Research paper

Strand test characteristics of non-self-combustible solid propellants in burning control with N_2O supply

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Abstract

To cope with the increasing number of diverse small satellite missions, development of new propulsion systems that can meet the demand of "rapid motion change" missions is a key issue in space programs. Electric thrusters, which are used almost exclusively now, cannot match the intrinsically high thrust of solid propellant motors. Hence, if solid propellant motors could be modified and equipped with the capability of burning control at will, such motors would be able to meet the increasing demand of rapid motion change; moreover, their simplicity and reliability would enable them to be used as effective reaction control systems. We have been conducting studies on methods to provide burning control features to solid propellants used for small/micro thrusters. Herein, we have proposed the use of an original fuel-rich solid propellant coupled with a flowing subsidiary oxidizer, such as nitrous oxide (N_2O). Experiments with a modified strand burner revealed that the flow of N_2O can control the burning of solid propellants used with a burning rate of 5–10 mm s^{-1}, increasing with the pressure value of up to 0.4 MPa. The results suggest that a reaction control motor using solid propellants with the proposed concept could achieve higher thrust than that is possible with electric propulsion.

Keywords: solid propellants, non-self-combustible, combustion control, nitrous oxide, strand test

1. Introduction

In recent years, a variety of small satellite missions have been proposed and planned, instigating the development of new propulsion systems. Electric thrusters have demonstrated successful mission performances[1], but they are not capable of coping with rapid motion change requirements, which are expected in more flexible future missions, due to their low thrust levels[2].

High thrust can be offered by chemical propellants because of large chemical energy stored within themselves; hydrazine currently holds an unchallenged position in space programs as a liquid propellant with reasonable burning controllability. However, hydrazine requires complex flow control systems, foreign-made decomposition catalysts with unstable availability, and tedious handling due to its toxicity and carcinogenicity[3]. On the other hand, solid propellants also generate high thrust and are advantageous because they contain a simple mechanism as they have fewer moving parts, which offer high propellant mass ratios and reliability.

However, solid propellants inevitably continue to burn once ignited, as seen in launchers from the ground, and are designed only to be used for faithful execution of pre-programmed combustion patterns. To minimize this constraint, we have been investigating new ways to control the burning of solid propellants and extend their use with regard to reaction control motors in upper stages[4)-6)]. We have anticipated the basic idea that solid propellants can achieve combustion control if they are manufactured in terms of intentionally separated from the ideal fuel-to-oxidizer ratios. This idea was confirmed in our previous studies with fuel-rich propellants in low chamber pressure conditions of less than 0.5 MPa. We refer to such fuel-rich propellants as non-self-combustible propellants

(NSC propellants).

To ignite and sustain combustion, we have tested several types of external power, such as arc discharges, plasma jets, and laser irradiation[7)−13)]. However, these methods necessitate external power usage during the operation throughout and cannot avoid deterioration of the combustion and specific impulse (*Isp*) due to the shift from the ideal mixture ratios of propellants. The idea proposed is to eliminate the above disadvantages by supplying N_2O, which would compensate for the insufficient oxidizer in the propellant subsidiary, eliminate external power, and augment the propulsion performance[14)]. Under the existence of N_2O, significantly less power would be needed to ignite NSC propellants, and the supplied N_2O could sustain as well as control the burning without external power. We will report the experimental results conducted with a strand burner apparatus modified for flowing N_2O to demonstrate the feasibility of the above proposal.

2. Background to the concept proposed and experimental

2.1 Concept and propellant selection

Figure 1 is a schematic diagram displaying how the proposed concept of a reaction control rocket motor is actuated using N_2O and our NSC solid propellant-fuel-rich hydroxy terminated poly-butadiene / ammonium-perchlorate (HTPB/AP) composite type. The *Isp* of the propellant changes with HTPB fuel content in HTPB/AP, as shown in Figure 2, which is obtained via calculation with a CEA program under the conditions of the chamber pressure, $Pc = 0.3$ MPa, and nozzle area ratio, $\varepsilon = 80$[15),16)]. The figure also depicts the zoning of self-combustible (hypergolic)/non-hypergolic regions, which have been confirmed by our preliminary experiments. Because the propellants are non-self-combustible for the proposed purpose, but are desired to have as high *Isp* as possible and to require less additional oxidizer, we judged that the propellant with a mass composition of HTPB/AP = 30/70 is appropriate. The AP used here is monomodal with a mean particle diameter of 5 µm. Figure 3 shows the sequence of the combustion start/stop cycle; the NSC propellant, ignited immediately after the supply of N_2O, continues to burn until N_2O cuts off. This cycle can be repeated as required.

