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Development of a University-Based Simplified H_2O_2/PE Hybrid Sounding Rocket at KAIST

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Abstract

This paper reports development process of a university-based sounding rocket using simplified hybrid rocket propulsion system for low-altitude flight application. A hybrid propulsion system was tried to be designed with as few components as possible for more economical, simpler and safer propulsion system, which is essential for the small scale sounding rocket operation as a CanSat carrier. Using blow-down feeding system and catalytic ignition as combustion starter, 250 N class hybrid rocket system was composed of three components: a composite tank, valves, and a thruster. With a composite tank filled with both hydrogen peroxide(H_2O_2) as an oxidizer and nitrogen gas(N_2) as a pressurant, the feeding pressure was operated in blow-down mode during thruster operation. The MnO_2/Al_2O_3 catalyst was fabricated for propellant decomposition, and ground test of propulsion system showed the almost theoretical temperature of decomposed H_2O_2 at the catalyst reactor, indicating sufficient catalyst efficiency for propellant decomposition. Auto-ignition of the high density polyethylene(HDPE) fuel grain successfully occurred by the decomposed H_2O_2 product without additional installation of any ignition devices. Performance test result was well matched with numerical internal ballistics conducted prior to the experimental propulsion system ground test. A sounding rocket using the developed hybrid rocket was designed, fabricated, flight simulated and launch tested. Six degree-of-freedom trajectory estimation code was developed and the comparison result between expected and experimental trajectory validated the accuracy of the developed trajectory estimation code. The fabricated sounding rocket was successfully launched showing the effectiveness of the simplified hybrid rocket propulsion system.

Key words: Hybrid sounding rocket, Hydrogen peroxide, Polyethylene, Flight test

1. Introduction

Sounding rockets have been used since the late 1950s for scientific experiments; currently, missions involving sounding rockets are becoming increasingly diverse. The demand for such diverse missions has arisen due to a mandatory inspection of the atmosphere by the UN and basic science experiments in micro gravity, supersonic combustion test, re-entry and aerodynamic test[1-8]. In addition, university-based small-scale sounding rockets have been used for educational purposes, e.g., rocket system studies and CanSat launches[9-18]. A general CanSat launcher can carry several

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can-sized satellites at once to an altitude of approximately 1 km and each CanSat performs its particular mission while gliding down from the relevant altitude after launch. For a low-altitude operation of a university-based sounding rocket as a CanSat carrier, more economical, simpler and safer rocket propulsion system configuration is essential.

Chemical rocket propulsions such as liquid, solid, and hybrid propellant rockets are commonly used for sounding rocket propulsion. Liquid propellant rocket system using two different liquid propellants as an oxidizer and a fuel is the common form of the rocket propulsion system for high thrust level and specific impulse performance. Liquid rocket system has relatively high system complexity, due to the requirement for pressurant tank or turbomachinery with a gas generator, flow controller, ignition system and propellant storage tanks[19]. On the contrary, a solid propellant rocket has system simplicity. In the solid rocket system, solid form propellant of the pre-mixed fuel and oxidizer is stored directly in the combustion chamber and igniter device triggers a combustion reaction of the propellant. However, difficulties of the thrust control and explosive danger of the pre-mixed propellant are major disadvantages of the solid propellant rocket[20]. Hybrid propellant rocket uses both a liquid and a solid propellant. Generally, a solid state fuel and a liquid state oxidizer are used. The separation of the oxidizer and the fuel improves safety from explosion or detonation of the propellant. In addition, thrust control and re-ignition are possible, with the higher specific impulse than that of the solid propellant rocket, at low cost stemming from safety features and inexpensively made fuel grain[21].

