The design and performance on 200N-class bipropellant rocket engine using decomposed H_2O_2 and Kerosene

Yang-Suk LEE[1], Jun Hwan JANG*[,2]

*Corresponding author
[1]Koreaaero Industry, Sacheon-si, Gyeongsangnam-do, Korea
icueyes@nate.com
*[,2]Mechanical Design Major Department of Mechanical Engineering, Yuhan University, Gyeonggi-do, Korea
bulbearj@gmail.com

DOI: 10.13111/2066-8201.2019.11.3.9

Received: 03 July 2019/ Accepted: 12 August 2019/ Published: September 2019

Abstract: *Mono-propellant thrusters are widely utilized in satellites and space launchers. In many cases, they are using hydrazine as a propellant. However, hydrazine has high toxicity and high risks in using for launch campaign. Recently, low-toxic (green) propellant is being highlighted as a replacement for hydrazine. In this paper, 200N bi-propellant engine using hydrogen peroxide/kerosene was designed/manufactured, and the spray or atomization characteristic and inflation pressure were determined by cold flow test, and combustion and pulse tests in a single cycle same as previous methods were conducted. As uniformly supplying hydrogen peroxide through plate-type orifice to a catalyst bed, the hot gas was created as a reaction with hydrogen and catalyst. And then, it was confirmed that the ignition is possible on the wide range of O/F ratio without additional ignition source. The liquid rocket engine with bi-propellant of hydrogen peroxide/kerosene and design/test methods which developed in this study are expected to be utilized as an essential database for designing of the ignitor/injector of bi-propellant liquid rocket engine using hydrogen peroxide/kerosene with high-thrust/performance in near future.*

Key Words: *Green Propellant, Hydrogen Peroxide, RCS(Reaction Control System), Space Launcher*

1. INTRODUCTION

The hydrogen peroxide (H_2O_2), as a propellant, was first utilized in turbine-driven system and gas generator for driving turbopump in German submarines in World War Ⅱ and later it was widely utilized in torpedo and thrust boost systems in various countries. However, in late 1960s, due to the performance-driven trend in rocket engine development, it was replaced with Hydrazine (N_2H_4), Liquid Oxygen (LOx), and Nitrogen Tetroxide (N_2O_4) which have relatively high specific impulse compared to hydrogen peroxide (H_2O_2) [1]. But this replaced propellant has high toxicity, it can have severe impact on human body or difficulty in storing in room temperature. Since late 1990s, environmental issues and handling reliability have been set as a critical agenda, the characteristic of hydrogen peroxide (H2O2) has been re-examined, it is environment friendly, easy to handle in room temperature and it has increased reliability. Various studies are being conducted in many countries [1].

The high concentrated H_2O_2 has an exothermic reaction which is decomposed into hot vapor and oxygen in a chemical reaction being contacted with catalyst. Due to this, H_2O_2 can be utilized as a mono-propellant using catalytic reaction in rocket engines with low thrust and simple structure [2,3].

In addition, the hot oxygen gas created by catalytic reaction can be utilized as an oxidizer in bi-propellant rocket engine and since it can have auto ignition of the fuel, it can also be utilized as an attitude controller in orbit correction engines of satellites or attitude controller of launch vehicles that require numerous reignition function [4].

Patel et al. [5] researched various materials and surface treatments on electro-kinetic pumping media and then, demonstrated an EK pumping of anhydrous hydrazine using all packed-capillary and larger sintered-monolith pump designs.

Anflo and Möllerberg [6] described performance, characteristics, design and verification of PRISMA HPGP propellant system and discussed interoperability issues related to using new propellant with COTS elements. Amri and Rezoug [7] focused on combustion of liquid propellant that is utilized in space launcher program (satellite and launch vehicle) and studied on combustion of propellant mix such as hydrogen/ oxygen, hydrocarbon fuel/ oxygen, hydrocarbon fuel/ hydrogen oxide, oxygenated hydrocarbon fuel/ hydrogen, hydrazine/ nitrogen tetroxide and mono-methyl hydrazine/ nitrogen tetrosin/ unsymmetrical de-methyl hydrazine/ nitrogen tetrosin/ nitride. Gohardani et al. [8] reviewed and studied on promising green space propellant selected for various space mission. Aforementioned in-depth system studies related to propellant system released potential approaches on advanced green propellant in near future.

Zhang et al [9] numerically studied liquid ADN-based ternary mixtures and the decomposition and combustion process for propulsion. Amrousse et al. [10] studied that liquid HAN-based single cell can be a replacement of toxicity of hydrazine and verified that the effect of additional methanol as fuel can bring out the improvement of combustion rate. Kang et al. [11] provided that the development of non-toxic hypergolic bi-propellent cab be a replacement of previous high-toxic propellant.

