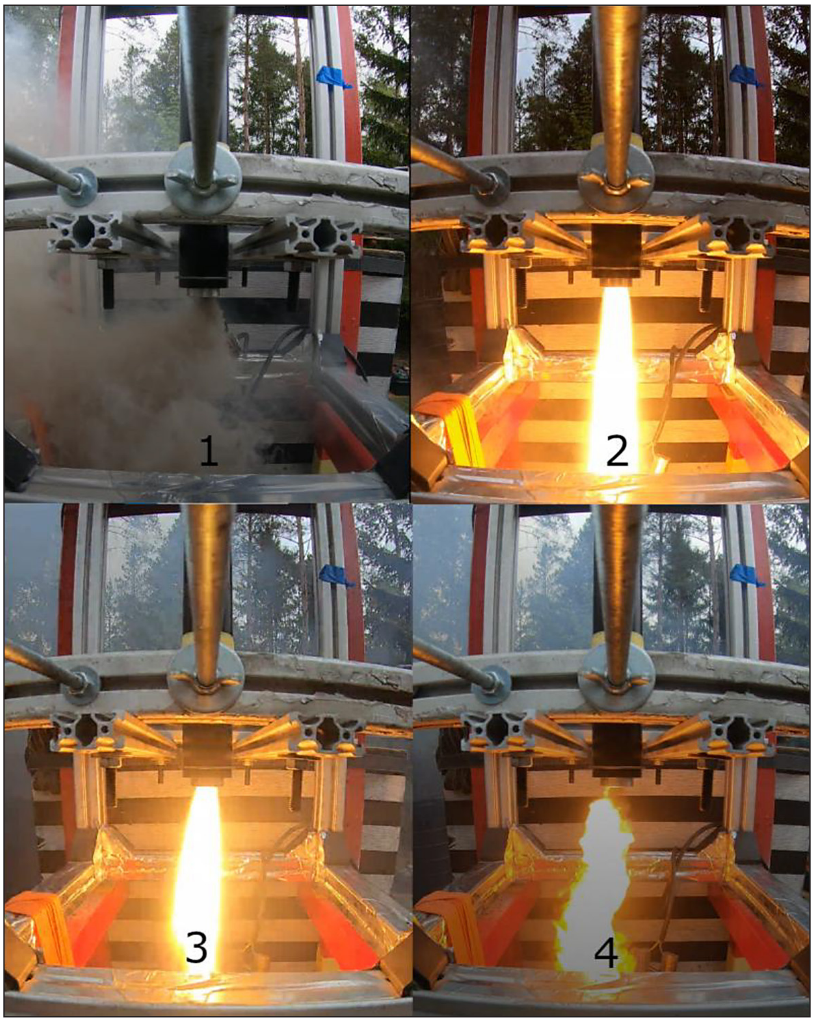

Figure 8. Motor hot fire test stages: 1 – ignition, 2 – liquid phase operation, 3 – gas phase operation, 4 – shutdown

correlation data for nitrous oxide, as reported by Ferreira and Lobo [13]. These curves provided input for estimating initial test conditions for more accurate simulation (Table 4).

Following the test, the motor was carefully removed from the test stand and disassembled to perform post-fire analysis. Thrust data was processed using analysis software to obtain the thrust curve and calculate specific impulse values (Figure 9).

The selected motor configuration achieved a specific impulse of 198 seconds at an average oxidizer to fuel ratio of 3.14. This corresponded to a volumetric specific impulse of approximately 1915 Ns/dm^3, indicating good although not perfect correlation with initial calculations with difference of around 15%.

Comparison of the measured thrust profile with simulation results indicates that the motor performed largely as expected. Notable discrepancies were observed primarily during the initial oxidizer boil-off and ignition transient, occurring within the first 800 ms of the burn (Figure 10). These phases are inherently difficult to model accurately in nitrous oxide hybrid rocket systems due to their strong dependence on empirical data and complex interactions. In the current simulations, ignition and oxidizer boiloff have not been modelled in detail due to the complexity of the two-phase flow and the limitations of the available empirical data. While this phase is relatively short compared to the steady state and does not significantly affect the test results, further work will include developing the model to include the dynamic effects of transient nitrous oxide evaporation and expansion, based on empirical literature data and computational fluid dynamics modelling capabilities. It is also planned to use calibration models based on data from subsequent