For a low-altitude sounding rocket application, a hybrid rocket propulsion system is desirable because it provides relatively high performance, system simplicity, and safety at low cost. A hybrid rocket system is considerably simpler than bi-propellant rocket system, and it has higher specific impulse than that of solid propellant rockets with throttle-ability and re-ignition-ability, which is difficult to achieve using a solid propellant rocket. In addition, a hybrid propellant has a lower explosion hazard (with the oxidizer and the fuel separated) than a solid propellant, and also it can be re-used after recovery. The hybrid propellant burns as macroscopic diffusion flame and the oxidizer-to-fuel ratio may vary depending on the length of the chamber and operation time, but the variation problem of the oxidizer-to-fuel ratio can be relieved by the short burning time and the small-scale chamber of a low-altitude sounding rocket.

There are spark ignition and catalyst ignition as the combustion starter for a hybrid rocket system. General spark ignition requires a gas mixture and electrical energy for ignition. In contrast, catalyst ignition requires no additional energy and gas mixtures; however, it does require a catalyst for propellant decomposition. A hybrid rocket system using catalyst ignition can be considerably simpler than a system using spark ignition. In addition, a catalyst ignition hybrid rocket system has higher ignition reliability. Therefore, hybrid rocket using catalyst ignition is a good candidate for a low-altitude sounding rocket propulsion system.

In this work, we report KAIST version university-based sounding rocket development process, which includes hybrid sounding rocket system design, fabrication, experimental performance test, and numerical estimations for internal and external ballistics. A simplified hybrid rocket system was developed with as few components as possible using catalyst ignition for low-altitude sounding rocket application. The developed hybrid rocket stand-alone system was fabricated and an experimental test was conducted to validate the performance of the designed rocket system. After the combustion test, the sounding rocket was designed and flight was simulated by developed trajectory estimation code. Finally, the fabricated sounding rocket was launch tested and effectiveness of this development process was evaluated.

2. Design of the propulsion system

2.1 Oxidizer and fuel selection

As a hybrid rocket oxidizer, nitrous oxide(N_2O), liquid oxygen(LO_2), and high-test hydrogen peroxide(H_2O_2) have been widely used. Nitrous oxide is the most commonly used oxidizer due to its self-pressurizing at the atmospheric temperature. Self-pressurization is a great advantage for a rocket system using N_2O as oxidizer due to non-necessity for pumps or external pressurization, which is desirable for simplifying the sounding rocket system. However, the N_2O pressure is notably sensitive to temperature and occasionally requires a heater to achieve the desired pressure and flow rate. In addition, there are detonation possibilities of N_2O from various ignition sources[22]. Liquid oxygen(LO_2) has shown good performance as an oxidizer and is easily available at low cost. As a cryogenic propellant, however, liquid oxygen has storage and handling problems. Even a small amount of water vapour in the feeding lines can become frozen and cause disturbance of the propellant flow, damaging the feeding lines. Furthermore, additional devices should be considered for the insulation and prevention of propellant boil off. In contrast, high-test hydrogen peroxide(H_2O_2) has high density and storability at the ambient temperature. A high density of H_2O_2 can reduce the size of rocket system by as much as half of the N_2O oxidizer rocket system[23], and the storage of H_2O_2 requires no additional devices for boil off prevention and insulation. In addition, H_2O_2 is a non-toxic propellant and is easy to handle. Therefore, H_2O_2 is a great alternative for a small-scale sounding rocket propellant and is chosen as the oxidizer in this study.

Hydrogen peroxide can be easily decomposed to oxygen and water vapour via a catalyst as in the equation below, with a heat energy of 2884.47 kJ/kg for 100 wt% H_2O_2, and the auto-ignition of fuel grain is possible via a high decomposition temperature.

$$2H_2O_2\,(l) \rightarrow 2H_2O\,(g) + O_2\,(g) + \text{Heat}.$$

The adiabatic temperature is 749˚C for the decomposition of 90 wt% H_2O_2, and this temperature is sufficient for auto-ignition of general carbon-based polymers, such as polymethyl-methacrylate(PMMA), polyethylene(PE), polybutadiene(PB), paraffin, and hydroxyl-terminated polybutadiene(HTPB). For fuel grain selection, the thermal strength of fuel grain should be considered. If the melting temperature of the a grain is too low, then the fuel grain can escape the combustor as a solid state without sufficient combustion, which may cause lower thruster performance[24]. We choose high density polyethylene(HDPE) as the hybrid rocket fuel, which has enough thermal strength for maintaining the fuel grain structure at the H_2O_2 decomposition temperature and has great compatibility with H_2O_2[24-26]. Polyethylene is also easily accessible at low cost.

