# Propulsive Performance of Rotating Detonation Engines in $CH_4/O_2$ and $C_2H_4/O_2$ for Flight Experiment

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## 1 Introduction

Detonative propulsion systems are promising candidates to replace deflagration in aerospace propulsion systems because of their high thermal efficiency and short combustor length to complete the combustion.[1] Compared to a pulse detonation engine (PDE), a rotating detonation engine (RDE) uses one or more detonation waves that continuously circle around its annular chamber to generate thrust. The combination of high flame speed, on the order of km/s, with continuous propellant flow rates can result in high thrust density, thrust-to-weight ratio, and volumetric efficiency.[1] In particular, the application of RDE as a spacecraft and rocket main thruster could enable smaller and more powerful propulsion system.

Frolov et al.[2] experimentally proved that the specific impulse of RDE was 6-7% higher than that in continuous combustion mode. Using a 2D simulation model, Schwer et al.[3] varied the pressure ratio of the

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stagnation and back-pressure, between 2.5 and 20, and found that it could affect the specific impulse and detonation wave heights. Kindracki et al.[4] measured the thrust and specific impulse of RDE under 0.5 bar. Frolov et al.[5] conducted large-scale RDE thrust measurement, and found that attaching a nozzle or reducing the air injection area could increase in the number of detonation waves and thrust. Shepherd et al.[6] constructed an analytical model for the thrust of a RDE based on control volume analysis, which could be used in the traditional quasi-one dimensional rocket motor methodology to estimate RDE's performance.

Considering the design of thrusters, the most important parameter should be specific impulse, $I_{sp}$. The specific impulse can be determined by the combination of propellants, state of combustion (combustion chamber stagnation temperature, $T_{o,c}$, specific heat ratio, $\gamma$, and gas constant, $R$), and the pressure ratio between combustion chamber stagnation pressure, $p_{o,c}$, and back-pressure, $p_b$. Ideal specific impulse under proper expansion, $I_{sp,opt}$, can be calculated as

$$I_{sp,\mathrm{opt}} = \frac{1}{g}\sqrt{\frac{2\gamma R T_{o,c}}{\gamma-1}\left\{1-\left(\frac{p_b}{p_{o,c}}\right)^{\frac{\gamma-1}{\gamma}}\right\}} \tag{1}$$

Considering the fundamental isentropic nozzle theory, combustion chamber stagnation pressure should be proportional to throat mass flux. Goto et al.[7] revealed that how RDE's combustor pressure ,$p_c$, almost proportinal to equivalent throat mass flux (i.e. minimum cross flow area), using gaseous ethylene / gasous oxygen. This implied that when throat and operating mass flow rates was determined, RDE chamber pressure could be determined, and expected specific impulse could be estimated for each back pressure.

However, it was not unclear whether we could apply that result for various fuels. Instead of ethylene, methane is a promising hydrocarbon fuel candidate for next-generation space chemical propulsion due to its good performance and good fuel storability. Our research group proposed a first flight experiment of detonation engine systems in space in 2020. From the point of view of making practical detonative propulsion system, broad design space exploring propulsive performance, detonatability of propellant and system configuration should be evaluated.

In this study, we conducted thrust measurements RDE of (1) methane / gas-oxygen and (2) ethylene / gas-oxygen with some throat geometries, including a throatless RDE, inside a 30.1 m$^3$ vacuum chamber to simulate various back-pressures.

## 2 Experimental Apparatus

In this study, we used several different RDE geometries, including throat, injector and combustor length. We defined the $z$-axis as the bottom of the combustor toward the downstream. We also defined the $r$-axis from the center of the RDE toward the radius. For throatless RDE, we defined equivalent throat area, $A_{th}$, as the detonation channel area. When the RDE had a geometric throat, $A_{th}$ was equal to the geometric throat area. The contraction area ratio, $\varepsilon_c$, was defined by the ratio of $A_{th}$ and detonation channel area, $A_{ch}$, as

$$\varepsilon_c = \frac{A_{ch}}{A_{th}} \tag{2}$$

Thrust measurement of the RDE-1 ($A_{ch}$ = 640.4 mm$^2$), shown in Figure 1, was conducted. The inner radius was $r_i$ = 30.25 mm, channel width was $\Delta$ = 3.2 mm, and combustion chamber length was $L_c$ = 48 or 70 mm. The RDE had a 30$^o$ conical plug, 72 or 120 fuel-injection holes 0.5 mm in diameter, and a 0.3-mm wide oxidizer injection slot. Pressure ports were located at the bottom of the combustor (0.5-mm diameter).

