Mixing Enhancement in Hybrid Propulsion

T. H. Chou, G. R. Lai
National Chiao Tung University
Hsinchu, Taiwan

Y. S. Chen, Bill Wu
National Space Organization
Hsinchu Science Park, Taiwan

J. S. Wu
National Chiao Tung University
Hsinchu, Taiwan

1 Abstract

Hybrid rocket combustion technology has recently been employed in the sounding rocket developments to support the science experiments and the establishment of a viable flight test platform for space components development. Due to its safety nature in propellant handling and combustion processes, hybrid rockets are suitable for university and research institute environments. In this paper, hybrid combustion technology in sounding rocket development approach and strategy are described. 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 rocket engine, which features in slow mixing characteristics of typical diffusion flames for conventional designs. Innovative design concepts are analyzed and improved using the present numerical model. The results are validated with hot-fire experiments.

2 Introduction

Over the past 60 years, sounding rockets have been demonstrated to be particularly useful for flight experiments above 20 km altitude, under which airplanes and balloons are also readily available. Although it has been mostly motivated by military applications, sounding rockets still serve as an R&D platforms for establishing the capability of launching satellites. Today, sounding rockets are often used for experiments in atmospheric studies, microgravity research, advanced aerospace flight tests 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 recent development of sounding rockets and space launch systems, hybrid rocket propulsion has drawn a lot of attention among researchers, especially in the civilian space tourism community, and has been demonstrated to become a viable alternative to the liquid and solid rockets. Hybrid rocket is a

Correspondence to: yenchen@nspo.narl.org.tw

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].

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 fuel and oxidizer combinations, 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]. 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.

For local availability of the materials, 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 worth investing in this research to push the thrust performance closer to its theoretical limit.

For the solid grain decomposition mechanism, an energy-balance model is incorporated [8,9]. Furthermore, 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 along the solid grain surface.

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.

3 Hybrid Rocket Technology Demonstration

In 2009, following 7 sounding rocket experiments since 1998, 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 the NTCU (National Chiao Tung University) team [9] and a N_2O-50%HTPB+50%Paraffin system of the NCKU (National Cheng Kung University) team [10]. The thrust level (around 300 kgf) and Isp (around 220 seconds) of these two approaches are comparable. 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 engine and compared with the experimental data.

4 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 multi-function sounding rockets in three years, capable of performing science experiments in altitudes between 100 km and 200 km.

With the mission descriptions for the next-phase of the sounding rockets using hybrid propulsion, one design concept has required that the booster thrust level for a 600 kg rocket is to be greater than 3,000 kgf. A two-stage hybrid rocket system is resulted based on the mission and system design analysis. The first-stage engine will be similar in design as that used in the previous study with direct scale-up and fine-tuned. For the second-stage engine, 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 provides 292 seconds of vacuum Isp, which represents a major improvement in performance that is much closer to the theoretical optimum value for a N_2O-HTPB hybrid system.

5 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. An efficient method for treating the real-fluid equations of state and fluid properties is employed to model the liquid propellant flows [9]. The convection terms of the governing equations are discretized with a second-order TVD scheme. Second-order central schemes are applied to the diffusion and source terms. A radiative heat transfer model with a finite-volume integration method [4,5] is employed.

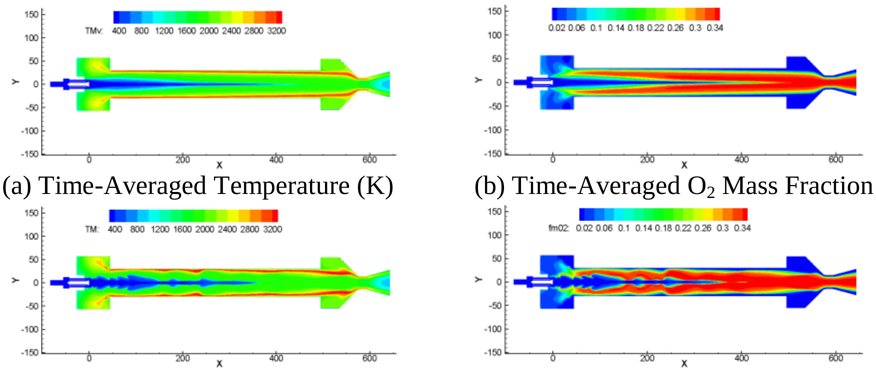
For accurate transient flow computations, the temporal terms of the transport equations are modeled with a second-order backward difference scheme. In order to have better computational efficiency for transient flows, the sub-iteration algorithm is replaced with a more efficient operator splitting technique [11-13]. This method consists of a predictor step plus two corrector steps for second-order accurate transient flow computations.