This result shows that there is a chance of new concept engine for space mission. Jing et al. [12] experimentally and numerically researched combustion process of liquid ADN-methanol that is liquid propellant in a small space thruster. A measurement of phase doppler is used to review injection process and its result is used as liquid boundary condition in simulation in order to minimize uncertainty.

Shindo et al. [13] applied A discharge plasma in an effort to execute combustion of hydroxylammonium nitrate-based propellant as a replacement of catalyst generally used. During launch test with short intervals, it was verified that thrust, thrust vs. output ratio and chamber pressure with a single discharge pulse.

Kumar [14] described detailed summary on physical and chemical characteristics, pyrolysis and combustion movement of ADN and ADN based propellant and discussed catalytic effect on pyrolysis of ADN, structure of combustion wave and combustion rate. Wilhelm et al. [15] conducted a preliminary test with two ignition methods by using ADN based liquid propellant FLP-106 and LMP-103S.

In this paper, the design and development test of 200N bi-propellant thruster engine using H_2O_2 and kerosene was conducted.

First, the 200N bi-propellant was designed and produced, verified spray performance of injector and then established ignition/ combustion test process using liquid rocket combustion test stand. Through this process, it was intended to establish a foundation for upcoming H_2O_2 kerosene engine.

2. DESIGN OF INJECTOR, COMBUSTION STAND/NOZZLE

2.1 Design of Injector

In this paper, it was intended to design/develop bi-propellant rocket engine with the same specification as in table 1 using H_2O_2 and kerosene as an oxidizer and fuel, respectively. As for ignition method, the catalytic ignition method was selected that sprays kerosene on hot gas disassembled from direct contact of H_2O_2 and catalyst (permanganate). The catalytic ignition method has the advantage of being able to ignite without additional instrument since it ignites by catalyst. Fig. 1 shows the schematic diagram of designed injector head, the kerosene sprayer is located in the center and the H_2O_2 sprayer which moved into catalyst bed is located in the outside of Kerosene sprayer.

It was designed that H_2O_2 injected into catalyst bed traveling from shower head type orifice with two stages, had a catalytic reaction and uniformly supplied into combustion stand. Kerosene was supplied into combustion chamber by using a swirl injector with great performance of atomization and combustion stability.

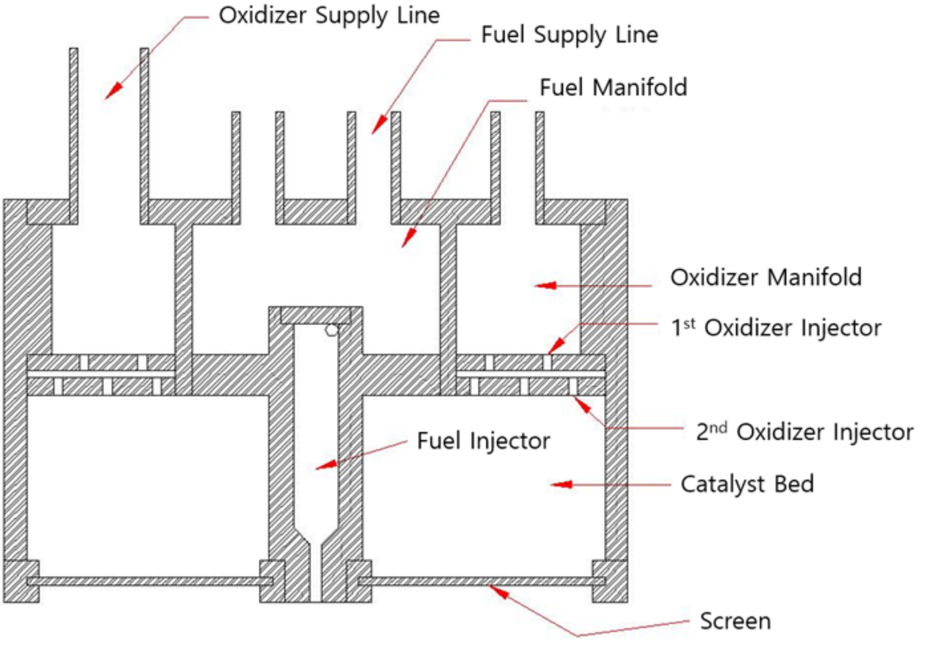


Fig. 1 Schematic of injector head

The swirl type fuel injector uses chemical equilibrium program (CEA) with the same process as in Fig. 2 and calculated main parameters value that requires configuration decision and then, main measurements for deciding injector configuration were calculated based on the general injector design method [16, 17].

In addition, the H_2O_2 injector has a shower head type orifice to uniformly supply H_2O_2 to the catalyst bed and the orifices were adequately allocated to maintain uniform supply to the catalyst bed, and to have sufficient reaction time with catalyst bed after calculation of entire area of orifice which make design mass flow rate supplied from design differential pressure.

The main parameters of design injector are shown in table 2 and the manufactured injector is presented in Fig. 3.

