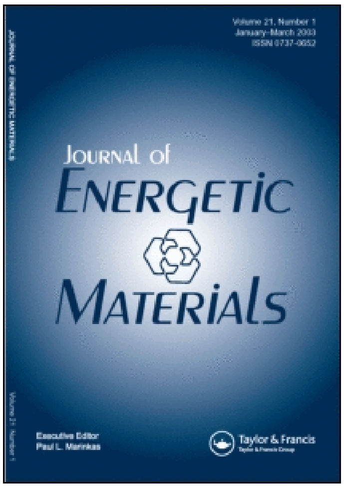

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Yoshino Oyumi[a]; Kikokazu Nagayama[a]
[a] Third Research Center, Technical Research and Development Institute, Tokyo, Japan

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DEVELOPMENT OF HIGH BURN RATE AZIDE POLYMER PROPELLANT

Yoshio Oyumi and Kiyokazu Nagayama

Third Research Center, Technical Research and Development Institute,
Japan Defense Agency, 1-2-10 Sakae, Tachikawa, Tokyo 190, Japan.

ABSTRACT

The combustion characteristics of high burn rate azide polymer composite propellant were examined by using ϕ 70 mm BMI composite rocket motor of L/D = 16. Azide polymer propellants have much higher specific impulse than HTPB propellant at below the AP content of 85%. AP/B/N propellants showed a plateau-mesa burning at a pressure range between 7 MPa and 15 MPa with a burn rate of approximately 28 mm/s. Very fine AP, however, diminished this favorable combustion characteristics. The AP/B/N also had excellent mechanical properties at an operational temperature range to enable the case bonded. The BMI motors burned stably and showed an excellent performance. The thickness of the insulation at the aft closure, 5.5 mm including a stress relief boot, was good enough.

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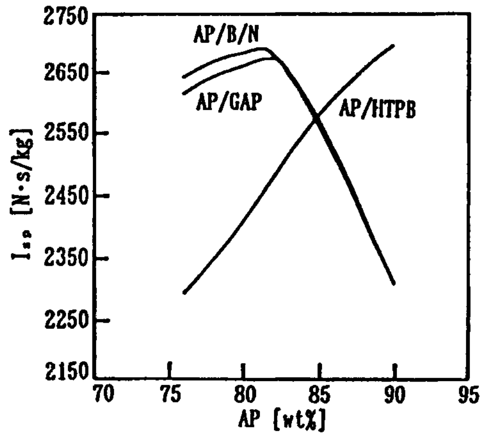

FIGURE 1 Specific impulse of ammonium perchlorate (AP) composite propellants at 20 MPa versus AP content.

INTRODUCTION

The energetic materials and the high operating pressure have become to be important to increase the performance of the reduced smoke rocket motors. The azide polymer propellants has become familiar to the propellant community to satisfy the insensitive munition characteristics. The high operating pressure also has been accomplished by the composite material motor case without any weight demerit. The operating pressure is dominated by the combination of the nozzle throat area, the burning surface area and the propellant burn rate. The high burn rate propellant was focused in order to design the high mass-ratio rocket motor in this study. A heat resist composite material, carbon/bismaleimide resin, was applied to flight weight rocket motors.

PROPELLANT SELECTION

The theoretical calculation at 20 MPa (1) shown in Fig. 1 indicated that the ammonium perchlorate (AP)/3,3-bis(azidomethyl)oxetane/3-nitrato methyl-3-methyl oxetane (7/3) copolymer (B/N) propellant (AP/B/N) and

TABLE 1 Propellant compositions

No	BDR	AP	Fe_2O_3	ZrC
1	21.9	73.3 *1	2.9	1.9
2	22.1	74.0 *2	1.9	1.9
3	22.5	75.4 *3	0.1	2.0

BDR: BAMO/NMMO(7/3) azide polymer binder
AP: ammonium perchlorate (average diameters were *1 : 106 μ m, *2 : 63 μ m, *3 :11 μ m)
Fe_2O_3 : ferric oxide ZrC: zirconium carbide

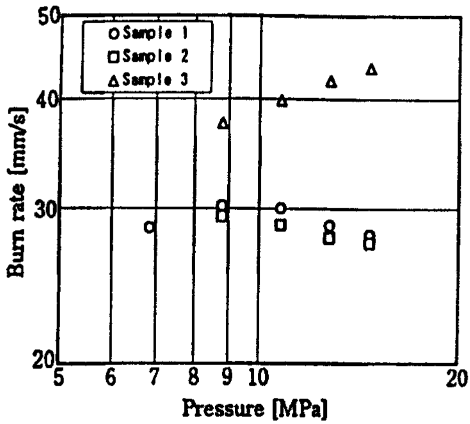

FIGURE 2 Burn rates of azide polymer composite propellants.

