Experimental Study on 100 N-scale C_2H_4/GO_2 Small RDE

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1 Introduction

Rotating Detonation Engine (RDE) is a combustor where fuel and oxidizer are consumed by self-sustained detonation waves propagating in the circumferential direction, generating the thrust in the axial direction. The RDE is considered as a pressure gain combustion (PGC) device that has an advantage in highly thermal efficiency by the addition compression by the detonation waves. A large number of the experimental studies have been carried during the last decades to realize the RDE and to understand the underlying physics [1~6]. Regardless, further efforts are still needed to stabilize the operational characteristics and to improve the performance [7]. Although the performance gain is theoretically possible by the effect of constant volume combustion (CVC), only few cases have presented the experimental performance [2,8] and the how much performance or efficiency can be improved than the conventional rocket engines and combustors [7,8].

In the present study, a small-scale rocket-type RDE is constructed as way to pursue of performance advantages over the conventional rocket engines. The experimental characteristics of the RDE is investigated and thrust performance is measured. The experiments are carried out over a range of equivalence ratio and total mass flow rate of up to 180 g/s. The measured thrust performances are compared with those of the rocket engine calculated by NASA CEA code [9].

2 Experimental Setup

Figure 1 is a schematic diagram of the designed RDE. Figure 2 is the schematic of propellant supply and control system, and the view of the experimental setup. The uncooled RDE design is fabricated with stainless steel for short operation time. Gaseous ethylene (C_2H_4) and oxygen (O_2) are used as propellants. The dimensions of the annular combustion channel are 50.0 mm I.D., 4.5 mm width, and 75.0 mm length. The fuel and oxidizer are injected from the plenums into the combustion channel through 0.3 mm and 0.5 mm slot, respectively. A conical center body of 30 degree angle is installed at the RDE exit for the performance comparison. To initiate the detonation in the combustion channel, a small-scale pre-detonator (micro PDE) of about 4.2 mm I.D. and 150.0 mm length is used [10]. It uses the same fuel and oxidizer as used by the RDE. Shchelkin spiral is not applied since the deflagration to detonation transition (DDT) distance is

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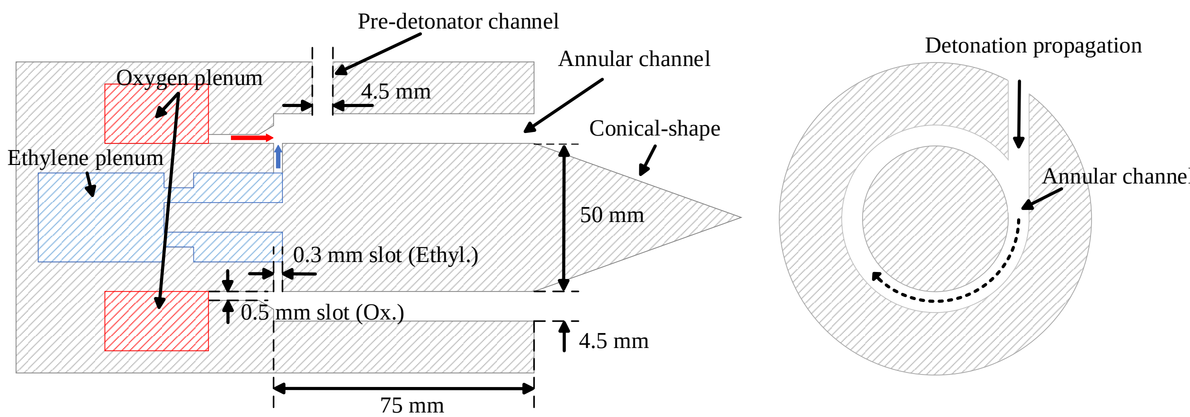
sufficient for the present oxygen-based mixture. The pre-detonator is installed in the tangential direction into the combustor to minimize the energy loss.

The total mass flow rates are calculated by measuring the weight of the 3.4 L gas cylinder before and after the cold flow test, and then calibrated with the feed pressure measured by the low frequency pressure sensors (Keller 23SY). These sensors are installed in each plenum, combustion channel, and pre-detonator for measuring the stagnation pressure. High frequency pressure sensors (PCB 113B) are used to calculate the detonation speed and to investigate the characteristics of the detonation wave. All of these pressure sensors are mounted in the recessed flat. The high frequency pressure sensors are mounted with a 4.0 mm recess length. A load cell (CAS SBA) for the thrust measurement is physically tightened the load cell and RDE. The measured thrust is compared with the theoretical rocket performance calculated by NASA CEA code with isentropic condition.



