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Thrust Enhancement of Rocket-Deployed Dual-Mode Ramjet Engine

Sangwook Jin[1], Haeseung Jeong[2], Minchan Kwon[3], Juhyun Bae[4], Seokjin Oh[5], Hojin Choi[6], Jaehoon Ryu[7], Suji Lee[8]

Abstract

In order to operate an aircraft across a wide range of altitudes and speeds, a combined cycle is required. This combined cycle comprises various engines, each optimized for specific altitude bands. However, in regions where different engines overlap, thrust may become insufficient. To address the issue of inadequate thrust during relatively low-speed, near-hypersonic flight, a dual-mode ramjet combustor with a rocket has been conceptualized. The combined combustor functions by injecting combustion gases generated by the ejector rocket into the dual-mode ramjet combustor through a strut nozzle. Experimental tests were conducted on the proposed engine to measure the combustor's thrust, which was then compared to thrust values calculated using theoretical formulas, quasi-one-dimensional method and pitot pressure rake at combustor exit.

Keywords: *Dual-mode ramjet, Thrust measurement, Rocket-based combined cycle*

Nomenclature

F – Thrust

p – Pressure

T – Temperature

h – Isolator height

A – Area

v – Velocity

V – Specific volume

Cp – Specific heat capacity

γ – Specific heat ratio

Δ – Difference

Subscripts

t – Throat or total

2 – Isolator inlet

3 – Combustor inlet

4 – Combustor exit

1′ – Rocket combustion chamber

2′ – Nozzle exit

3′ – Atmosphere

TMS – Thrust measurement system

Theoretical – Theoretical method

Q1D – Quasi-one-dimensional method

Rake – Pitot pressure measurement

1. Introduction

Aircraft require engines to obtain propulsion, and the flight envelop is determined by the thrust they can generate. For aircraft flying in the atmosphere, they typically utilize a method of intaking air from

[1] *Agency for Defense Development, Yuseong P.O.Box 35, Daejeon, 34186, Republic of Korea, engine@add.re.kr*
[2] *Agency for Defense Development, Yuseong P.O.Box 35, Daejeon, 34186, Republic of Korea, jeonghs@add.re.kr*
[3] *Agency for Defense Development, Yuseong P.O.Box 35, Daejeon, 34186, Republic of Korea, mckwon@add.re.kr*
[4] *Agency for Defense Development, Yuseong P.O.Box 35, Daejeon, 34186, Republic of Korea, phizbjh@add.re.kr*
[5] *Agency for Defense Development, Yuseong P.O.Box 35, Daejeon, 34186, Republic of Korea, seokjin5@add.re.kr*
[6] *Agency for Defense Development, Yuseong P.O.Box 35, Daejeon, 34186, Republic of Korea, 5genie@add.re.kr*
[7] *Agency for Defense Development, Yuseong P.O.Box 35, Daejeon, 34186, Republic of Korea, ryu0804@add.re.kr*
[8] *Agency for Defense Development, Yuseong P.O.Box 35, Daejeon, 34186, Republic of Korea, sujilee@add.re.kr*

the atmosphere and burning fuel to generate thrust, known as air-breathing propulsion systems. However, when flying in rarefied air regions, such as outer space, thrust is obtained by burning fuel with onboard oxidizers, with rockets being a prime example. For aircraft operating at a wide range of speeds and altitudes, various engines are combined, forming what is known as a combined cycle. Examples are the Turbine Based Combined Cycle (TBCC), which combines a turbojet engine with a dual-mode ramjet engine, and the Rocket Based Combined Cycle (RBCC), which combines a rocket engine with a dual-mode ramjet engine.

Each engine in a combined cycle is responsible for specific flight regimes, allowing for efficient flight capabilities. In TBCC, the gas turbine operates at low speeds while the dual-mode ramjet engine operates at high speeds. In RBCC, the rocket engine handles low-speed and rarefied air regimes, while the dual-mode ramjet engine manages high-speed regimes, although the rocket can operate in almost all regimes. While rockets may not be as efficient as gas turbines in terms of specific impulse (Isp), they can supplement thrust in specific flight regimes when combined appropriately with other engines.