AP based glycidyl azide polymer propellant (AP/GAP) have much higher specific impulse than AP based polybutadiene propellant (AP/HTPB) at below the AP content of 85%. The azide polymers combustion is more exothermic than polybutadiene and contributes to the better performance. The less reactive products, such as nitrogen and hydrogen cyanide, from the azide polymer decomposition, however, decrease the specific impulse at the high oxidizer content of greater than 82% because of the lower oxidation reaction. AP/B/N propellant has higher specific impulse than AP/GAP propellant at from 75% to 30% of AP content, which are the most feasible AP content.

The strand burn rate–vs–pressure curve was shown in Fig. 2. The propellant formulations were listed in Table 1. The average particle size of AP was varied to reduce the content of the ballistic modifier of ferric oxide and to try to keep the burn rate 27 mm/s at 15MPa. 2% zirconium carbide was added to the propellant as a pressure oscillation suppression agent. Sample 1 and Sample 2 showed a plateau–mesa burning characteristics and almost the same burn rate. The plateau–mesa burning characteristics is the most favorable to the motor design. Very fine AP, however, diminished the plateau burning at a pressure range between 9 MPa and 15 MPa. This result indicated that the combustion mechanism of the propellant tested here was dominated by the particle size of AP. The structure of combustion wave near the burning surface was influenced by the modification of AP particle size. Both propellants, Sample 1 and Sample 2, showed similar combustion characteristics but Sample 2 was selected to the full scale combustion test because of its lower content of the catalyst.

HEAT RESIST COMPOSITE ROCKET MOTOR COMBUSTION TEST

In consideration of an aerodynamic heating and an insensitive munitions characteristics carbon/bismaleimide (BMI) filament wound composite structure was used as a rocket motor case. The design pressure was 24 MPa with a fiber of IM600 and 7 plys of composite layer. The wall thickness was approximately 1.5 mm. The selected azide polymer propellant, Sample 2, was case bonded. Separate end closures were selected at the both ends. Manufacture of a case bonded charge necessitated a relatively large opening at one end for ease of propellant filling and core removal so that a separate closure was required at one end. For small diameter motors, such as 70 mm, there was little difference mass wise between separate and part integral closures and body manufacturing considerations became the criteria. As aft closure had to be insulated the method of application of the insulation was considered. Rigid insulant of

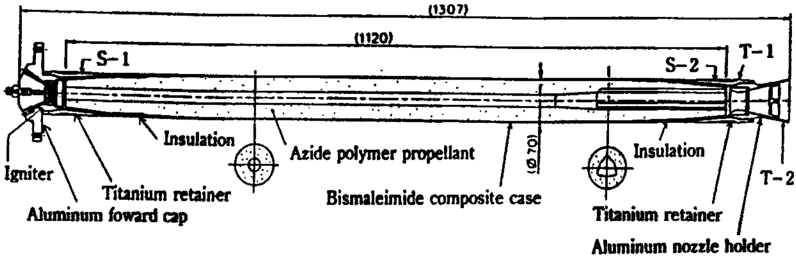

FIGURE 3 Schematic drawing of the BMI composite rocket motor

silica-phenolic resin and graphite nozzle throat insert were bonded into the component with an aluminum nozzle holder. The component was finish machined prior to application of the insulation and a gap filling resin was also used.

The cut-view of the BMI composite rocket motor was shown in Fig. 3. The ratio of chamber length to diameter, L/D, was 16. The propellant grain consisted of three types of the configurations, which were a cylindrical part, a tapered part and a three slots part from the forward. Although an equal or a gradual decrease in core diameter was reported to be effective to control the pressure oscillation [2], a gradual increase type was selected here to obtain a high volumeric mass fraction. The three slots part plays to ease an erosive burning at an aft part of the propellant grain. A cross sectional area at the three slots part should be greater than the nozzle throat area to keep choking at the nozzle throat.