The RDE-2 ($A_{ch}$ = 1759 mm$^2$), had an inner radius of $r_i$ = 31 mm, and a channel width of $\Delta$ = 8 mm. We tested four throat configurations of $\varepsilon_c$ = 1, 1.5, 2.5, 8 for this study. For $\varepsilon_c$ = 1.5, the width of the asymmetric throat was 5.5 mm, and the outer exit width was 29.6 mm ($A_e/A_{th}$ = 5.7). For $\varepsilon_c$ = 2.5, the width of the symmetric throat was 3.2 mm, and the outer exit width was 8 mm ($A_e/A_{th}$ = 2.5). For $\varepsilon_c$ = 8, the width of the symmetric throat was 1 mm, and the outer exit width was 16 mm ($A_e/A_{th}$ = 16).

We used two types of injector geometries, doublet and triplet, for RDE-2. One injector scheme was doublet-impinging injection, which had 120 sets of 1-mm diameter fuel-injection holes and an oxidizer hole (1 mm diameter). The combustion chamber length $L_c$ was 70 mm, and had a truncated conical plug nozzle. Measuring chamber pressure, pc, was taken through the hole (2 mm) located at the bottom of combustor, z = 0. The other injector scheme was triplet, it had 72 sets of two fuel injection holes (1-mm diameter) and an oxidizer hole (1.4-mm diameter) between fuel holes. It had a combustion chamber length of $L_c$ = 75 mm, and a truncated conical plug nozzle. Measurements of chamber pressure $p_c$, was taken through the hole (2 mm) located at z =5 mm, which was as close to the bottom of combustor as possible since the entire bottom surface of the triplet injector had injection holes.

