Hybrid Rocket Propulsion Technology for Sounding Rocket Development

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ABSTRACT

Hybrid combustion technology has recently been employed in the sounding rocket developments to support the science experiments of academic applications and the establishment of a viable flight test platform for space components development. Due to its safety nature in material handling and combustion processes, hybrid rockets are suitable for university and research institute environments for advanced technology developments and to enable the hands-on experience in rocket science education. In this paper, hybrid combustion sounding rocket development approach and strategy are described with its future plan also outlined. The present hybrid sounding rocket development program involve multi-disciplinary design practice that covers the areas of propulsion, aerodynamics, thermal protection materials, structure, trajectory/attitude, flight computer avionics, telemetry, ground support and payload processing. Among these disciplines, advancement in hybrid propulsion technology presents the major contribution to the hybrid rocket designs in this research. Computational fluid dynamics (CFD) methodology is employed as an efficient and effective tool in the design and analysis of hybrid rocket engine concepts. The main objective is to improve the overall combustion efficiency of hybrid combustion, which features in slow mixing characteristics of typical diffusion flames. Innovative design concepts are analyzed and improved with advanced multi-physics CFD models using parallel computing to improve the combustion efficiency of the design. The numerically designed hybrid rocket engines are validated with hot-fire experiments. Two mixing enhancing designs are obtained as a result of this investigation.

Keywords: Sounding Rocket, Hybrid Propulsion, Mixing Enhancement, Computational Fluid Dynamics, Combustion Modeling.

1. INTRODUCTION

Over the past 60 years, sounding rockets have been demonstrated to be particularly useful for flight experiments above 50 km altitude, under which airplanes and balloons are also readily available. Although it has almost always been motivated by military applications, for many space faring countries, sounding rockets also served as rocket technology research and development platforms for the establishment of their space launch capabilities. Today, sounding rockets are often used for science experiments in atmospheric studies, microgravity environment, advanced aerospace technologies and space components developments and qualifications. And, solid rockets are commonly employed for these applications, which are mostly fabricated by military or well-established aerospace organizations in technically qualified facilities with skilled personnel.

In the recent development of sounding rockets and space launch systems, hybrid rocket propulsion has drawn a lot of attention, especially in the civilian space tourism community, and has been demonstrated to become a viable alternative to the liquid and solid rockets. The hybrid rocket is a combination of both the solid and liquid systems with half of the plumbing of the liquid rockets but retaining the flexibility of operation and avoiding the explosive nature of the solid rockets [1]. It is therefore suitable for advanced hybrid technology developments in universities and research institute environments for academic research and educational purposes.

In joining the strength of both theoretical and experimental approaches, the hybrid rocket development programs have been demonstrated to be very effective and efficient. For the theoretical approach, since the early 1980s, computational modeling approaches have been gradually adopted in the aerospace community in the development of combustion devices and space launch systems. Numerical models using computational fluid dynamics (CFD) methods have been applied to liquid and solid rocket combustion systems with successful results in supporting the technical programs [2-7]. These computational tools allow design studies in hybrid rocket propulsion, rocket aerodynamics, rocket structural dynamics and flight mechanics, etc. Among these disciplines, rocket propulsion and combustion is the key to deliver the overall performance and involves the most complicated technology with multi-physics phenomena.

There are many types of hybrid combustion systems, in which fuel is a solid and the oxidizer is a liquid or gas in classical hybrid designs. Typical examples of combination of fuel and oxidizer with optimum O/F ratio, specific impulse Isp (184 to 326 sec) and characteristic velocity (1224.38 to 2118.36 m/sec) are summarized and discussed in [1]. Some examples for 34 bars chamber pressure at sea-level conditions are given in Table 1 below. And, the ideal vacuum Isps for HTPB-LOX and HTPB-N_2O hybrid rocket engine with nozzle expansion ratio of 80 are 366 sec and 323 sec, respectively.

