Design Study on a Variable Intake and a Variable Nozzle for Hypersonic Engines

Hideyuki TAGUCHI, Hisao FUTAMURA†, Kazuo SHIMODAIRA‡,*
Tetsuya MORIMOTO†, Takayuki KOJIMA‡ and Keiichi OKAI‡
Institute of Space Technology and Aeronautics, Japan Aerospace Exploration Agency (ISTA-JAXA)
Chofu, Tokyo 182-8522, Japan
E-mail: taguchi.hideyuki@jaxa.jp

Keywords : Hypersonic, Intake, Nozzle, Space Plane

Abstract

Variable air intake and variable exhaust nozzle of hypersonic engines are designed and tested in this study. Dimensions for variable geometry air intake, ram combustor and variable geometry exhaust nozzle are defined based on the requirements of a pre-cooled turbojet engine. Hypersonic Ramjet Engine is designed as a scaled test bed for each component. Actuation forces of moving parts for variable intake and variable nozzle are reduced by balancing the other force in the opposite direction. A demonstrator engine which includes variable intake and variable nozzle is designed and the components are fabricated. Composite material with silicone carbide is applied for high temperature parts under oxidation environment such as leading edge of the variable intake and combustor liner. Internal cooling structure is adopted for both moving and static parts of the variable nozzle. Pressure recovery and mass capture ratio of the variable intake at Mach 5 is obtained by a hypersonic wind tunnel test. Flow characteristics of the variable nozzle are obtained by a low temperature flow test. Wall temperature and heat flux of the nozzle at Mach 3 is obtained by a firing test. As results, the intake and the nozzle are proved to be used at designed pressure and temperature environment.

Introduction

Basic concepts for TBCC (Turbine Based Combined Cycle) space planes have been proposed.[1-3] Figure 1 shows a conceptual drawing of TBCC space plane. Pre-cooled turbojet engine, scramjet engine and rocket engine are assumed to be combined for SSTO space planes.. Pre-cooled turbojet engine is also assumed to be solely used for the first stage of TSTO space planes.

Pre-cooled turbojet engine was analytically proved to have a capability to be operated under conditions between sea level static and Mach 6 flight. Several thermodynamic cycles such as expander cycle, staged combustion cycle and non-cooled brayton cycle were compared regarding the engine.[3] Figure 2 shows a diagram for a staged combustion cycle of pre-cooled turbojet engine.

As a result, pressure ratio and equivalence ratio was found to be principal parameters to determine thrust to mass ratio and specific impulse. However, the difference of the thermo-dynamic cycle appeared to have small effect to the payload injection capability, especially in the TSTO analysis. Additionally, it was found that the mass of the variable geometry air intake and the variable geometry exhaust nozzle has serious effect to the payload injection capability.

In this design study, the shape and aerodynamic performance of a variable geometry air intake and a variable geometry exhaust nozzle are defined. At the beginning, basic shapes for variable geometries are selected. Then, detailed shapes for the air intake and the exhaust nozzle are analyzed by aerodynamic calculations. Thermo-dynamic performances of pre-cooled turbojet engine and ramjet engine with the intake and the nozzle are also analyzed. Finally, scaled experimental models to confirm the performance and heat resistant structure are designed and tested.

Fig.1 Conceptual Drawing of a TBCC Space Plane

* Future Space Transportation Research Center
† Aeronautical Environment Technology Center
‡ Aeroengine Testing Technology Center

Fundamental Design

Requirements

Requirements of aerodynamic and functional performances to air intakes and exhaust nozzles from each engine cycles are almost the same. In order to enhance the thrust of the engines for whole flight speed range, air intakes and exhaust nozzles are required to adapt the throat area to attain adequate air flow rate for the core engine. Furthermore, air intakes are required to minimize pressure loss, which occurs during the conversion process from kinetic energy to static pressure through shock waves. Spillage flow should also be reduced to avoid excessive drag. Static pressure on the compression surface of air intakes rises by shock waves. However, if a part of flow is spilled out of the cowl, the pressure relative to the spillage flow causes some drag to the engine.

Exhaust nozzles are required to convert the internal energy of combustion gas to kinetic energy, efficiently. Boat tail drag, which is generated at the low pressure base area of the cowl, should be minimized.