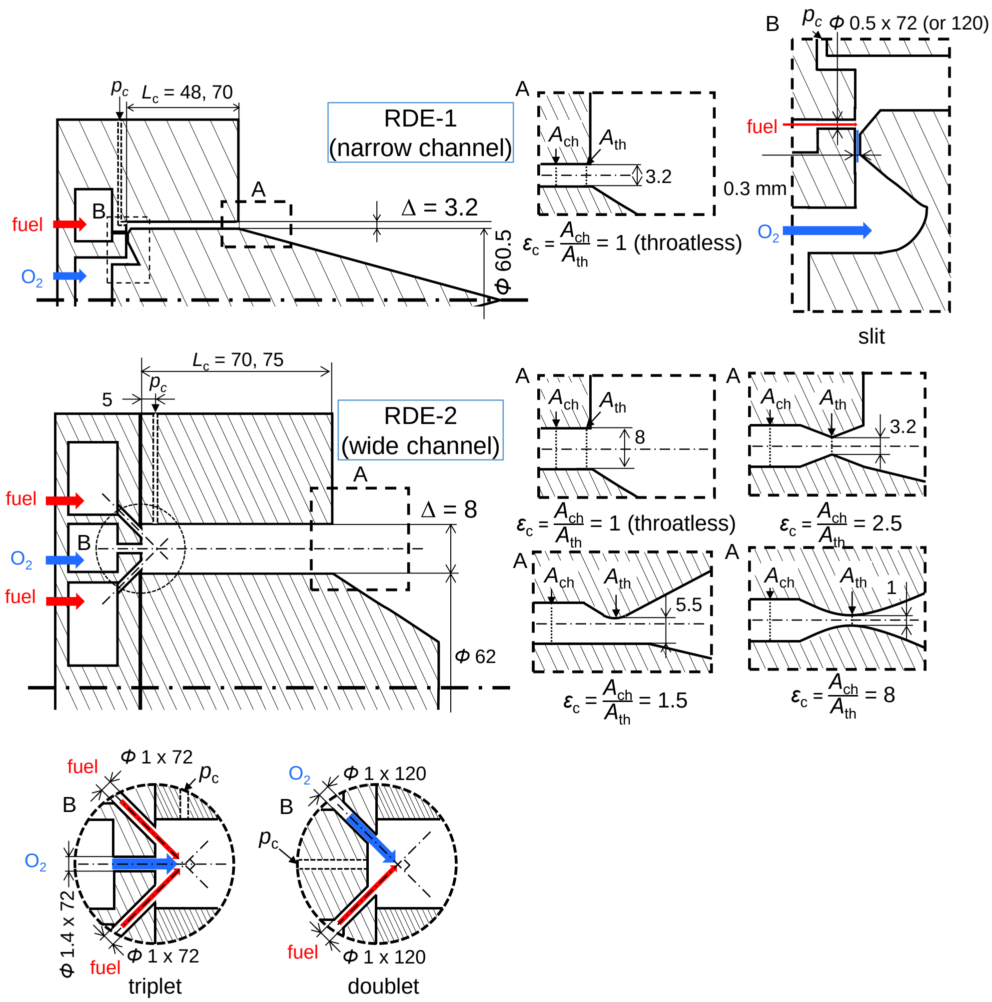


Figure 1. Schematic of annular RDEs, their throat geometry and injector shapes in this study.

Figure 2 presents a schematic view of the thrust measurement stand in a vacuum chamber. In this study, we measured thrust, pressure, and recorded high-speed imaging of the detonation channel from the aft end of the RDE, using a high-speed camera (SA5, Photron). Thrust was measured with a load cell attached to the thrust stand inside the vacuum chamber. The 30.1 $m^3$ volume vacuum chamber was connected to a vacuum pump and allowed simulation of initial back-pressure conditions ranging from sea level to 0.6 kPa.

We used two combination of propellants in this study. One combination is gaseous methane and gaseous oxygen, and the other is gaseous ethylene and gaseous oxygen. Each mass flow rate was controlled and determined by choking orifices upstream of the RDE feeding line.

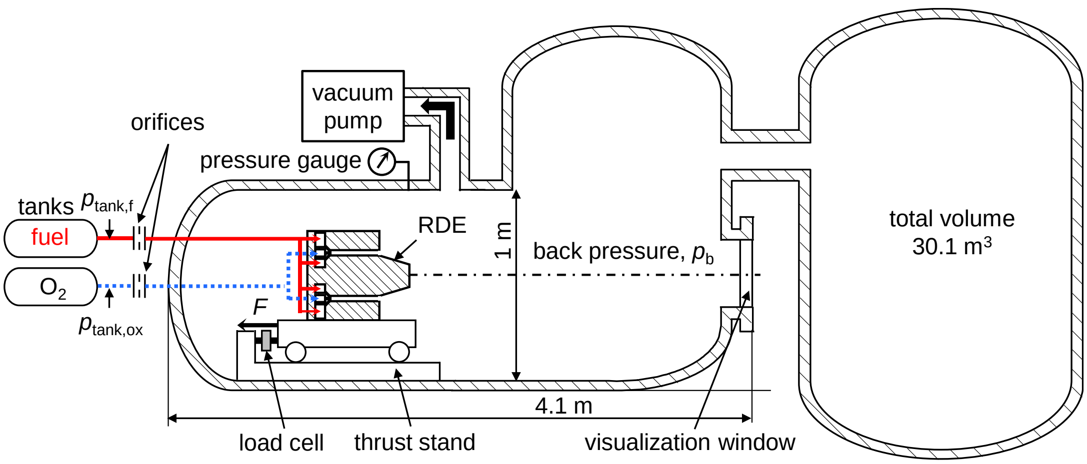


Figure 2. Schematic of experiment to measure RDE thrust.