(a) (b)

Fugure 1. Schematic of the RDE: (a) sectional side view and (b) sectional front view

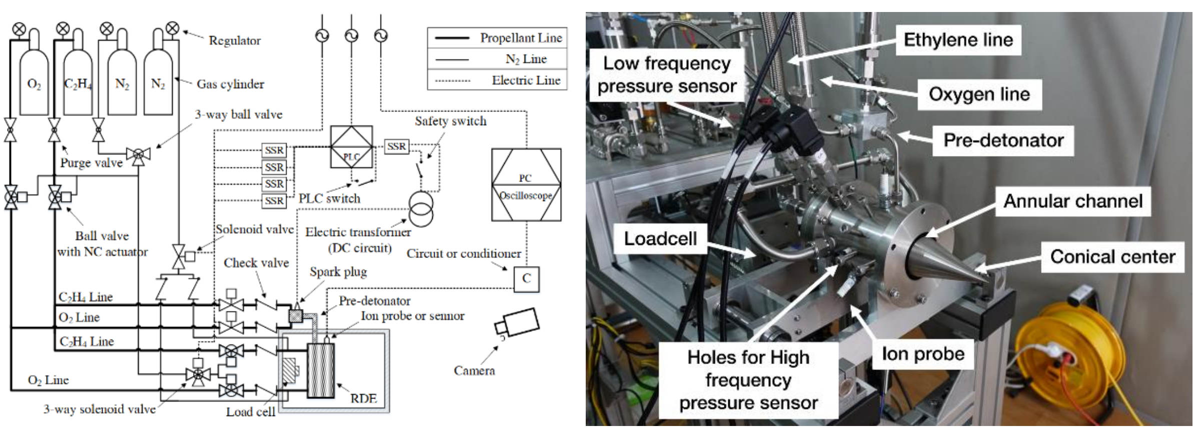
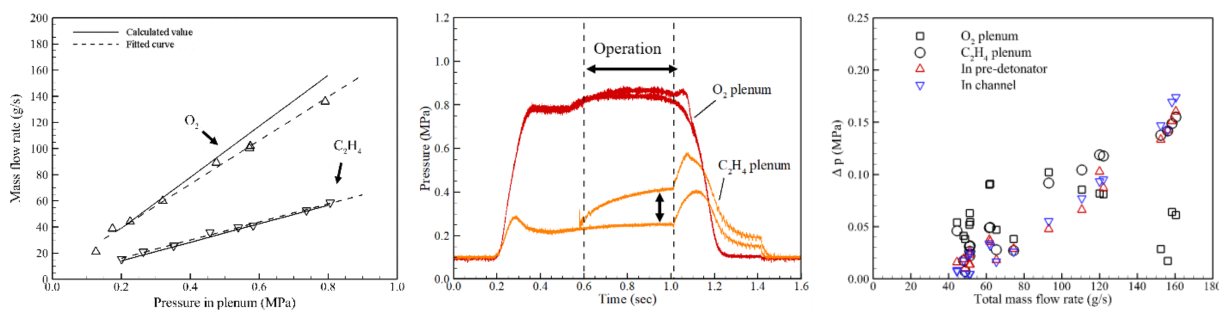


Figure 2. Schematic of propellant supply and control system (left) and a picture of the experimental setup (right)

3 Results

Figure 3a shows the measured and calculated mass flow rates within the measurement range that is 0.2 ~ 0.8 MPa. The difference from the calculated values showed 0.3 ~ 16.5 g/s (0.8 ~ 10.0%) for the oxygen, and 1.2 ~ 2.0 g/s (2.1 ~ 14.0 g/s) for the ethylene. Figure 3b shows the pressure variation in the plenum for cold (non-reacting) and hot (burning) flow test cases. The RDE operates between 0.6 and 1.0 sec duration (i.e., combustor operation is about 0.4 sec). The pressure rise in this section is caused by the pressure rise in the combustion channel. The sudden pressure rise after 1.0 sec is due to the purge gas (N_2) supply. Figure 3c shows the stagnation pressure in the combustion channel, pre-detonator, and each plenum with respect to the mass flow rate. These pressure values are the maximum values during the operating time. It seems that the choking occurs at the RDE exit from the mass flow rate of about 120 g/s or more. The maximum pressure rise of 0.17 MPa occurred under the experimental conditions.
Figure 4 shows the pressure record of the rotating detonation in combustion channel. From this result, it was confirmed that the detonation wave propagates along the combustion channel wall, but it seems that the unstable detonation wave propagates under the low mass flow rate condition. The dynamic pressure measurements under the high mass flow rate conditions were temporarily interrupted due to the ablation phenomenon as shown in the Figure 5. This ablation occurred the hole of the pre-detonator and sensors of the combustion channel. It appeared from the choking condition ($\dot{m} \geq 120$ g/s) and lean condition of the combustor.



(a) (b) (c)

Figure 3. Measured operational flow parameters: (a) mass flow rate, $\dot{m}$, (b) pressure variation in plenum and (c) pressure differences at flow locations.

