Research paper

Plasma torch igniter for hybrid rockets

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Abstract

In this study, a plasma torch is used as an igniter for hybrid rockets. This igniter contains no explosive and can be used multiple times to restart a rocket. This plasma torch is not cooled with water or other fluids for easy use in space. Ignition experiments were conducted using a small lab-scale unchoked hybrid rocket. FT-0070 wax, FT-105 wax, polymethylmethacrylate, and polyethylene were used as fuels, and gaseous oxygen was used as the oxidizer. These fuels were ignited successfully by using the plasma torch with an ignition assistant wax fuel, which was mounted at the igniter exit. Subsequently, the rocket was ignited twice in succession by using the plasma torch. Finally, the plasma torch was used to ignite a lab-scale choked hybrid rocket with a rated chamber pressure of 3 MPa during firing tests. The ignitions were achieved by combustion of the ignition-assistant fuel vapor or melted droplets produced by the plasma torch. Thus, we demonstrated that a plasma torch igniter is suitable for use in space flights.

Keywords: plasma torch, igniter, hybrid rocket

1. Introduction

The fuels used in hybrid rockets, such as wax, HTPB, and polyethylene, are generally not explosive. Therefore, a hybrid rocket has little explosion risk and a high level of safety advantage. In addition, a hybrid rocket can possibly be restarted. Hence, an igniter without explosives that can restart a rocket is required for hybrid rockets. Recently, a restart-capable arc igniter was developed by Whitmore et al.[1]. In the present study, a plasma torch is tested as an igniter for a hybrid rocket.

A plasma torch is composed of a cathode rod along its centerline; working gas flows around the cathode and is ejected through the anode nozzle. Arc discharge occurs between these electrodes, and a part of the working gas changes to plasma. The temperature of this plasma is 3000−6000 K. The igniter electrodes are exposed to high-temperature plasma and, as a result, wear out easily. Therefore, the electrodes are generally cooled, for example, with water. In the past, plasma torches required heavy power production units. However, recently, owing to recent advances in battery technology, the mass per unit capacity has decreased dramatically, for example, in the case of lithium−ion batteries. Therefore, the use of plasma torches to ignite hybrid rockets, in addition to an initial launch vehicle stage, is desirable.

Ignition and combustion using plasma torch devices have been studied by many researchers since Weinberg et al.[2), 3)]. Kimura et al. succeeded in igniting hydrogen gas in a supersonic airflow by using a plasma torch[4)]. In the present study, we attempt to ignite a small lab-scale hybrid rocket under atmospheric pressure by using an uncooled plasma torch. This is followed by firing tests of a hybrid rocket at a rated pressure of 3 MPa by using a plasma torch.

2. Plasma torch for hybrid rocket ignition

The plasma torch used in this study is shown in Figure 1. It consists of a copper anode, stainless steel body, tungsten cathode, and stainless holder. The outer diameter of the body is 20 mm, and the diameter of the anode exit hole is 3 mm. The total length of the torch is approximately 120 mm. The working gas used here is Argon (Ar); however, Ar can be replaced with helium for a practical space rocket because space rockets generally use helium to supply oxidizers. Here, Ar is used owing to its low cost and because its thermal properties are similar to those of helium given that both gases are monoatomic molecules. The plasma torch has no cooling device to

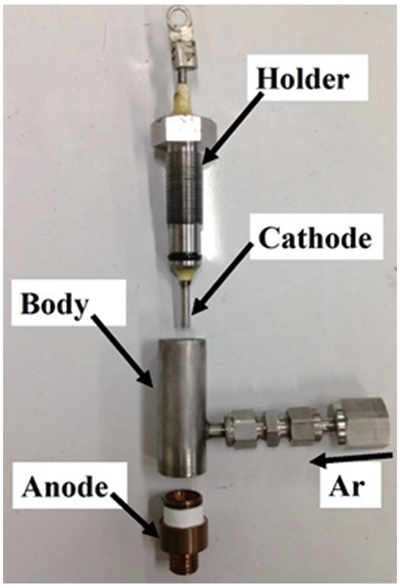


Figure 1 Plasma torch.

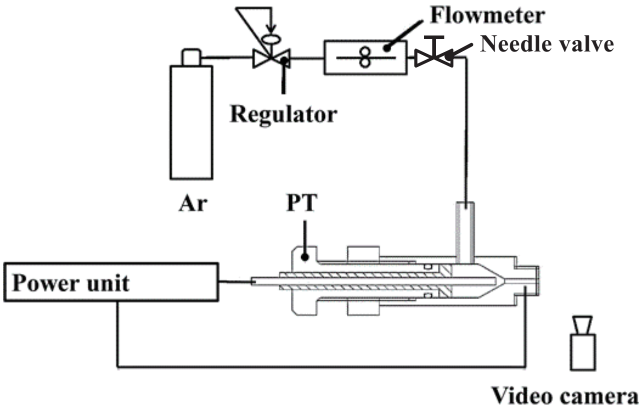


Figure 2 Plasma torch operation test setup.