Shape Selection

In order to satisfy the above mentioned requirements, two types of shapes can be adopted. One type is using variable position ramp, and the other type is using variable angle ramp.

The variable position type has a little capability to change throat area by moving the ramp part. Air intakes with the type can not adapt the angle of oblique shock waves. The first shock wave is formed at far front of the cowl at low Mach number, if the air intake is designed to attain high performance at high Mach number. A large amount of air flow spills over the cowl at low Mach number and serious drag occurs especially at transonic speed. As for exhaust nozzles, if the variable position type is selected, cowl angle should be large enough to attain high turning angle for high expansion ratio. The large angle of the cowl causes a large boat tail drag especially at transonic speed.

The variable position type can easily be adopted for axisymmetric shape intakes and nozzles. The intakes and nozzles have a merit that it can be fabricated with small mass compared to rectangular shapes. However, the axisymmetric intake also has a risk that it may cause serious pressure distortion at the entrance of the core engine, and it may give some damage to the core engine. The distortion is generated by shock interactions between two intakes or between an intake and a wing. Therefore, the axisymmetric air intakes should be installed with enough distance from other intakes and wings. This feature is not preferable for space planes because it needs large thrust to attain enough acceleration and it needs a large number of engines to be installed under small area wings.

The variable angle type can be operated with designated ramp angle even at off design speed. The throat area also changes along with the variation of the angle. Therefore, nearly designated throat area variation can be also attained by selecting the ramp length. These features are ideal to keep high pressure recovery and high flow capture ratio even at off-design speed.

The variable angle type is difficult for axisymmetric application. Then, it can be used only for rectangular shape, with normal designs. The rectangular shape has a demerit that the structural

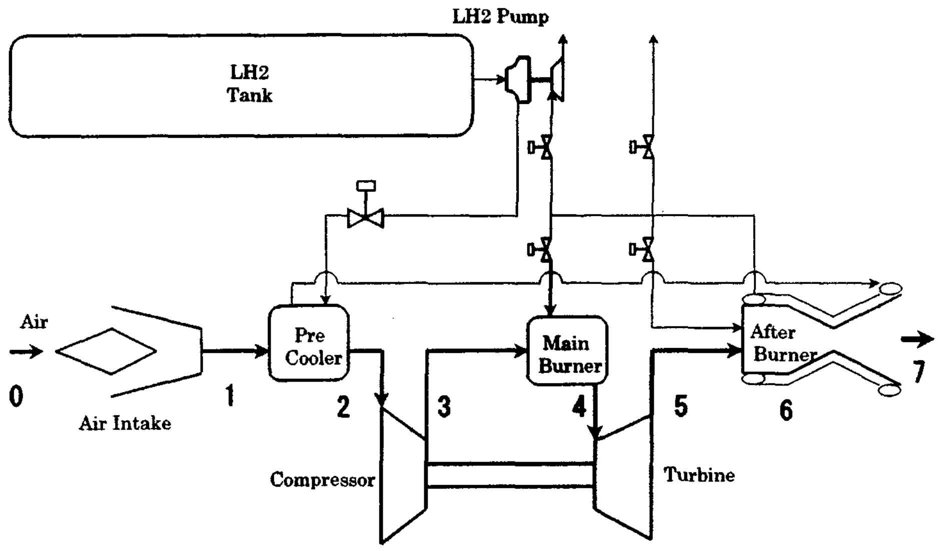

Fig. 2 Diagram of Pre-cooled Turbojet Engine with Staged Combustion Cycle

mass tends to be large. However, it features high pressure recovery, high air capture ratio, and low risk for pressure distortions. Then, this type is selected for further investigations. The principal technical challenges of this type are reduction of the structural mass and the actuator mass as well as aerodynamic design to adapt requirements from the core engine.

Engine System Design

Rectangular bifurcated shape for the air intake and the exhaust nozzle are adopted for a pre-cooled turbojet engine. The shape also has a merit that it can be easily integrated with scramjet engines, if needed.

Figure 3 shows the cross section of a pre-cooled turbojet engine with the rectangular bifurcated variable intake and nozzle.[4] The variable air intake is composed of 4 moving ramps. The ramps are connected each other and the links are designed to cancel principal forces at whole Mach number range. The variable exhaust nozzle is composed of 2 moving parts. The angle of the ramps is designed to form internal expansion nozzle with the outer cowl. Expanded flow downstream of the internal expansion nozzle is further expanded by an external expansion nozzle adapting to the ambient pressure.