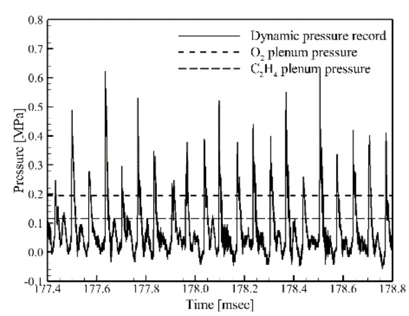


Figure 4. Pressure record of the rotating detonation in combustion channel at $\dot{m}$ = 48.6 g/s, Φ = 0.91

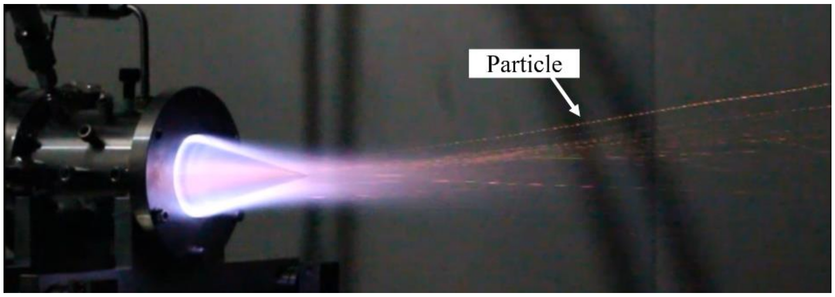
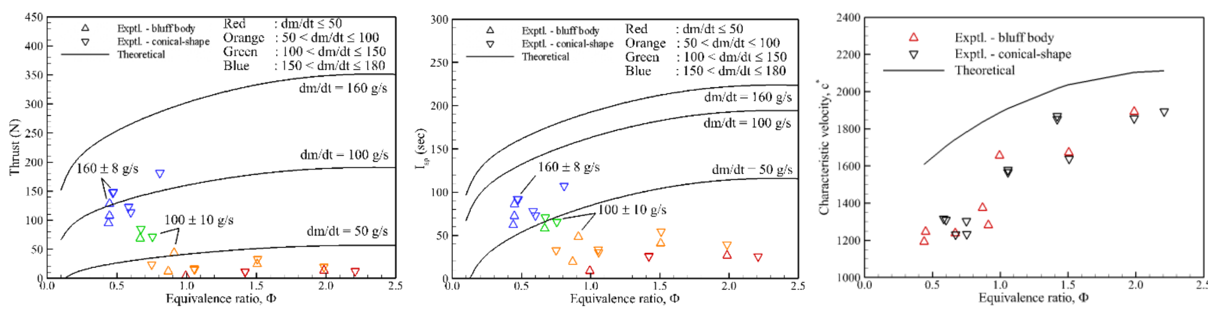


Figure 5. Direct picture of RDE plume on ablation phenomenon at $\dot{m}$ = 163.0 g/s, Φ = 0.47 (3,840x2,160 resolution using Sony RX-100M5 camera)

Figure 6a and 6b shows the thrust and the specific impulse performance with respect to the equivalence ratio for several mass flow rate conditions. Th theoretical values are the performance of the ideal rocket engine. The measured values show a loss of 40% or more than the theoretical value. The loss is considered to be a caused by the absence of a C-D nozzle or a plug nozzle at the RDE exit. In this work, maximum thrust obtained is 181.1 N (specific impulse of 107.1 sec). Figure 6c shows the characteristic velocity (c^*). The characteristic velocity is calculated as follows:

$$c^* = p_c A_t / \dot{m} \tag{1}$$

Where, p_c is pressure in combustion channel, A_t is throat area, and $\dot{m}$ is total mass flow rate. The experimental values of characteristic velocity are 73 ~ 90% (80% in average) of theoretical values.



(a) (b) (c)

Figure 6. Performance parameters with respect to the equivalence ratio, Φ: (a) thrust, F, (b) specific impulse, I_{sp}, and (c) characteristic velocity, c^*

4 Conclusion

A small-scale RDE is constructed and tested to understand the operational and performance characteristics. The measured thrust performance was compared with that of the ideal rocket engine calculated by the NASA CEA code. The results suggest that C-D nozzle or plug nozzle is essential to improve the thrust performance of RDE. This suggestion is also shown in the simple thrust comparison with conical center body and bluff body. The thrust with conical center body was improved by 10 ~ 30% than with bluff body in present experiment range. For more accurate comparison of the RDE and the conventional rocket engine, it would be necessary to conduct an experimental comparison with equivalent shape and operating condition. A technical issue arose during the experiments should be compensated for further experiments. The occurred ablation at around holes appeared to be prominent from the lean condition ($\Phi < 1.0$) and the choking condition ($\dot{m} \geq 120$ g/s) of the combustor. It would be necessary to minimize the inner diameter of the pre-detonator and other things to reduce the ablation.

5 Acknowledgment

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