2.2 Catalyst preparation

Hydrogen peroxide is decomposed by several catalysts, such as silver[27], platinum[28-31], iridium[32], and manganese dioxide[25, 33-35]. Among the possible catalyst alternatives, we choose MnO_2, which can be easily fabricated using the impregnation method with low cost and showed good performance for hydrogen peroxide decomposition. With alumina pellets as the catalyst support, which has sufficient thermophysical strength and good adhesion with metal, the MnO_2 catalyst was fabricated for hydrogen peroxide decomposition in this work. For MnO_2/Al_2O_3 catalyst fabrication, γ-alumina support was crushed to have a sufficient support size because an appropriate catalyst support size is required for the high performance of the catalyst reactor. If the catalyst support is overly large, then the catalyst surface for the chemical reaction is not sufficient for oxidizer decomposition at the catalyst reactor. However, if it is overly small, then a high pressure drop occurs at the catalyst bed, which results in the loss of the propulsion performance and causes low thrust efficiency. The catalyst support size was determined as 10 - 16 mesh, 2.00 - 1.19 mm size, considering previous work[24, 34, 36]. Ground γ-alumina support was washed with water and dried at 120˚C in a convection oven for 24 hours. Forty wt% sodium permanganate($NaMnO_4$) solution was used as the precursor, and the dried γ-alumina support was wetted at the solution for MnO_2 loading. The wetted γ-alumina support was dried at 120˚C in the convection oven for 24 hours, and the calcination process was conducted at 500˚C furnace for 5 hours. The calcination process was performed to burn off the unwished organic matter on the support. After the calcination process, the catalyst was washed with tap water to remove sodium ions on the support. Finally, the catalyst dried again at 120˚C in the convection oven for 24 hours and fabrication was completed. After the catalyst fabrication, the MnO_2/Al_2O_3 catalyst was introduced into the catalyst bed of the thruster.

2.3 Thruster design

The thruster consisted of the following: injector, catalyst bed, combustor, and nozzle. Injector supplies oxidizer into catalyst bed uniformly for efficient decomposition of the oxidizer, and it was designed with 45 holes of 500 μm size diameter, referring previous work[24]. The catalyst bed contains a catalyst for oxidizer decomposition and was designed for containing the fabricated MnO_2/Al_2O_3 catalyst with catalyst holder to prevent the catalyst from washing away at the catalyst bed. The catalyst capacity, which means the decomposable propellant mass flow rate per catalyst bed volume, was considered as 2.0 g/s cm^3[36], and the catalyst bed volume was determined for the required oxidizer mass flow rate. For low-altitude sounding rocket propulsion, the required thrust was 250 N, and the designed chamber pressure was 15 bar. According to the chemical equilibrium calculation result based on the CEA code[37], the theoretical maximum specific impulse occurred at the oxidizer-to-fuel ratio of 7 of 90 wt% H_2O_2/HDPE, which was 223 sec at frozen equilibrium condition. Therefore, the required oxidizer H_2O_2 was 114 g/s. For the catalyst bed design, however, the decomposition capacity for the oxidizer flow rate as high as 120 g/s was considered for adequate decomposition efficiency, and the designed catalysts bed volume was 60 cm^3 with diameter 6.0 cm and length 2.1 cm. The aspect ratio was 1/3, which was determined to be less than 1 to reduce the pressure drop in the catalyst bed[36]. The decomposed oxidizer product at the catalyst bed is supplied to the combustor. Combustor contains fuel grain and provides the volume for combustion of the fuel with the decomposed oxidizer. The reaction products at the catalyst bed are water