Engine Installation

Mass of hypersonic components are estimated based on an empirical mass estimation method.[5] Then, total system mass for a reference TSTO space plane are calculated with a flight analysis method.[3] Payload of 1Mg and orbit diameter of 200km are assumed as the requirements to the flight analysis.

Figure 4 shows a 3 dimensional view of the optimum size TSTO space plane. Eight engines are assumed to be installed using four sets of intakes and nozzles. By using rectangular shape, cluster engine installation is easily attained and the number of engines can be increased to reduce total length and mass. If the smaller engine can be applied, development cost and risk will be extremely reduced.

Fig. 4 Engine Installation to a TSTO Space Plane

Engine Performance Analysis

Hypersonic Ramjet Engine

Basic shapes of a variable air intake and a variable exhaust nozzle are defined. However, some serious problems regarding to real fabrication and operation may happen. Then, a sub-scale model is fabricated to check the aerodynamic performance and the heat resistant structure in a simulated environment.

Figure 5 shows the cross section of a Hypersonic Ramjet Engine. The engine is composed of a variable geometry air intake and a variable geometry exhaust nozzle. Rectangular ram combustor is also designed to simulate the exhaust

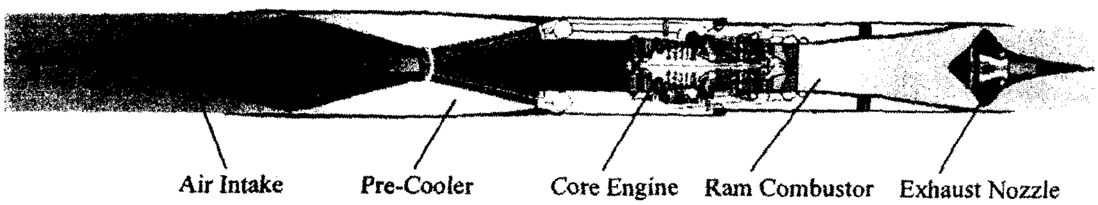

Fig. 3 Cross Section of Pre-Cooled Turbojet Engine

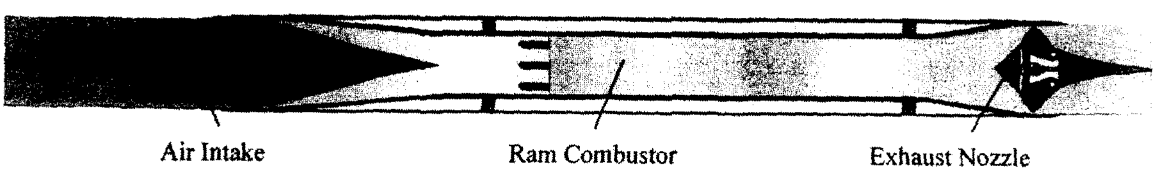

Fig. 5 Cross Section of Hypersonic Ramjet Engine (H: 10cm, W: 15cm, L: 190cm)

