CURRENT STATUS OF ROCKET DEVELOPMENTS IN UNIVERSITIES - DEVELOPMENT OF A SMALL HYBRID ROCKET WITH A SWIRLING OXIDIZER FLOW TYPE ENGINE

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Abstract

To develop an experimental small hybrid rocket with a swirling gaseous oxygen flow type engine, we made a flight model engine. Burning tests of the engine showed that a maximum thrust of 692 N and a specific impulse of 263 s (at sea level) were achieved. We designed a small hybrid rocket with this engine. The rocket measured 1.8 m in length and 15.4 kg in mass. To confirm the flight stability of the rocket, wind tunnel tests using a 1/2-scale model of the rocket and simulations of the flight attitude and trajectory were carried out. A flight test was conducted at Taiki-cho, Hokkaido, Japan on March 2001. The rocket reached an altitude of about 600 m, thus recording the first successful flight of a hybrid rocket in Japan. For the next stage, future issues to develop larger hybrid rockets using a swirling liquid oxygen flow type engine are discussed, and preliminary burning tests of the engine have been carried out.

Nomenclature

a	acceleration	[m/s^2]	M_Z	pitching moment	[N・m]
b	maximum span of fins	[m]	P_a	atmospheric pressure	[MPa]
C_D	drag coefficient		P_c	chamber pressure	[MPa]
$C_{L\alpha}$	inclination of lift coefficient		$\dot{r}$	regression rate	[mm/s]
$C_{M\alpha}$	inclination of pitch moment coefficient		S	flight distance from launcher	[m]
d_i	injection hole diameter	[mm]	S_g	geometric swirl number	
D	drag	[N]	V	velocity of rocket	[m/s]
	body diameter occurring with b	[m]	V_w	head wind velocity	[m/s]
F	thrust	[N]	α	angle of attack	[deg]
I_{SP}	specific impulse	[s]	β	angle between the axis of rocket and the axis of thrust	[deg]
G_o	oxidizer mass flux	[kg/(m^2・s)]			
h	altitude	[m]	δ	distance from the center of gravity of rocket to the axis of thrust	[mm]
L	grain length	[mm]			
	lift	[N]	γ	angle of direction of flight	[deg]
L_ℓ	launcher length	[m]	γ_d	$\gamma - \gamma_\ell$	[deg]
Lw	lift of fins alone	[N]	γ_ℓ	launch elevation angle	[deg]
$\dot{m}_o$	oxidizer mass flow rate	[g/s]	θ	angle between the axis of rocket and the horizontal reference	[deg]
$\dot{m}_f$	fuel mass flow rate	[g/s]			
mg	weight of the rocket	[N]	φ	equivalence ratio	

1. Introduction

The concept of hybrid rocket engines was proposed about half a century ago.[1] These engines have many advantages over solid and liquid rocket engines: safety with respect to explosion and storability, less cost of manufacturing and operation, clean burning for environmental impact, and re-ignitable capability for upper-stage propulsion systems. However, the hybrid rocket engine has not yet been used in practical rocket systems. This is mainly due to the low fuel regression rate of hybrid combustion, which makes it difficult to achieve a theoretical maximum specific impulse obtained in the fuel rich side.

The regression rate of hybrid combustion can be determined by convective heat transfer from the flame to the fuel surface.[2] To increase the fuel regression rate, therefore, the thickness of the boundary layer over the fuel surface must be reduced. It is expected that a swirling flow field in the combustion chamber may reduce the boundary layer thickness due to the increase in velocity by the additional tangential velocity component near the fuel surface. The centrifugal force of the swirling motion also has favorable effects on the reduction due to the increase in pressure. In addition, the swirling motion may increase the residence time of the combustion gas in the chamber, causing the combustion efficiency of the hybrid engine to increase. Based on this, we have proposed a new unique technique, a swirling oxidizer injection flow, to improve the poor combustion property of hybrid combustion.[3]

In our previous study of gaseous O_2/PMMA (transparent acrylic) small hybrid rocket engines, it was found that applying swirl to an oxidizer flow increased the fuel regression rates of the engine, and varying the swirl intensity of the oxidizer flow could control the equivalence ratio of the engine even at a constant oxidizer mass flow rate.[3,4]

Concurrently, a joint research team of universities in Japan was organized in 1998 to develop small-scale reusable launch systems based on hybrid rockets.[5] Our laboratory, which had been conducting a project to launch an unguided small hybrid rocket, also joined the research team. The objective of our launch was to demonstrate a swirling oxidizer engine with a performance equivalent to that of solid propulsion in a realistic environment.