Table 1. Sea-level Performance of Some Hybrid Rocket Propellants

Fuel	Oxidizer	C*, m/s	Isp, sec
HTPB	LOX	1820.27	280
HTPB	N_2O	1604.47	247
Paraffin	LOX	1804.42	281
Paraffin	N_2O	1605.69	248
PE	LOX	1791.31	279
PE	N_2O	1599.59	247
Li/LiH/HTPB	OF_2	2118.36	326
Carbon	Air	1224.38	184

For material local availability, N_2O-HTPB propulsion system is selected in the present hybrid rocket development. The maximum vacuum Isp demonstrated in pratice so far for the N_2O-HTPB propulsion system is only fair around 250 seconds while its theoretical limit can be as high as 323 seconds. This indicates that it is worthwhile to invest in this research to push the thrust performance close to the theoretical limit.

Very few [8,9] have attempted to model the complicated reactive flow phenomena of a realistic hybrid propulsion system that an energy-balanced surface decomposition model is employed. The modeling efforts have reached successes to some extent based on fitting of experiments and numerical simulations with real-fluid effects considered in the models. The real-fluid properties affect the overall flow structure in the combustion chamber, especially near the injectors, and affect the combustion processes and heat transfer characteristics, which is the key to good prediction of the regression rates of the solid grain.

Thrust performance, propellant mass fraction, reliability and cost are among the major factors that determine the overall performance of a rocket system. Theoretically, hybrid rocket systems are advantageous among many of these factors as compared to solid and liquid rocket systems. However, the thrust performance aspect of hybrid rockets still need further investigations to improve their combustion efficiency, Isp and stable O/F ratio control, etc. And, these are good research topics for universities and research institutes.

2. HYBRID ROCKET TECHNOLOGY DEMONSTRATION

The sounding rocket program of National Space Organization of Taiwan was initiated in 1998 with the aim to establish a flight test platform for ionosphere studies and space technology developments. The long-term goal of this program is for building up the technical capabilities for designing space launch systems. So far, 7 solid sounding rockets, with a two-stage design, have been launched. The success rate was 6 out of 7. These solid rockets were produced by CSIST, a military research institute.

In 2009, a hybrid rocket development program was initiated by the National Space Organization (NSPO) with two university research teams selected to bring lab-scale hybrid combustion devices to their maturity that can be integrated into flight-worthy small-scale sounding rocket systems. This technology demonstration program gave the university teams research opportunities in extending the results of their fundamental studies to multi-disciplinary systems engineering practices in order to realize the system integration, testing and flight operations. This program is unprecedented in Taiwan's research community and has broadened the views and hands-on experience of the researchers and students involved in the program. As part of the outcome of the program, successful hybrid rocket flight tests were demonstrated in 2010 and 2011 to altitudes around 10 km.

In this program, two combustion design concepts were employed. They are a N_2O-HTPB system with mixing enhancer of NTCU (National Chiao Tung University) [9] and a N_2O-50%HTPB+50%Paraffin system of NCKU (National Cheng Kung University) [10]. The thrust level (around 300 kgf) and Isp (around 220 seconds) of these two approaches are comparable. The rocket systems are completed with composite N_2O pressure tanks, flight computers and telemetry data links. During the propulsion system developments, hybrid rocket motors with designed flight specifications and configurations were hot-fire tested in the laboratory to measure the thrust performance, solid regression rates, thermal protection environment and nozzle erosion characteristics. In the meantime, a comprehensive numerical model with finite-rate chemistry and real-fluid properties [9,11] was developed and employed by the NCTU team to study the internal ballistics of the hybrid rocket motor and compared with the hot-fire test data.

3. HYBRID SOUNDING ROCKET DEVELOPMENT

Based on the successful program of the hybrid technology demonstration, the National Science Council of Taiwan has approved proposals of the two university teams in 2011 for continuing developments of the sounding rocket technology in the next phase. The main goal of the new program is to develop viable sounding rockets in three years, capable of performing science experiments in altitudes between 100 km and 200 km.

With the mission descriptions of the next-phase sounding rocket using hybrid propulsion, one design concept has required that the thrust level for a 600 kg rocket is to be greater than 3,000 kgf. For the new rocket motor development, the comprehensive numerical model is employed in an extensive design analysis effort while a hot-fire ground test facility is being built. A final design of the hybrid motor will be selected based on theoretical and experimental studies.