Expanding the flight envelope to include Earth-to-orbit missions necessitates redefining the concept of aircraft, with various proposals such as SSTO (Single-Stage-To-Orbit) and TSTO (Two-Stage-To-Orbit). However, the key to realizing such concepts lies in propulsion systems. Choi et al. [1] proposed a new concept, MJCC (Multiple Jointed Combined Cycle), integrating existing TBCC and RBCC designs. It combines a gas turbine, dual-mode ramjet engine, and rocket, with the gas turbine operating at low speeds, the dual-mode ramjet engine at high speeds, and the rocket handling sparse air regimes and speeds above Mach 6–7. The goal is to enable operations up to an altitude of 60 km and speeds of Mach 10 from the ground.

However, not all engines may generate sufficient thrust in every flight regime. To address this, a combined ejector rocket engine was conceived to augment thrust where it may be insufficient. Flames generated by the rocket are directed through nozzles into the flow path of the dual-mode ramjet engine to produce additional thrust. Research on such ejector rockets, utilizing gaseous oxygen and kerosene, was conducted by Zhuo et al. [2], while Shi et al. [3] observed ignition and flame-holding states in RBCC engines comprising ejector rockets and multiple cavity type of flame holder. While thrust measurements for individual rocket engines are relatively achievable using thrust stands, deriving thrust when combined with dual-mode ramjet engines is challenging, and previous studies have not addressed this. Thus, this study aims to construct a combined ejector rocket for the dual-mode ramjet combustor and calculate the resulting thrust during ejector rocket operation using various methods such as thrust measurement, theoretical calculations, quasi-one-dimensional analysis, and exit pitot pressure measurement. The analysis aims to provide a practical thrust measurement method for directly connected experiments.

2. Experimental Setup and Test Procedure

Fig. 1 and Fig. 2 respectively represent the conceptual diagram and the actual configuration of the combined combustor with ejector rocket application.

