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Short communication

Design specifications of H_2O_2/kerosene bipropellant rocket system for space missions

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ABSTRACT

Over the last decade, interest has been rekindled on hydrogen peroxide as a rocket propellant. As a result, 1000-N-class H_2O_2/hydrocarbon bipropellant rocket systems are in a stage requiring a serious exploration for application. Application to orbit-raising maneuvers is proposed here as the most appropriate use of the 1000-N-class thrusters. In this study, optimal design specifications of a 90% H_2O_2/kerosene rocket system for an apogee kick motor were derived. The optimal design specifications are configured with a thrust of 742 N, a chamber pressure of 11.7 bar, a mixture ratio of 7.36, and a nozzle expansion ratio of 580. It was found that, in the case of the latest Korean geosynchronous satellite, replacing the current apogee kick motor with the H_2O_2/kerosene rocket system would result in a mass increase of less than 0.5% of the total mass.

1. Introduction

Over the last decade, interest has been rekindled on hydrogen peroxide as a rocket propellant [28,29,2,7,12,8,16,23,24,13,14,5,25]. Hydrogen peroxide of a high concentration becomes an oxidizer to a hydrocarbon fuel in a bipropellant rocket system. Numerous experiments have achieved a fast ignition delay of less than 100 ms and a high combustion efficiency of about 95% with different concentrations of hydrogen peroxide and different hydrocarbon fuels [2,7,12,8,16,23,24,13]. Efforts are being made to increase the combustion efficiency even higher and to make combustion reliable [13,14,5,25]. There are two types of ignition method in using hydrogen peroxide as an oxidizer: hypergolic ignition method which uses liquid hydrogen peroxide and a fuel mixed with a catalyst [2,7], and autoignition method which uses gaseous oxygen obtained through catalytic decomposition [12,8,16,23,24,13]. For the autoignition type, silver [8,16,23,24] or manganese dioxide [12,13] have been used as the catalyst. The researches can be also classified according to the thrust levels: 10-tonf [16,23], 1000-N [2,12,24,13], and 10-N classes [7,8]. Traditionally nitrogen tetroxide and hydrazine derivatives have been used as the oxidizer and the fuel for the 1000- and 10-N-class bipropellant rocket system. Though the H_2O_2/hydrocarbon rocket system has a lower specific impulse than that of the N_2O_4-derivative/N_2H_4-derivative rocket system, the former has a definite advantage of being environmentally friendly and therefore of lower development cost. The 10-tonf-class rocket system would be appropriate for an upper-stage engine for a space launch vehicle, and the 10-N class would be applicable for a reaction control system of a large satellite.

It is possible for one to conceive application of the 1000-N class in weapon systems. For civilian applications, application to orbit-raising maneuvers is proposed here as the most appropriate use of the 1000-N-class thrusters. An apogee kick motor is needed for geosynchronous satellite missions as well as planetary missions. Apogee kick motor is best fit for such a use. The existing bipropellant rocket systems for orbit-raising maneuvers using N_2O_4 and N_2H_4 derivatives to have design specifications that are similar, i.e. in thrust, thrust chamber pressure, propellant mixture ratio, and nozzle expansion ratio [3,9,18,4,10,17]. For the H_2O_2/hydrocarbon rocket system, there do not exist an optimum design specifications to emulate the performance achieved by the existing systems. The objective of the present study is to derive optimal design specifications of a 90% H_2O_2/kerosene rocket system used for an apogee kick motor. It seeks to minimize the total mass of a spacecraft for a given mission scenario.

Before proceeding to the design analysis, it is necessary to identify the status of the 1000-N-class H_2O_2/hydrocarbon rocket system. Currently, there are at least two research groups, Purdue University [2,24] and Korea Advanced Institute of Science and Technology (KAIST) [12,13] developing the rocket system. The groups achieved a similar level of technology maturity. The rocket system in operation developed by the KAIST group is shown in Fig. 1, and its characteristics are presented in Table 1 [13,11]. The group is nearing development of a 2500-N-class rocket system. Thus the

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Fig. 1. Hot-fire test of a state-of-the-art in 1000-N-class H_2O_2/hydrocarbon rocket system [13].