A two-stage hybrid rocket system is resulted based on the system design analysis. The first-stage motor will be similar in design as that used in the previous study with direct scale-up and fine-tuned. For the second-stage motor, a low slenderness form factor design using dual vortical flow concept is proposed. This new design also ends up with higher combustion efficiency based on the numerical optimization process. The new design gives 292 seconds vacuum Isp, which is one big step closer to the theoretical optimum value for a N_2O-HTPB hybrid system. This finding will be confirmed by hot-fire experimental data later during the preliminary design phase of the program.

The 8 meter long sounding rocket will carry three nano-size satellite engineering models to be deployed during rocket ascend at around 100 km altitude after nose fairing separation. The scientific instruments onboard will conduct ionosphere measurements and send the data to the ground stations through the downlink telemetry system. The second-stage of the rocket is designed to deploy parachute after reentry and splash down the Pacific Ocean about 50 km off the southeast coast of Taiwan. The second-stage rocket, which contains the flight computer and some position and attitude sensors, will be recovered from the sea. To accomplish the mission, reliable flight computer avionics and telemetry subsystems are also important in the hybrid sounding rocket system design.

Furthermore, as technical developments are invested for commercial applications in recent years, the overall competitiveness of hybrid rocket propulsion becomes quite encouraging. Figure 1 shows comparisons of the overall performance between some known solid, liquid and hybrid propulsion technologies. It shows a clear trend that the hybrid rockets produce Isp better than solid systems and feature higher impulse density than hydrogen-oxygen systems. And, the performance of hybrid propulsion is close to that of the LOX-hydrocarbon systems. As for the present development efforts, the research results have shown Isp improvements from 222 sec in the beginning, to 292 sec for the latest dual-vortical flow designs and will move toward the 300 sec mark when the designs are further optimized and validated with hot-fire experiments. Establishing good reliability of the present hybrid propulsion system is also an important task in future developments.

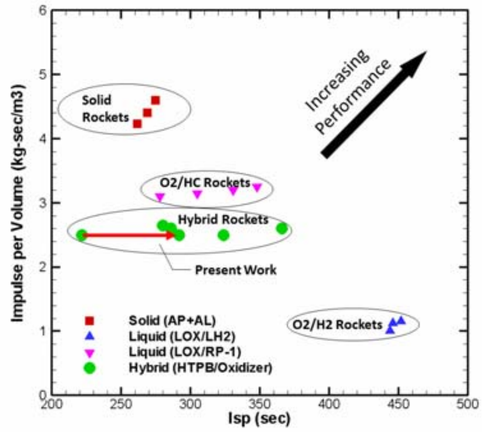


Fig. 1. Performance comparisons of solid, liquid and hybrid rocket propulsion systems.

4. Hybrid Rocket Propulsion Modeling

The present numerical method solves a set of governing equations describing the conservation of mass, momentum (Navier-Stokes equations), energy, species concentration and turbulence quantities, for the flow variables of density (ρ), species mass fraction (α_i), mean velocities (u_i), total enthalpy (H), turbulence kinetic energy (k) and its dissipation rate (ε). The governing equations are written as:

$$\frac{\partial \rho}{\partial t} + \frac{\partial}{\partial x_j}\left(\rho u_j\right) = 0$$

$$\frac{\partial \rho \alpha_i}{\partial t} + \frac{\partial}{\partial x_j}\left(\rho u_j \alpha_j\right) = \frac{\partial}{\partial x_j}\left[\left(\rho D + \frac{\mu_t}{\sigma_\alpha}\right)\frac{\partial \alpha_i}{\partial x_j}\right] + \omega_i$$

$$\frac{\partial \rho u_i}{\partial t} + \frac{\partial}{\partial x_j}\left(\rho u_j u_i\right) = -\frac{\partial p}{\partial x_i} + \frac{\partial \tau_{ij}}{\partial x_j}$$