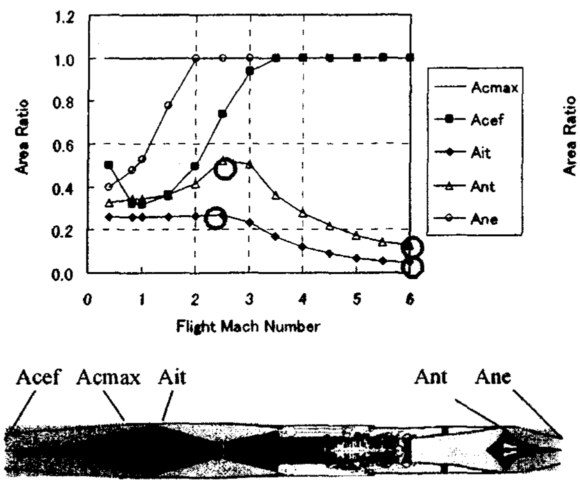

Fig. 6 Area Variations of Pre-Cooled Turbojet Engine

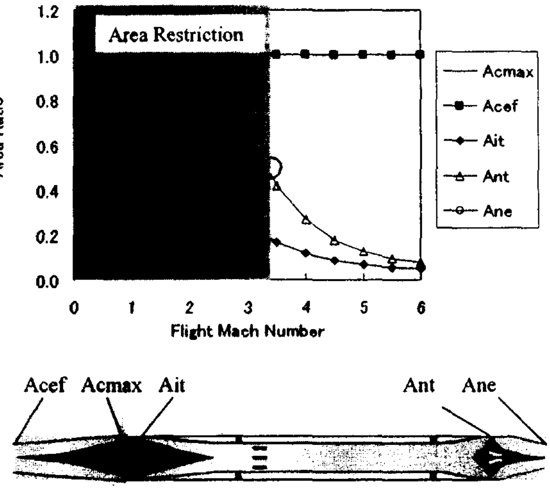

Fig. 7 Area Variations of Ramjet Engine

gas temperature of pre-cooled turbojet engine. The combustor is designed in rectangular shape in order to connect to the rectangular intake and the rectangular nozzle. Gas hydrogen is assumed as a fuel to raise the temperature at the nozzle entrance up to 2173K. Fuel rich combustion is assumed in the final target engine. However, fuel lean combustion is selected as the early stage experiments to keep safety.

This model can be used as a normal ramjet engine within limitations of the air speed at the combustor entrance and the variation of the nozzle throat area.

Analysis Method and Conditions

In order to clarify the required moving range of each ramp, performance analysis of pre-cooled turbojet engine is performed.[3)] The analysis of ramjet cycle is also performed to check the thrust performance of the Hypersonic Ramjet Engine.

Flight dynamic pressure is assumed as 50 kPa. Thermodynamic parameters such as pressure, temperature, and area at each point in the engine are analyzed. Flow balance, thermal balance and power balance are satisfied by varying throat area and fuel flow rate. Combustion temperature, intake capture area, and nozzle exit area are defined as limitation parameters.

Area Variations

Figure 6 shows required variation schedule for the intake throat area and the nozzle throat area based on a performance analysis of pre-cooled turbojet engine. The maximum throat area ratio is defined as the ratio of throat area to the intake capture area. The intake throat area should vary up to 5 times larger than the minimum area, and the nozzle throat area should vary up to 7 times larger than the minimum area, to satisfy the requirements. Dimensions for the variable air intake and the variable exhaust nozzle are decided taking account of the requirements and aerodynamic characteristics.

Figure 7 shows required variation schedules based on a performance analysis of a ramjet engine without any restriction of variable area. If the air intake is designed to satisfy the requirement of pre-cooled turbojet, the maximum throat area ratio of the intake is about 0.3. The maximum area ratio of the nozzle throat is about 0.5. The operating range of the Hypersonic Ramjet Engine can be obtained using these values and plots in fig. 7. The engine with a design for the pre-cooled turbojet engine has a potential to be operated from Mach 3.5 to Mach 6.0 with no limitation.

Ramjet Operation

Comparing the analytical results of the pre-cooled turbojet engine and the ramjet engine, maximum pressure and maximum temperature are nearly the same. Then, pressure and temperature environment for the intake and the nozzle of the pre-cooled turbojet engine can be simulated by the ramjet operation.

Figure 8 shows thrust per capture area of the pre-cooled turbojet engine. The maximum value is about 120 kN/m^2 at subsonic speed. Figure 9 shows thrust per capture area of the ramjet engine. The maximum value is about 60 kN/m^2 at Mach 3.5. The thrust per capture area at the speed lower than Mach 3.5 should be very low because the air flow is limited by the maximum nozzle throat area.