The combustion tests were conducted four times at three different propellant initial temperatures. The results were listed in Table 2. The typical pressure-vs-time curve and thrust-vs-time curve were shown in Fig. 4. The motor burned out within one second and showed no combustion instabilities. There was no severe ignition pressure peak at −30 °C. These results indicated that there was good relationship between an amount of igniter powder and a free volume of the rocket motor.

TABLE 2. Results of BMI composite rocket motor combustion tests

Test No.	1	2	3	4
T_{ini} [K]	243	243	293	333
D_t [mm]	27.45	27.53	27.37	27.48
t_b [s]	0.844	0.866	0.796	0.732
P_{av} [MPa]	14.45	14.16	15.59	17.11
P_{max} [MPa]	15.53	15.24	16.91	18.57
F_{av} [kN]	12.69	12.67	13.76	15.02
F_{max} [kN]	14.19	14.13	15.39	17.17
I_t [kN·s]	11.46	11.55	11.56	11.60
I_{sp} [N·s/kg]	2329	2336	2352	2365
r_b [mm/s]	27.3	26.6	28.9	31.4

T_{ini} : initial temperature of the propellant, D_t : nozzle throat diameter, t_b : burn time, P_{av} : average chamber pressure, P_{max} : maximum chamber pressure, F_{av} : average thrust, F_{max} : maximum thrust, I_t : total thrust, I_{sp} : specific impulse, r_b : average burn rate.

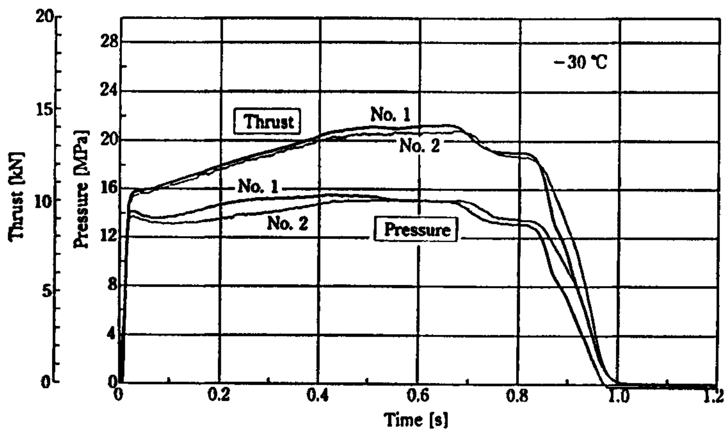

FIGURE 4 Pressure-vs-time curve and thrust-vs-time curve at −30 ℃

The effect of the propellant initial temperature was shown in Fig. 5. The lower the initial temperature, the burn time became long. Although there was no influence on the total impulse, the maximum thrust was varied

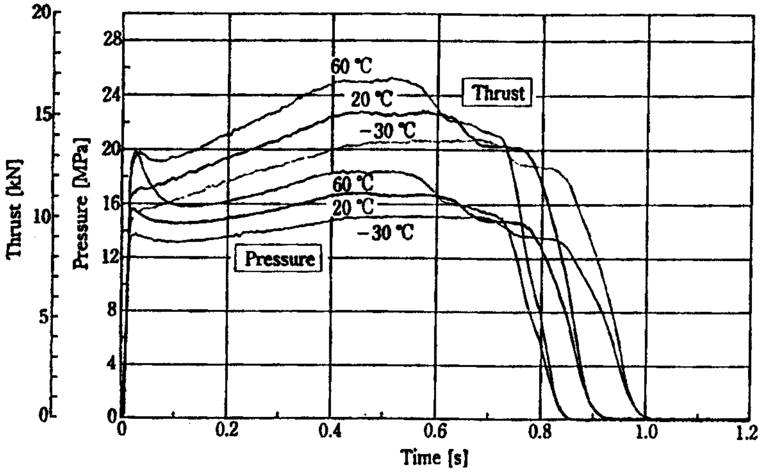

FIGURE 5 Effect of the propellant initial temperature on the chamber pressure and the thrust