Figure 3 Power unit of plasma torch.

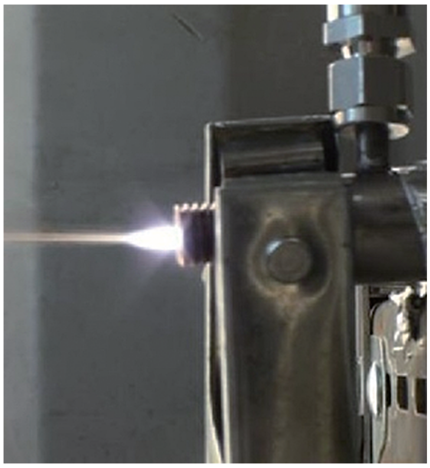


Figure 4 Ar plasma jet.

ensure ease of use in space; this means its working duration is limited.

Plasma torch operating tests were conducted using the setup shown in Figure 2. The electrical power unit used was an off-the-shelf plasma-cutting machine CT-312, as shown in Figure 3. This machine comprises a high-frequency, high-voltage power source that initiates an arc discharge as well as a direct current (DC) power source. After the discharge starts, the power source switches to the DC power source automatically. The DC can be controlled from 15 A to 30 A by using a knob mounted on the unit.

The Ar working gas was supplied from a 7 Nm^3 gas cylinder and regulated to a pressure lower than 0.2 MPa. The Ar flow rate was controlled using a needle valve from a flow meter.

A stable plasma jet was realized by adjusting the Ar flow rate and electric current. Figure 4 shows the stable plasma jet with an Ar flow rate of 80 NL min^{-1} and electric current of 30 A.

3. Ignition tests under atmospheric pressure conditions

As shown in Figure 5, a small lab-scale hybrid rocket engine was used to conduct the ignition tests. The outer and inner diameters, respectively, of the fuel case and the post-combustion chamber were 50 mm and 40 mm. Both the fuel case and the post-combustion chamber were made of polymethylmethacrylate (PMMA). Therefore, the post-combustion chamber was transparent, which facilitated confirmation of ignition. A stainless-steel pipe was used to connect the fuel case and the post-combustion chamber. The nozzle was made of graphite, and its throat diameter was 20 mm, which is sufficiently large to avoid choking. Hence, the chamber pressure was approximately equal to the atmospheric pressure. The oxidizer was gaseous oxygen (GOX). GOX was injected from two horizontally opposed ports, as shown in Figure 5. The port diameter was 4 mm. The outer diameter of the fuel grain was 40 mm, inner diameter was 10 mm, and length was 50 mm. Four types of fuels were used: FT-0070 wax, FT-105 wax, PMMA, and polyethylene. FT-0070 wax and FT-105 wax were made by Nippon Seiro Co., Ltd.

Figure 6 shows a schematic of the ignition test setup. The electric current was adjusted to approximately 30 A, and the voltage was set to approximately 36 V. The Ar flow rate was adjusted to 80 NL min^{-1}. Before ignition, GOX was supplied into the engine at a low flow rate, and it

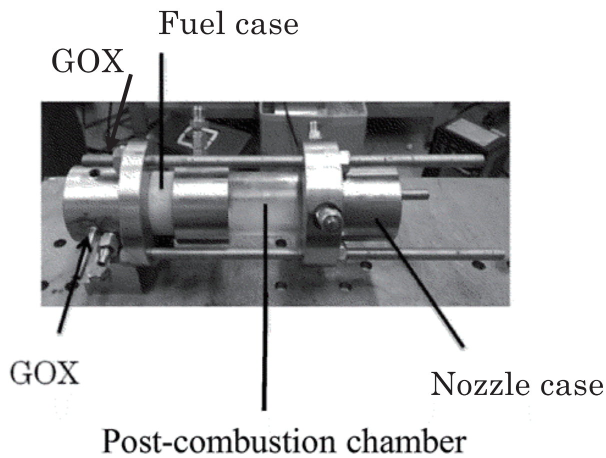


Figure 5 Small lab-scale hybrid rocket engine.