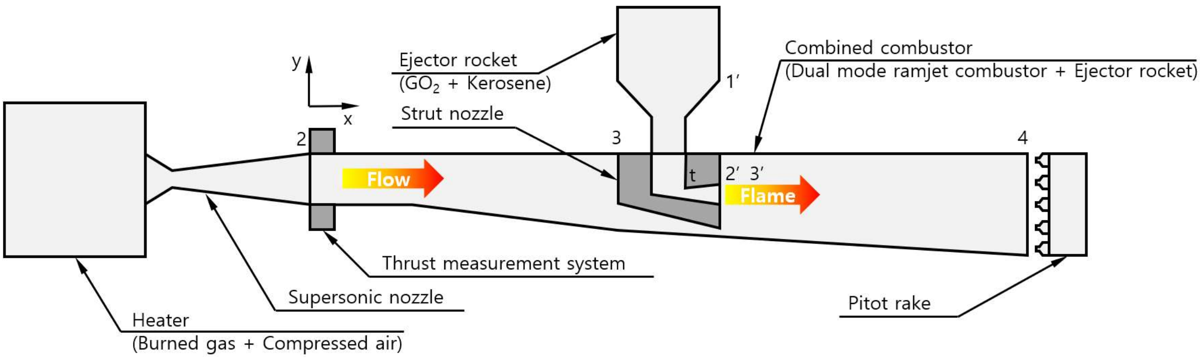

Fig. 1 Schematic diagram of a combined combustor

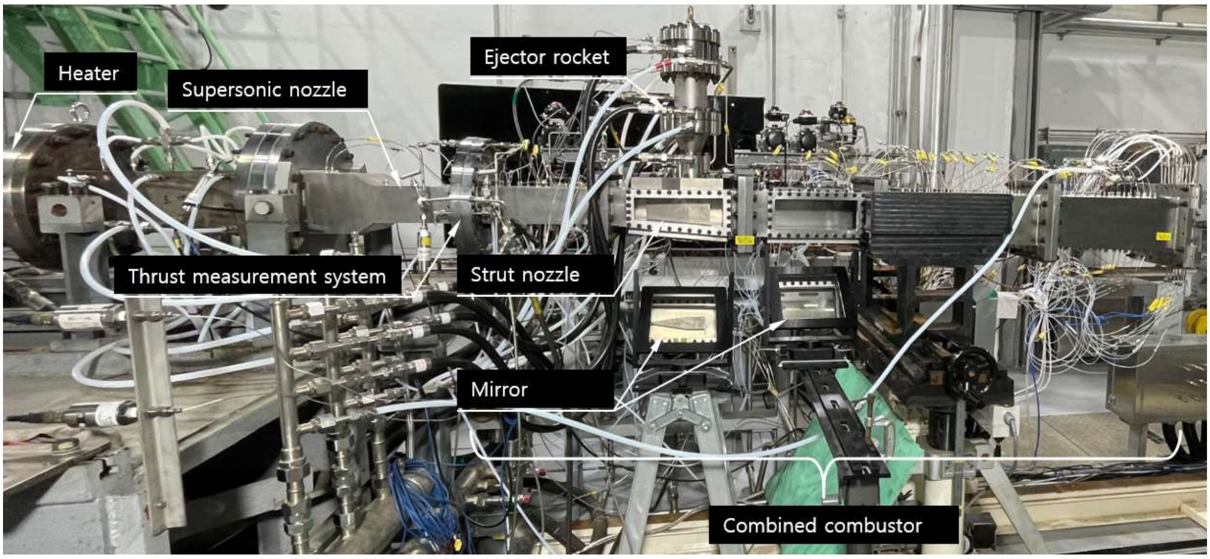

Fig. 2 Actual configuration of a combined combustor with test facility

To simulate the enthalpy conditions of the aircraft flying at a specific altitude, combustion heaters and compressed air were utilized. The flow of isolator inlet, recovered by external intake, and decelerated, was simulated using a supersonic nozzle. Following the isolator, a combustor combining the ejector rocket and dual-mode ramjet engine was connected, with no separate engine nozzle attached as the tests were conducted at ground altitude. At the main inlet, conditions corresponding to Mach 4 and an altitude of 20 km were supplied with airflow of about 4 kg/s, at a total temperature of 875 K and a total pressure of 647 kPa, at a speed of Mach 2.2. In Fig. 2, flow direction is from left to right, with the forward section excluding the part labeled "combined combustor" serving as a simulation equipment, and the combusted air exiting through a discharge approximately 1.5 m away. Following the supersonic nozzle, a combustor with isolator and expansion angles is sequentially connected, with a strut nozzle positioned centrally at the beginning of expansion part, with the rocket combustion chamber attached above.

The dimensionless length of all components based on the height of isolator, h as unit. The width of the combined combustor is 2.5h, and the length of the isolator are 5.67h. The combustor with dual expansion angles of 2.5 degrees and 5 degrees each has a length of 6.67h and 24.3h, resulting in a total length of 36.67h.

The ejector rocket uses kerosene (ADFS-1) as liquid fuel and gaseous oxidizer, with a maximum supply rate of 300 g/s for both fuel and oxidizer, and a combustion pressure of 35 bar. Although higher pressures and fuel/oxidizer quantities could be supplied theoretically, a single operating condition was chosen for stability during testing. The rocket has three injection ports, with an internal diameter of 1.33h and a length of 3.17h for the combustion chamber, and a throat diameter of 0.47h. The rocket combustion chamber is connected to a conical strut nozzle, composed of two nozzles with throat diameters of 0.17h and exit diameters of 3.7h, resulting in an exit velocity of M2.77.

The x-axis represents the flow direction, with 0 at the start of the combustor and increasing in the flow direction, while the y-axis has 0 at the center of the supersonic nozzle, increasing upwards. Key stations are defined, with the combustor starting at 2, the strut nozzle beginning at 3, and the end of the integrated combustor at 4. The ejector rocket is defined internally as 1', the nozzle throat as t, the nozzle exit as 2', and the nozzle exterior as 3'.

The test procedure is as follows: once the combustion heater ignites and the flame stabilizes, additional compressed air and oxygen are supplied to match the test conditions. The prepared air flows into the main inlet of the combined combustor. After stabilizing the airflow, fuel and oxidizer are supplied to the ejector rocket, and ignition is initiated using a methane-oxygen torch igniter. The combustion gas from the ejector rocket flows into the main inlet through the strut nozzle, and the test concludes by sequentially shutting off the ejector rocket's fuel, oxidizer, and main airflow.

3. Method of Acquiring the Thrust

3.1. Thrust measurement

Fig. 3 is a schematic diagram of the devices installed to measure the thrust generated by the combined combustor. There is a thrust disk with rings on the front and back, and the interior of the rings is filled with a fluid for pressure measurement. The thrust disk is attached at the beginning of the isolator of the combined combustor and can move freely back and forth under axial force. The pressure inside the rings in the front and back changes according to the extent of movement of the thrust disk. In other words, if the thrust disk moves forward due to the generated thrust, the pressure in the front ring increases, while the pressure in the rear ring decreases. By understanding this relationship in advance and deriving the equation, the measured differential pressure during the actual test can be substituted to calculate the generated thrust. Since the integrated combustor does not have a nozzle, measuring its own thrust is meaningless, but in the case of a combined combustor with an ejector rocket combustion chamber, differences in thrust due to the operation of the rocket combustion chamber can be observed.