Table 1. Requirement for engine design

Propellant	**Oxidizer**	**Fuel**
	H_2O_2 (96%)	**Kerosene**
Thrust	200 N	
Chamber pressure (P_c)	10 bara	
P_c / P_e	1000	

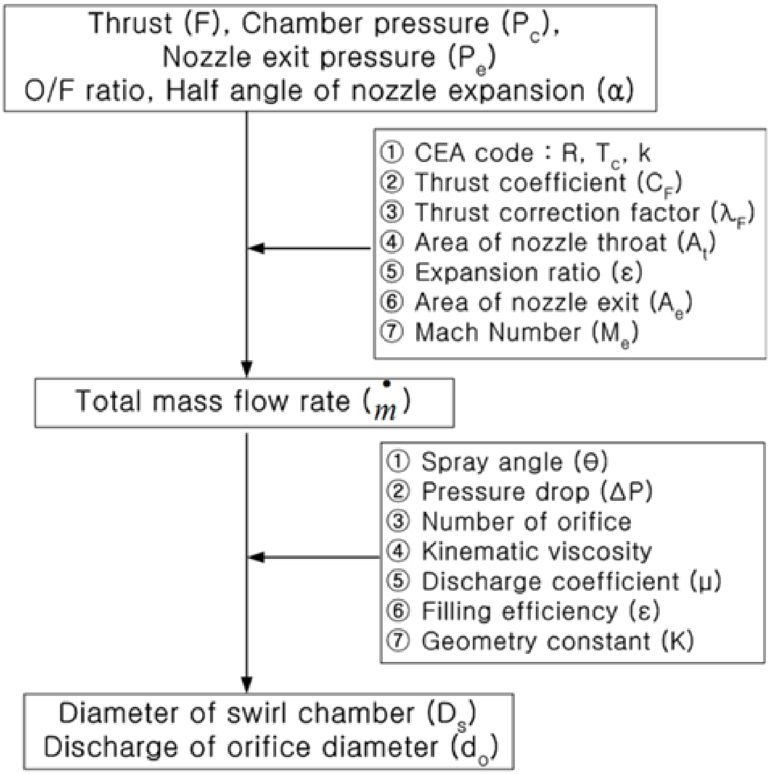


Fig. 2 Flow chart of injector design (Fuel)

Fig. 3 Manufactured injector head (200N)

3. TEST INSTRUMENT AND METHOD

3.1 Cold flow test method

Prior to combustion test, cold flow test was conducted in order to set spray angle of injector designed/manufactured using simulation propellant, measure mass flow rate and set supply pressure. The test instrument of cold flow consists of simulation propellant supply instrument, test stand and patternator.

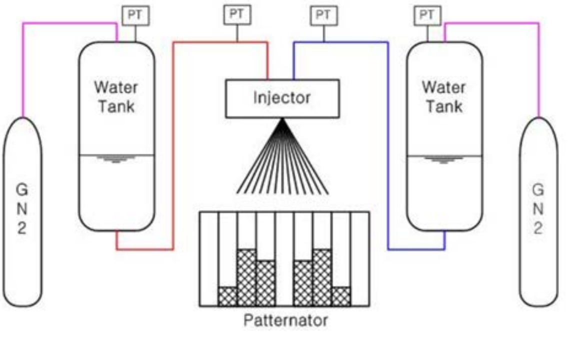
In cold flow test, the water was utilized as simulation propellant for H_2O_2 and kerosene since it is easy to handle and safe. When using simulation propellant instead of actual propellant, the mass is different from the actual propellant, the flow rate of real fluid was calculated by density correction on measured flow rate.

The designed/manufactured injector was installed on the upper part of cold flow test stand as shown in Fig. 4 and mass flow rate test was conducted for each propellant based on differential pressure.

In cold flow test, each propellant manifold pressure was confirmed by pressure sensor which installed on the upper part of injector by supplying simulation propellant into the injector and the mass flow value was calculated by measuring of total weight and time and collecting simulation propellant which passed through injector.

In addition, the uniform atomization characteristic and spray angle were measured by using Trouble Shooter HRMS (FASTEC) to confirm the uniform atomization characteristic of the propellant.

The data of mass flow rate and pressure can be stored and monitored in real time by Labview and data collection used NI PCI-6254 board which is able to use up to 18 channels.