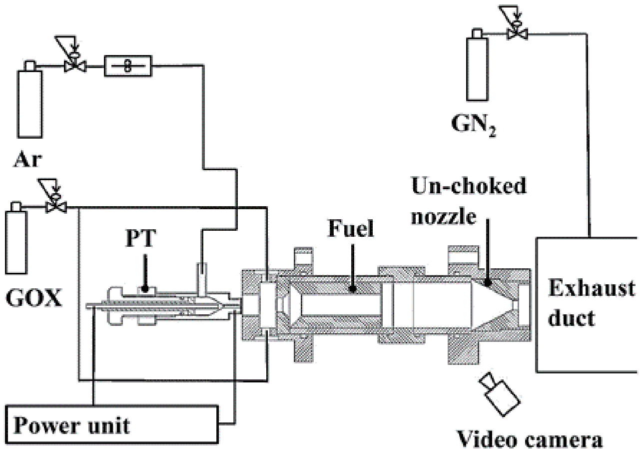
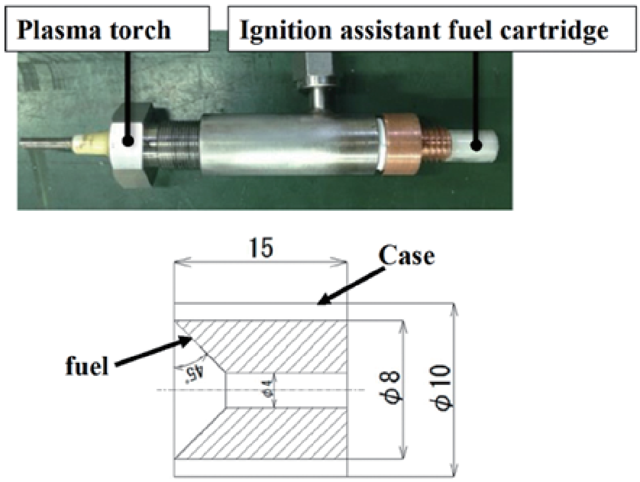


Figure 6 Schematic of ignition test setup.

filled the chamber. The plasma torch was then turned on, and the GOX flow rate was increased to approximately 0.01 kg s^{-1}. It is possible to determine whether ignition occurs via visual observation and video camera recording.

Our first set of attempts to ignite the rocket by using the plasma torch were unsuccessful, possibly because the length of the plasma jet flame was only 10 mm and could not reach the fuel surface. Then, an ignition-assistant fuel cartridge was attached at the plasma torch exit, as shown in Figure 7. The ignition-assistant fuel was set in the PMMA pipe, the outer diameter of which was 10 mm. The outer diameter, inner diameter and length of the ignition-assistant fuel cartridge were 8 mm, 4 mm, and 15 mm, respectively. The plasma jet flowed through the port of the ignition-assistant fuel, and vaporized or melted assistant ignition fuel was injected into the engine with the plasma jet. The vaporized or melted assistant ignition fuel was expected to serve as a good ignition source. Two types of ignition-assistant fuels were used: one was composed of FT-0070 and the other of PMMA.

The ignition test results obtained using the ignition-assistant fuel are listed in Table 1. When using the FT-0070 ignition-assistant fuel, ignition succeeded for all types of fuels. However, when using the PMMA ignition-assistant fuel, ignition was unsuccessful in the case of FT-0070 fuel. Figure 8 shows a photograph of successful ignition. Figure 9 shows a photograph of unsuccessful ignition.



(unit: mm)

Figure 7 Ignition-assistant fuel cartridge.

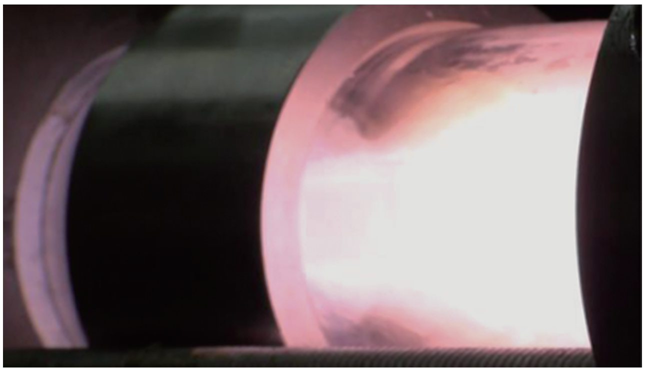
Table 1 Ignition tests results.