## 3 Result and Discussion

The trend of time averaging chamber pressure during burn time (less than 1 s), $p_c$, obtained through increasing the throat mass flux is shown in Figure 3 (left). The solid line is a regression line generated by our measured experimental data plots. As expected, chamber pressure was approximately proportional to throat mass flux, even though RDE geometries and propellant combination were different. Especially, the data of throatless RDE with methane also on the same line of throat RDE with methane. This was similar trend of previous study using ethylene[7].

Then we nondimensionalized chamber pressures by the ideal $p_{c,Ref}$ from NASA-CEA[8] using the same mass flow rate and throat area for each condition assuming choking condition at equivalent throat area for each case as shown in Figure 3 (right). Nondimensionalized combustor pressures are similar to $c^*$ efficiency, used in conventional rocket engines. The result implied that $c^*$ efficiency was almost close to 1.

From a qualitative perspective, an increase in the ratio between combustion chamber pressure and back pressure should give a higher specific impulse. Figure 4 (left) shows the relation between specific impulse and pressure ratio, defined as $p_c/p_b$. The solid slope represents the ideal specific impulse curve under correct expansion calculated by NASA-CEA[8] as a reference (calculated initial condition was determined by the average value of all experimental conditions). To assess the true performance of RDEs, we nondimensionalized $I_{sp}$ by the ideal specific impulse, $I_{sp,opt}$, of constant pressure engine operating with the measured mass flow and back-pressure conditions as shown in Figure 4 (right). In Figure 4 (right), normalized specific impulse gradually reached unity as $p_c/p_b$ increased because each nozzle exit conditions approached to the optimum condition. Opimum expansion conditions for each nozzle expansion ratio ($A_e/A_{th}$) can be calculated from folllowing equation ($\gamma$ = 1.1 assumed).

$$\left(\frac{A_e}{A_{th}}\right)_{opt} = \sqrt{\left(\frac{\gamma-1}{2}\right)\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma-1}} \Bigg/ \left\{\left(\frac{p_b}{p_{o,c}}\right)^{\frac{2}{\gamma}} - \left(\frac{p_b}{p_{o,c}}\right)^{\frac{\gamma+1}{\gamma}}\right\}} \tag{3}$$

RDE with methane-oxygen in this study could achieve more than 80% of that of the optimum expansion when they had divergent nozzles for high pressure ratio cases. The deficit of the $I_{sp}$ efficiency might be due to the mismatch of nozzle operating conditions or the different propellant combination or slight decrease in combustor pressure. The thrust measurement at the lower back-pressure should be needed to evaluate the performance in space.