This paper reports the progress of how to develop this hybrid rocket, that is, 1) the performance of the flight model of a hybrid rocket engine with a swirling gaseous oxygen flow, 2) the design concept and outline of the unguided small hybrid rocket fabricated and its flight stability, and 3) the flight demonstration of the rocket. Furthermore, we comment the extraction of issues for scaling up the swirling oxidizer hybrid rocket, and show the outline of the current experiment using swirling liquid oxygen.

2. Combustion Experiment of Flight Model Engine

The flight range and altitude of the hybrid rocket in this flight program were limited due to safety reasons of the launch site. In addition to this limitation, considering the oxygen tank size, weight of the engine parts, and the burning duration of propellant, the initial mass of the hybrid rocket for the flight demonstration was preliminarily estimated to be 15 kg. To achieve a sufficient initial acceleration of the rocket at the launch, a thrust of over 600 N was required.

Using the same configuration of the hybrid engine and the empirical equation between G_o and $\dot{r}$ for ground tests,[3,4,6] where G_o is defined as the oxygen mass flow rate per fuel-grain port section area, the values of $\dot{m}_o$ and $\dot{m}_f$ necessary for obtaining a thrust of 600 N were estimated to be $\dot{m}_o$=138 g/s and $\dot{m}_f$=101 g/s at S_g=19.4 and L=500 mm. The

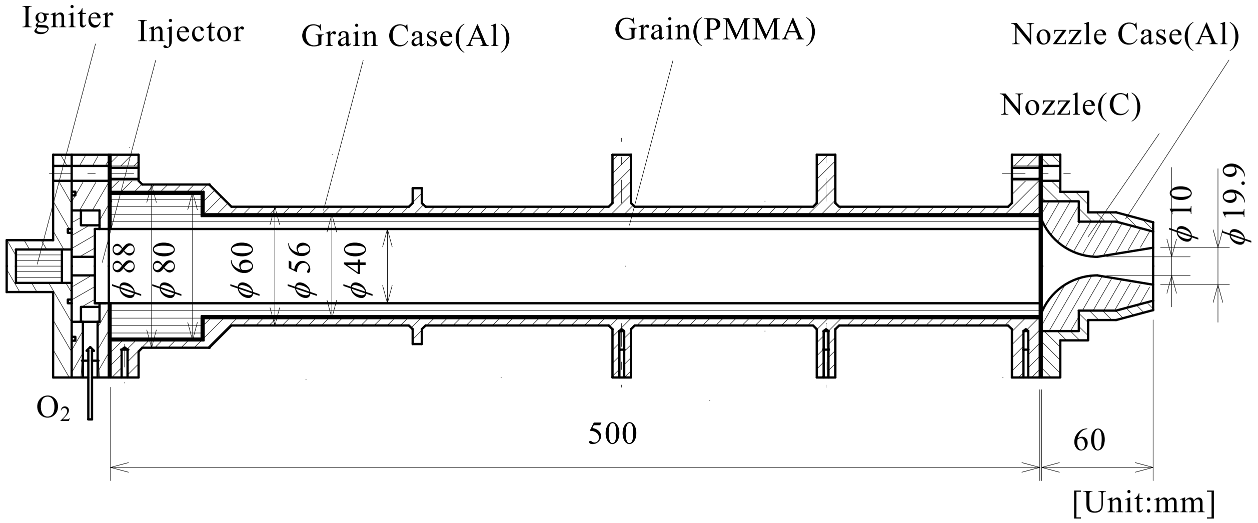

Fig.1 Flight model of hybrid rocket engine

corresponding P_c and I_{SP} were 5.01 MPa and 256 s, respectively. Based on the estimation, a hybrid rocket engine for the launch was designed and fabricated.

Figure 1 shows a schematic of the hybrid rocket engine used for the flight test. The flight model engine had essentially the same structure as the previous engine.[6] The fuel grain made of PMMA was 500 mm in length and had a single port with an inner diameter of 40 mm. The first 50 mm of the grain had an outer diameter of 80 mm that was larger than the rest of the section, which had a 56-mm outer diameter, to prevent the grain case from flame exposure, because the swirling oxygen flow remarkably increased the fuel regression rates at the leading edge. A gaseous oxygen tank for medical purposes (charged O_2 pressure: 20 MPa, charged O_2 mass: 0.56 kg) was used for weight reduction. The gaseous oxygen was supplied tangentially through eight inlet slits of the injector from the head of the engine in the blowdown mode. The geometrical swirl number S_g of the oxygen injector was 19.4. The nozzle was made of carbon and had a throat diameter of 10 mm without cooling. The nozzle exit diameter was decided so that the optimum expansion of the nozzle was obtained when P_c=2 MPa, P_a=0.098 MPa and $\dot{m}_o$=97 g/s. The burning duration was varied from 7 to 10 s.