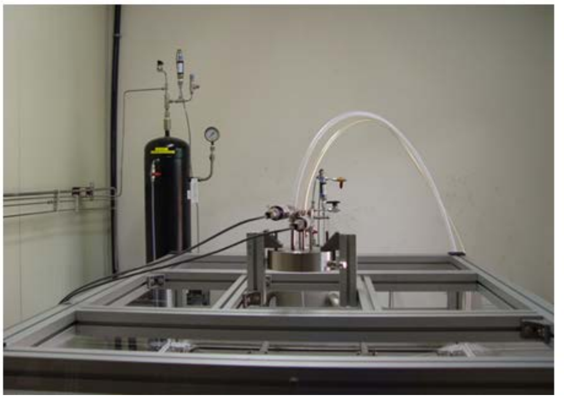


Fig. 4 Schematic diagram of cold flow test % Installed injector

3.2 Combustion test method

Based on the result from the cold flow test, the operating requirements of designed / manufactured injector were determined and as shown in Fig. 5, the injector head, combustion chamber and nozzle were assembled horizontally on the combustion test stand. The combustion test was conducted by varying kerosene flow while maintaining H_2O_2 flow as shown in table 2 and combustion performance based on various O/F (6.5 ~ 10.8) ratio was reviewed.

In addition, the time of combustion was set as 3 seconds in consideration of the time when the pressure of combustion chamber reached at normal state and the pulse mode combustion test was also conducted to see the characteristics of reignition.

The supply of H_2O_2 was led for 2 seconds in order to secure sufficient temperature for stable combustion with ignition of kerosene on the high temperature created by decomposition of H_2O_2.

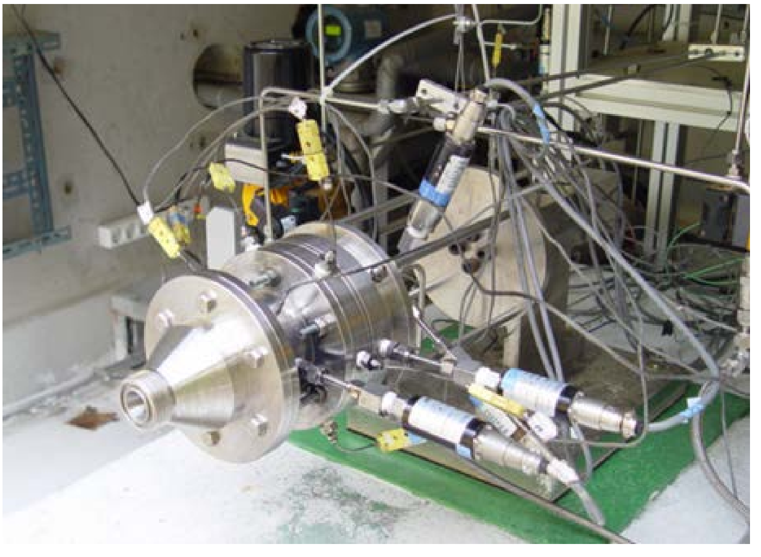


Fig. 5 Installed engine for hot test

Table 2. Test conditions of hot firing test

Test Number	Mass flow rate (g/s)		O/F ratio	Time
	H2O2	Kerosene		
1	124	11.5	10.8	3 Sec
2	124	14.0	8.9	
3	123	15.0	8.2	
4	123	16.0	7.7	
5	124	19.0	6.5	
6	124	15.0	8.2(Pulse)	

4. TEST RESULT AND ANALYSIS

4.1 Cold flow test results

As mentioned above, to measure the mass flow of designed/manufactured H_2O_2 and injector based on differential pressure, the cold flow test was conducted using water, simulation propellant and the mass flow was converted through density correction. Fig. 6 shows the configuration of simulation propellant sprayed from the injector.

As shown in Fig. 6, it was confirmed that the spraying of the simulation propellant was uniformly executed, and it can be inferred that smooth and stable catalytic reaction can be made by uniform supply of H_2O_2 to the catalyst bed.

Fig. 7 shows the mass flow rate of H_2O_2 injector based on differential pressure and it was confirmed that the mass flow value which meets the designed value of 108.8g/s can be achieved when the differential pressure is at 1.5 bar lower than the designed differential pressure.

This is due to the fact that there is a small artificial error since the orifice is very small as 0.67mm.

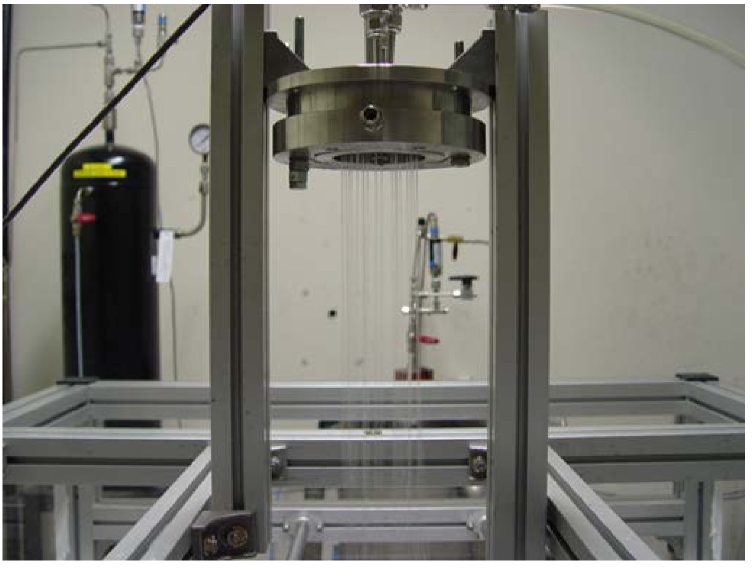


Fig. 6 H_2O_2 injector cold flow test

The kerosene injector which is designed as swirl type plays the role of maintaining the combustion process through atomization/ vaporization/ mix of kerosene supplied to high gas which phase changed by catalytic reaction of H_2O_2.