Table 1
Characteristics of a state-of-the-art in 1000-N-class H_2O_2/hydrocarbon rocket system [5].

Parameter	Value
Propellant	90% H_2O_2/kerosene
Thrust at sea level	900 N
Calculated thrust in vacuum	1200 N
Specific impulse at sea level	221 s
Calculated specific impulse in vacuum	290 s
Tested operation time	30 s
Mass	1.8 kg

technology is in a stage requiring a serious exploration for application of the 1000-N class rocket system.

2. Calculation procedure

2.1. Specific impulse estimation

The specific impulse is calculated by analyzing the nozzle flow using one-dimensional approximation. Because of its relatively small size and low chamber pressure, two-temperature nonequilibrium nozzle flow analysis was applied [1,21,19]. Some results are shown in Fig. 2. As shown in the figure, the theoretical specific impulse exceeds 300 s. In the experiment shown in Fig. 1, which used an area ratio of 4.95 and a chamber pressure of 30 bar, the measured specific impulse was 221 s (Table 1), which is compatible with the results shown in Fig. 2. Note that in the mass calculation, the specific impulse was implemented with an efficiency of 95%.

2.2. Propulsion system mass estimation

To estimate the thrust-chamber mass, the thrust chamber was conceptually designed using the data on an existing 1000-N-class H_2O_2/kerosene rocket system [11,26]. Calculated results are shown in Fig. 3. It is the case for a 1000-N-class thrust chamber with the nozzle expansion ratio of 100 and the mixture ratio of 7.4. As the figure shows, the thrust-chamber mass decreases with increasing chamber pressure. In the pressure range expected for an in-space system, the expected mass is about 4 kg. Thus thrust-to-weight ratio is about 25, well within the useful range.

From the propellant mass deduced from the mission scenario, the size and the mass of the propellants and the helium pressurant tanks were calculated. A titanium alloy, Ti-6Al-4V, was chosen as the material for the propellant tanks. For the pressurant tank, a composite overwrapped pressure vessel made with T700/epoxy overwraps on a load-sharing liner made of an aluminum alloy, Al 6061-T6, was chosen [27]. It was assumed that all the tanks are cylindrical with a safety factor of 1.3. It was also assumed that the

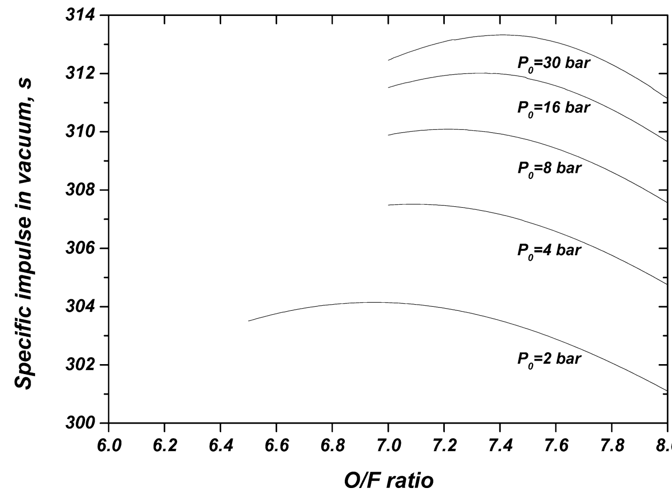

Fig. 2. Mixture ratio vs. specific impulse in vacuum for nonequilibrium flow (nozzle expansion ratio of 60).