Ignition-assistant fuel	Fuel			
	FT-0070	FT-105	PMMA	PE
FT-0070	Good	Good	Good	Good
PMMA	NG	–	–	–

Tests without fuels were conducted using the same setup to reveal the effects of the two types of ignition-assistant fuels under the same plasma torch operation conditions as those in the ignition tests. When using the FT-0070 ignition-assistant fuel, strong light emission from the combustion flame was observed, as shown in Figure 10. However, when using the PMMA ignition-assistant fuel, light emission was weak, as shown in Figure 11.

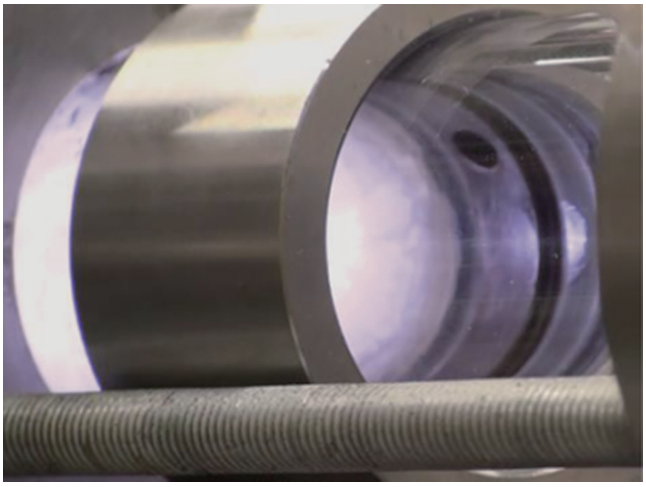
Then, to investigate ignition-assistant fuel consumption, tests were conducted without fuel by changing the duration of the plasma torch operation. Figure 12 shows the relationship between the reduction mass of the FT-0070 ignition-assistant fuel and the duration of plasma torch operation. The reduction mass reached 0.47 g in approximately 5 s; here, 0.47 g is the theoretical total mass of the FT-0070 ignition-assistant fuel. Therefore, FT-0070 was considered to have been consumed completely in 5 s. Figure 13 shows the regression rates of FT-0070 as a hybrid rocket fuel and ignition-assistant fuel. The regression rate of FT-0070 as a hybrid rocket fuel is considerably higher (approximately 10 times higher) than that as an ignition-assistant fuel. This means the amount of heat transferred in the case of the ignition-assistant fuel was considerably lower than that in the case of the hybrid rocket fuel.

When using PMMA as an ignition-assistant fuel, the amount of reduction of PMMA in 5 s was too low to measure; accordingly, the amounts of fuel ingredients were too small, combustion energy was too low, and the amount of heat transfer to the fuel surface was not adequate for igniting a hybrid rocket. The regression rate of PMMA is approximately 10 times lower than that of FT-0070 in the case of hybrid rocket fuel because PMMA is not liquefied fuel, such as wax[5]. Furthermore, the



(Fuel: FT-0070
Ignition-assistant fuel: FT-0070)

Figure 8 Successful ignition of hybrid rocket.



(Fuel: FT-0070
Ignition-assistant fuel: PMMA)

Figure 9 Unsuccessful ignition of hybrid rocket.

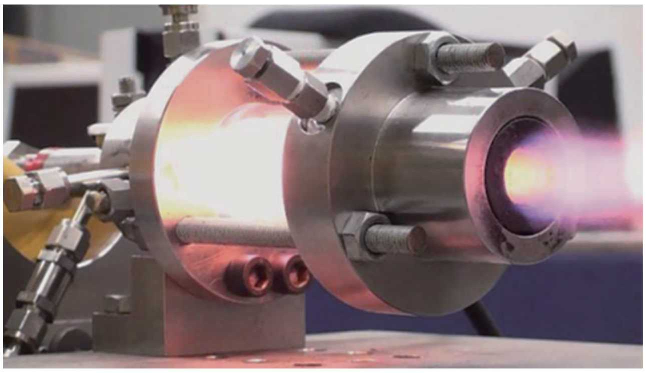


Figure 10 Observed combustion flame using FT-0070 ignition-assistant fuel.

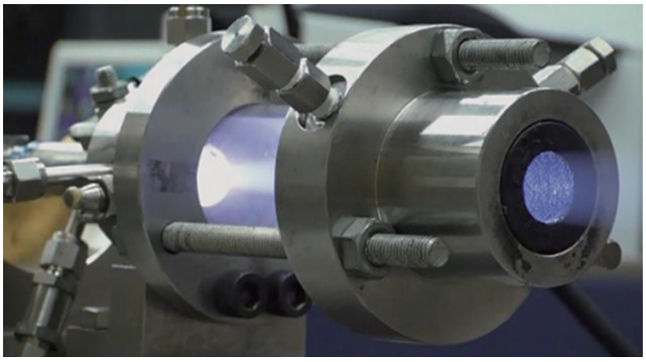


Figure 11 Emission of plasma jet only when using PMMA ignition-assistant fuel.