Fig. 9 shows the configuration of spraying propellant through swirl type kerosene injector and the angle of dispersion was measured approximately as 40 degree in designed differential pressure as a result of measuring angle of dispersion using high speed camera. The reason why the measured angle of dispersion was lowered that the designed angle of dispersion (80 degree) is that the length of swirl chamber and orifice was extended due to the interference of catalyst bed compared to the length of general swirl type injector.

Therefore, it can be estimated that the actual angle of dispersion was formed lower than the designed angle of dispersion because of the loss of each speed due to the viscosity while the propellant supplied to the swirl chamber traveling through swirl chamber and orifice. In addition, due to the high O/F ratio of H_2O_2/ kerosene compared to O/F ratio of kerosene/ liquid oxygen the relatively small amount of fuel was supplied therefore, it was estimated that the actual angle of dispersion was formed lower due to the difficulty of process and loss of viscosity because of small diameter of injector orifice.

Fig. 9 shows the mass flow rate of kerosene injector based on differential pressure of kerosene injector and it was confirmed that fuel swirl injector has approximately 14.7g/s of mass flow at the approximately 3.5 bar of differential pressure which is similar to designed mass flow (approximately 14.7 g/s).

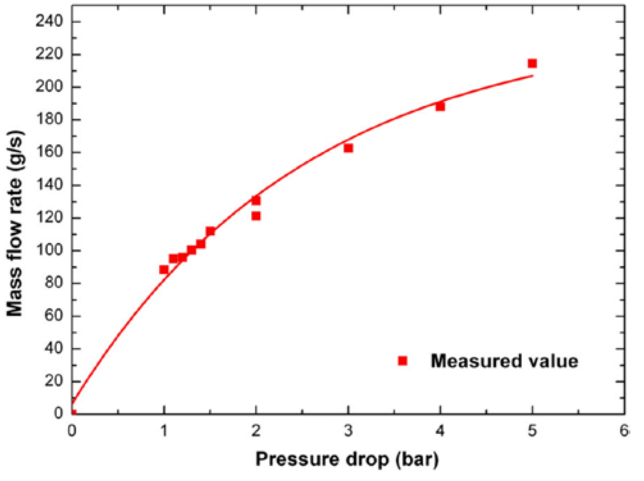


Fig. 7 Mass flow rate of H_2O_2 injector

Fig. 8 Kerosene injector cold flow test

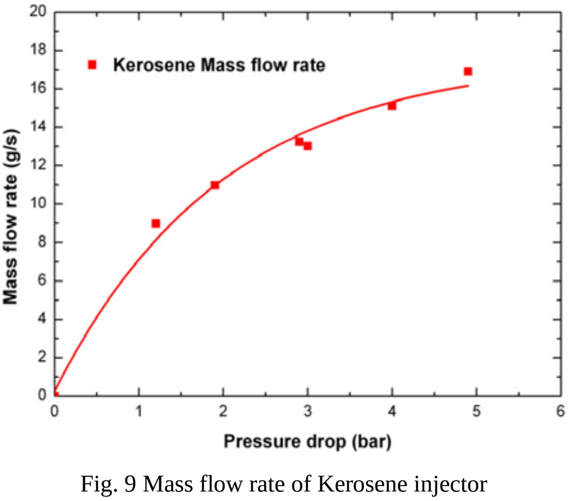


Fig. 9 Mass flow rate of Kerosene injector

4.2 Combustion test results

Fig. 10 shows the changes of pressure combustion chamber and mass flow of propellant over time; it can be found that 170g/s of H_2O_2 mass flow was supplied to combustion chamber which is larger than the design value (108.8g/s).

This is due to the fact that the mass flow was controlled by a plate-type orifice not by venturi, the differential pressure of H_2O_2 manifold and combustion chamber is larger than the designed differential pressure before the initial pressure of combustion chamber reaches to designed pressure (10bara).

It could be also found that the pressure of combustion chamber increases to approximately 8.5bara when H_2O_2 dissembled into high temperature gas through catalyst bed. It means that the pressure of combustion chamber is very high since the H_2O_2 firstly supplied in case of H_2O_2/ kerosene combustion as compared to the kerosene/ liquid oxygen combination which has little impact on the pressure of combustion chamber when one of two propellants supplied first. Therefore, this suggests that we need to be careful of setting purge pressure to prevent from the firstly supplied hot propellant backflowing to the fuel injector when it comes to development of high performance of H_2O_2/ kerosene engine with additional ignitor and multi-injector. Later, as the kerosene supplied, the pressure of combustion chamber reached to 10.4bara which is similar to design pressure and it was found that mass flow which is similar to initially intended mass flow was supplied.