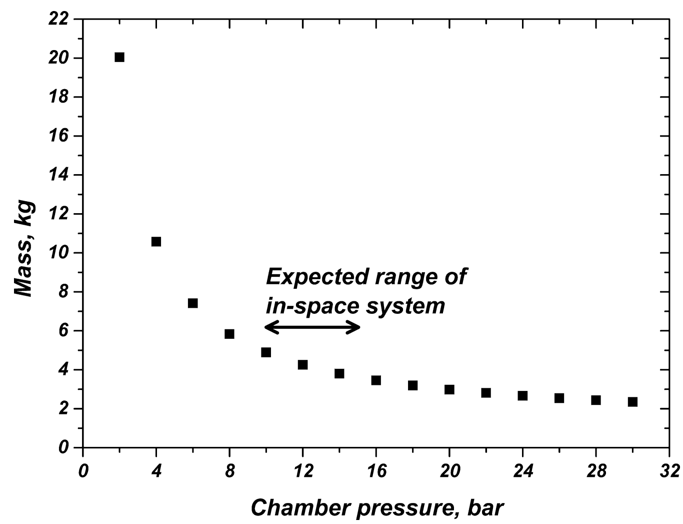

Fig. 3. Chamber pressure vs. thrust chamber mass (thrust of 1000 N, nozzle expansion ratio of 100, and mixture ratio of 7.4).

masses of the valves, pipes, and fittings are 10% of the calculated propulsion system mass.

2.3. Optimal design specifications

The optimal design specifications are found to minimize the cost function

$$J = m_0 \tag{1}$$

subject to

$$\begin{aligned} V_f - V_i &= I_{sp} g_0 \ln\left(\frac{m_0}{m_0 - m_p(1-r)}\right) - \bar{g}\Delta t \\ &= -I_{sp} g_0 \left(\ln\left(1 - \frac{m_p(1-r)}{m_p + m_{prop} + m_{other}}\right) \right. \\ &\quad \left. + \bar{g}\frac{m_p(1-r)}{F}\right) \end{aligned} \tag{2}$$

where m_0 is the total mass of a spacecraft at launch in kg, V_f is the target velocity in m/s, V_i is the velocity at the beginning of the engine burn in m/s, I_{sp} is the specific impulse in s, g_0 is the standard gravitational acceleration, 9.80665 m/s^2, m_p is the required

Table 2
Design specifications of the current apogee motors and optimal ones of H_2O_2/kerosene rocket system for GEO insertion mission [28,29,7,12,14,10].

Design specification	N_2O_4 or MON-α/N_2H_4	N_2O_4 or MON-α/MMH	H_2O_2/kerosene
Thrust	400–500 N		742 N
Thrust chamber pressure	7–15 bar		11.7 bar
Propellant mixture ratio	~1	~1.65	7.36
Nozzle expansion ratio	160–400		580
(Specific impulse)	(310–330 s)		(312 s)

MON-α: α% NO in N_2O_4.

propellant mass in kg, r is a fraction of a residual mass, $\bar{g}$ is the average gravitational acceleration during engine burn in m/s^2, Δt is burning time in s, m_{prop} is the propulsion system mass in kg which consists of a pressurant tank, propellant tanks, valves, pipes, fittings, and thrust chamber, m_{other} is the dry mass except the propulsion system mass in kg, and F is the thrust in N. The problem was solved using the fmincon function in MATLAB.

It is noted that there are typical values for the required velocity increment, $V_f - V_i$, and $\bar{g}$ given a mission scenario. Therefore the optimal design specifications would be different from the different missions. It is also noted that only the thrust out of the design specifications is expressed explicitly in Eq. (2), and the others are implicitly affecting the variables.

3. Results and discussion

In this study, the design specifications for geosynchronous orbit (GEO) insertion were derived. To compare the derived specifications to those for the existing apogee motors, it was assumed that the propulsion system of Communication, Ocean and Meteorological Satellite (COMS), the latest Korean geosynchronous satellite, is replaced with the H_2O_2/kerosene propulsion system.