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vapour and oxygen at high temperature, which is sufficient for auto-ignition of the fuel grain in the combustor. The fuel grain was designed considering polyethylene regression rate[24] with decomposed hydrogen peroxide for the oxidizer-to-fuel ratio of 7. The fuel grain was of cylindrical shape, with 6 cm outer diameter and 12 cm length, and with a cylindrical hole of 1.2 cm diameter at the center of the cross section. The product of combustion is accelerated in the nozzle. The converging-diverging nozzle was used with a 45˚ contraction half angle at the converging section and with a 15˚ conical expansion half angle at the diverging section, considering the recommendations[19, 20]. All the component materials were chosen as stainless steel. However, due to the high temperature of the product gas of combustion, the graphite nozzle was used for product gas acceleration, and a nozzle case was required for integration of two different components. Designed thruster specification is listed in Table 1.

2.4 Tank, valves, and pipes configuration

For the simplified rocket propulsion system, we chose the blow down type feeding method, for which additional rocket system components, such as a tank, regulators, and valves for the pressurant, are not required. Three main components composed simplified hybrid rocket system: tank, valve, and thruster. The tank was used for both the oxidizer and the pressurant storage; therefore, a sufficient strength was required for high inner pressure. A cylindrical composite tank made of a 6061-T6 aluminum alloy as the liner and carbon/glass fiber as the reinforcement material was used, which has a maximum filling pressure of 250 bar. For the case of unexpected high pressure resulting from the unintended decomposition of hydrogen peroxide at the tank, relief valve was added for safety. To fill the tank with pressurant and oxidizer, two ball valves were used in a series at the tank inlet. As the main valve for rocket system operation, a latch valve was installed. The latch valve requires only an initial electrical signal input for actuation and no additional constant signal input is required to keep its position, which is desirable to be used for system simplicity of the sounding rocket. Stainless steel, which is compatible with hydrogen peroxide, was used for the oxidizer supplying lines. Feeding lines with 1/4” and 3/8” size were used for the required oxidizer flow rate. A drawing of the designed rocket stand-alone propulsion system is shown in Fig. 1.

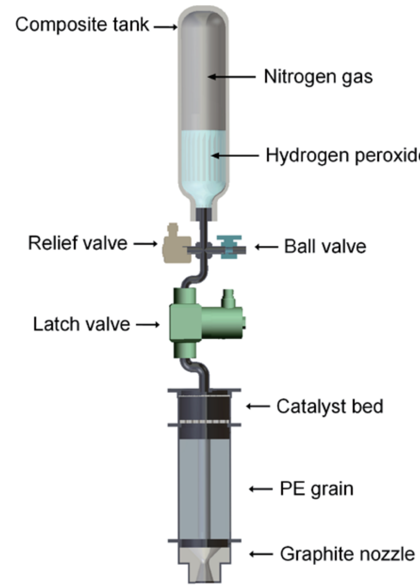

Fig. 1. Hybrid rocket stand-alone system configuration

Table 1. Hybrid thruster design result

Thruster specification	
Thrust	250 N
Chamber pressure	15 bar
Propellant	90wt% H_2O_2/PE
Specific impulse	223 sec
Oxidizer to fuel ratio	7
Oxidizer flow rate	120 g/s
Catalyst capacity	2.0g/cm^3
Catalyst/support	MnO_2/Al_2O_3
Catalyst support size	10-16 mesh