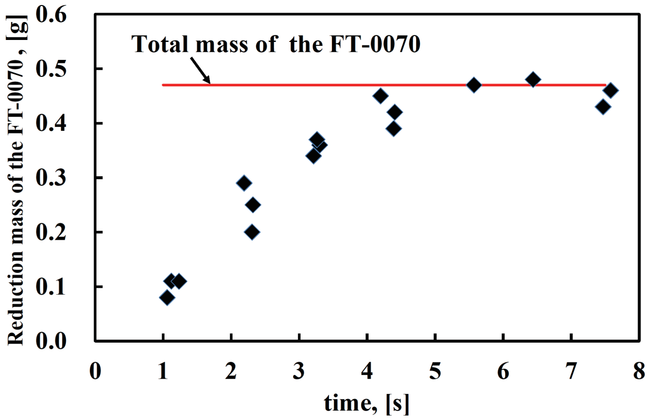


Figure 12 Reduction in mass of FT-0070 ignition-assistant fuel vs. duration of plasma-jet torch operation.

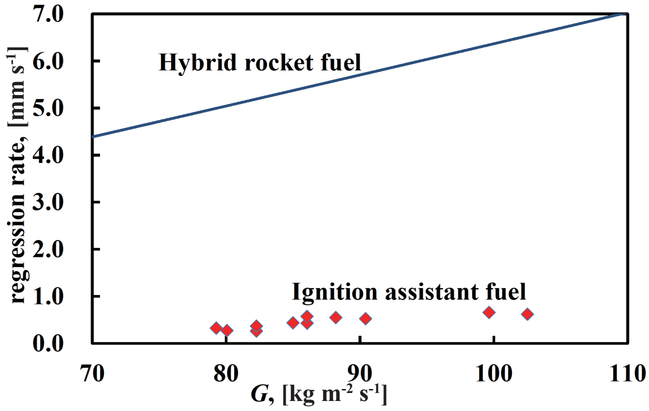


Figure 13 Regression rate of hybrid rocket fuel FT-0070 vs. oxygen mass flux, and regression rate of ignition-assistant fuel FT-0070 vs. argon mass flux.

amount of heat transfer in the case of the ignition-assistant fuel is considerably lower than that in the case of the hybrid rocket fuel, as mentioned. Thus, the amount of reduction of PMMA was negligible. This difference was considered to affect the ignition performance of the small lab-scale hybrid rocket. Wax-based fuels, such as FT-0070, which have a higher regression rate, are suitable for use as ignition-assistant fuels.

Next, a re-ignition test was attempted using the FT-0070 fuel and the FT-0070 ignition-assistant fuels. Before ignition, GOX was supplied into the engine at a low flow rate, and it started to fill the chamber. The plasma torch was then turned on, and the GOX flow rate was increased to approximately 0.01 kg s^{-1} and maintained for almost 3 s. Thereafter, the torch was shut down. Nitrogen gas was then supplied to the engine, and the combustion was extinguished. After 30 s, the chamber was filled with GOX and the plasma torch was turned on again. Then, the GOX flow rate was increased to approximately 0.01 kg s^{-1}. In this experiment, the hybrid rocket was successfully ignited twice.

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Table 2 Test conditions employed in six firing tests.

	No. 1	No. 2	No. 3	No. 4	No. 5	No. 6
Ar flow rate [NL min^{-1}]	80	80	80	80	80	80
Electric current [A]	30	30	30	30	30	30
Electric voltage [V]	36	36	36	36	36	36
GOX flow rate [g s^{-1}]	95	92	91	72	78	92
Initial nozzle throat diameter [mm]	10.9	9.8	10.5	9.0	10.5	10.6

4. Hybrid rocket firing tests

Hybrid rocket firing tests were conducted using the plasma torch. A schematic of the hybrid rocket engine used in the tests is shown in Figure 14. The injector was the shower-head type, and it contained 7 holes, each measuring 3 mm in diameter. The outer diameter of the fuel grain was 80 mm, inner diameter was 40 mm, and length was 140 mm. The fuel contents were 92.5% microcrystalline wax, Hi-Mic-2095 made by Nippon Seiro Co., ltd., and 7.5% stearic acid. A baffle plate (with five holes) was set at the rear of the fuel grain to improve combustion efficiency[6)]. The same plasma torch that was used in the ignition tests was mounted at the front end of the engine. The rated pressure was adjusted to 3 MPa. Table 2 lists the test conditions employed in the six firing tests.