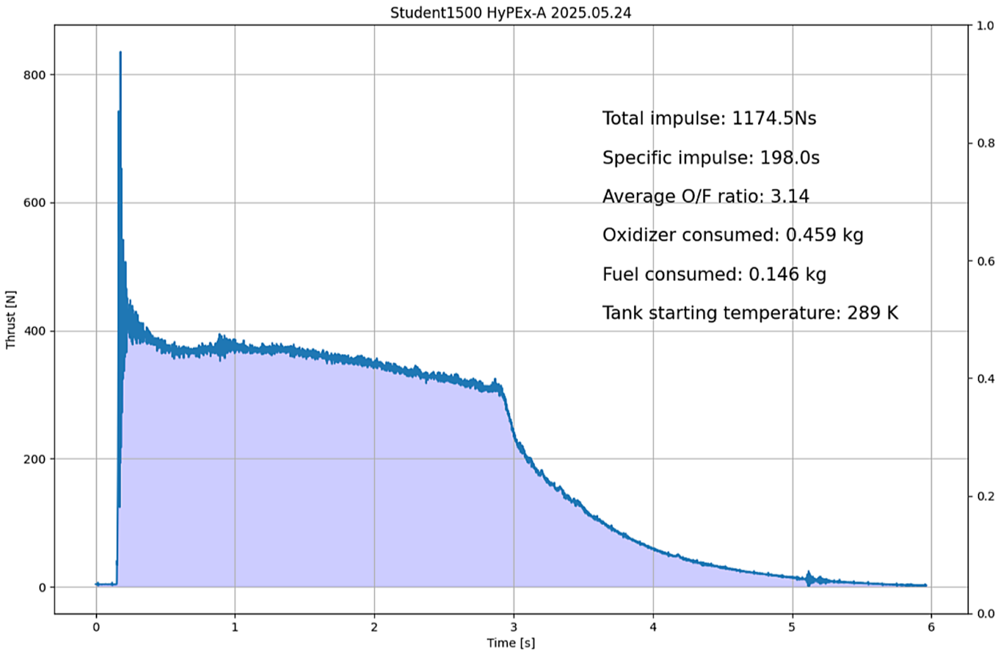

Figure 9. Thrust curve obtained from the test

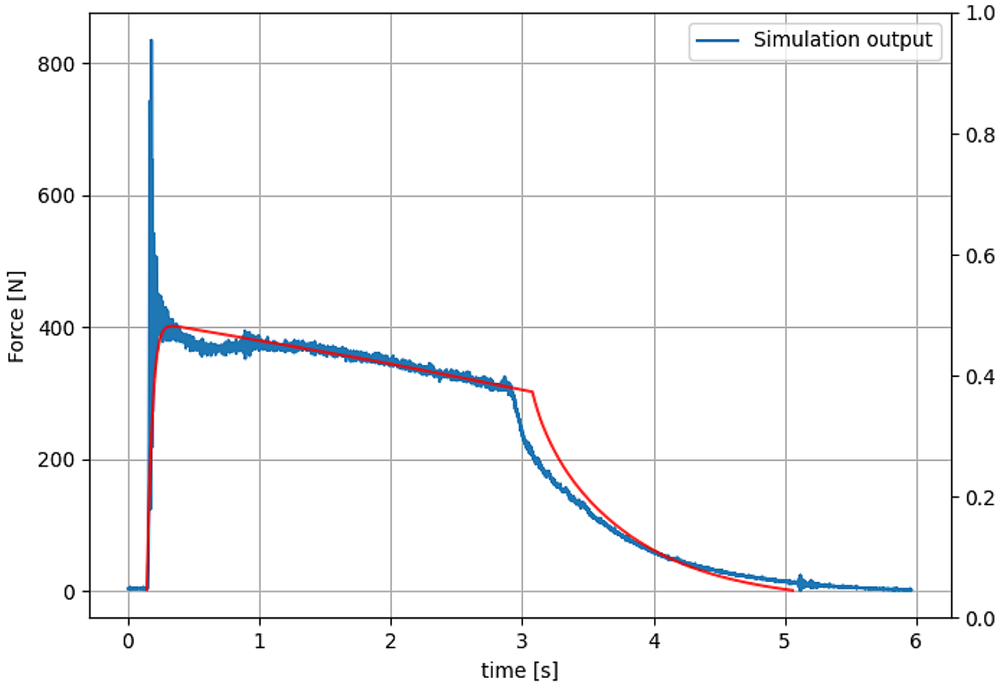

Figure 10. Comparison between motor simulation and hot fire results

fire tests, allowing a more accurate representation of this phase in future simulations.

In parallel with test data analysis, a post-fire examination of the fuel grain was conducted. The cleaned and prepared HTPB grain was sectioned longitudinally to expose its internal cross-section. This inspection provided valuable insight into the burn profile along the grain, revealing nonuniform regression (Figure 11). Notably, regression was particularly visible near the mixing plates, where minor burn-through was observed. On the scale of larger motors this could potentially affect combustion stability. In the current tests, this effect did not cause noticeable anomalies in the thrust profile or significantly affect performance, partially due to small scale and good

Figure 11. Fuel grain cross section after hot fire test

mixing along whole motor length. Nevertheless, in order to improve combustion efficiency, it is planned to analyze critical points inside the combustion chamber and modify the geometry of the mixing plates in future designs at bigger scale to prevent potential catastrophic failure.

In addition, a sample of the charred deposit collected from the nozzle inlet was analyzed (Figure 12) to determine its composition. Fourier-transform infrared spectroscopy using the KBr pellet method was performed on the sample. Comparison of the observed spectral peaks (Figure 13) with literature data [14] revealed a significant presence of aluminum oxide. This finding confirms that aluminum was at least partially combusted within the motor during operation (Figure 5).