The reason why the pressure of combustion chamber is slightly increased at the initial combustion is that the kerosene is supplied through the orifice at the time of the combustion pressure was reached at 8.5bara and the initial differential pressure is formed higher than the design differential pressure.

As such, the kerosene supplied through the orifice is the result of larger amount of supply than the amount of design mass flow. Fig. 11 shows the picture of combustion test at a normal state with 8.9 O/F ratio.

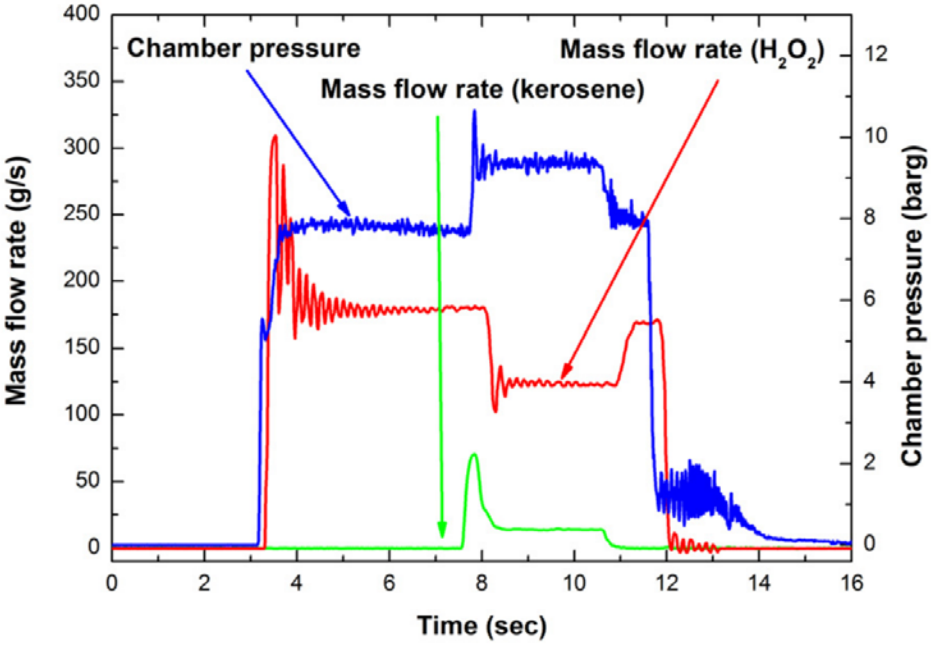


Fig. 10 Chamber pressure and mass flow rate

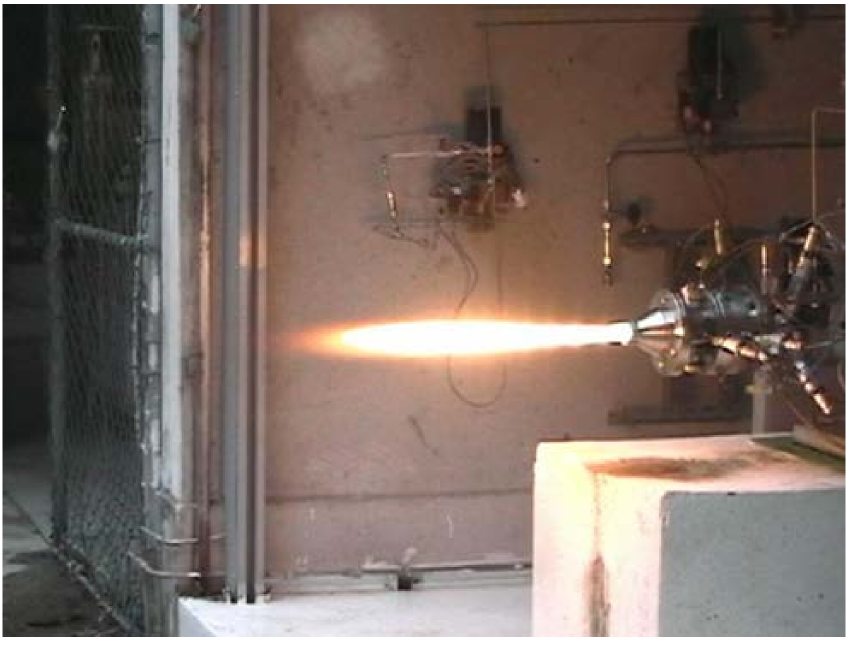


Fig. 11 Hot firing test (O/F ratio = 8.9)