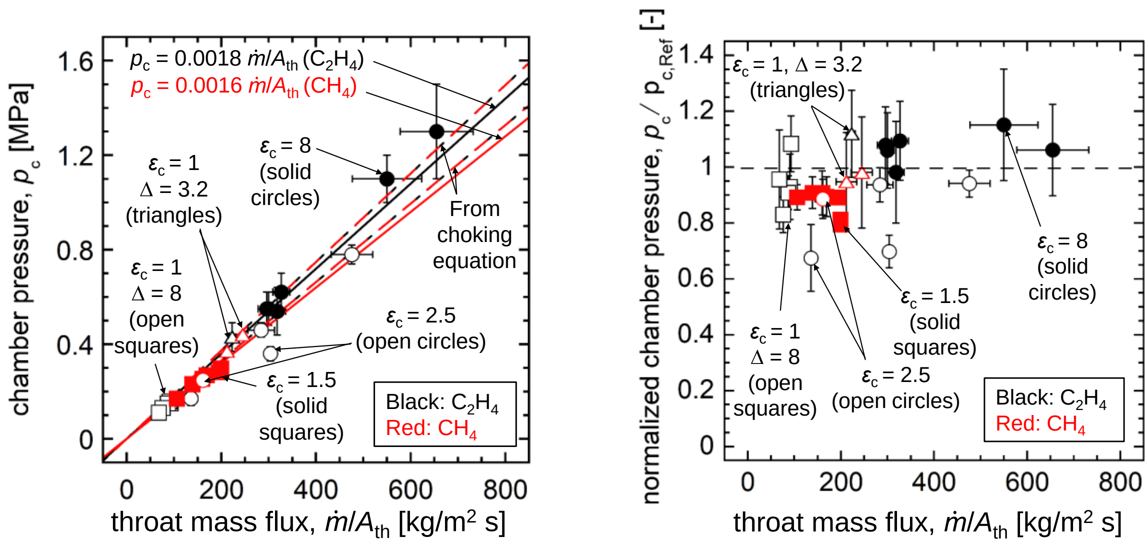


Figure 3. Left: RDE combustor pressure and throat mass flux, right: Normalized RDE combustor pressure and throat mass flux. Ethylene-oxygen data were from Goto et al.[7]

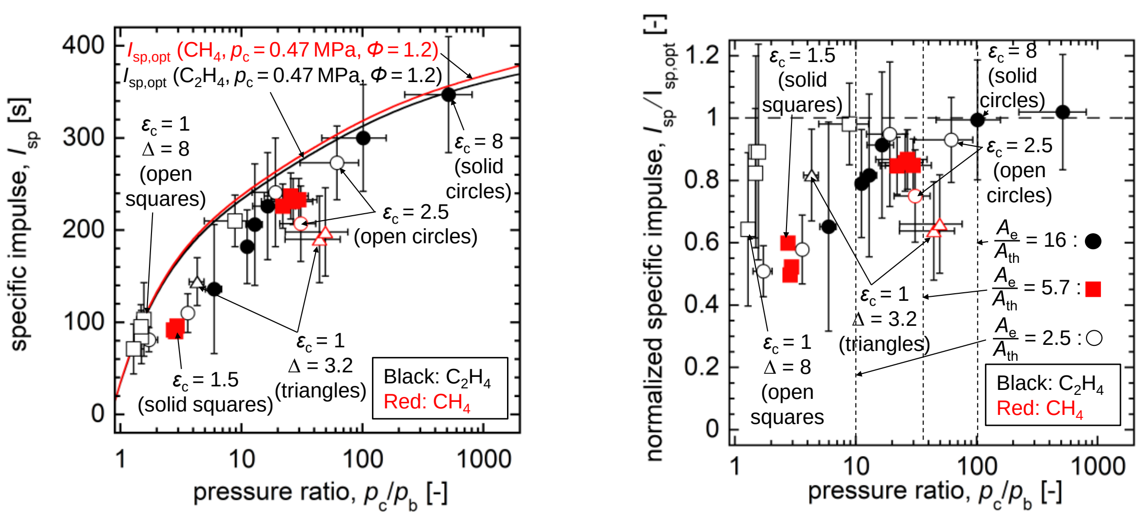


Figure 4. Left: Specific impulse and RDE combustor pressure ratio, right: Normalized specific impulse and RDE combustor pressure ratio. Ethylene-oxygen data were from Goto et al.[7]

# 4 Conclusions

Thrust measurements of RDE of (1) methane / gas-oxygen and (2) ethylene / gas-oxygen with various throat geometries in a vacuum chamber to simulate different back-pressure conditions ranging from 1.1-104 kPa were conducted. For throatless RDE, equivalent throat area was defined as the detonation channel area, and then four nozzle contraction ratios of 1, 1.5, 2.5, and 8 were tested.

We measured that time averaged RDE combustor pressure with methane / oxygen mixture and compared the previous results using ethylene / oxygen mixture. It was revealed that it was almost proportional to the RDE throat mass flux regardless of contraction ratios for methane / oxygen mixture, and this was similar results of previous study using ethylene / oxygen mixture.

Specific impulse of RDE increased along with $p_c/p_b$ and approached optimum expansion conditions for each nozzle expansion conditions for both methane / oxygen and ethylene / oxygen mixture case. In this study, using methane / oxygen, specific impulse could achieved at least 80% of optimum specific impulse.

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