Figure 2 shows typical traces of $\dot{m}_o$, P_c and F for the engine during combustion. Rapid ignition and sure shutdown were confirmed. A maximum F of 692 N and a maximum I_{SP} of 263 s were achieved just after ignition, when the initial $\dot{m}_o$ was 164 g/s and the initial G_o was 130 kg/(m^2·s). Although F and I_{SP} gradually decreased as the oxygen tank pressure decreased, combustion oscillation did not occur. In spite of the comparatively low theoretical performance of hybrid rocket engines

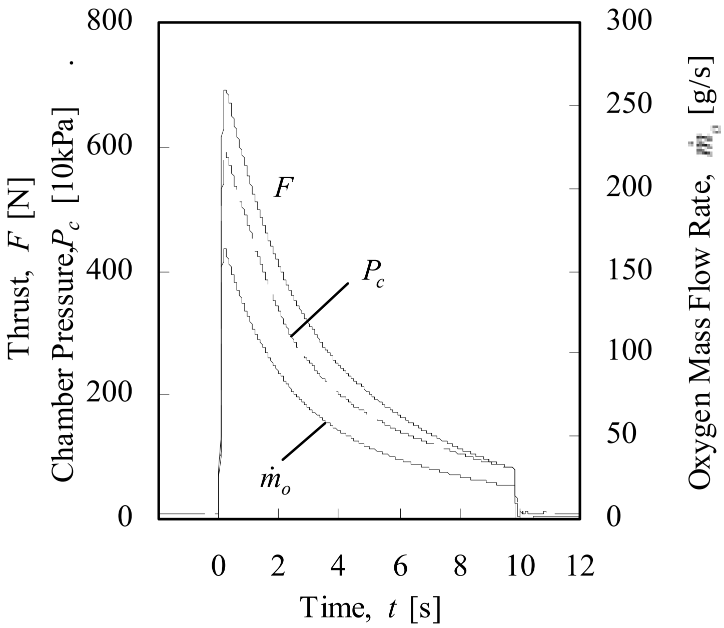

Fig.2 Typical traces of a full-scale engine performance

with O_2 and PMMA, the maximum value of I_{SP} obtained here was the same level as those of recent solid rockets with composite propellants. After burning, the fuel grain, except for the leading edge, almost uniformly regressed over the longitudinal axis, and no erosion at the throat of the carbon nozzle occurred. These results suggest that our hybrid rocket engine with the swirling oxidizer flow had reached a sufficient level for practical use. The thrust of this engine could be enough to launch a rocket with an initial mass of 15 kg with sufficient initial acceleration.

3. Design of Small Hybrid Rocket

3.1 Design Concept

We designed and fabricated a small hybrid rocket, with the rocket engine shown in Fig. 1, which follows the criteria shown here:

(1) Low cost:
using mostly commercial products and processing developed mainly by ourselves at TMIT (Tokyo Metropolitan Institute of Technology).

(2) Safety:
using non-explosive and non-pyrophoric propellants.

(3) Environmental cleanliness:
using gaseous O_2/PMMA propellants, which do not include chlorine and nitrogen.

(4) Reusability:
recovering the rocket by a parachute and replacing the gaseous O_2/PMMA propellants.

3.2 Airframe

Figure 3 shows a schematic of the hybrid rocket designed according to the above concepts. The diameter of the rocket was decided from the diameter of the gaseous oxygen tank. The rocket was a simple structure without a flight control mechanism so that we could fabricate it cheaply in our institute. In order to reuse the rocket, the grain could be easily exchanged. The airframe of the rocket was made of aluminum alloy to reduce weight. The