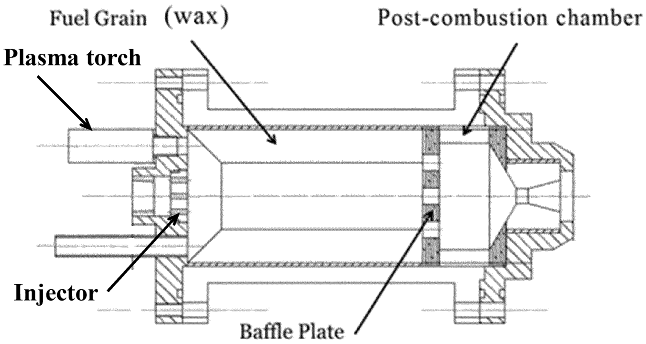

Figure14 Schematic of hybrid rocket engine.

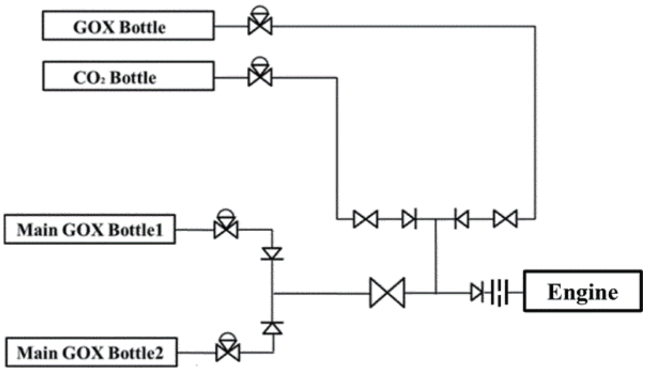

Figure15 Schematic of test setup.

Figure 15 shows a schematic of the test setup. GOX was supplied as the oxidizer through the regulator, valves, and choke orifice into the engine. The GOX flow rate is estimated by measuring the pressure (denoted as P_o) and temperature (denoted as T_o) at the upper side of the choke orifice. The flow coefficient was calibrated before the tests. Another GOX line was integrated to fill the chamber with GOX just before ignition. A CO_2 line was integrated to extinguish combustion after the tests. The pressure (denoted as P_c) at the end of the chamber was measured as well.

The test sequence was as follows. Before ignition, GOX was supplied into the engine at low flow rate through the sub-GOX line, and it started to fill the chamber. The plasma torch was turned on, and after approximately 5 s, the main GOX flow was started and plasma torch was turned off. After 1 s, the sub-GOX flow was shut down. After 2 s, the main GOX flow was shut down, and the CO_2 flow line valve was opened.

Ignition succeeded in all the six firing tests. An example of the pressure curves of the tests (test No. 3) is shown in Figure 16. The ignition point was defined as the inflection point of the initial increase in the P_c curve shown in Figure 16. The time of ignition is denoted t_i. The first point of increase in P_c is defined as first point at which the pressure was higher than 0.15 MPa. We considered that the main GOX flow began to fill up the chamber at that time and denoted this time t_c. t_c is adjusted to 0 s in Figures 16, 17, and 19. The ignition delay-time (t_d) was defined as follows:

$$t_d = t_i - t_c \quad (1)$$

Figure 17 shows an example of the P_c curves of firing test No. 3 and the cold flow test that was conducted just before firing test No. 3. The cold flow test was conducted under the same test conditions as the firing test but with no igniter operation. The inflection point of the P_c curve of the firing test shows the turning point of ignition or no ignition; hence, this point can be considered the ignition point.

Figure 18 shows the P_c at the ignition points against t_d. All t_d values are shorter than 0.25 s. This is sufficient to satisfy general design requirements. This result implies that ignition delay is not a problem from the viewpoint of practical use. All P_c values at the ignition point were higher than 0.3 MPa, meaning the nozzle was choked at

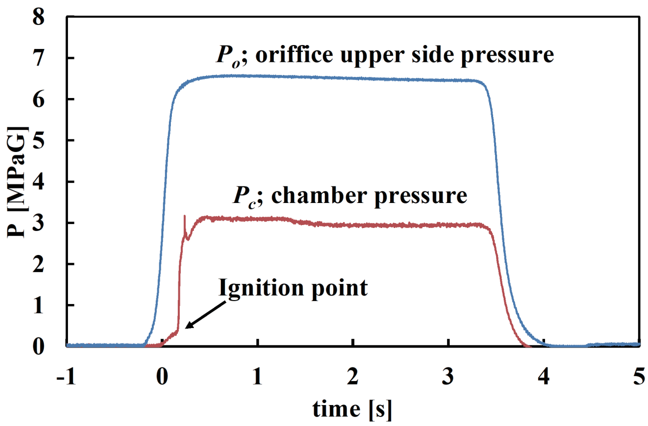

Figure16 Example of pressure curves (test No. 3).