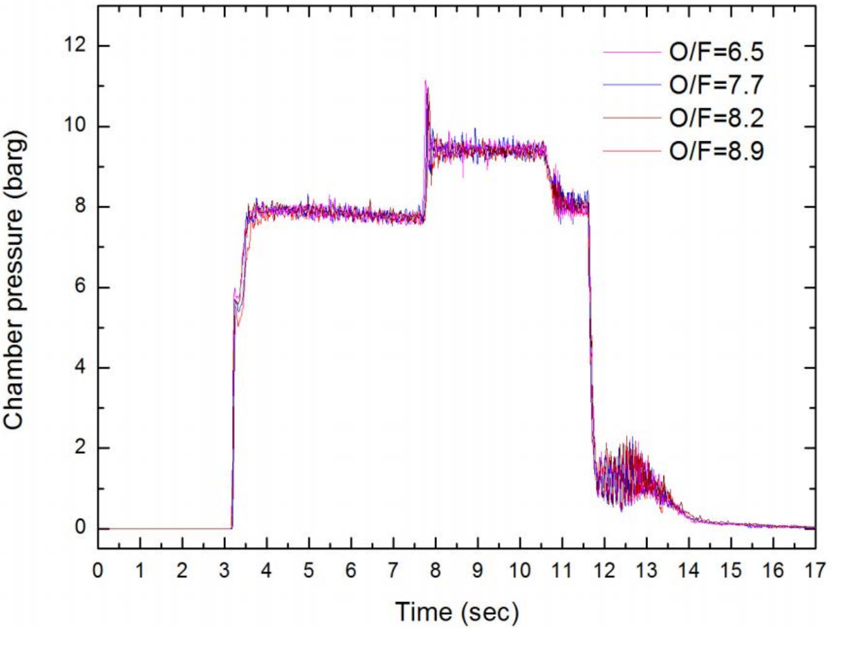


Fig. 12 Chamber pressure vs time curve (O/F ratio)

Fig. 12 shows the changes of pressure in combustion chamber on various O/F ratio over the time.

Around all time frame, there is little changes of pressure in the combustion chamber based on the combustion test process and it was found that the combustion test can be conducted without issues on various O/F ratio.

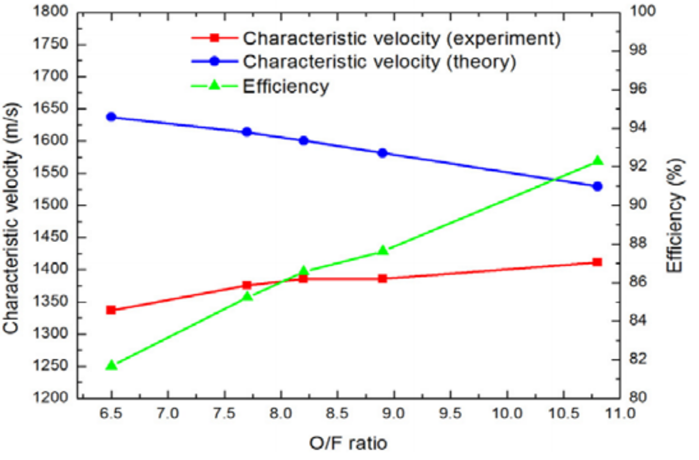


Fig. 13 O/F vs C* and efficiency

Fig. 13 shows the characteristic velocity(C*) and characteristic velocity efficiency(ηC*) which usually are utilized as parameters for evaluation of combustion performance in order to evaluate the combustion performance based on O/F ratio change. The characteristic velocity (C*,exp) measured from the test was calculated by using the pressure in the combustion chamber and the total mass flow supplied to the combustion chamber from the combustion test and the characteristic velocity efficiency was calculated by the ratio of theoretically calculated characteristics of velocity(C*,theory) by using CEA code. As we can see in Fig. 13, the characteristic velocity (theory) shows highest O/F ratio at 6.5 and it shows a tendency of gradual reduction based on the increase of O/F ratio. Whereas, the characteristic velocity (experiment) shows a tendency of little increase based on O/F ratio change. But, the increment is very little and it can be assumed that the efficiency of spraying is reduced due to the loss of the spraying angle of the fuel injector and error of production. It can be a reason that the efficiency of characteristic velocity efficiency was calculated in 82 ~ 92% but, it is not facing any problems for this engine related study to be utilized as database for advance study of H_2O_2/kerosene bi-propellant rocket in near future.