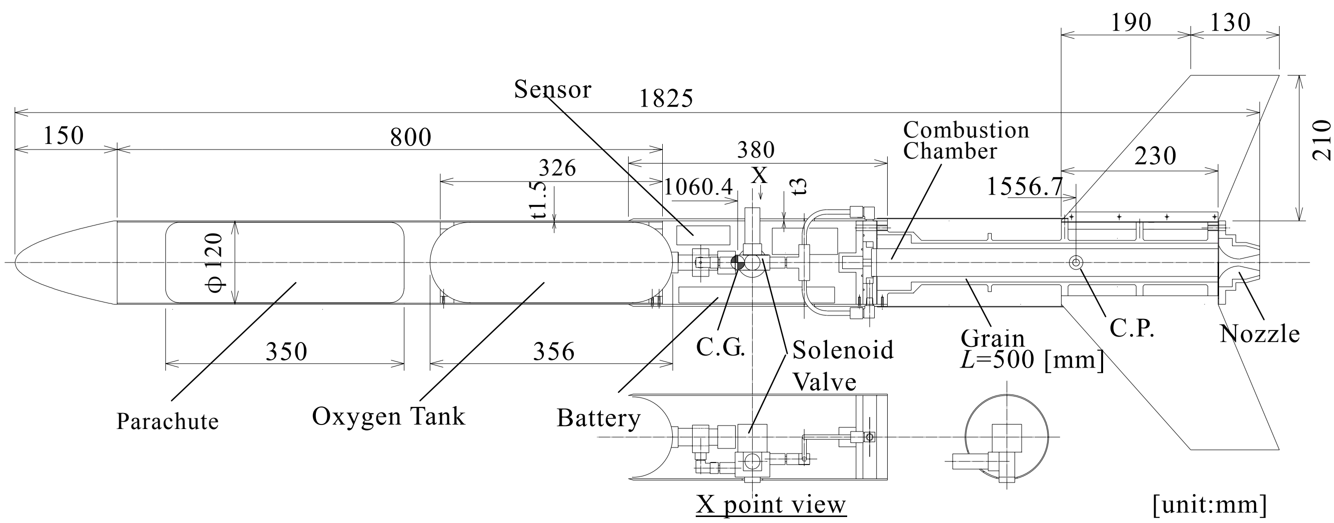

Fig.3 Small hybrid rocket developed by TMIT.

strength of the rocket body was evaluated according to the method reported in previous studies.[7,8] Calculation results showed that the rocket was sufficiently safe with respect to the bending strength and divergence speed of the rocket body.[8] The total mass of the rocket was 15.4 kg and the center of gravity of the rocket was about 1060 mm from the top. Since this rocket was stabilized by tail fins only, the aerodynamic characteristics of the rocket played a crucial role in its flight stability. To determine the lift coefficient of the rocket, we considered the components of the body in the absence of fins, the fins alone, and the interference force between the fins and body. The slender body theory[9] was used to predict the lift coefficient of the body without fins. The total combination lift distributed between the fins and the body was estimated by the relationship $(1+D/b)^2Lw$.[9,10] In addition, fin-body interference was assumed to have little effect on the center of pressure for a fin.[9,10] The drag coefficient of the rocket at subsonic speed was estimated from an early report,[11] in which the nose and body friction drag, base drag, fin surface drag, and fin interference drag were considered. The following aerodynamic results were obtained: $C_{L\alpha}$=28.07 rad^{-1}, $C_{M\alpha}$=−7.66 rad^{-1}, C_D=0.51. These aerodynamic coefficients were similar to those of Tokyo University's rockets,[10,12] which suggested that the flight of our rocket would be stable. The rocket was equipped with a multi-sensor recovery deployment system to record the acceleration of the rocket and barometric pressures during flight, and a recovery parachute, which was ejected by a trigger from a multi-sensor or mechanical timer in the rocket.

Before the final design was decided, wind tunnel tests of a 1/2-scale model of the airframe were conducted to confirm aerodynamic stability using the low-speed wind tunnel at TMIT. The drag, lift and moment around the center of rotation were measured at a wind velocity of 35 m/s. It was found that the attitude of the rocket was stable between α=±15 deg, and the aerodynamic characteristics were measured to be similar to those of the estimation.[13] In addition, a parachute release test, an airframe rotation test by swirling oxygen jets without combustion, and various other tests were carried out to verify the safety of the rocket in flight.

4. Safety of Flight

4.1 Flight Attitude

Because we fabricated the rocket, slight thrust misalignments could not be avoided. The flight attitude of the rocket was analyzed to examine the effects of δ and β by using the flight model shown in Fig. 4 and the analytical theory of rocket flight,[14] with consideration of the dispersion during burning. In our rocket, a different distance between the center of gravity and the axis of the rocket may arise from the uneven allocation of asymmetric parts, such as the battery and solenoid valve, and the deviation angle of thrust with the axis of the rocket may arise from the nozzle, which was drilled on a slant or installed by