Despite a malfunction during the first static fire test, the second test successfully confirmed many of the predicted performance outcomes. The motor achieved a specific impulse of 198 seconds and a volumetric specific impulse of approximately 1915 Ns/dm^3 at an average oxidizer to fuel ratio of 3.14. This 15% difference in volumetric Isp

Figure 12. Slag deposit obtained from the nozzle converging section

Table 5. Motor performance summary

Parameter	Value	Unit
Total impulse	1174	Ns
Specific impulse	198	s
Average O/F ratio	3.14	-
Volumetric specific impulse	1915	$\frac{Ns}{dm^3}$

comparing to theoretical value represent a notable improvement over the conventional HTPB baseline. Comparing test results to those demonstrated by Stephen Whitmore and Zachary Peterson [15], they achieved mean Isp of 204.1 seconds which corresponds to volumetric specific impulse of around 1643 Ns/dm^3 at demonstrated oxidizer to fuel ratios. This 16% difference in favor of aluminium doped fuel further reinforces that adding energetic additives might be a viable strategy in applications where volume requirements are more important than mass constraints.

Additionally, the measured average regression rate over the burn period was increased by about 10% compared to HTPB baseline presented by Xingliang Sun et al [16]. HyPEx-A exhibited regression rate at given mass flux more aligned with heavily doped formulations.

CONCLUSIONS

This study presents a systematic investigation into methods for improving the volumetric specific impulse of HTPB based hybrid rocket motors using nitrous oxide as the oxidizer. By combining energetic additives such as aluminum powder with halogen donors and potentially high-density metal oxides, two fuel formulations were developed and partially evaluated. Thermochemical simulations, supported by tools like NASA CEA and the AFAL performance code, guided the selection of these formulations based on predicted gains in both specific impulse and propellant density.

Comparison between simulated and experimental thrust profiles indicated a high level of

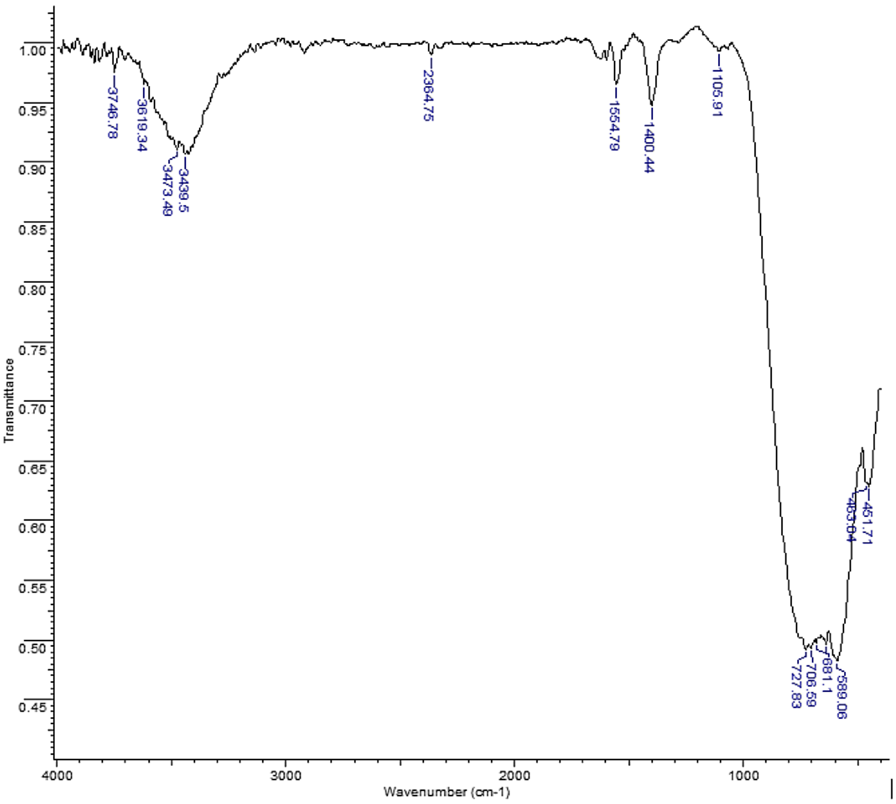

Figure 13. FTIR spectrum of nozzle deposites

conformity, especially during steady-state operation. Deviations in the initial 800 ms were attributed to complex phenomena which are notoriously difficult to reliable model.

Post fire analysis provided further insight into the combustion process. Examination of the fuel grain cross-section revealed nonuniform regression patterns, particularly near the mixer plates, suggesting areas for improved flow mixing and combustion uniformity. FTIR spectroscopy of nozzle deposits confirmed the presence of aluminum oxide, indicating at least partial combustion of metallic additives, and validating their active role in boosting energy release during operation.

Overall, this study demonstrates that with targeted formulation strategies, it is possible to substantially enhance the performance characteristics of hybrid rocket motors while retaining simplicity, safety, and scalability qualities that are essential for suborbital and volume constrained aerospace applications. These findings lay the groundwork for continued development and optimization of hybrid propulsion systems tailored to compact, efficient, and cost-effective missions.

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