$$\begin{aligned} \frac{\partial \rho H}{\partial t} + \frac{\partial}{\partial x_j}\left(\rho u_j H\right) = \frac{\partial p}{\partial t} + Q_r &+ \frac{\partial}{\partial x_j}\left(\left(\frac{K}{C_p} + \frac{\mu_t}{\sigma_H}\right)\nabla H\right) \\ &+ \frac{\partial}{\partial x_j}\left(\left((\mu + \mu_t) - \left(\frac{K}{C_p} + \frac{\mu_t}{\sigma_H}\right)\right)\nabla\left(V^2/2\right)\right) \\ &+ \frac{\partial}{\partial x_j}\left(\left(\frac{K}{C_p} + \frac{\mu_t}{\sigma_H}\right)\left(u_k \frac{\partial u_j}{\partial x_k} - \frac{2}{3} u_j \frac{\partial u_k}{\partial x_k}\right)\right) \end{aligned}$$

$$\frac{\partial \rho k}{\partial t} + \frac{\partial}{\partial x_j}\left(\rho u_j k\right) = \frac{\partial}{\partial x_j}\left[\left(\mu + \frac{\mu_t}{\sigma_k}\right)\frac{\partial k}{\partial x_j}\right] + \rho\left(\Pi - \varepsilon\right)$$

$$\frac{\partial \rho \varepsilon}{\partial t} + \frac{\partial}{\partial x_j}\left(\rho u_j \varepsilon\right) = \frac{\partial}{\partial x_j}\left[\left(\mu + \frac{\mu_t}{\sigma_\varepsilon}\right)\frac{\partial \varepsilon}{\partial x_j}\right] + \rho\frac{\varepsilon}{k}\left(C_1 \Pi - C_2 \varepsilon + C_3 \Pi^2/\varepsilon\right)$$

An efficient method for treating the real-fluid equations of state and fluid properties is employed herein for liquid propellant combustion flows [9]. The convection terms of the governing equations are discretized with a second-order upwind scheme. Second-order central schemes are applied to the diffusion and source terms. For complete description of the thermal environment in the combustion chamber, a radiative heat transfer model with a finite-volume integration method [4,5] is also employed in the present model. In the combustion chamber, the main participating species in the radiation model are carbon-dioxide and hot steam. For transient flow computations, an efficient second-order time-marching scheme, which has been validated for vortex shedding and transient start-up nozzle flows [7], is employed in the present study. These numerical models are important for high fidelity simulations of combustion physics. They are described in the following sections.

4.1 Transient Time-Marching Scheme

In the present numerical model, the temporal terms of the transport equations are modeled with a second-order backward difference scheme. There are two ways of performing time-marching procedures. First, the momentum, energy and pressure-correction equations are grouped together within an iteration loop and drive to convergence for each time step. Although the transient sub-iteration method described in the previous section is accurate for transient flow applications, it requires many sub-iterations (more for highly dynamic cases) to advance one time step, especially for low-speed flows. In order to save computational effort in transient flow applications, this sub-iteration algorithm can be replaced with a more efficient method, such as the operator splitting technique (Kim et al. [11]; Chen at al. [12]; Wang and Chen [13]). This method consists of a predictor step plus two corrector steps to drive the discretization errors to second-order accuracy.

4.2 Finite-rate Chemistry Model

The present combustion system involves liquid N_2O and solid HTPB propellants. With a pyro grain to serve as an ignition heat source and some catalytic effects for the nitrous oxide, a diffusion flame is established upon the injection of nitrous oxide into a combustion chamber with a single port HTPB grain. The generated heat from the diffusion flame continues to decompose the nitrous oxide and HTPB through convective and radiative heat transfer. Then, the decomposed gas species are mixed and combusted in the diffusion flame to produce mainly water vapor and carbon dioxide. The decomposition rate of HTPB is modeled using empirical correlations for energy balance through the pyrolysis process to produce mainly C_4H_6 (70%) and some C_2H_4 (30%) on the solid grain surface. Next, the butadiene and ethylene are further decomposed into CO and H_2 as irreversible global reactions as part of the gas reaction steps. The gas reaction steps mainly involve the nitrous oxide decomposition and the wet-CO mechanisms. The present chemistry system involves 16 species and 28 reaction steps as summarized in Table 2.