3. Propulsion system performance test

3.1 Experimental setup

The designed propulsion system was experimentally tested while vertically installed on the ground as shown in Fig. 2. Two aluminum plates support both the top and bottom sides of the thruster, and the other plate supports the part of the feeding lines. Ground test setup structures were made of aluminum and fixed on the ground. To measure the pressure, the PSH model pressure sensor with 0.054% accuracy from the Sensys Corporation was used, and the pressure sensors were installed at five points: the tank inlet, tube before injector, catalyst bed, before the combustor, and after the combustor. A K-type thermocouple was installed at the catalyst bed to measure the hydrogen peroxide decomposition temperature and determine the decomposition efficiency. A K-type thermocouple is inexpensive and the most commonly used sensor to measure temperatures over the range of -200˚C to 1350 ˚C with approximately 41 μV/°C sensitivity, and it was appropriate for the adiabatic temperature of the 90 wt% hydrogen peroxide decomposition, 749 ˚C. To record the experimental test, two image devices were used, focusing on both the overall propulsion system and the nozzle. Before the combustion test, several water tests were conducted to determine the required initial pressurant filling pressure, which was expected to determine the mass flow rate of the oxidizer. The water test results showed various flow rates depending on the initial pressurant pressure. In addition, the relationship between the average mass flow and the pressure difference at the injector was attained. Using the flow rate characteristics, it was possible to predict the required pressurant filling pressure, considering the density of the hydrogen peroxide as the oxidizer. The required pressurant initial pressure was 33 bar, for the target average oxidizer flow rate of 120 g/s, which was higher than the designed oxidizer flow rate as much as 6 g/s, in preparation for degradation of the specific impulse performance by ignition delay and a larger oxidizer flow rate at the initial operation under no additional mass flow rate controller. After filling 360 grams of 90 wt% hydrogen peroxide into the composite tank, the entire system was assembled, and then 99.99% nitrogen gas filled the remainder of the composite tank with approximately 33 bar initial filling pressure. The operation valve was triggered by a signal input over a long distance for safety. The pressures and temperature data were acquired by a data acquisition system of National Instruments Co., Ltd.

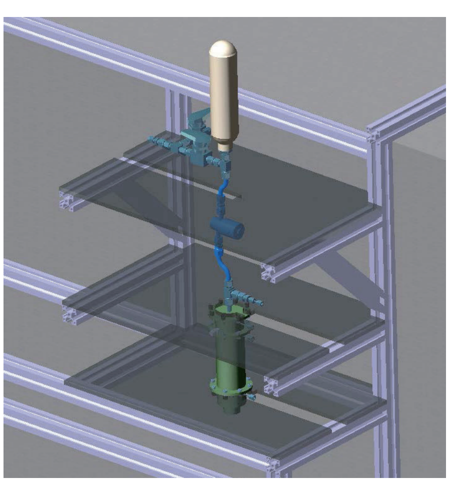

Fig. 2. Experimental setup of the combustion test on the ground

3.2 Numerical and experimental test results

Before the ground test, numerical internal ballistics was conducted to estimate the performance of the propulsion system. First, the pressure variation of the pressurant depending on the oxidizer consumption in the tank was calculated using experimental nitrogen gas properties. After that, oxidizer flow rate was estimated considering pressure difference between feeding and chamber pressure. Experimental discharge coefficient of feeding line was used, which was acquired using water injection tests. Finally, combustion chamber pressure was estimated using experimental regression rate in previous work[24] and chemical equilibrium properties obtained by CEA code[37] developed by NASA. Under the assumption of flow choking at the nozzle throat, the chamber pressure was calculated using the equation of c* definition. Repeating these calculations at each time step, feeding and chamber pressure were estimated and these values were compared with experimental data after the test. Calculation procedure for propulsion system performance estimation is summarized in Fig. 3.