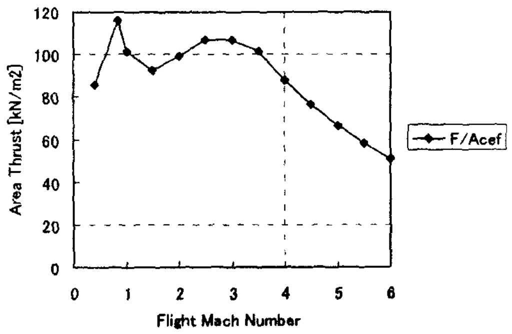

Fig. 8 Area Thrust of Pre-Cooled Turbojet Engine

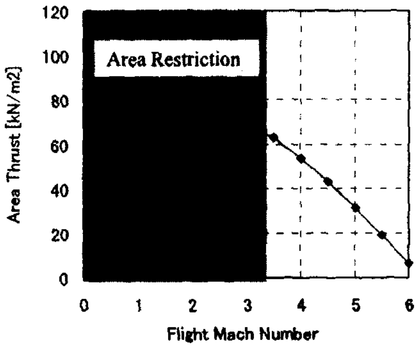

Fig. 9 Area Thrust of Ramjet Engine

Fabrication of Test Models

Experimental models for the Hypersonic Ramjet Engine are fabricated. Height, width and length of the engine are 10cm, 15cm and 190cm, respectively. Main objective of the engine is to confirm principal technologies required for hypersonic engine components by real pressure, temperature and velocity environments. Composite material with silicone carbide (SiC/SiC) is applied for high temperature parts under oxidation environment such as leading edge of air intake and combustor liner. Normal carbon/carbon composite is inadequate for the oxidation environment. Thermal properties of the SiC/SiC composite material are obtained by preliminary experiments.[6] The shape of connecting parts with metal structure is designed based on a finite element method analysis.

Wind Tunnel Test of Air Intake

Air Intake Design

The air intake is designed at Mach 6 with 7 oblique shock waves and a normal shock wave system. Figure 10 shows the shock wave pattern at the design point. The hypersonic flow is decelerated through 7 oblique shock waves and finally become to subsonic speed through a weak normal shock wave. The ratio between external compression and internal compression is 2 : 8 at the Mach 6 design point. Pressure recovery analysis is performed using oblique shock wave equations and normal shock wave equations.[7]

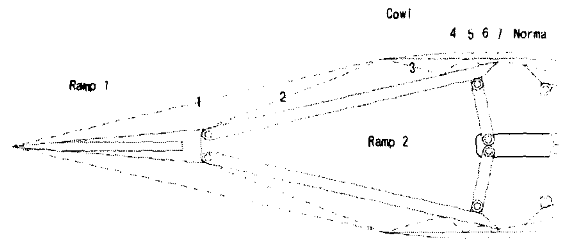

Fig. 10 Shock Pattern of Variable Air Intake

Variable Air Intake Model

A variable air intake model for a hypersonic wind tunnel test is fabricated. Figure 11 shows the relation between the Hypersonic Ramjet Engine and the fabricated wind tunnel test model. Upper half part of the bifurcated intake is included in this model, because of the blockage area limitation of the wind tunnel facility. Width of the model is 100mm and Height of it is 75mm. Variable geometry ramps are connected with links to cancel principal forces. The ramps are actuated by an electrical servo motor. A load cell is installed between the links and the motor to measure the actuation force. A flow plug to regulate air flow rate is installed at the exit of the intake. The plug is also actuated by another electrical servo motor. Wall pressure distribution along with the center line of ramps and pitot pressure distribution at the exit of the intake are measured. The model has optical windows for Schleren photography to visualize the complicated flow field around the throat.

A metallic leading edge and a SiC/SiC composite material leading edge are fabricated. The SiC/SiC leading edge is fabricated to endure the high temperature condition of Mach 6 flight. The temperature is about 1650K.

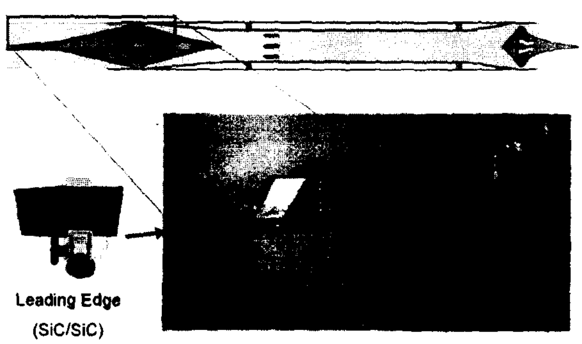

Fig. 11 Variable Air Intake Model

Wind Tunnel Test Results

Wind tunnel test is conducted at the hypersonic wind tunnel facility in ISTA-JAXA. Nominal stagnation conditions are total pressure of 1.0MPa and total temperature of 653K. Maximum test duration is 60 seconds. Ramps and a flow plug are actuated by a programmed schedule during the tests. Figure 12 is a Schleren photograph of air intake under Mach 5 condition. Thick boundary layer is observed on the second ramp. Because of the boundary layer, the intake should be operated with a larger throat height than the designed height. The intake becomes unstart condition, if the throat height is reduced to the designed value. The throat height can be reduced by using a throat bleeding.