The present hybrid rocket combustion model include a 580 mm combustion chamber with the design of forward-end and aft-end mixing chambers. A simple convergent-divergent conical nozzle is attached to the end of the combustion chamber. A single-port simple solid grain of HTPB is cast in two sections and assembled into a single segment through bonding. To boost the mixing efficiency, a mixing enhancer (patent pending) is also installed near the forward corner of the solid grain. A pintle-type injector made of stainless steal is employed for steady injection of the N_2O oxidizer. The nitrous oxide tank upstream of the injector and control valve is setup in a vertical position. Due to the properties of the oxidizer and the absence of thermal control for the oxidizer tank and without using a presurant tank upstream, the injection flow rates of this pressure-fed injection system depend directly on the temperature of the environment. A small pyro grain is attached to the forward face of the solid grain, which is ignited at engine start-up to melt the HTPB solid and start the combustion process after the N_2O control valve is commanded to open.

Table 2. N_2O-HTPB chemical kinetics

Grain Surface Global Reaction: HTPB $\rightarrow$ 0.7 C_4H_6 + 0.3 C_2H_4			
Gas-Phase Reactions:			
1	$C_4H_6 + O_2 \rightarrow 2\,CO + C_2H_4 + H_2$	15	$OH + HO_2 = H_2O + O_2$
2	$C_2H_4 + O_2 \rightarrow 2\,CO + 2\,H_2$	16	$CO + OH = CO_2 + H$
3	$O_2 + H_2 = 2\,OH$	17	$CO + O_2 = CO_2 + O$
4	$H_2 + OH = H_2O + H$	18	$O + CO + M = CO_2 + M$
5	$2\,OH = H_2O + O$	19	$N + NO = N_2 + O$
6	$H_2 + O = H + OH$	20	$N + O_2 = NO + O$
7	$O_2 + H = O + OH$	21	$N + OH = NO + H$
8	$O + H + M = OH + M$	22	$N_2O + O = N_2 + O_2$
9	$2\,O + M = O_2 + M$	23	$N_2O + O = 2\,NO$
10	$2\,H + M = H_2 + M$	24	$N_2O + H = N_2 + OH$
11	$H + OH + M = H_2O + M$	25	$N_2O + OH = N_2 + HO_2$
12	$H + O2 + M = HO_2 + M$	26	$N_2O + M = N_2 + O + M$
13	$H + HO_2 = 2\,OH$	27	$HO_2 + NO = NO_2 + OH$
14	$H + HO_2 = H_2 + O_2$	28	$NO + O = NO_2$

Typically, a hot-fire test of this experiment runs between 15 to 20 seconds. During the test, motor thrust, chamber pressures and HTPB port temperatures are measured. The oxidizer flow rates and the regression rates of the solid grain are obtained through post-processing after the hot-fire tests. Overall, the experimental data show that the averaged solid regression rate is around 1.2 mm/sec for the current design, which is slightly lower than the correlation burning rate equation of Lohner et al.[25] The measured specific impulse, sea-level Isp, of the motor with the mixing enhancer is around 213 sec (or vacuum Isp of 222.18 sec). From test cases without the mixing enhancer, the measured sea-level Isp is around 178 sec (or vacuum Isp of 187.18 sec) for the same motor geometry.

To model this test case, an axisymmetric mesh system with 91,960 elements is employed, after a grid independence study, to model the geometry of the combustion chamber, pintle injector, solid grain and the conical nozzle. The mixing enhancer is not modeled in this case due to the axisymmetric condition simplification. A real-fluid property database is created to represent the nitrous oxide fluid in the numerical model. Figure 2 shows the nitrous oxide thermodynamics properties, where the pressure P is in ATM.