The experimental test results successfully showed the performance of the hybrid rocket system and auto-ignition of the fuel grain. Two image devices recorded the experiment, and the results are shown in Fig. 4. The pressures at five points and the temperature at the catalyst bed were measured during the ground combustion test. As shown in Fig. 5, the temperature of the catalyst bed was approximately 746˚C, which was 3˚C lower than the adiabatic temperature of the 90 wt% hydrogen peroxide decomposition indicating sufficient decomposition efficiency of the catalyst bed. Among the five measured pressures, the highest pressure(feeding pressure) and the lowest pressure(after combustor pressure) are shown in Fig. 5. The feeding was in blow-down mode due

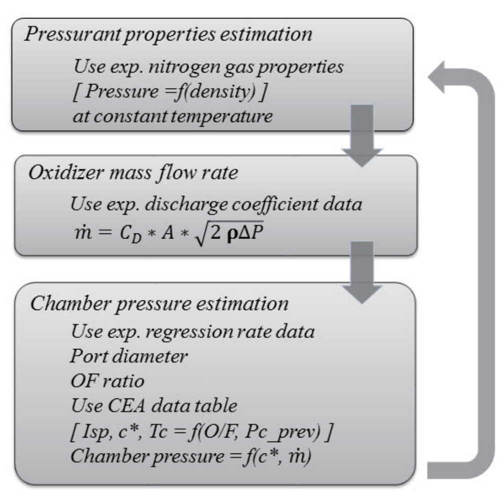

Fig. 3. Numerical estimation procedure for internal ballistics

to pressurant expansion as the oxidizer consumed at the tank. From approximately 33.3 bar, the feeding pressure was reduced to approximately 13.0 bar during operation. The pressure after the combustor, which was located just before the converging nozzle inlet, was also varied from 16.6 bar to 10.0 bar. The time-averaged pressure was 17.4 bar at the feeding line and 12.1 bar at after the combustor. The ignition delay was 0.38 sec, and the rising time to the 90% of the maximum pressure at after the combustor was 0.46 sec. Since the ground test was conducted without mass flow meter in order to have same system configuration as flight test, the oxidizer mass flow rate was calculated using the mass flow equation below, with the injector discharge coefficient C_d, the injector hole total area A, the oxidizer fluid density ρ, and the measured injector pressure difference ΔP.

$$\dot{m} = c_d A \sqrt{2\rho \Delta P}.$$

For thrust estimation, propellant mass flow rate and theoretical specific impulse regarding measured chamber

Fig. 4. Static combustion test of the hybrid rocket stand-alone system

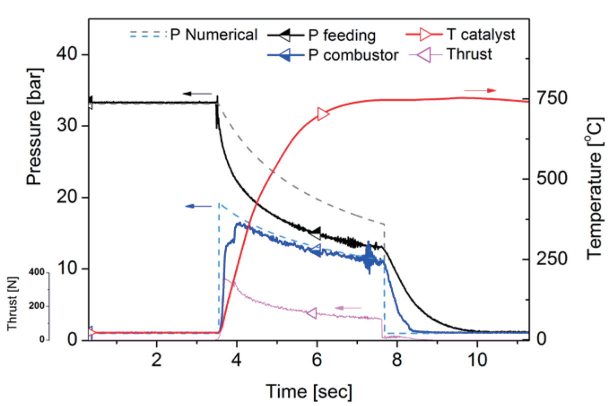

Fig. 5. Ground performance test results: catalyst bed temperature, feeding pressure, combustor pressure and estimated thrust generation with numerical pressures

pressure were considered. Chemical equilibrium code[37] was used for the specific impulse calculation at each chamber pressure. Oxidizer-to-fuel ratio required for the theoretical specific impulse estimation was determined using oxidizer mass flow rate and fuel grain mass variation during operation. The initial and final mass of the fuel grain was measured and approximately 30 g mass variation was observed after 3 sec firing. The time-averaged fuel grain consumption rate was approximately 10 g/s and the oxidizer mass flow rate was 120 g/s. Therefore, the oxidizer-to-fuel ratio was approximately 12. The estimated thrust using the propellant mass flow rate and the calculated specific impulse is shown in Fig. 5. The experimental total impulse was 700 N•sec, which was 93.3% of the designed total impulse, 750 N•sec.