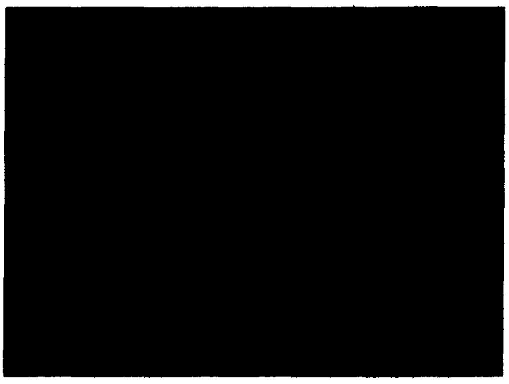

Fig. 12 Schleren Photograph of Air Intake (Mach 5)

Figure 13 shows pressure recovery variations. The dotted line means pressure recovery of MIL-E-5000B[8)], which is normally used at preliminary engine performance analyses. Design analysis is performed with the intake size of 450mm width and 300mm height, which size adapts to a sub-scale pre-cooled turbojet engine. Basic pressure recovery of the intake is plotted with open circles. It appears to be high enough, comparing to the MIL specification. However, pressure recovery of the wind tunnel experiment under Mach 5 condition is smaller than the design value. This is because the boundary layer thickness is larger than the expected value and the ramp angle have to be smaller to attain larger throat height. It causes larger Mach number at upstream of the normal shock wave and the shock wave causes larger pressure loss. Pressure recovery improves by using throat bleeding.

Figure 14 shows mass capture ratio of the design and wind tunnel test. Mass capture ratio of the test is smaller than that of design by the throat bleeding. Pressure recovery should improve with larger size and larger Reynolds number intake. However, if the size of the wind tunnel test model is used, trade off study between pressure recovery and mass capture ratio will be needed.

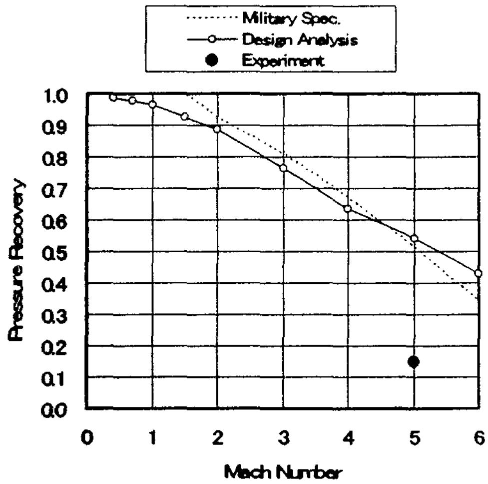

Fig. 13 Pressure Recovery

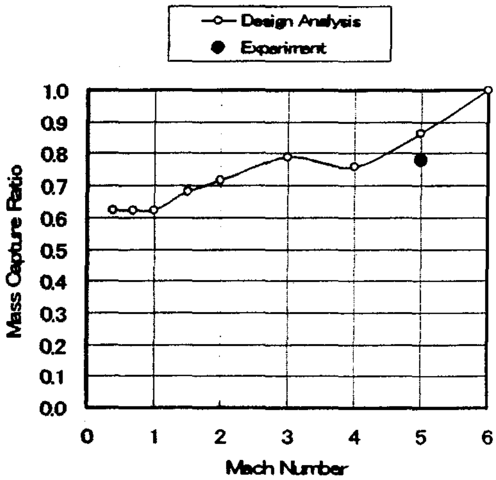

Fig. 14 Mass Capture Ratio

Firing Test of Exhaust Nozzle

Exhaust Nozzle Design

The exhaust nozzle is also designed at Mach 6 with an internal expansion and an external expansion system. Figure 15 shows the expansion fan pattern for the exhaust nozzle. The variable exhaust nozzle is composed of 2 moving ramps and 2 fixed ramps. The room between 2 moving ramps is supposed to be pressurized by the air from the compressor exit. Then the principal force to the moving ramp is canceled by both room side pressure and gas side pressure. Ambient pressure for the pre-cooled turbojet engine largely changes because it is used from sea level to 26 km altitude. If a normal convergent divergent nozzle is used for a high altitude design nozzle, gas flow at downstream of the nozzle tends to separate from the wall at low altitude. It causes reduction in the thrust coefficient.