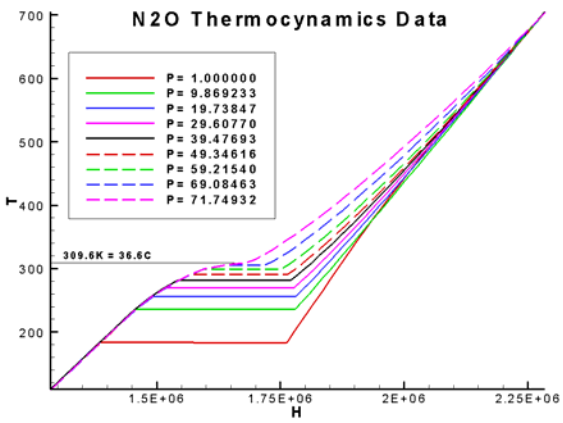
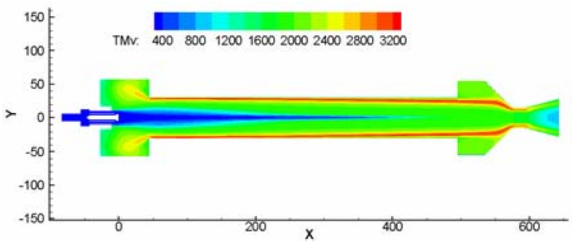


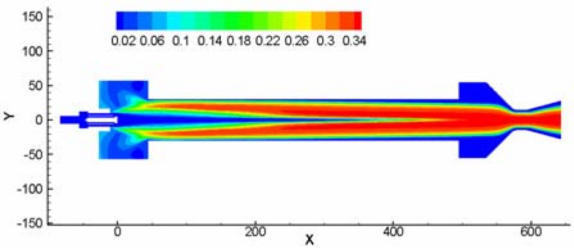
Fig. 2. Real-fluid property of N_2O used in the present model.

A radiation model with finite-volume integration method is used to calculate the radiative heat transfer effects. The turbulence fields are predicted with an extended two-equation model[3]. The total pressure (30 ATM) and total temperature (283 K) boundary conditions are imposed at the injector inlet with the consideration of estimated total pressure losses between the tank and the injector inlet. However, this setup still can not represent the true test conditions because the real inlet conditions are transient in nature. During the test, the tank pressure and temperature are reduced rapidly due to high flow rate and the expansion effects. Therefore, in the present model, we compare the numerical predictions to the test data when the oxidizer injection stays in the liquid phase, i.e. between 2 and 7 seconds after ignition.

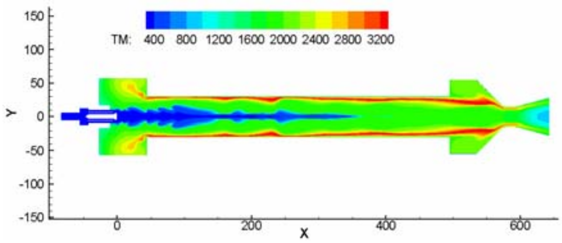
The numerical simulation takes 80,000 time steps with 1 microsecond time step size to obtain a quasi steady-state solution of the flowfield. Figure 3 shows the predicted hybrid rocket motor combustion flowfield using the present numerical model. It is clear that well-organized shear-layer oscillation structure, similar to the Taylor-Goertler type instability effects of thin shear layers, is predicted for the injection system. This oscillation is then coupled with the oxidizer decomposition process and the flame along the solid grain surface. Clearly, the axisymmetric simplification of the present model can not completely represent the real physics of shear layer instabilities. And, it is plausible that the oxidizer evaporation may also contribute to the oscillation of the predicted flowfield. The numerical solution indicates that further improvement in the injector design or mixing enhancement method in the chamber is required in order to increase the overall performance of this combustion system. The predicted averaged sea-level Isp is 181.2 sec (or vacuum Isp of 191.18 sec), which is lower than the measured data as expected due to the fact that a mixing enhancer is not present in the numerical model.



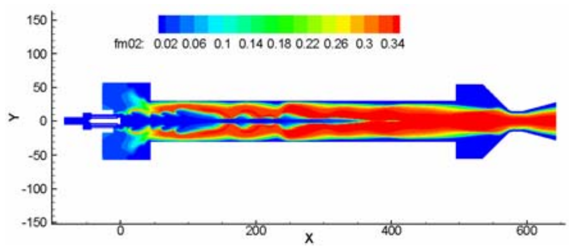
(a) Time-Averaged Temperature (K)



(b) Time-Averaged O_2 Mass Fraction



(c) Instantaneous Temperature (K)



(d) Instantaneous O_2 Mass Fraction

Fig. 3. Computational results of the hybrid rocket combustion (axisymmetric model).

