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Performance Characteristics of the Hundreds-Newton Class Thruster Using Gaseous Methane and Liquid Oxygen as Propellants

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Abstract
A ground hot-firing test was conducted to examine the performance characteristics of gaseous methane and liquid oxygen bipropellant thruster. Experimental setup is depicted by a piping and instrumentation diagram as well as a perspective view of test apparatus. Ground firing conditions are described in detail, first. As results, effects of combustion chamber pressure on thrust performance are clarified. Meanwhile, varying aspect ratio of the combustion chamber does not give any remarkable influence on thrust performance. Strategy and test results for film cooling are mentioned, additionally.

1. Introduction

A new space era has come up: Rocket stages are returned, retrieved, refurbished, and relaunched; space tourism seems to be realized very soon from commercial sector; recently NASA has admitted the commercial access to ISS along with commercial crew program (CCP); and so-called Lunar Gateway for the exploration of other planets is supposed to be 'opened' in years. A phrase, space economy, is read on the News. Nowadays, when something starts being mentioned, it becomes a reality in this globalized planet, more rapidly than bit-by-bit.

A key for the access to space has been in the transportation capability. The new space era necessitates an additional efficiency and sustainability in the space transportation. This paradigm is directly linked to the energy resource or type of propulsion.

For the human roundtrip exploration to near planets, there are a few candidates among the many propulsion technologies developed up to date. Electromagnetic propulsion that has appreciable vacuum thrust-level and high efficiency, for instance VASIMR, is still under way for its maturity; nuclear propulsion is techno-ideologically out of concern. On the other hand, some propulsion-leading groups have been enchanted by methane among the hydrocarbon propellants from the aspects of its relatively-high specific impulse, echo-friendliness, cost-effectiveness, consistent physical properties originating from single molecule, and a higher coking margin especially favourable to engine reusability. In-situ resource utilization (ISRU) may be another decisive motivation for the choice of methane-fueled rocket engines in prolonged exploration because of the abundant prevalence of methane in the so-called Jovian planets and their satellites.

There have been many studies for methane-fueled engines up to now. Shuichi et al. [1] carried out a hot-fire test of methane rocket engine under high altitude condition obtaining a high efficiency of characteristic velocity. Craig et al. [2] also presented the result for the development testing of LOx-Methane engine emphasizing an excellent combustion stability of LOx-Methane compared to LOx-Ethanol. Johannes and Oskar [3, 4] studied a flame stabilization and effect of recess in coaxial injector of high pressure LOx-Methane rocket engine. Delphine et al. [5] compared the functioning of shear and swirl-coaxial injector in LOx-GCH_4 rocket combustion using light-scattering diagnostics. Christopher et al. [6] studied LOx-LCH_4 engine utilizing coaxial injector. Hot-fire testing of 100 lbf LOx-LCH_4 reaction control engine at various altitude conditions were presented by Marshall et al. [7].

Performance characteristics for GCH_4-LOx thruster is primarily examined in the following sections. Although this study has been conducted as a pathway to the design and development of LCH_4-LOx thruster, the gas-liquid combination may simulate an injection condition of the expander cycle or fuel-rich staged cycle (FRSC) of liquid rocket engine (LRE). Experimental setup will be depicted first.

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2. Experimental Setup

Thrust chamber assembly (TCA) consists of an upper chamber, a liquid oxygen manifold, a gaseous methane manifold, a film-cooling skirt, a combustion chamber, and a supersonic nozzle, etc. Assembly of the two propellant manifolds forms a swirl-coaxial injector. A torch ignitor using gaseous methane and gaseous oxygen is also incorporated into the injector assembly. Fig. 1 shows P&ID (Piping and Instrumentation Diagram) of test facility and a perspective view of test apparatus along with a rendering image of TMR (thrust measurement rig) assembly. Setup of the strict calibration steps for the measurement of thrust and propellant mass flow is essential to the precise and accurate performance evaluation of thrusters.