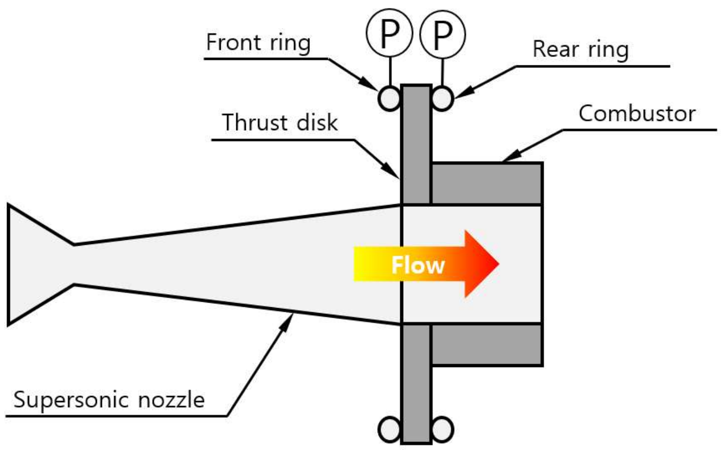

Fig. 3 Schematic diagram of thrust measurement system

Fig. 4 presents the dimensionless pressure difference measured on the front and back rings for the given loads. The relationship between pressure difference and thrust can be determined with a linear regression analysis, as depicted in Fig. 4, and its value is expressed in Eq. 1.

$$\Delta F = a\,\Delta p + b \tag{1}$$

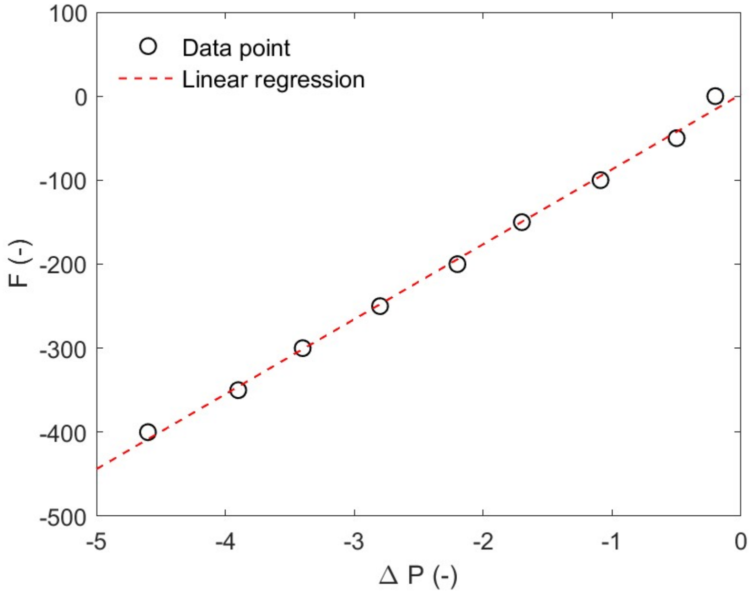

Fig. 4 Calibration graph of the thrust

3.2. Theoretical Calculation

The thrust of the ejector rocket, when combined with the strut nozzle, can be calculated using the following Eq. 2 [4]. The Mach number at the nozzle exit, depending on the ratio of nozzle throat to exit area, can be determined by Eq. 3 [5]. The subscripts and English terms used below follow the definitions shown in Fig. 1. Here, F, A, v, p, V, and γ represent force, area, velocity, pressure, specific volume, and specific heat ratio. Subscripts t, 1′, 2′, and 3′ denote throat, combustion chamber, nozzle exit, and external air, respectively.

$$\mathrm{F} = \frac{A_t v_t v_{2'}}{V_t} + (p_{2'} - p_{3'})A_{2'} = A_t p_{1'} \sqrt{\frac{2\gamma^2}{\gamma-1}\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma-1}}\left[1-\left(\frac{p_{2'}}{p_{1'}}\right)^{\frac{\gamma-1}{\gamma}}\right]} + (p_{2'} - p_{3'})A_{2'} \tag{2}$$

$$\frac{A_t}{A_{2'}} = \left(\frac{\gamma+1}{2}\right)^{\frac{\gamma+1}{2(\gamma-1)}} M_{2'}\left(1+\frac{\gamma-1}{2}M_{2'}^2\right)^{-\frac{\gamma+1}{2(\gamma-1)2(\gamma-1)}} \tag{3}$$