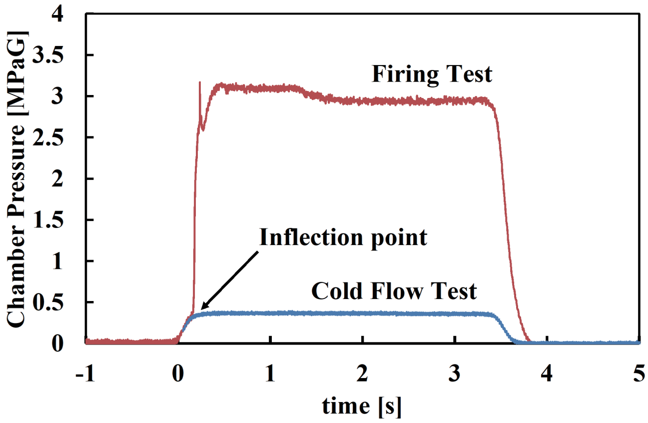

Figure17 Chamber pressure (P_c) curves obtained in firing test No. 3 and cold flow test.

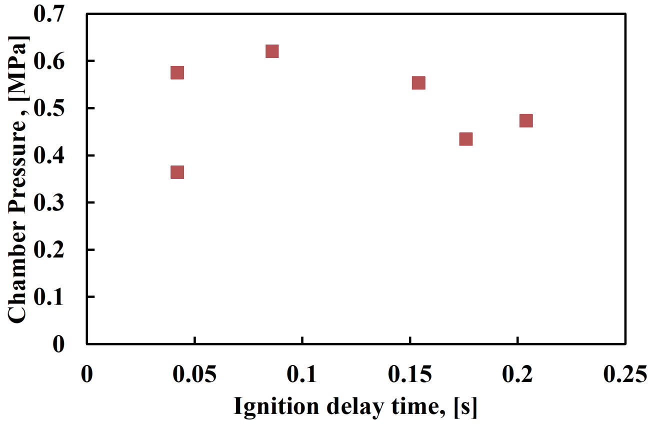

Figure18 Chamber pressure (P_c) at ignition points vs. ignition delay time (t_d).

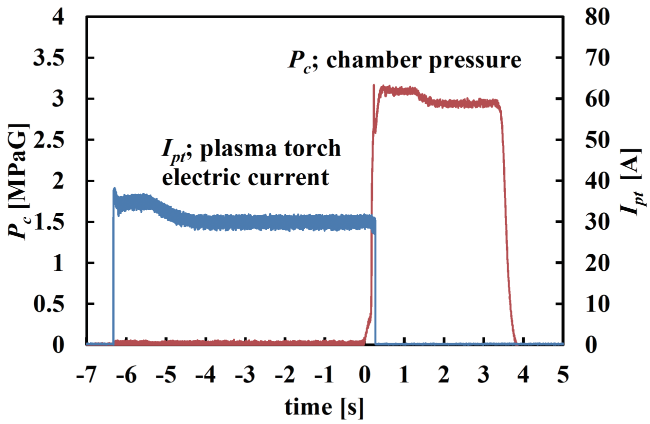

Figure19 Example data (test No. 3) of electric current of plasma torch and P_c.

ignition. Therefore, the ignition condition in the chamber was not affected by environmental pressure conditions, and the ignition was considered to feasible even in the vacuum conditions prevalent in space.

As stated already, the timing of plasma torch turn-on is approximately 5 s before the timing of the main GOX valve open signal. Figure 19 shows example data (test No. 3) of the electric current of the plasma torch and P_c. After ignition, the plasma torch was turned off manually.

The average electrical voltage of the plasma torch (E_{pt}) was 36 V, as measured in the pretests. The electric current of the plasma torch (I_{pt}) was measured in each test. The electrical power of the plasma torch (W_{pt}) was calculated using the following equation:

$$W_{pt} = E_{pt} I_{pt} \tag{2}$$

The electrical energy of the plasma torch (Q_{pt}) was calculated using the following equation:

$$Q_{pt} = W_{pt} t_{pt} \tag{3}$$

where t_{pt} denotes plasma torch operation duration.