The present combustion system involves liquid N_2O injection over HTPB solid-grain. With a pyro grain as an ignition heat source, a diffusion flame strrucute is established upon the injection of nitrous oxide into the 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 an empirical correlation 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 [14]. The present finite-rate chemistry system involves 16 species and 28 reaction steps.

The present hybrid rocket engine includes a 580 mm combustion chamber with forward-end and aft-end mixing chambers. A convergent-divergent nozzle is attached to the end of the combustion chamber. A single-port solid grain of HTPB is cast and assembled in the chamber. To boost the mixing and combustion efficiency, a mixing enhancer (patent pending) is installed near the forward corner of the solid grain. A pintle-type injector is employed for steady injection of the N_2O oxidizer.

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 burning rate correlation equation of Lohner et al. [15] The measured sea-level specific impulse, Isp, of the engine with the mixing enhancer is around 213 sec (or vacuum Isp of 222.18 sec). From the test cases without mixing enhancer, the measured sea-level Isp is around 178 sec (or vacuum Isp of 187.18 sec) for the same engine geometry.

An axisymmetric model with mesh size of 91,960 elements was first employed to simulate this case. The mixing enhancer is not modeled in this case due to the axisymmetric condition. The total pressure (30 ATM) and total temperature (283 K) boundary conditions are imposed at the injector inlet. However, this setup still can not represent the true test conditions because the real inlet conditions are transient in nature. Therefore, we compare the numerical predictions with the test data when the oxidizer injection stays in the liquid phase, i.e. between 2 and 7 seconds after ignition.



(a) Time-Averaged Temperature (K) (b) Time-Averaged O_2 Mass Fraction

(c) Instantaneous Temperature (K) (d) Instantaneous O_2 Mass Fraction

Figure 1. Computational results of the hybrid rocket combustion (axisymmetric model).

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 1 shows the predicted hybrid rocket combustion flowfield using the present numerical model. It is clear that well-organized shear-layer oscillating structure, similar to the Taylor-Goertler type instability for thin shear layers, is predicted for the injection system. Clearly, the axisymmetric simplification of the present model can not completely represent the real physics of shear layer instabilities. 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 without mixing enhancer in the numerical model.

For modeling the 3-dimensional flowfield, solutions for cases with and without mixing enhancer are sutdied. A grid-independent study was performed and decided that a mesh with 5.2 million cells gives converged solutions for this problem. The overall performance data comparisons are summarized in Table 1, which shows that the present solutions are in good agreement with the test data.

Table 1: 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 model 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 the result of the axisymmetric model. This shows the modeling consistency between the 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 numerical model as compared to the measured data. The predicted averaged O/F ratios 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 stoichiometric value of 8.963 for the N_2O/HPTB system, and consequently gives higher chamber temperature and motor Isp. Figure 2(a) shows the calculated temperature field inside the combustion chamber of the single-port hybrid rocket engine. The mixing of the diffusion flame structure is enhanced with vortex structure downstream of the mixing enhancer. In order improve the form factor and combustion efficiency of hybrid rocket motors, another innovative dual-vortical-flow chamber design (patent pending), e.g. Figure 2(b), is numerically analyzed that produces a vacuum Isp of 292 sec. This design will be validated experimentally in future study.

Figure 2. Predicted instantaneous temperature field in hybrid engines. (a) Left: Single-port with an 8-blade mixing enhancer. (b) Right: Dual-vortical flow design.

The solid grain regression rate is also important in anchoring the present computational model. Figure 3 shows regression rate data comparisons of the present results and empirical correlations of Sutton [1] and Lohner [15]. 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.

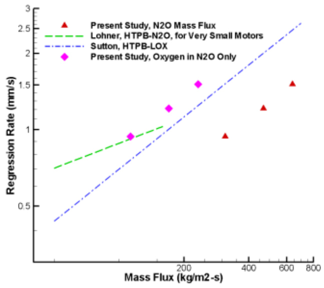


Figure 3. Comparisons of regression rate data of single-port chamber designs.

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. It is worth noting that, for hybrid rocket system, improving 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 engine, 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.

6 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 capability has played an important role in the development of the N_2O-HTPB hybrid sounding rocket system.

In the present phase of the sounding rocket R&D program using hybrid propulsion technology, a two-stage rocket has been designed and analyzed using the numerical model and will verified in flight tests.

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.

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. As a result, mixing and combustion efficiency are greatly improved for the diffusion flames in the hybrid rocket engine. Future study is planned for further improvements of the hybrid propulsion.

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