3.3. Quasi-One-Dimensional Analysis

To calculate the axial quantity, the following equation is utilized

$$\frac{dM}{dx} = -\frac{M}{2}\frac{1}{A}\frac{dA}{dx} - \frac{1+\gamma M^2}{2\gamma M}\frac{1}{P}\frac{dP}{dx} - M\frac{C_f}{D_h} \tag{4}$$

$$\frac{dT_t}{dx} = \frac{T_t}{\left(1+\frac{\gamma-1}{2}M^2\right)A}\frac{dA}{dx} - \frac{T_t(1-M^2)T_t(1-M^2)}{\gamma M^2\left(1+\frac{\gamma-1}{2}M^2\right)}\frac{1}{p}\frac{dp}{dx} - \frac{2T_t}{\dot{m}}\frac{d\dot{m}}{dx} - \frac{T_t[1+(\gamma-1)M^2]T_t[1+(\gamma-1)M^2]}{2\left(1+\frac{\gamma-1}{2}M^2\right)}\frac{4c_f}{D} + \frac{T_t}{\left(1+\frac{\gamma-1}{2}M^2\right)}\frac{(T_{aw}-T_w)(T_{aw}-T_w)}{2TPr^{\frac{2}{3}}}\frac{4C_f}{D} \tag{5}$$

The Mach number and total temperature are calculated using the drag coefficient derived from the Blasius relationship in equations, Eq. 5 and Eq. 6 [6]. The friction coefficient is derived from the Eckert et al. [7] Eq. 7 to obtain the dynamic friction coefficient based on the wall temperature. The wall temperature is set to 200 °C, with superscript * and subscript w, denoting reference and wall values, respectively. The quantity of combustion gas at Eq. 8 is derived using the lower heating value (LHV) [8].

$$C_f = \frac{0.074}{(Re_x^*)^{0.2}} = 0.074\left(\frac{\rho^* ux}{\mu^*}\right)^{-0.2} \tag{6}$$

$$T^* = T\left[1 + 0.032M^2 + 0.58\left(\frac{T_w}{T} - 1\right)\right] \tag{7}$$

$$\dot{m}_c = \frac{(\dot{m}_2 + \dot{m}_f)c_p T_{t4} - \dot{m}_2 c_p T_{t3}}{LHV} \tag{8}$$

By utilizing the inlet conditions and wall pressure, the partial differential equation can be solved, from which the axial temperature and Mach number can be calculated. The force due to the operation of the ejector rocket can be calculated, and from there, the thrust can be derived by Eq. 9 and Eq. 10.

$$\mathrm{F} = \dot{m}\mathrm{v}_4 = \dot{m}\mathrm{aM}_4 = \dot{m}\sqrt{\gamma \mathrm{RT}_4}M_4 \tag{9}$$

$$\Delta F_{1D} = \left[F_{onR} - F_{offR}\right]_{1D} \tag{10}$$

In the above equation, "on" represents the case where there is a flame in the Ejector rocket, while "off" represents the case where there is no flame.

3.4. Pitot Rake Measurements.

The Rayleigh-Pitot formula, can be utilized to calculate the Mach number at the measured point of the pitot tube using the pitot pressure and wall pressure. By averaging the Mach numbers calculated in this way, the thrust due to the operation of the ejector rocket can be obtained, similar to the quasi-one-dimensional calculation method, Eq. 12.

$$\frac{p_{t4'}}{p_4} = \left[\frac{(\gamma+1)M_4^2}{2}\right]^{\frac{\gamma}{\gamma-1}}\left[\frac{\gamma+1}{2\gamma M_4^2 - (\gamma-1)}\right]^{\frac{1}{\gamma-1}} \tag{11}$$

Where, the superscript ′ means the downstream of the normal shock at the pitot probe.

$$\Delta F_{Rake} = \left[F_{onR} - F_{offR}\right]_{Rake} \tag{12}$$

The rake is attached vertically to the outlet of the combined combustor. It has a total length of 2h and features nine holes with a diameter of 1 mm, spaced at intervals of 0.25h, oriented in the opposite direction of the flow. These holes are protruded in a manner that prevents interference with each other. Since the entire structure is directly exposed to flames, it is cooled using water.