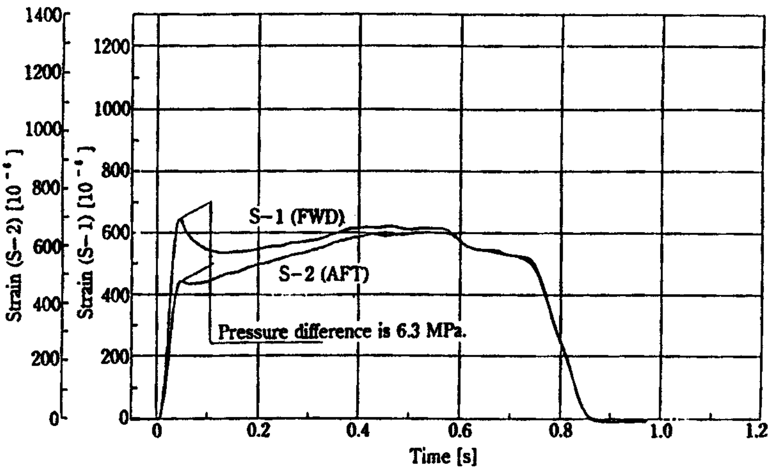

FIGURE 6 Strain data of S−1 and S−2 at 60 °C

with the chamber pressure change. The azide polymer propellant showed relatively small temperature sensitivity of the burn rate. That of igniter powder, however, was large and this caused a large ignition peak at 60 °C .

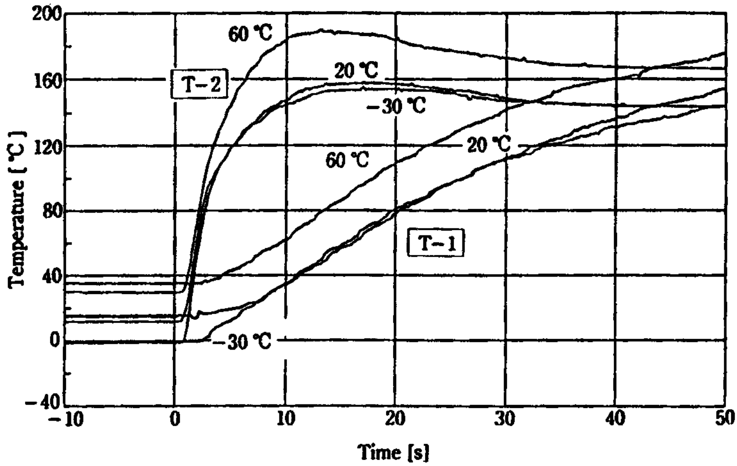

FIGURE 7 Temperature profile near the nozzle

TABLE 3 Ablation data at an initial propellant temperature of 60 ℃ [mm]

Position	a	b	c	d	e	f	g	h	i	j	k	l	m	n	o	p
Ti	5.5	5.5	5.5	3.0	0.5	0.5	0.5	0.5	0.5	0.5	2.0	3.5	5.5	5.5	5.5	5.5
Tf	5.5	5.5	5.5	3.0	0.5	0.5	0.5	0.5	0.5	0.4	1.3	2.3	3.4	1.9	1.7	0.0

The locations of "a" to "p" were 0 mm, 10 mm, 80 mm, 160 mm, 300 mm, 400 mm, 500 mm, 670 mm, 885 mm, 935 mm, 985 mm, 1020 mm, 1070 mm, 1105 mm, 1120mm, and 1130mm from the head-end of the propellant grain, respectively. "a" to "h" located at the cylindrical part , "i" existed in the tapered part and "j" to "p" belonged to the slot part. "Ti" is an initial thickness of the insulation and "Tf" is a final thickness.

The combustion tests of the BMI composite rocket motor were conducted stably at a temperature range from −30 ℃ to 60 ℃ .

Relatively large L/D rocket motor tends to have a large pressure difference between aft and fwd parts of the chamber. Strain data of both ends at 60 ℃ were shown in Fig. 6. The pressure difference at an initial stage was 6.3 MPa. However, with progressing the combustion and increasing the cross sectional area of the propellant grain, the pressure difference rapidly decreased.