2.2 Reaction between NSC main solid propellants and supplemental oxidizer

When N_2O is supplied as a subsidiary oxidizing propellant, the *Isp* will be augmented to some extent with the mass ratio of N_2O to the burned NSC solid propellant, m_{N_2O}/m_{SP}, as shown in Figure 4. The mass ratio yielding the best *Isp* is at $m_{N_2O}/m_{SP} = 0.9$, which is the target value. While the mass flow rate of N_2O, $\dot{m}_{N_2O}$, can be set arbitrarily at will, the mass consumption rate of the NSC propellant, $\dot{m}_{SP}$, depends on the reaction with N_2O. It is important to note that the mass ratio m_{N_2O}/m_{SP} and the mass rate ratio $\dot{m}_{N_2O}/\dot{m}_{SP}$ have the same value for steady state cases assumed here.

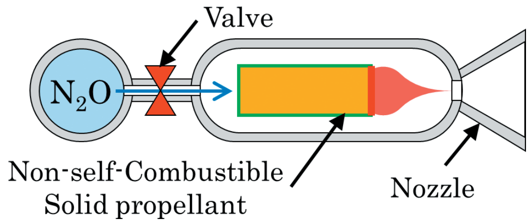


Figure 1 Concept of the proposed reaction control rocket motor.

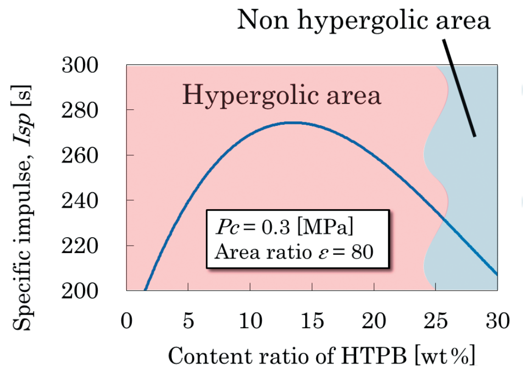


Figure 2 Specific impulse change with HTPB fuel content in HTPB/AP.

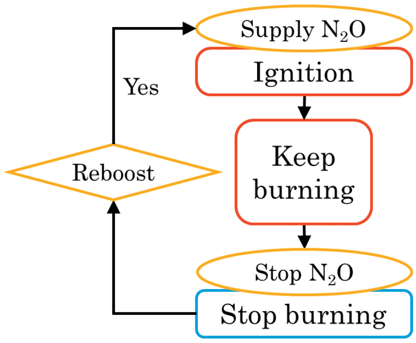


Figure 3 Sequence of burning cycle.

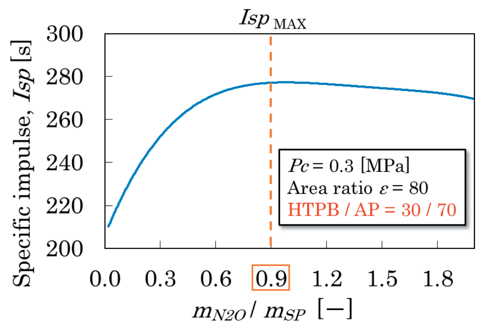


Figure 4 *Isp* augmented with m_{N_2O}/m_{SP}, mass ratio of N_2O to burned NSC propellant.