The comparison between estimated and experimental pressures of the propulsion system presented mostly well-matched pressure variations at the tank and combustion chamber showing the effectiveness of algorithm for propulsion system performance estimation. However, a small difference between expected and measured feeding pressure occurred because feeding pressure was measured not in the oxidizer tank, but near the inlet of the tank due to difficulties of inner pressure sensing in the commercialized composite tank.

4. Sounding rocket system configuration and flight testing

4.1 Sounding rocket system

With the developed hybrid propulsion as a propulsion system for a low-altitude flight mission, a sounding rocket was designed, fabricated, flight simulated and tested. Considering mass distribution and size of the developed propulsion system, the sounding rocket was designed as compact as possible. We estimated the mass and moment of inertia of the sounding rocket using a 3D CAD program, which is essential for sounding rocket static and dynamic stability design. Using the 3D CAD software, the sounding rocket total mass was estimated to be 5.2 kg, and the final mass was estimated to be 4.9 kg. The moment of inertia was estimated to be 0.02 kg·m^2, 0.6 kg·m^2, and 0.6 kg·m^2, for the axial direction along the x-axis, y-axis, and z-axis, respectively. The center of gravity(CG) position of the sounding rocket was 63 cm from the nose cone tip when the oxidizer filled the tank, and the value changed to 64 cm when the oxidizer was completely consumed. The center of pressure position(CP) was estimated using the Barrowman equation[38, 39] and it was 82 cm from the nose cone tip.

Therefore, the static margin at the initial condition was two caliber, i.e., the distance between CG and CP was two times longer than the diameter of the rocket body, indicating that the rocket was statically stable[40, 41]. The sounding rocket total length was 111 cm, and the body diameter was 11 cm with four fins. The fins were designed considering the appropriate center of pressure position based on the Barrowman equation. The nose cone was designed using the Von Karman nose cone curve for the lowest aerodynamic drag. The specifications of the designed sounding rocket are summarized in Table 2. The designed sounding rocket was fabricated using polycarbonate material for the casing, and the casing was reinforced by stainless steel support.

4.2 Flight simulation and experimental testing

Before the flight test of the sounding rocket, flight simulation was conducted to estimate the dynamic stability of the sounding rocket as well as the flight altitude and velocity. For the flight simulation of the designed sounding rocket, a trajectory estimation code was developed considering 6 degree-of-freedom motion. As the first step for the trajectory calculation, an angular velocity of the rocket for each axis was acquired by the equation of rotation motion, followed by Euler angle calculation using the equation of kinematic relation. The attitude of the rocket expressed by Euler angle made gravitational force be divided into each direction of the body fixed axis. Finally, velocity for each direction of body fixed frame was determined considering a translational motion of the rocket and repeating this calculation for each time step showed 6 degrees of freedom flight trajectory of the sounding rocket. The trajectory simulation algorithm is shown in Fig. 6. The developed code was validated by comparing the results with that of open trajectory calculation code, OpenRocket[42]. The two estimated trajectories were almost the same for altitude, velocity and flight time, which verified good estimation accuracy of the developed code. The simulation was conducted at the launch angle of 85° without additional attitude control and thrust vector control. After the launching at the ground, the sounding rocket was accelerated to approximately 33 m/s until the end of the burning of 3 sec. During the coasting flight, the velocity gradually decreased until the sounding rocket reached the maximum altitude, and then the velocity increased again until it reached the ground with the maximum velocity of approximately 45 m/s. The flight time was 11 sec, and the maximum altitude was approximately 100 m in 6 sec after the launch.