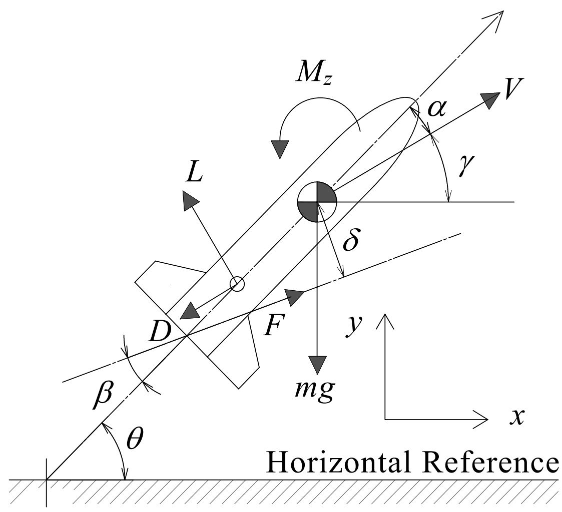

Fig.4 Model for calculating flight attitude

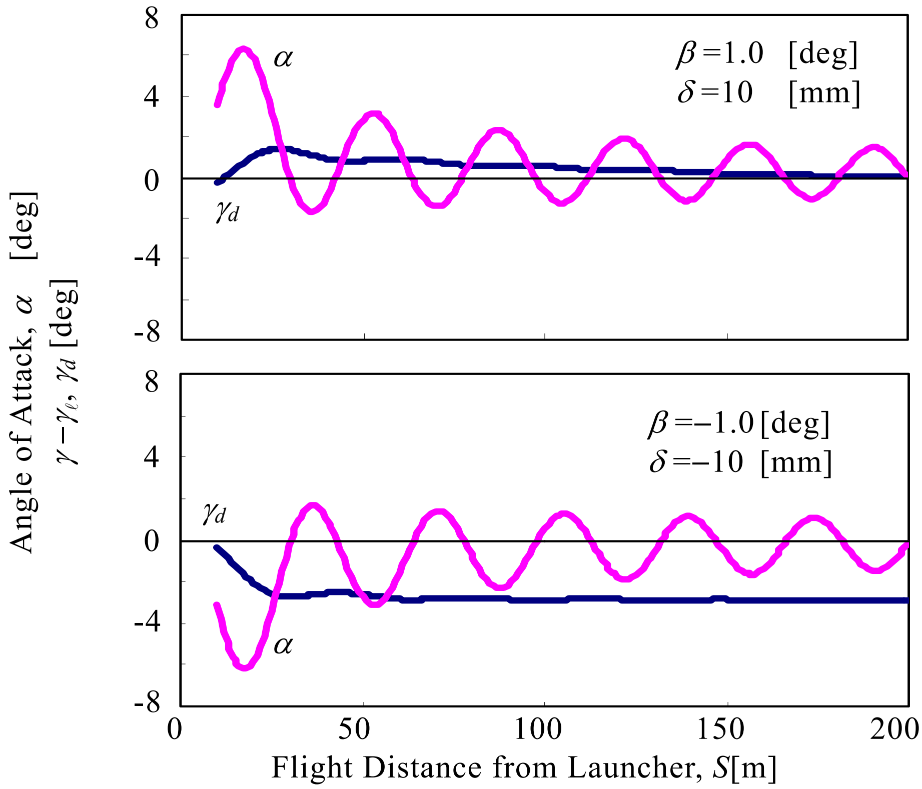

Fig.5 Estimated flight attitude at γ_ℓ =86 deg, L_ℓ=4 m and F=700 N

shifting of the rocket axis. The thrust were estimated even for a large case, $|\delta|$=5 mm and $|\beta|$=0.5 deg. To evaluate the flight attitude under more severe conditions, the maximum thrust misalignments of our rocket were assumed to be $\delta=\pm10$ mm and $\beta=\pm1.0$ deg. Figure 5 shows a typical example of the flight attitudes with these misalignments at γ_ℓ =86 deg, L_ℓ=4 m and F=700 N. Just after the launch, the absolute value of α has a maximum of about 6 deg, and it decreases gradually with S. Since the flight stability of the rocket was verified within $|\alpha|$=15 deg by the wind tunnel tests, the rocket was confirmed to remain stable during flight.

4.2 Flight Trajectory

The landform around the launch site restricted the radius of the flight of the rocket to within 1 km from the launch site. Flight trajectories of the rocket without misalignments were simulated at various launch angles and wind velocities using the step-by–step integration method, assuming two-dimensional motion for cases in which the parachute did or did not open. The calculation conditions were as follows: L_ℓ=4 m, V_w<10 m/s, the aerodynamic characteristics estimated in Section 3.2, and the thrust time trace shown in Fig. 2.

When V_w exceeds 8 m/s and the parachute will not open, the rocket would land outside of the limited area independent of γ_ℓ. These trajectories show that when V_w=5 m/s, the optimum γ_ℓ giving the most distant landing point within the restricted area for the safety of the staff at the launch site is 86 deg, at which the maximum velocity of the rocket is 89.7 m/s and the maximum altitude is 681 m.

5. Flight Test

Before the launch, a ground test was carried out at the launch site at Taiki-cho, Hokkaido during cold weather in order to check the burning of the engine, the ejection of the parachute and the sequence of the launch. The results showed that there was no essential problem with the launch. The flight test was conducted on March 9, 2001 at conditions of γ_ℓ = 85.5 deg and burning duration of 7 s. Figure 6 shows views of the small hybrid rocket after the launch. The flight was stable and no smoke from the engine was observed. This indicates that the swirling oxidizer flow did not disturb the flight stability and the exhaust of the engine was environmentally clean.