2.3 Test apparatus

This study requires such a combustion test apparatus that measurement can be obtained under N_2O flow supply. Figure 5 is a schematic and photo of the modified strand burner used in this study; the volume inside the burner is 1.56 L, and it has an observation window through which the burning is monitored and recorded to evaluate burning rates. Oxidizer N_2O is flowed to the surface of the NSC propellant, set in a heat-resistant glass tube 100 mm in length without being mixed with the inert N_2 gas flow used for setting ambient pressures. The shape of the NSC

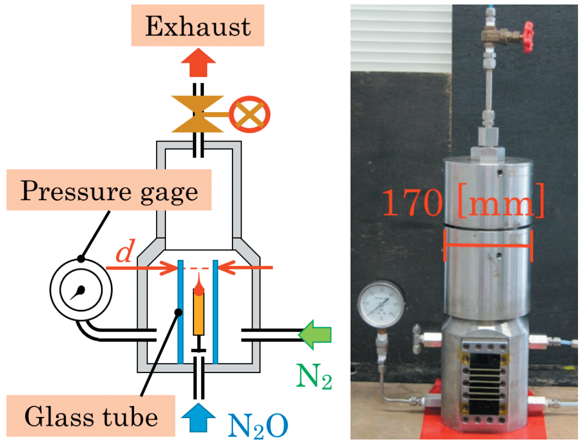
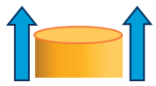
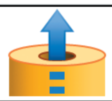
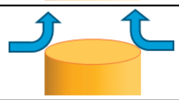


Figure 5 Schematic and photo of the modified strand burner.

Table 1 Three burning schemes examined.

N_2O supply method		Propellant shape
Upward		Filled
Central		Hollow
Side		Filled

propellant used was a 100 mm long rod with a 7 mm diameter. Ignition was conducted with an electrically heated wire.

2.4 Test procedure

The test procedure is different from that with a conventional strand motor; N_2O flow rates are set by anticipating that the amount of NSC propellant reacted and consumed will be in terms of favorable mass ratios with N_2O mentioned above, i.e. $m_{N_2O}/m_{SP} = 0.9$. Combustion experiments are repeated until a satisfactory m_{N_2O}/m_{SP} is obtained by changing the N_2O flow rate at each ambient pressure set by the N_2 flow. The ambient (chamber) pressures, Pc, are not affected significantly by the NSC propellant-N_2O reactions because N_2 flow rates are high enough. Comparative experiments with simulated gas $N_2 + 1/2\ O_2$ were also conducted to measure the heat release effect in N_2O decomposition. The part of the NSC propellant used for the burning measurement is the section of 50 mm in length from the point where ignition is fully assured. The extinction was judged by video observation of the propellant burning.

3. Results and discussion

The effect of the N_2O flow scheme on the NSC propellant burning was examined in three burning cases, as shown in Table 1. Figure 6 displays the relationship between the N_2O flow rate and burning rate, r, with • symbol in upward flow scheme. The cases with central flow (△) and side flow (□) were unstable and unreliable in

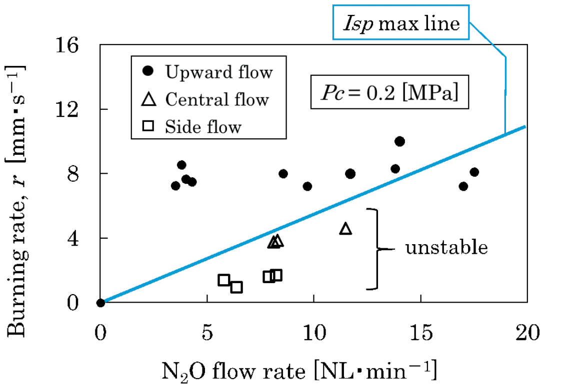


Figure 6 Relation between N_2O flow rate and burning rate.

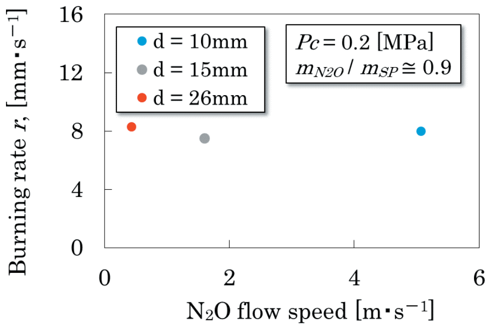


Figure 7 Influence of N_2O flow speed on the solid propellant burning rate.
※N_2O flow ratio ≈ 15 NL·min^{-1}

most cases; hence, excluded in the rest of this study, as aforementioned. The *Isp* max line corresponds to the ideal burning rates, at which the maximum *Isp* is obtained providing that the reaction between N_2O and the NSC propellant is completed. In other words, the points above the line indicate a fuel-rich situation, while the points below the line indicate a fuel-lean situation, which are increasingly less ideal as they stray further away from the line.