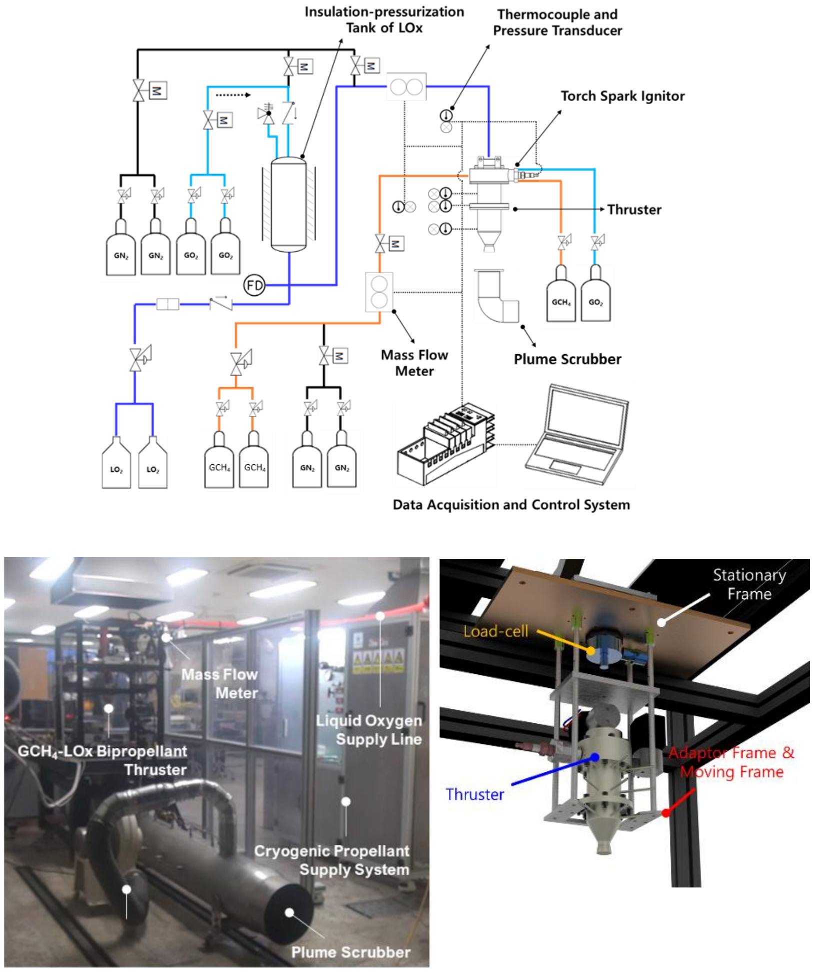

Fig. 1: Experimental setup

3. Results and Discussion

Ideal target performance of the 200 N-class thruster employing gaseous methane (GCH_4) and liquid oxygen (LOx) as bipropellants is summarized in Table 1 with its design baseline and operation condition. The fuel-rich O/F ratio is derived from the calculation of theoretical rocket performance utilizing NASA's CEA and also reflects the feature of film cooling using fuel against overheating of thrust chamber wall. However, when the combination GCH_4 and LOx is employed, as is the situation to be presented in this paper, the film cooling using the cryogenic oxygen is a matter of course and thus an oxygen-rich injection is inevitable. On the ground firing, the practical parameters are varied except for the expansion ratio of 3.4 in the cut-off bell-shaped nozzle and the characteristic length of 1.35 m. The expansion ratio even lower than 50 of vacuum-nozzle was chosen to get away from any possibility for the shockwave and flow separation which may occur inside nozzle-diverging section under the atmospheric ambient condition.

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Table 1 : Target performance of 200 N-class GCH_4/LOx thruster

Parameter	Specification
Vacuum thrust, F_{vac}	200 N
Specific impulse, $I_{sp,vac}$	371.5 s
Total propellant flow-rate, $\dot{m}_{prop}$	54.9 g/s
(O/F)mass ratio ($(O/F)_{stoich}$=4.0)	3.0
Nozzle expansion ratio, A_e/A_t	50
Chamber pressure, P_c	1.79 MPa (260 psi)
Characteristics length, L^*	1.35 m

Three (3) aspect ratios for the combustion chamber configuration, i.e., 1.5, 1.8, and 2.1 are derived from the fixed characteristic length which, irrespective of the aspect ratio, results in the same chamber volume with a fixed throat area. Nominally, LOx is supplied with 220, 255, and 320 psia, respectively. The LOx flow undergoes pressure drop of about 20 to 50 psia due to flow control components ahead of injector and does additional drop of 40 to 80 psia through a swirled injection to combustion chamber. These overall pressure drops of LOx flow determine its mass flow rate resulting in 38 to 50 g/s. In the present study, mass flow rate of GCH_4 was fixed at 10.5 g/s, therefore O/F mixture ratio is entirely determined by the mass flow rate of LOx.