Plug nozzles use a plug shape structure in the center of nozzle part, and it utilizes the movement of free boundary line to achieve static pressure balance between the exhaust gas and the ambient air. This effect prevents a flow separation and regarding thrust loss. However, if the exhaust speed at the cowl exit is set to be sonic speed, angle of the cowl should be very large. It may cause serious boat tail drag

Internal expansion part with a variable geometry mechanism is introduced to reduce the boat tail drag. The exhaust gas is accelerated in the internal expansion part to supersonic speed. The average vector of the flow is set to incline to the inside. Then the flow can expand in the direction of outside at the external expansion part. The cowl angle can be set to 0 degree by this idea. This angle is ideal to eliminate the boat tail drag.

This nozzle can adapt both wide expansion ratio range and required throat area variation. If the only external expansion part is used, initial ramp angle and cowl angle must be around 70 degree and throat area variation is strictly limited by the shape. Then, the nozzle size must be very large in order to satisfy the throat area requirement at low Mach number. By introducing the internal expansion part, the maximum ramp angle can be reduced to around 48 degree and the cowl angle can be set to 0 degree.

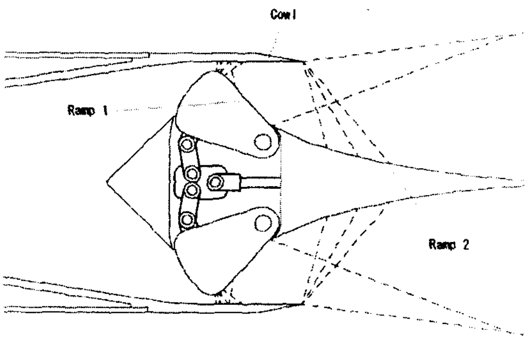

Fig. 15 Expansion Fan Pattern of Variable Nozzle

Exhaust Nozzle Test Model

Ram combustor and variable exhaust nozzle model for the Hypersonic Ramjet Engine is fabricated. Nozzle exit sizes are 150mm width and 100mm height. Figure 16 shows test setup for ram combustor and variable exhaust nozzle combination test. Two stage injectors for a ram combustor is designed to satisfy deep throttling requirements through wide flight Mach number range. SiC/SiC composite material liner is installed on the combustor wall surface. Water cooling structure is applied for both moving and static parts of the variable exhaust nozzle on the first test model. Actuation devices in the nozzle are protected by pressurized cold air. The heat resistant structure is designed to endure the hot combustion gas with 2173K.

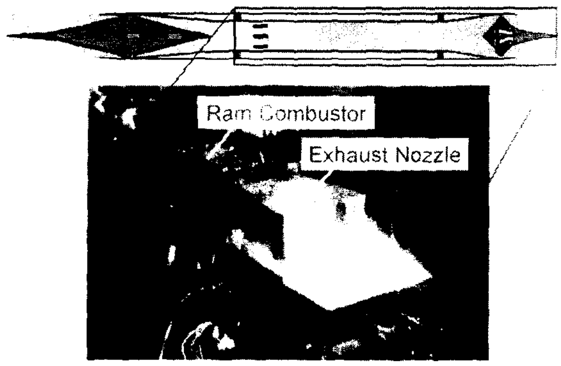

Fig. 16 Test Setup of Ram Combustor and Variable Exhaust Nozzle

Firing Test Results

Firing test of the ram combustor and variable exhaust nozzle model is conducted at Continuous Ramjet Combustion Facility of ISTA-JAXA. The facility has a capability to supply high pressure and high temperature air, which simulates combustor inlet conditions up to Mach 5. Flow characteristics of the variable nozzle are also obtained with a constant air temperature testing without combustion.