For modeling the 3-dimensional flowfield, typical instantaneous flowfield solutions for cases with and without mixing enhancer installed near the forward section of grain port are sutdied. The effects of the mixing enhancer in modifying the flame shape downstream gives slight increase in the HTPB regression rate and improves the overall combustion efficiency. A grid independence study using three meshes with 3.5 million, 5.2 million and 6.5 million cells for the case without mixing enhancer was first carried out. It was decided based on the time-averaged solutions that the mesh with 5.2 million cells gives grid independent solutions for this problem. The overall performance data comparisons are summarized in Table 3, which show that the present solutions are in good agreement with the test data.

Table 3. Comparisons of hybrid rocket motor performance

(Vacuum Isp, O/F)	**w/o Mixing Enhancer**	**with Mixing Enhancer**
2D Axisymmetric	(191.38, 10.05)	- - -
3D Model	(190.82, 10.11)	(223.77, 9.72)
Experiment	(187.18, 11.03)	(222.18, 10.91)

The numerical solutions of the 3D models show an averaged vacuum Isp of 223.77 sec and 190.82 sec for the cases with and without the mixing enhancer, respectively. Notice that the predicted Isp for the case without mixing enhancer is close to that predicted by the axisymmetric model. This shows the modeling consistency between 2D and 3D approaches. And, the 3D model produces slightly lower Isp than the axisymmetric model, which can be attributed to the differences in the turbulent mixing behavior in the combustion chamber. That is, the 3D gives more dissipation losses as the oxidizer flowing through the solid grain port. The results also show consistently higher predicted Isp of the present numerical models as compared to the measured data. The predicted averaged O/F ratios of the present model are 9.72 and 10.11 for the cases with and without mixing enhancer, respectively, as oppose to the measured O/F ratios of 10.91 and 11.03, respectively. Thus, the numerically predicted mixture ratios are closer to the optimum value of 8.963 for the N_2O/HPTB system, and consequently give higher chamber temperature and motor Isp. Figure 4 illustrates the vorticity decay effects downstream of a mixing enhancer with 8-blade vortex generators of the present design using a cold-flow computational model. Figure 5 shows the calculated temperature field inside the combustion chamber of the present hybrid rocket motor. The mixing of the diffusion flame structure is enhanced with vortical flow patterns downstream of the mixing enhancer. Based on these results, the effects of multiple sets of mixing enhancers will be analyzed and tested in the subsequent performance optimization study. Also, in order improve the form factor and combustion efficiency of hybrid rocket motors, another innovative dual-vortical-flow chamber design, e.g. Fig. 6, is numerically analyzed that gives a vacuum Isp of 292 sec. This design will be validated experimentally in future study.

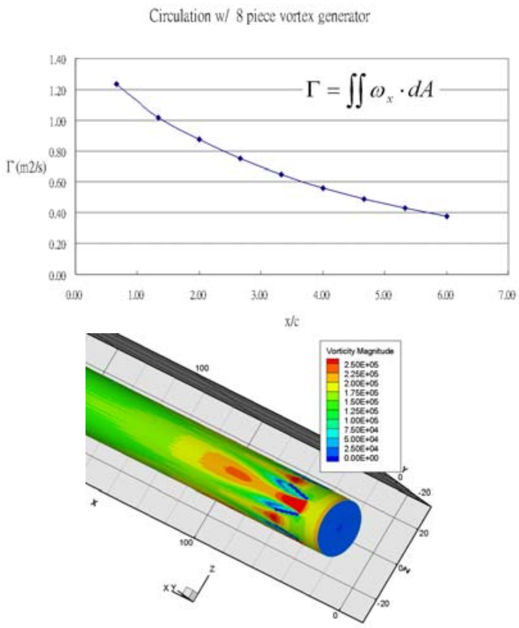


Fig. 4. Vorticity decay cold-flow study downstream of an 8-blade mixing enhancer.

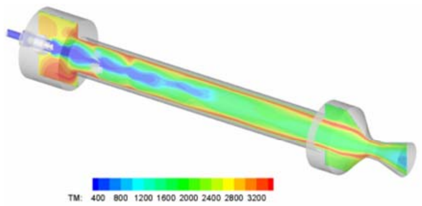


Fig. 5. Predicted instantaneous temperature contours of 3-D computational result for an axial single-port design with mixing enhancer.