4. Result and Discussion

The inlet flow has a pressure, temperature, and mass flow rate of 666.5 kPa, 916 K, and 4.14 kg/s, respectively. The oxygen and fuel supplied to the rocket combustion chamber are 0.2084 kg/s and 0.1196 kg/s, respectively, with a total of 0.328 kg/s. The pressure in the combustion chamber is 3558.76 kPa. Some parts of the strut nozzle are made of graphite for heat resistance, and the test proceeded normally for about 1 second after ignition.

Fig. 5 shows the pressure values of the front and rear rings of the thrust disk measured during the test, along with the calculated thrust values using the conversion equation. Due to significant oscillations in the thrust values, the values were averaged over a 0.5-second moving window to determine the values at specific points in time. At the 20-second mark, before stabilization of the inlet flow and operation of the rocket combustion chamber, and at the 30-second mark, after stabilization of the strut nozzle flame, the thrust values were determined to be -528.6 N and 419.5 N, respectively. And its difference is 948.2 N.

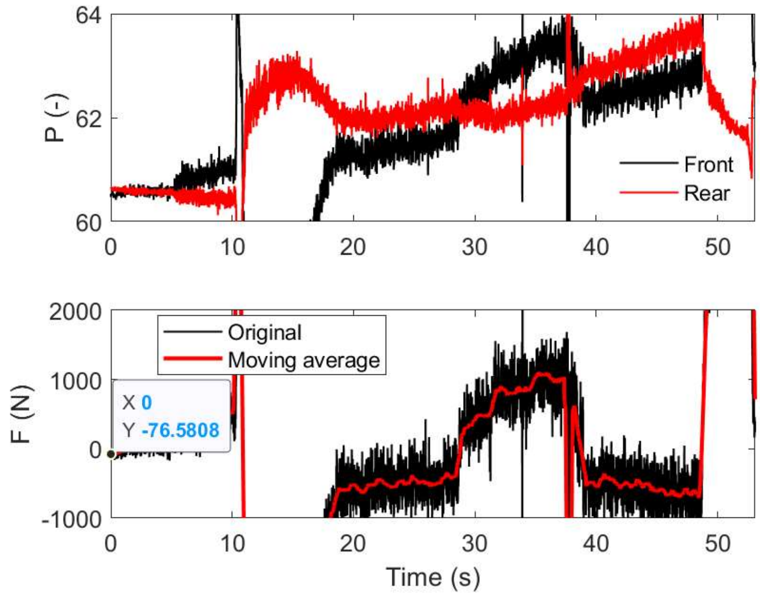

Fig. 5 Thrust and pressure

The thrust value calculated using the theoretical equation is 896.3 N.

Fig. 6 shows the measured axial wall pressure during the test, while Fig. 7 displays the Mach number obtained through quasi-one-dimensional analysis. In the case where there is no flame in the ejector rocket, the combustion chamber maintains a high-speed, low-pressure state internally and undergoes expansion at the exit where the pressure is relatively high, resulting in pressure recovery. As a result, the Mach number at the exit is also calculated as subsonic. However, since separation is not considered in the assumptions of quasi-one-dimensional analysis, the calculation may not be accurate from the point of separation. The calculated thrust values before and after the ejector flame are 1811.4 N and 3611.1 N, respectively, resulting in a difference of 1799.7 N in thrust depending on the presence of the ejector rocket flame.