Based on these estimations, experimental flight test was conducted. First, the fabricated sounding rocket was installed at the launch pad with a barometer as a payload for altitude measurement, and then latch valve was operated by the signal through the electrical wire, the loose connection

Table 2. Sounding rocket design specification

Specification	
Thrust	250 N
Burn time	3 sec
Final mass	4.9 kg
Initial mass	5.2 kg
Ix, Iy, Iz	0.02, 0.6, 0.6 kg m^2
Static margin	2 caliber

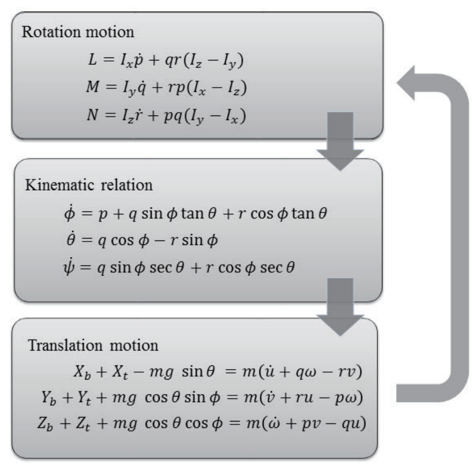

Fig. 6. Algorithm for flight trajectory estimation


Fig. 7. Sounding rocket installed on the launch pad (left) and flight testing (right)

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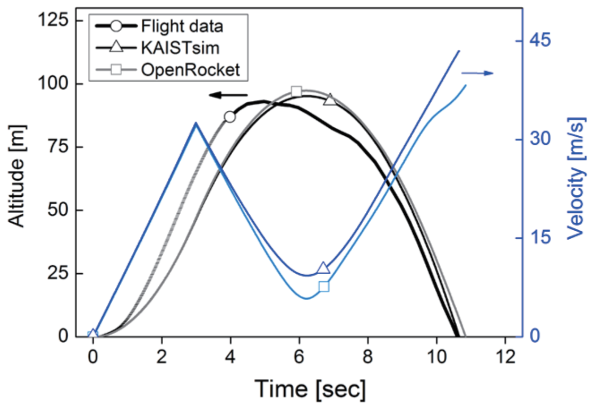

Fig. 8. An experimental flight trajectory with estimations using KAISTsim and OpenRocket

of which made it cut immediately just after the launch. Fig. 7 presents the sounding rocket installed at the launch pad ready for flight testing. During majority of flight, altitude measurement by the barometer was good. However, for the first 4 sec of flight there was sensing error due to effect of pressure variation from the thruster. Fig. 8 shows the acquired flight altitude data with modification during the time of first 4 sec based on trajectory estimation code. As expected in flight simulation, maximum altitude of the sounding rocket was approximately 100 m with the flight time of 11 sec.

Both flight trajectory simulation and testing showed that the designed sounding rocket reached around a hundred meter altitude with dynamic stability using the developed hybrid rocket system, successfully showing the effectiveness of the simplified hybrid rocket as a propulsion system for small-scale sounding rocket operation. The 3 sec burning time of the hybrid rocket propulsion system made these results, but the maximum altitude can be changed depending on the initial propellant mass and the burning time according to the sounding rocket mission profile. We expect various low-altitude mission capacities of the sounding rocket as a CanSat carrier using the developed simplified hybrid rocket system.

5. Conclusion

A Simple hybrid rocket system was designed, and its performance was verified successfully; these results were essential for the low-altitude operation of the simpler and safer small-scale sounding rocket at low cost. The hybrid rocket system consisted of as few components as possible. Catalyst ignition operation eliminated additional ignition devices on the rocket system, and blow-down operation with both pressurant and oxidizer in the one tank required no additional tank, valve, and feeding line for pressurant feeding and storage. Using 90 wt% hydrogen peroxide as an oxidizer, auto-ignition of the polyethylene grain was successfully achieved via decomposed hydrogen peroxide at the experimental test. Numerical internal ballistics result was well matched with experimental propulsion performance showing the effectiveness of suggested algorithm, and external ballistics result presented validity of the numerical estimation. The sounding rocket was designed and fabricated using the developed hybrid rocket propulsion system. The sounding rocket was successfully launched and validated the developed rocket system and the development procedure. Various missions of the sounding rocket are expected in the form of a CanSat carrier using the simplified hybrid rocket system according to initial propellant mass and burning time.

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