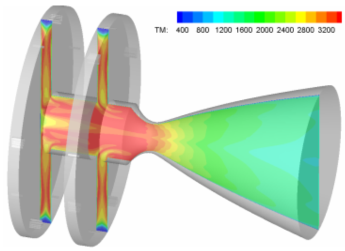


Fig. 6. Predicted instantaneous temperature contours of a dual-vortical-flow chamber design.

For the single-port chamber design without mixing enhancer, the solid grain regression rate is also important in anchoring the present computational model. Figure 7 shows regression data comparisons of the present computations and empirical correlations of Sutton [1] and Lohner [14]. Since Lohner's experiments used very small scale motors, the data measured are not as reliable as those of Sutton. When the N_2O mass flux is used, the present model shows lower regression rates than Sutton's correlation. However, if only the oxygen content in N2O is used to calculate the oxidizer mass flux, the present model correlates well with Sutton's data. This result validates the present numerical model in predicting the solid regression rate.

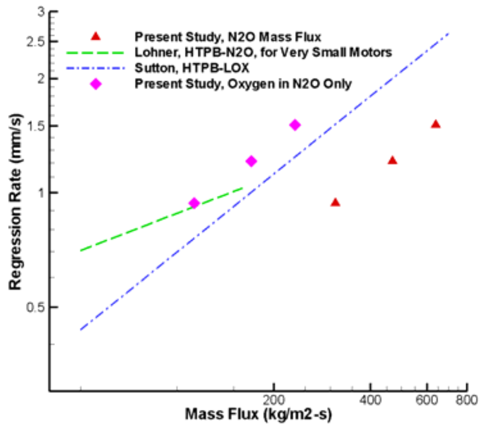


Fig. 7. Comparisons of regression rate data of single-port chamber designs.

This study shows the effectiveness of the mixing enhancer numerically and experimentally. The results of the Isp data indicate that the mixing enhancer improves the overall combustion efficiency, increases the chamber temperature and pressure, and therefore produces higher Isp. The O/F ratio data, on the other hand, indicate that the mixing enhancer also increases slightly the overall regression rate of the solid grain. The vortex structures created downstream of the mixing enhancer has boosted the mixing effects of the diffusion flame along the solid grain surface. The resultant increased heat flux at the grain surface also enhances the regression rate of the solid fuel. It is worth noting that, for hybrid rocket system, improvement in combustion efficiency is far more important than the issue of increasing the solid grain regression rate. After all, Isp is the key design parameter in determine the overall performance of a rocket propulsion system. Obtaining high combustion efficiency and, at the same time, keeping good mechanical/thermal properties of the solid grain for long burn time of the motor, is a plausible design practice. The mixing enhancer proposed in the present study is heading in the right direction based on the results of the present numerical and experimental study.

5. CONCLUSIONS

The sounding rocket development program of Taiwan using hybrid rocket propulsion has been described in the paper. A robust design practice with comprehensive computational models and hot-fire test data validation capabilities has played an important role in the development of the N_2O-HTPB hybrid sounding rocket system.

In the 2nd-phase of the sounding rocket R&D program using hybrid propulsion technology, a two-stage rocket has been designed and analyzed using the numerical models and will be verified using experimental data. A new hybrid rocket motor has been designed numerically for the second-stage of the rocket that has shown good combustion efficiency and Isp value. The overall thrust performance of the new design will be further validated using hot-fire test data.

A comprehensive multi-physics CFD model has been developed and validated for the designs of high-efficiency hybrid rocket engines. This numerical model enables efficient and effective design optimizations of the mixing enhancement devices in the hybrid combustion chamber. It has also assisted in this study in identifying the key mechanism for boosting the combustion efficiency along the solid grain surface. As a result, mixing efficiency and combustion efficiency are greatly improved for the diffusion flames in the hybrid rocket engine. Further study is planned in the future to come up with more advanced designs in order to push the thrust performance of hybrid propulsion closer to its theoretical limit.

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