After reaching an apogee altitude, the parachute was opened. However, the opening shock broke the connections between the parachute and the airframe, and the rocket fell into a snowfield. The falling point was 620 m from the launch site. Figure 7 shows the estimated flight data based on the falling locations of some parts of the rocket and video images of the flight. This estimation indicates that the maximum altitude was about 600 m, which is in good agreement with the value predicted in Section 5.2. Although the rocket failed to land softly and could not be reused, we recorded the first successful flight of a hybrid rocket in Japan.

Fig.6 Views of the small hybrid rocket after launch

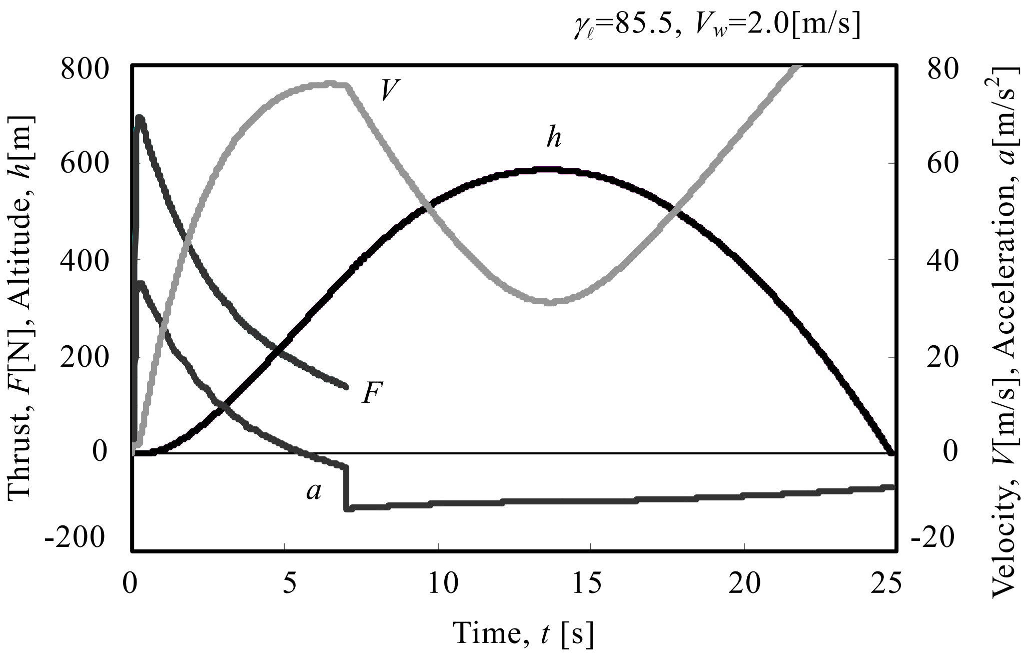

Fig.7 Calculated flight data

6. Future Issues and Swirling Liquid Oxygen Experiment

With regard to scaling up hybrid rockets in the future, unfortunately, using gaseous oxygen for hybrid rocket engines has critical demerits: it is difficult to increase the oxygen mass flow rate, and the burning duration is short due to the low density of gaseous oxygen stored in a tank. It is essential to scale up hybrid rockets to use liquid oxygen, which has a higher density. However, hybrid combustion of solid fuel in swirling liquid oxygen could result in uncertain phenomena, listed below:

- ＊ whether solid fuel can ignite and burn,
- ＊ whether applying swirl to the liquid oxygen flow has the same effects on the fuel regression rate as the gaseous oxygen flow,
- ＊ whether combustion oscillation occurs, and so on.

To clarify issues in the development of hybrid rocket engines with swirling liquid oxygen, a liquid oxygen feed system on a laboratory scale was constructed and preliminary burning tests of the hybrid rocket engine were conducted using various injectors with different swirl intensities of liquid oxygen flow. To prevent that the supplied liquid oxygen was vaporized by heat of the feed line before reaching the injector and the vaporized oxygen choked the line, the feed line and injector were pre-cooled sufficiently just before injection of liquid oxygen to the engine. Figure 8 shows the hybrid rocket engine used in this study, having the similar configuration and size as the one with gaseous oxygen [3,4]. The spacer, made of a carbon tube, was adopted to prevent remarkable regression of the fuel at the leading edge because of direct impingement of liquid oxygen and to promote vaporization of liquid oxygen. Five swirling flow type injectors for liquid oxygen with the similar angular momentum to that for gaseous oxygen, which differed in diameter and number of the injection holes, were used at injection pressures from 1.5 – 5 MPa. Details of the experiment apparatus were described elsewhere.[15]