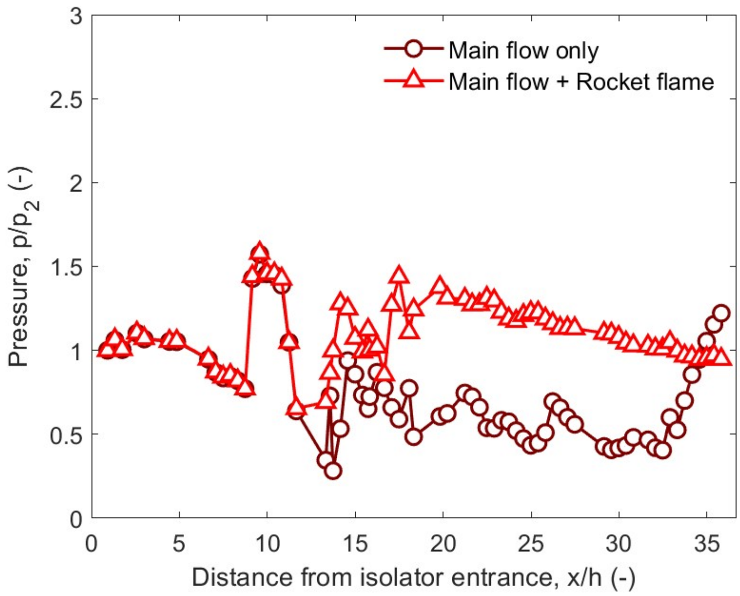

Fig. 6 Axial wall pressure

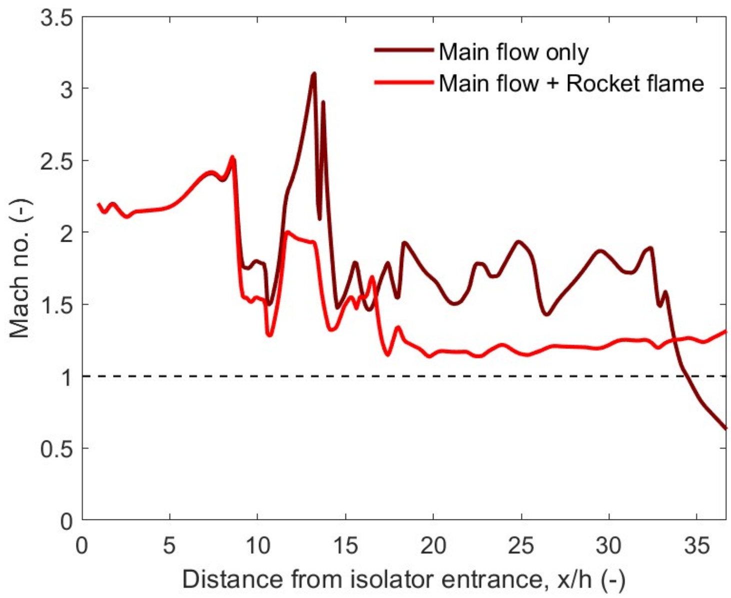

Fig. 7 Axial Mach no.

Fig. 8 shows the pressure values measured at the rake, while Fig. 9 displays the Mach numbers calculated using the rake pressure and the stagnation pressure at the exit of the combustion chamber. The Mach numbers obtained through area averaging are 1.03 and 1.33, respectively. The Mach number in the case of flame presence is similar to the calculated exit Mach number obtained through quasi-one-dimensional analysis. The calculated thrust values before and after the flame are 2963.4 N and 3640.9 N, respectively.