A typical test result is depicted in Fig. 2. This case comes under Test No. A17 in Table 2 to appear soon. It is found in the figure that (1) the injection pressure of LOx reaches 229.9 psia of steady state pressure right after the ignition, resulting in 43.49 g/s of mass flow rate, (2) about 200 psia of GCH_4 injection pressure causes 10.27 g/s of mass flow rate, (3) the mixing and combustion of GCH_4 and LOx build up 168.9 psia of combustion chamber pressure, and (4) average thrust of 99.3 N are produced, resultantly. Both the rise-up and tail-off of thrust are less than 500 ms. Flow-rate curves reveal fairly slow response of Coriolis mass flowmeter: this tardy characteristic may not be adequate to sensing the mass flow rate in pulse-mode operation of thruster. The measured thrust, 99.3 N is 24% lower than the ideal thrust that is 130.9 N in this test case of oxygen-rich (O/F mass ratio = 4.23) and reduced chamber pressure: it seems that this poor efficiency originates from a poor combustion under too low chamber pressure.

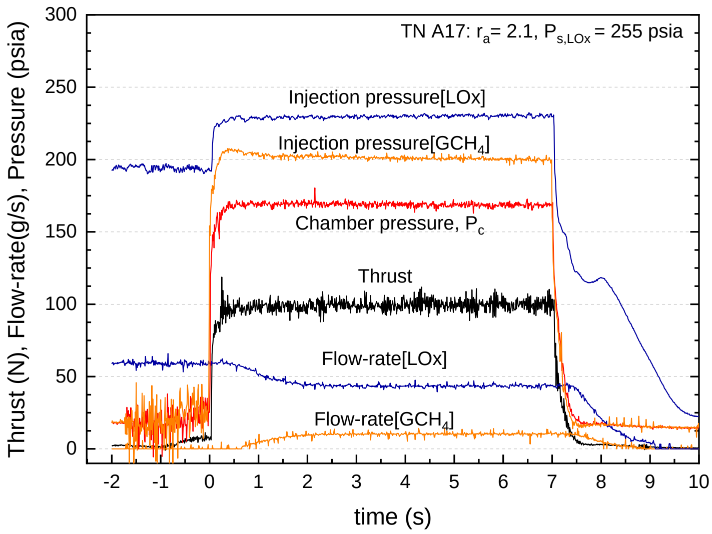

Fig. 2: Test result of case A17

Table 2 lists up the test cases and performance results according to varying aspect ratio and chamber pressure. The performance data are pictorially presented in the following. It should be noted that a relatively-low thrust and characteristic velocity of Test No. A18 were caused by the leakage at a sensing port on combustion chamber.

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Table 2: Ground hot-firing test condition

		Pressure (psia)				Flow-rate (g/s)					
Test No.	Aspect Ratio (r_a=L_c/D_c)	LOx Supply $P_{s,LOx}$	LOx Injection $P_{i,LOx}$	GCH_4 Injection P_{i,CH_4}	Chamber P_c	LOx $\dot{m}_{LOx}$	GCH_4 $\dot{m}_{CH_4}$	$(O/F)_{mass}$	Thrust (N)	$I_{sp,vac}$ (s)	C^* (m/s)
A05	1.8	220	198.2	191.2	156.2	39.34	10.68	3.68	89.2	181.78	1494.17
A07	1.8	255	229.0	202.8	170.2	43.70	10.52	4.15	99.9	187.76	1501.98
A08	1.8	320	280.6	225.1	199.1	52.33	10.50	4.98	120.8	196.05	1516.24
A16	2.1	220	198.7	185.9	151.7	39.43	10.27	3.84	88.0	180.49	1460.47
A17	2.1	255	229.9	200.5	168.9	43.49	10.27	4.23	99.3	188.33	1503.52
A18	2.1	320	223.4	217.9	190.3	54.61	10.15	5.38	114.8	180.72	1405.77
A21	1.5	220	200.0	188.0	154.7	39.61	10.14	3.91	89.0	182.36	1487.57
A25	1.5	255	226.5	202.6	170.4	44.05	10.42	4.23	101.8	190.46	1496.49
A27	1.5	320	279.0	226.9	198.4	51.70	10.65	4.85	120.6	197.22	1522.77

Figure 3 compares the thrust performance along with the mass-flow rate of GCH_4 and LOx according to varying chamber pressure. The increase of LOx flow rate at a fixed GCH_4 flow rate directly affects the increase of chamber pressure resulting in the thrust augmentation.