Figure 9 shows typical traces of combustion chamber pressure, liquid oxygen mass flow rate, and thrust of the hybrid rocket using the injector with 0.6-mm injection holes. It was found that ignition occurred rapidly and definitely, and stable combustion of PMMA with swirling liquid oxygen was verified. The combustion behavior, observed by using a transparent combustion chamber, was very similar to that using gaseous oxygen. However, the average *Isp* was 156 s, which was rather lower than that of the engine with gaseous

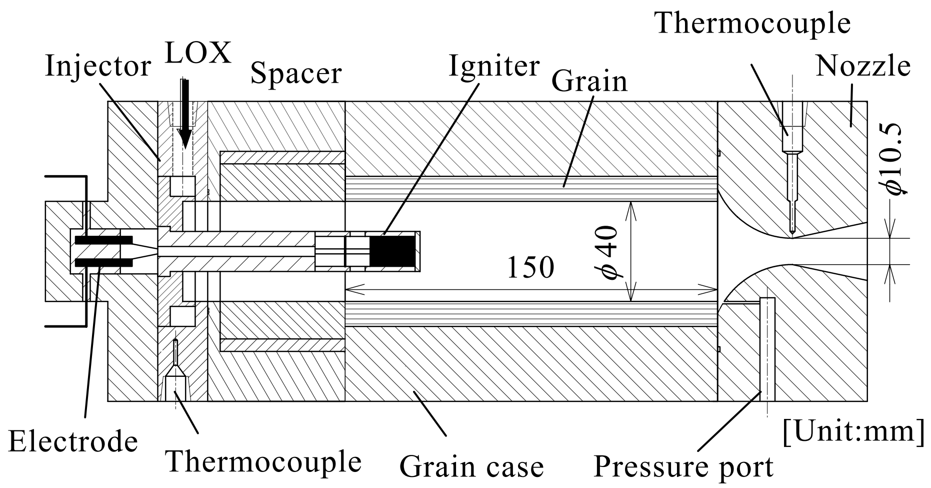

Fig.8 Schematic of hybrid rocket engine

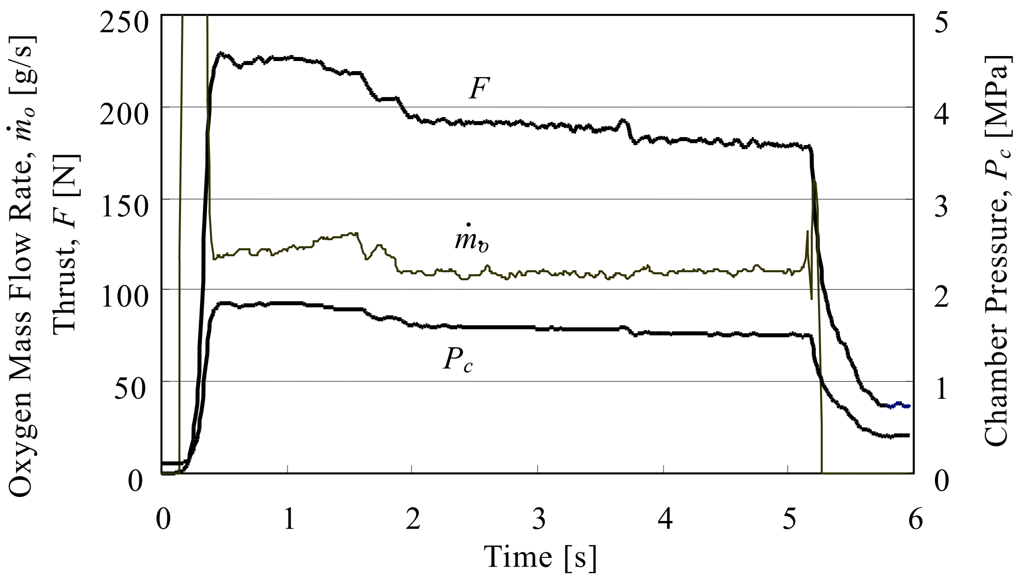

Fig.9. Time traces of engine parameters
(*d* = 0.6x8, *L* = 150 [mm])