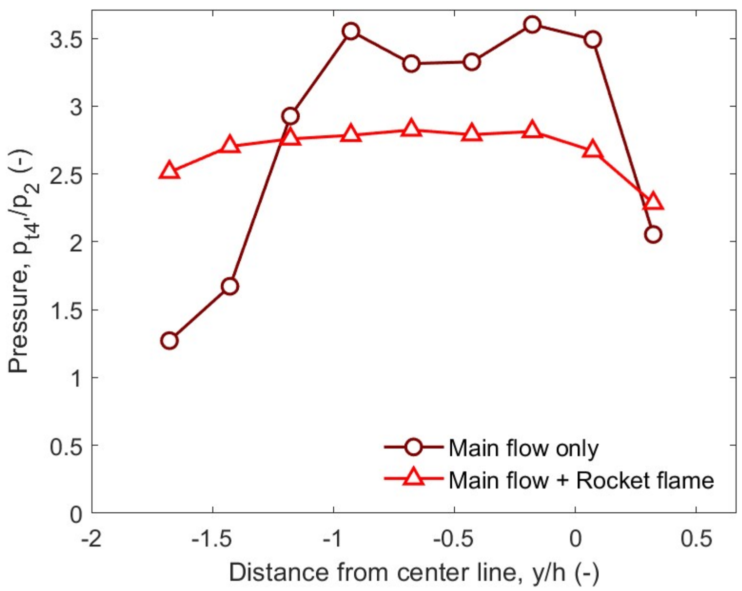

Fig. 8 Pitot pressure

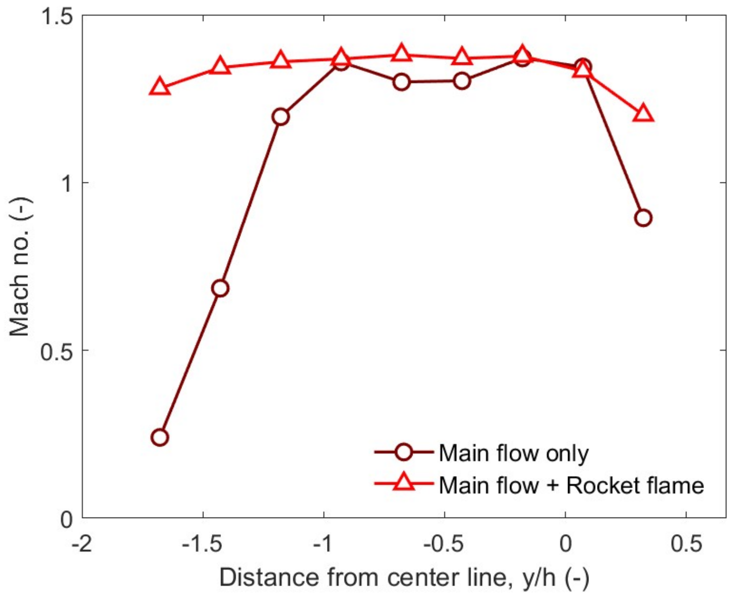

Fig. 9 Mach no. at combustor exit

In Table 1, we compared the forces obtained using each method before and after the flame, as well as the increase in thrust due to the operation of the ejector rocket, and the differences according to each calculation method, with the theoretical equation as the reference. The theoretical equation only analyzes the flame of the ejector rocket itself, so it cannot account for the ejecting effect. Therefore, it may also be reasonable to use the values obtained by the thrust measurement device as a reference, although the magnitudes of the values are similar. However, the value obtained by the quasi-one-dimensional method is difficult to compare because it cannot accurately determine the value due to separation at the exit of the combustion chamber in the case of no flame. The value obtained by the lake method is relatively low at 677.5 N. This result may be due to the relatively low specific heat ratio and stagnation temperature used in calculating the exit velocity in the case of flame presence. In cases where precise temperature measurements and knowledge of the composition of the combustion gases are lacking, such results are inevitable. Nevertheless, despite this, it can be observed that the quasi-one-dimensional method with the same stagnation temperature, specific heat ratio, and gas constant applied has a similar value to the thrust in the case of flame presence.

Table 1 Comparison of thrust

	TMS	Theoretical	Quasi-1D	Rake	Ejector Rocket
$\mathbf{F_{off}}$ (N)	-528.6	-64.1	1811.4	2963.4	off
$\mathbf{F_{on}}$ (N)	419.2	833.2	3611.1	3640.9	on
$\mathbf{\Delta F}$ (N)	947.8	897.3	1799.7	677.5	
$\mathbf{\Delta}$(%)	100%	95%	190%	71%	

5. Conclusion

The rocket-integrated combustor test was conducted using a dual-mode ramjet engine equipped with a strut nozzle inside the combustor to supply combustion gas from the rocket engine, thereby obtaining additional thrust from the ejector rocket.

Air at an enthalpy condition corresponding to Mach 4 and an altitude of 20 km, with a pressure of 666.5 kPa, a temperature of 916 K and air mass flow rate of 4.14 kg/s, was supplied to the combustor, simulated using a supersonic nozzle after passing through an external intake. The simulated air entered the dual-mode ramjet combustor of the integrated combustor, and the increase in thrust was confirmed by generating combustion gas through the rocket when the flow stabilized. The values were confirmed using various methods such as 1) theoretical calculation method, 2) thrust measurement, 3) quasi-one-dimensional analysis, and 4) exit pressure measurement.

Except for the quasi-one-dimensional analysis, which cannot calculate the normal exit velocity in cases where separation occurs in the exit flow under conditions without flame, the sizes of the thrust obtained through three calculation methods showed a similar distribution. Excluding the theoretical calculation method, which does not consider the ejecting effect, the thrust value calculated using the thrust plate can be considered the most accurate. From the perspective of accuracy, utilizing a thrust measurement device would be the best method. However, setting up such a device incurs significant additional costs, and there are difficulties in designing, manufacturing, and installing the test device to ensure that only thrust is acting, excluding additional forces.

As an alternative, using a Pitot tube lake to calculate thrust and verify the flow distribution at the exit can be considered. However, special attention must be paid to cooling design to prevent damage to the rake exposed to the flame. Finally, the easiest method is the quasi-one-dimensional approach, which can calculate values using inlet conditions and wall pressure, and its accuracy is somewhat guaranteed as it is similar to the thrust calculation results from the lake. However, errors are inevitable due to the significant assumptions included in the calculation of physical properties.

In conclusion, various methods can be used to calculate the thrust of the integrated combustor, and applying the appropriate calculation method for the purpose of the test can sufficiently derive thrust even in direct connection tests.

Acknowledgement

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