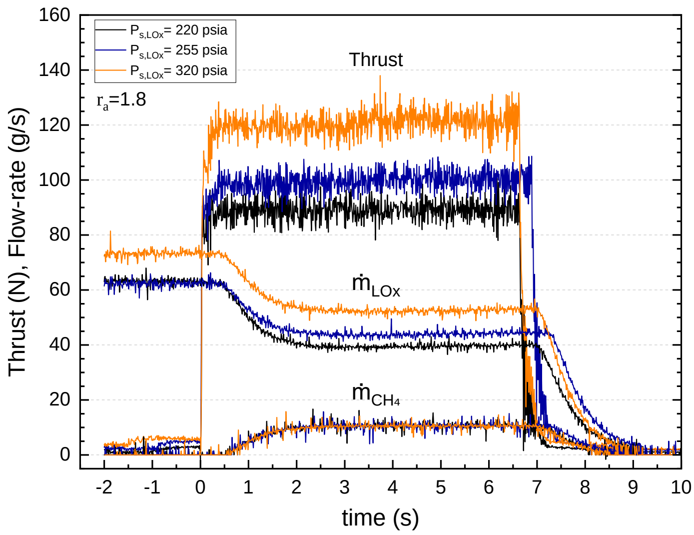

Fig. 3: Variation of thrust and propellant flow rate according to oxidizer supply pressure

As can be found in Fig. 4, it is noticeable that chamber aspect ratio (L_c/D_c) is not that influential in thrust performance at the similar chamber pressures: this might be caused by the fixed chamber volume originating from the fixed characteristic length as mentioned earlier, or by mixing characteristics of the gaseous fuel and liquid oxidizer which have a severe difference of mass diffusivity between the two. More scrutiny needs to be made for this indefiniteness.

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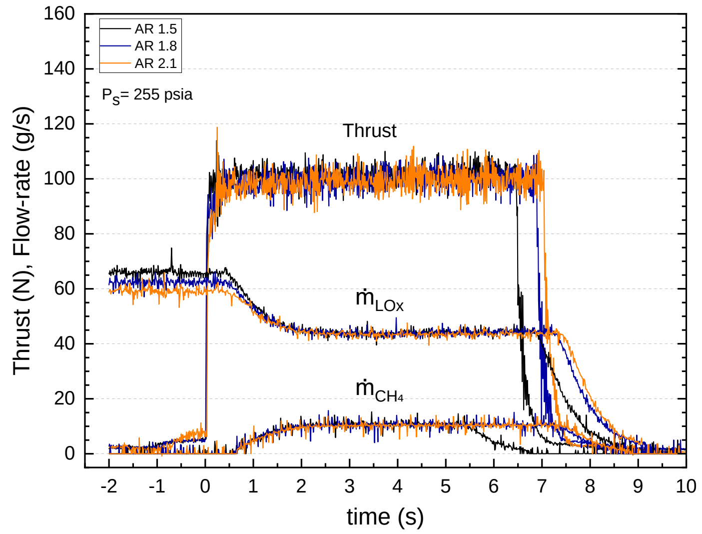

Fig. 4: Variation of thrust and propellant flow rate according to aspect ratio

Specific impulse and its efficiency based on theoretical performance for each test case of Table 2 are again represented along with characteristic velocity in Fig. 5. The figure evokes the thruster performance strongly-governed by chamber pressures and poorly-relevant to aspect ratios. An unexpected performance of Test No. A18 is attributed to the leak problem, as noted before. It is certainly inferred from the performance results that efficiency of the thruster with current design configuration can be approached to its ideal value by more chamber-pressurization.