5. CONCLUSIONS

In this paper, auto-ignition 200N bi-propellant liquid rocket engine using H_2O_2/ kerosene as propellant and catalyst was designed/manufactured; the spray characteristic and inflation pressure were decided by the cold flow test; the single cycle of combustion test, which is the same as for the traditional method and the pulse test were conducted. By uniformly supplying H_2O_2 through the panel orifice to the catalyst bed, hot gas was created as the reaction of H_2O_2 and catalyst and then the kerosene was injected to see the ignition can be made without additional ignition source at a wide range of O/F ratio. After observing the combustion performance based on various range of O/F ratio, 82 ~ 92% range of the characteristic velocity was achieved, and it was confirmed that the combustion was stable within approximately 2.3% of pressure perturbation. In addition, the producibility was excellent per each pulse cycle and the re-ignition was possible through the pulse mode combustion test. The H_2O_2/kerosene bi-propellant rocket engine developed and the design/test method in this paper can be utilized as viable basis data for the design of high thrust/performance of ignitor/spray of H_2O_2/ kerosene bi-propellant liquid rocket engine.

REFERENCES

[1] M. Ventura and P. Mullens, *The Use of Hydrogen Peroxide for Propulsion and Power*, 35th Joint Propulsion Conference and Exhibit, pp. 1-9, 1999.

[2] P. Gallier and X. Pages, *200N Newton Bipropellant Thruster Development*, presented at the Second European Spacecraft Propulsion Conference, 1997.

[3] J. C. Sisco, B. L. Austin, J. S. Mok, and W. E. Anderson, Autoignition of Kerosene by Decomposed Hydrogen Peroxide in a Dump Combustor Configuration, *American Institute of Aeronautics and Astronautics*, vol. **21**, pp. 450-462, 2003.

[4] J. C. Sisco, B. L. Austin, J. S. Mok, and W. E. Anderson, Ignition Studies of Hydrogen Peroxide and Kerosene Fuel, *American Institute of Aeronautics and Astronautics*, pp. 2003-0831, 2003.

[5] K. D. Patel, M. S. Bartsch, M. H. McCrink, J. S. Olsen, B. P. Mosier, and R. W. Crocker, Electrokinetic pumping of liquid propellants for small satellite microthruster applications, *Sensors and Actuators B: Chemical*, vol. **132**, pp. 461-470, 2008.

[6] K. Anflo and R. Möllerberg, Flight demonstration of new thruster and green propellant technology on the PRISMA satellite, *Acta Astronautica*, vol. **65**, pp. 1238-1249, 2009.

[7] R. Amri and T. Rezoug, Numerical study of liquid propellants combustion for space applications, *Acta Astronautica*, vol. **69**, pp. 485-498, 2011.

[8] A. S. Gohardani, J. Stanojev, A. Demairé, K. Anflo, M. Persson, N. Wingborg, and C. Nilsson, Green space propulsion: Opportunities and prospects, *Progress in Aerospace Sciences*, vol. **71**, pp. 128-149, 2014.

[9] T. Zhang, G. Li, Y. Yu, Z. Sun, M. Wang, and J. Chen, Numerical simulation of ammonium dinitramide (ADN)-based non-toxic aerospace propellant decomposition and combustion in a monopropellant thruster, *Energy Conversion and Management*, vol. **87**, pp. 965-974, 2014.

[10] R. Amrousse, T. Katsumi, N. Itouyama, N. Azuma, H. Kagawa, K. Hatai, H. Ikeda, and K. Hori, New HAN-based mixtures for reaction control system and low toxic spacecraft propulsion subsystem: Thermal decomposition and possible thruster applications, *Combustion and Flame*, vol. **162**, pp. 2686-2692, 2015.

[11] H. Kang, D. Jang, and S. Kwon, Demonstration of 500 N scale bipropellant thruster using non-toxic hypergolic fuel and hydrogen peroxide, *Aerospace Science and Technology*, vol. **49**, pp. 209-214, 2016.

[12] L. Jing, X. You, J. Huo, M. Zhu, and Z. Yao, Experimental and numerical studies of ammonium dinitramide based liquid propellant combustion in space thruster, *Aerospace Science and Technology*, vol. **69**, pp. 161-170, 2017.

[13] T. Shindo, A. Wada, H. Maeda, H. Watanabe, and H. Takegahara, Performance of a green propellant thruster with discharge plasma, *Acta Astronautica*, vol. **131**, pp. 92-95, 2017.

[14] P. Kumar, An overview on properties, thermal decomposition, and combustion behavior of ADN and ADN based solid propellants, *Defence Technology*, vol. **14**, pp. 661-673, 2018.

[15] M. Wilhelm, M. Negri, H. Ciezki, and S. Schlechtriem, Preliminary tests on thermal ignition of ADN-based liquid monopropellants, *Acta Astronautica*, vol. **158**, pp. 388-396, 2019.

[16] L. P. Bayvel, Z. Orzechowski, *Liquid Atomization*, Taylor & Francis: Washington, DC, USA, ISBN 9780891169598, 1993.

[17] D. K. Huzel and D. H. Huang, Modern Engineering for Design of Liquid-Propellant Rocket Engines, *Progress in Astronautics and Aeronautics*, Volume *147*, ed. Seebass, A. R., American Institute of Aeronautics and Astronautics, Washington, DC, 1992.