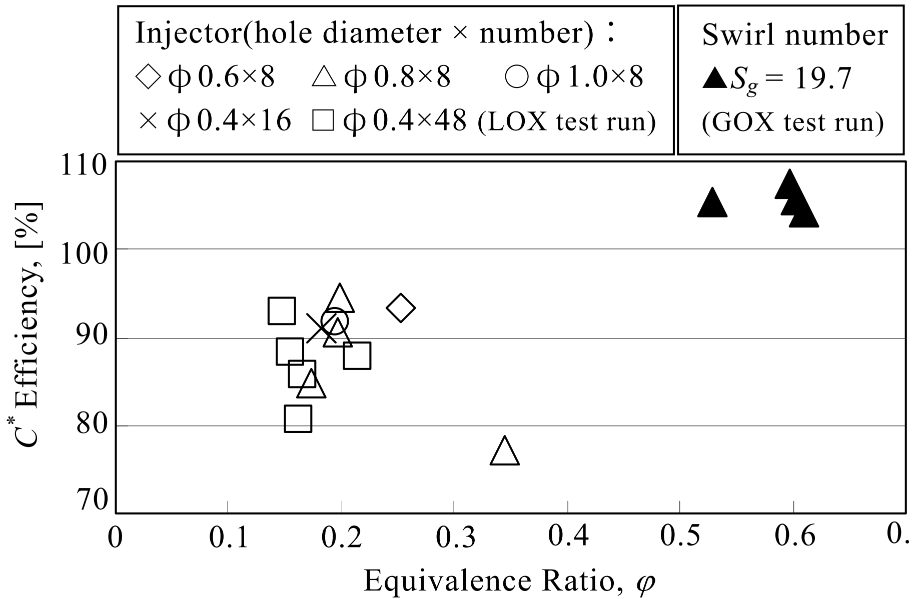

Fig.10 C^* Efficiency versus equivalence ratio

oxygen, and φ of O_2/PMMA was 0.25, significantly smaller than 1.0. Figure 10 shows the C^* efficiencies of burning runs of the hybrid rocket engines with a 150-mm combustion chamber length using swirling liquid oxygen and gaseous oxygen as a function of φ. In this study, the C^* efficiency in hybrid rocket engines was defined as the ratio between the experimental C^* based on P_c and $\dot{m}\,(=\dot{m}_o+\dot{m}_f)$, the propellant mass flow rates, of burning runs and the corresponding theoretical C^* based on φ and $\dot{m}$.[4] The C^* efficiencies with swirling gaseous oxygen were nearly 100%, but those with swirling liquid oxygen were considerably lower. (For the burning runs with gaseous oxygen, the C^* efficiencies exceeded 100%. This may be due to additional chamber pressures by the strong centrifugal swirling flow.) In addition, φ with liquid oxygen were much smaller than those with gaseous oxygen. These results indicate that the fuel regression rates of PMMA with swirling liquid oxygen were considerably low and the combustion reactions proceeded insufficiently. This was attributed to that the swirling motion of the liquid oxygen flow remarkably decreased during vaporization.

In addition, when using the injectors with larger injection holes and low injection pressures, a chugging-mode combustion oscillation frequently appeared at the end of burning. This fact implies that the injected liquid oxygen required a certain amount of time to evaporate and burn, and the time was coupled with delays in the oxygen injection system. These processes might result in poor combustion of PMMA with liquid oxygen. Therefore, we decided to vaporize liquid oxygen before injecting into the combustion chamber, while cooling a nozzle of the hybrid rocket.[16,17] The most important current issues for the swirling oxidizer flow type hybrid rocket engine with liquid oxygen is to design a liquid oxygen evaporation nozzle and to clarify the evaporation processes of liquid oxygen through the nozzle.

7. Concluding Remarks

The flight model of a swirling gaseous oxygen hybrid rocket engine using PMMA fuel developed in this study achieved maximum values of F of 692 N and I_{SP} of 263 s. The I_{SP} and chamber pressure of the engine were high enough to be on the same order as recent conventional rockets. This suggests that using this type of engine will potentially be applied to practical rockets with compactness and high performance.

A small hybrid rocket with the swirling O_2 /PMMA engine was designed and fabricated. The rocket measured 1.8 m in length and 15.4 kg in mass. Aerodynamic characteristics were theoretically estimated and evaluated by wind tunnel tests using a 1/2-scale rocket model. Flight simulations of the rocket with realistic misalignments and estimated aerodynamic characteristics showed that the rocket flies stably and falls within a restricted area. On March 9, 2001 the rocket was launched in Hokkaido, Japan, and reached an altitude of about 600 m. This was the first launch of a hybrid rocket in Japan. This successful launch demonstrated the advantages of the hybrid rocket engine with a swirling oxidizer flow: that is, high safety, low cost, easy operation and clean exhaust. This means that a practical propulsion performance of the hybrid rocket was achieved.

Using liquid oxygen is essential to scale up hybrid rockets. A swirling liquid oxygen engine system on a lab-scale was constructed, and burning tests were conducted. They showed that direct injecting of swirling liquid oxygen into the combustion chamber lowered the performance of the engine compared to that with gaseous oxygen due to poor combustion of PMMA with liquid oxygen. To increase the performance, we proposed the re-generative cooling nozzle method using liquid oxygen, that is, liquid oxygen is vaporized before injecting into the combustion chamber through the nozzle of the engine.

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