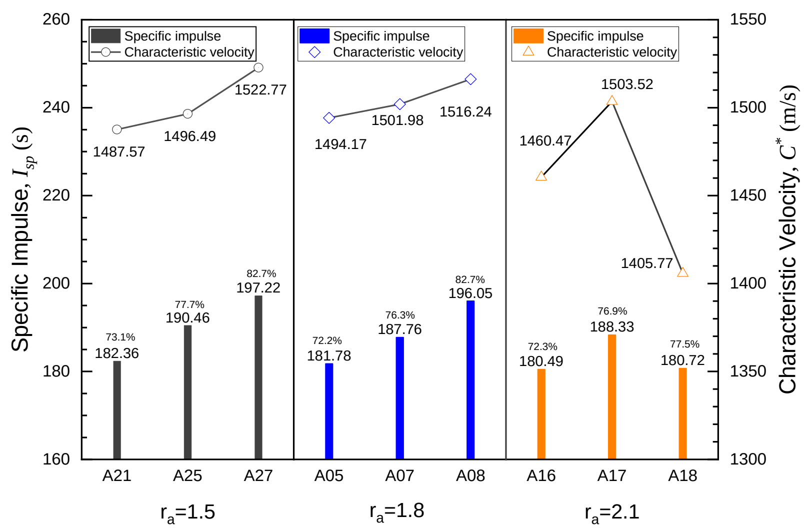

Fig. 5: Characteristic velocity and specific impulse with its efficiency

A higher pressure of combustion chamber will result in higher chemical reaction rate and higher product-gas temperature which necessitates a cooling of thrust chamber wall for the long-term or repeated operation of bipropellant thruster. Film cooling is the only way to cool down the wall for these small thrusters. As a first trial, a skirt which is to generate a coolant film along the chamber wall was designed, fabricated and exquisitely installed on around injector face. A coolant channel supplying the cryogenic oxygen up to the inlet between skirt and chamber wall was also incorporated onto the injector assembly. Thruster performance with film cooling is compared to the case without the cooling in Fig. 6 and summarized in Table 3. At a glance, it is caught from the results that the flow rate of LOx in film-cooled operation is higher than in the case without cooling, producing a slight increase of the thrust caused by additional pressurization and a decrease of the specific impulse and characteristic velocity caused by the more propellant injection. Additional test and performance evaluation are needed for the configuration optimization of film cooling.

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Table 3: Test result of performance with film cooling

Parameter	w/o Film cooling A07	w/ Film cooling B03
Thrust	99.9 N	106.5 N
$I_{sp,exp}$	187.8	180.3
$I_{sp,ideal}$	251.7	237.6
C^*	1502.0	1406.3

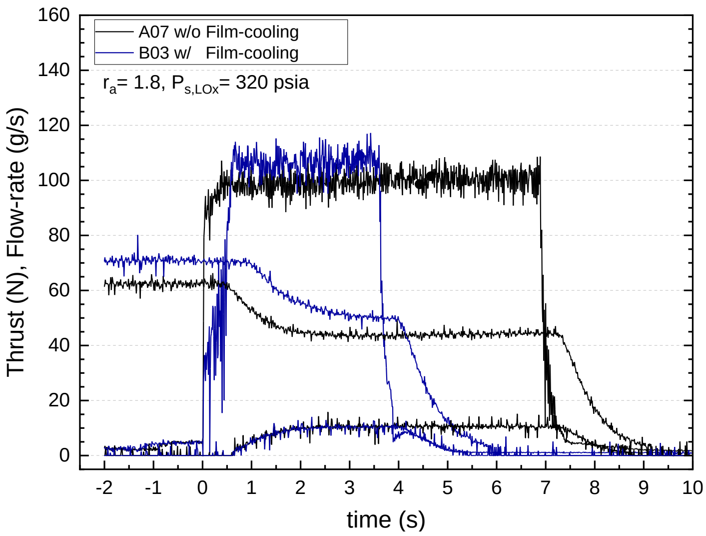

Fig. 6: Effects of film cooling on thrust performance

4. Conclusion

Performance characteristics of the thruster employing GCH_4 and LOx as bipropellants were scrutinized. Experimental setup was depicted by P&ID (Piping and Instrumentation Diagram) of test facility and a perspective view of test apparatus along with a rendering image of TMR (thrust measurement rig) assembly. Pressure and aspect ratio of combustion chamber were parameterized in the ground performance test. It was found that the increase of LOx flow rate at a fixed GCH_4 flow rate directly affected the increase of chamber pressure resulting in the thrust augmentation. On the other hand, chamber aspect ratio (L_c/D_c) was not that influential in thrust performance at the similar chamber pressures. It was inferred from the performance results that efficiency of the thruster under consideration would be enhanced by more chamber-pressurization. Flow rate of LOx in film-cooled operation was higher than in the case without cooling, producing a slight increase of the thrust caused by additional pressurization and a decrease of the specific impulse caused by the more propellant injection.

Acknowledgements

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