Typical mission scenario is that a satellite with an apogee motor is inserted into a geosynchronous transfer orbit (GTO), and the motor is ignited several times at near the apogee of the orbit to be inserted into GEO. The design specifications of the current apogee motors are shown in Table 2 [3,9,18,4,10,17]. A mass of the satellite for GEO mission is usually between 2 ton and 5 ton at launch, but the thrust of the apogee motor is usually between 400 N and 500 N as described in Table 2; gravity loss which is the second term in Eq. (2) is quite high if the satellite is inserted into GEO with only one maneuver. Multiple-burn is also attractive for the precise positioning. Therefore typical GEO insertion is conducted with three or four maneuvers. As an example, COMS used three maneuvers [22,15]. It weighed about 2497 kg at launch [6], and carried propellants of about 1292 kg, and about 934 kg out of the mass was for GEO insertion [20]. The m_{prop} was assumed to be about 124 kg based on the tank mass estimation explained in Section 2.2. Considering multiple-burn, the $\bar{g}$ was assumed to be about 0.0084 m/s^2, and then the required velocity increment is about 1418 m/s.

The calculation results are shown in Fig. 4 and are compared with the design specifications of the current liquid apogee motors in Table 2. The chamber pressure is similar to that of the current ones. The mixture ratio is the value which the maximum specific impulse can be achieved with. The thrust and the nozzle expansion ratio are higher than those of the current values. It is thought that the higher values are to reduce gravity loss due to the lower specific impulse performance of H_2O_2/kerosene rocket system. The minimized mass of the simulated satellite is about 2508 kg which is heavier than the COMS by about 11 kg due to the lower specific impulse performance. However, considering the environmental hazards, higher cost, and higher risk using the toxic propellants, the mass increase may be forgiven.

For sensitivity analysis, contour maps were presented in Fig. 4. The upper plot in the figure is the contour map with the optimal thrust and nozzle expansion ratio, and the bottom is for the optimal chamber pressure and mixture ratio. The white asterisks in the plots are the optimal locations. The results show that the chamber pressure and the mixture ratio are affecting more to the total mass, so there are more margins in designing the thrust and the nozzle expansion ratio. It is more sensitive when the chamber pressure decreases compared with increasing the chamber pressure. Also lowering the nozzle expansion ratio with a low thrust is more sensitive than lowering it with a high thrust. Therefore if off-optimal design is required, it should be configured with near-optimal chamber pressure and mixture ratio, a lower nozzle expansion ratio, and a higher thrust. If a thrust of about 450 N is required for compatibility with the current design, the nozzle expansion ratio should be as high as possible with the optimal chamber pressure and mixture ratio.

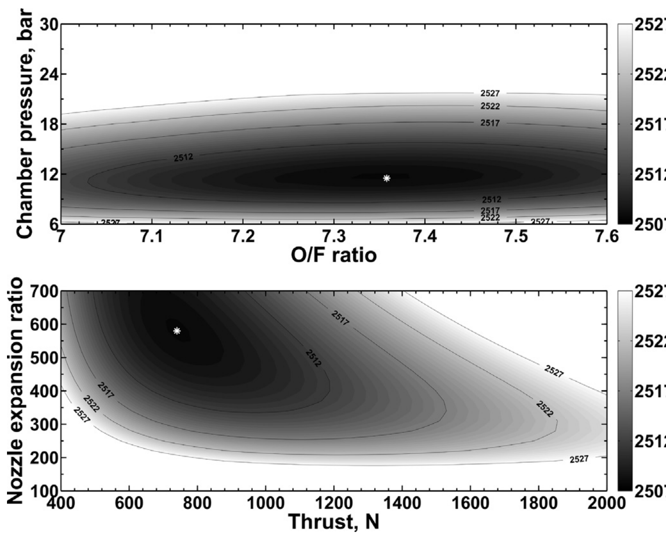

Fig. 4. Contour maps of GEO satellite mass.

4. Conclusion

A design calculation is made for a 1000-N-class H_2O_2/kerosene bipropellant rocket system which is nearing its development. The optimal design specifications of such a system for an apogee kick motor used for GEO insertion were determined. The optimal design specifications are configured with a thrust of 742 N, a chamber pressure of 11.7 bar, a mixture ratio of 7.36, and a nozzle expansion ratio of 580. It was found that, in the case of the latest Korean geosynchronous satellite, replacing the current apogee kick motor with the H_2O_2/kerosene rocket system would result in a mass increase of only about 11 kg, less than 0.5% of the total mass.

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