The typical value of I_{pt} was 30 A and that of t_{pt} was 6 s. Therefore, the typical value of Q_{pt} was 6.48 kJ. The ignition-assistant fuel heat value (H_{ia}) was estimated to be 40 MJ/kg, which is on par with that of petrochemical products[7]. The total mass of the ingnition-assistant fuel (m_{ia}) was 4.7 g. Therefore, the energy of the ignition-assistant fuel (Q_{ia}) was calculated using the following equation:

$$Q_{ia} = H_{ia} m_{ia} \tag{4}$$

Q_{ia} is 18.77 kJ. The total ignition energy of the plasma torch with the ignition-assistant fuel (Q_T) is

$$Q_T = Q_{pt} + Q_{ia} = 25.3 \text{ kJ} \tag{5}$$

The typical gunpowder energy is 4 MJ kg^{-1} [7]. Therefore, the equivalent gunpowder mass (M_T) of Q_T is

$$M_T = \frac{Q_T}{4} \times 10^3 = 6.0 \times 10^{-3} \text{ kg} \tag{6}$$

Figure 20 shows the empirical relationship between the initial chamber free volume and the igniter powder mass (m) for the solid rocket employed herein:

$$m = 0.27 V_F^{0.7} \tag{7}$$

where V_F is the initial chamber free volume[8], and M_T vs, the initial chamber free volume point in these tests, is plotted. M_T is greater than m, but its order is same as that of the m of the solid rocket. This implies that the order of the energy level for hybrid rocket ignition with the plasma torch in these tests was the same as that for the solid rocket. The combustion mechanism of a hybrid rocket is boundary layer combustion. Therefore, ignition is initiated on the fuel surface under oxidizer flow, and energy is required to vaporize fuel and increase the temperature of the vaporized fuel gas to the flush point. This is quite

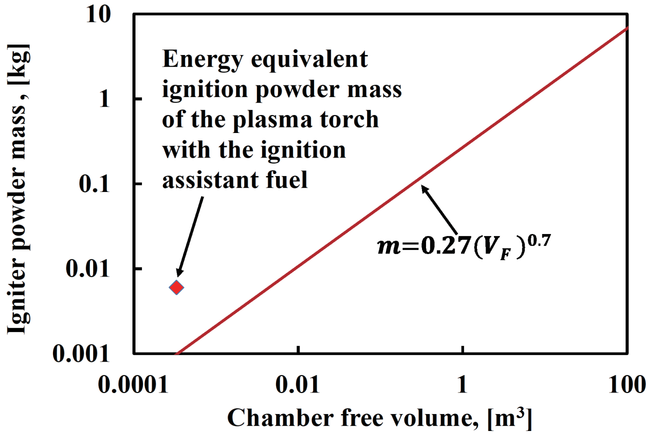

Figure20 Relationship between chamber free volume and ignition powder mass.

different from that of a solid rocket; hence, equation (7) is considered meaningless in the case of a hybrid rocket. However, the order of M_T is same as that of m for a rocket with the same V_F.

5. Conclusions

A plasma torch that has been studied and developed as an igniter for reciprocating engines and as a flame holder for a supersonic combustion ramjet engine or other combustion promotion devices was used as a hybrid rocket igniter in this study. Ignition tests were conducted using a lab-scale hybrid rocket engine with a no-cooling plasma torch igniter, and successful ignition of a hybrid rocket was achieved. Then, ignition succeeded twice in a sequence test. Subsequently, hybrid rocket firing tests were conducted using this plasma-torch. The results showed that the ignition delay time was less than 0.25 s, which is suitable for practical use. The chamber pressure at ignition was higher than 0.3 MPa due to oxidizer flow. It exceeded the nozzle choke critical pressure under atmospheric pressure. Therefore, we believe that ignition can be achieved even under the vacuum conditions in space.

Thus, a plasma torch can be used as a hybrid rocket igniter. The igniter does not use explosives or dangerous substances, which means it retains the safety advantage of a hybrid rocket and can reduce the administrative costs of launching rockets. Moreover, restart-capability even under the vacuum conditions in space makes coasting possible and improves the accuracy with which a satellite can be placed into an orbit.

The ignition-assistant wax fuel mounted at the plasma torch exit made hybrid rocket ignition possible with a small amount of electrical energy. This can be ascribed to combustion of the ignition-assistant fuel vapor or melted droplets produced by the plasma torch. The energy-equivalent gunpowder mass of the plasma-torch igniter with the ignition-assistant fuel was higher than the powder mass of a general solid rocket igniter; however, the order of the two masses was the same. This result shows that the power production unit used in this ignition system is light, and the plasma torch system is suitable for space flight applications.

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