Temperature profiles near the nozzle were shown in Fig. 7. The locations of the thermocouple were shown in Fig. 3. T-1 located at a backside of a throat insert and T-2 located 5 mm far from the nozzle cone end. Therefore, a temperature rise rate at T-2 was much higher than that of at T-1. Since fins are installed at near the T-1, it is important to know the temperature profile around this part. According to Fig. 7, temperature increased with time at the initial stage but the temperature of T-2 at 5 seconds after the ignition was approximately 140 °C. These results indicated that an expected electronics was able to survive in the operational conditions.

The combustion gas velocity at an initial stage of L/D = 16 rocket motor was expected to be approximately 700 m/s. Therefore, the insulation was very important. The combination of Kevlar filled EPDM rubber and silica-phenolic resin were excellent insulation material.

The ablation data were listed in Table 3. These data were measured after the removal of char layer. The severe ablation was observed at aft part of the insulation whose material was Kevlar filled EPDM. The insulations at aft end and fwd end were thick because of the stress relief boot. As shown in Fig. 3, the chamber at aft end was tapered because of the filament wound of titanium retainer. Relatively large L/D rocket motor tends to have a large pressure difference between aft and fwd parts of the chamber. This pressure difference causes the high combustion gas velocity, which dominates the erosive burning. The regression rate of the propellant at the aft part becomes greater by the erosion and aft part insulation was exposed to hot gas stream earlier than expected. Since the burning surface reached to the insulation surface about 0.3 s later, the insulation at three slots part severely damaged. No damage was observed at the cylindrical part and the tapered part. At the position "p" all insulation materials were ablated away but the temperature profile at the T-1 was good enough to the

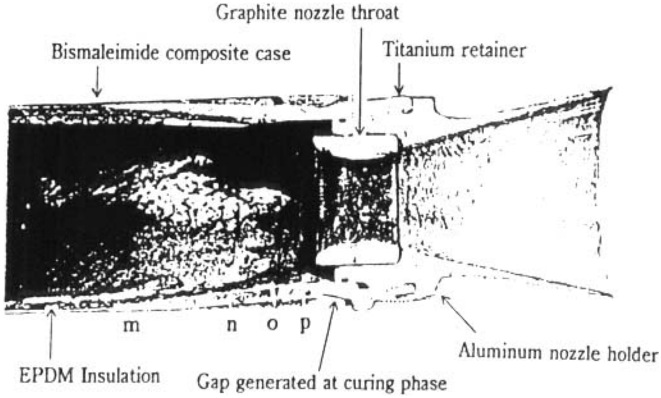

FIGURE 8 Photo–picture of the aft closure part tested at 60 °C

operation. Therefore, there was no need to increase the insulation thickness at the three slot part. If propellant liner coating or some humidity resist coating material can act as a pressure seal, insulation is not needed at the cylindrical part and the tapered part, except a stress relief boot.

A photo–picture of the aft closure part of the cut away motor tested at 60 °C was shown in Fig. 8. The positions, "n", "o", "p" in Table 3, were also shown. There was no damage in the titanium retainer, aluminum nozzle holder, and BMI case. although approximately 1 mm gap was generated at the curing process of BMI, BMI showed an excellent adhesive properties with the titanium retainer. Additional hoop layer at the retainer also showed an excellent reliability. This joint design has a continuous carbon fiber and shows an advantage in the mechanical properties and in the manufacturing. But the difference in the heat expansion ratio between the composite material and the metal part, this design has a limitation in the curing temperature. The motors used here were cured at 180 °C .

CONCLUSIONS

The combustion characteristics of high burn rate azide polymer composite propellant were examined by using ϕ 70 mm BMI composite rocket motor of L/D = 16. Azide polymer propellants have much higher specific impulse than HTPB propellant at below the AP content of 85%. AP/B/N propellants showed a plateau-mesa burning at a pressure range between 7 MPa and 15 MPa with a burn rate of approximately 28 mm/s. Very fine AP, however, diminished this favorable combustion characteristics The AP/B/N also had excellent mechanical properties at an operational temperature range to enable the case bonded. The BMI motors burned stably and showed an excellent performance. The thickness of the insulation at the aft closure, 5.5 mm including a stress relief boot, was good enough.

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