The effect of the N_2O flow "speed" was examined with different inner-diameters of glass tubes, $d = 10$ mm, 15 mm, and 26 mm, at a flow rate of nearly 15 NL·min^{-1}. The result shown in Figure 7 assumes that the influence of the N_2O flow speed on the burning rate is small; hence, the rest of this study was conducted only with a tube of 10 mm diameter.

Shown in Figure 8 is the case in which N_2O yielded much higher burning rates than with its simulated gas N_2 $+1/2\ O_2$. This result is primarily caused by the effectiveness of heat release in N_2O decomposition as

$$N_2O = N_2 + 1/2\ O_2 + Q\ \ (81.6\ \mathrm{kJ{\cdot}mol^{-1}}).$$

However, increase in N_2O flow rate has almost no effect on the NSC propellant regression. This result can be attributed to the fact that higher oxidizer flow cannot necessarily induce more reaction with NSC propellants. The intersection of the *Isp* max line and solid symbols designates the desirable combination of N_2O supply and NSC propellant consumption, $\dot{m}_{N_2O}/\dot{m}_{SP} = 0.9$.

The relation between combustion chamber pressure, Pc in MPa, and burning rate, r in mm·s^{-1}, is shown in Figure 9. Same as the ordinary solid propellants, the burning rate

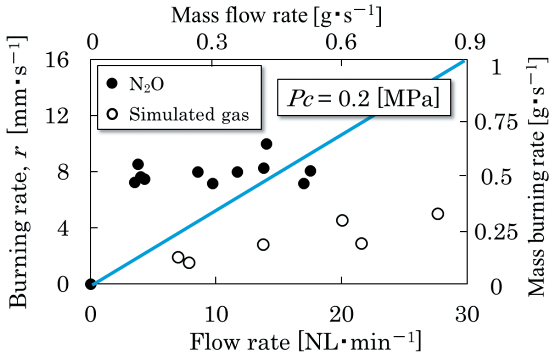


Figure 8 Comparison of burning rates in the cases with N_2O and its simulated gas.

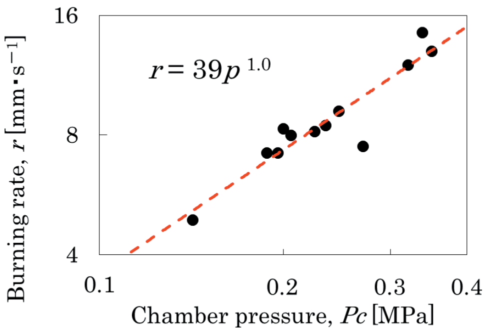


Figure 9 Relation between combustion chamber pressure and burning rate.

depends on the pressure; the empirical formula of the burning rate obtained is $r = 39\ Pc^{1.0}$ in the pressure range under 0.4 MPa. We should note here that the pressure index does not have to be less than unity for stable burning control unlike ordinary propellants that have hypergolicity because burning, including extinguishment, can be controlled with N_2O flow.

4. Conclusions

Through combustion and extinction tests using a strand burner apparatus modified for flowing nitrous oxide, the overall result has demonstrated the viability of our proposed concept. The following are specific conclusions obtained from the experiment:

1) Fuel-rich HTPB/AP = 30/70 composite solid propellant, which is non-self-combustible under 0.5 MPa, can be burned and extinguished at will with the N_2O flow and cut-off.
2) The N_2O flow rate does not significantly influence the resultant solid propellant consumption; $m_{N_2O}/m_{SP} = 0.9$, which gives the theoretically ideal Isp, is realized at $\dot{m}_{N_2O} = 15$ NL·min^{-1} and resultant $r = 8$ mm·s^{-1} with the specific configuration and size used in this study.
3) The relation between the burning rate and the pressure can be expressed by the same form of formula, $r = a\ Pc^n$, as is the case with ordinary solid propellants, but the technique proposed is capable of combustion control including extinction in a pressure range up to 0.4 MPa.

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