The first firing test is performed with Mach 3 condition with maximum throat height. Figure 17 shows the time history of temperature at the test. Combustor inlet air temperature is kept to 600K. Hydrogen fuel is injected and nozzle inlet temperature is raised to 2140K. Nozzle surface temperature is stabilized at about 940K with the water cooling structure.

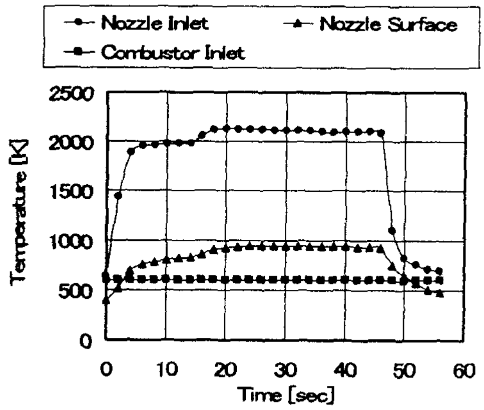

Fig. 17 Time history of Temperature

Heat flux at each part is calculated by measuring cooling water temperature rise and mass flow rate of the water. Figure 18 shows history of heat flux on the Mach 3 test. Time delay of heat flux rise is observed comparing to the temperature history. The delay may correspond to the heat capacity of metallic structure. Heat flux becomes a stable condition after 30 seconds. Left throat part has the largest heat flux on the test. Difference between left and right throat may be caused by one sided flow, because the exit low pressure is not simulated in this test and the throat is not choked. Downstream ramp part has the smallest heat flux as expected.

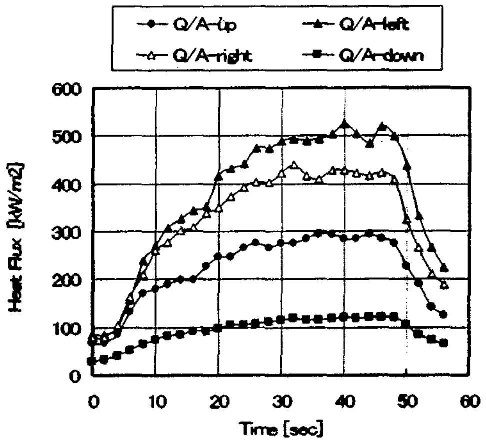

Fig. 18 Time history of Heat Flux

Summary of Experimental Results

Air Intake Performance

Figure 19 shows designed area variation of the Hypersonic Ramjet Engine. Experimental results are also plotted on the graph. Intake throat area (Ait) on the wind tunnel test is larger than the design value. This is because the boundary layer thickness on the test is larger than the design value.

Figure 20 shows pressure variation of Hypersonic Ramjet Engine. Based on the experimental result, intake inlet pressure (Pii) and intake exit pressure (Pie) for the analysis condition is calculated. Intake exit pressure is very low comparing to the design value. This is because the Reynolds number of the wind tunnel test model is very low.

Exhaust Nozzle Performance

Nozzle throat area is set on the design value at each test as shown in Fig. 19. Flow characteristics test is conducted under conditions between Mach 3.5 and Mach 6.0. Air temperature is fixed to 400K on the test. Nozzle inlet pressure (Pni) on the test is plotted in Fig. 20. Variable mechanism is proved to work normally on the designed pressure environment.

Figure 21 shows designed air flow rate of exhaust nozzle. Mass air flow at designed combustion temperature is calculated based on the experimental results of flow characteristics test. The experimental value is also plotted on the graph. The flow rates agree with each other. It means ideal flow around the nozzle throat is attained at whole variation range.

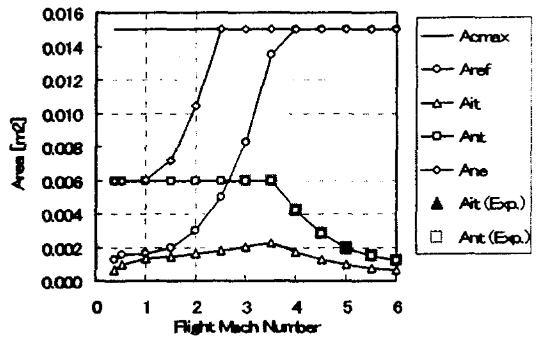

Fig. 19 Area Variation of HRE

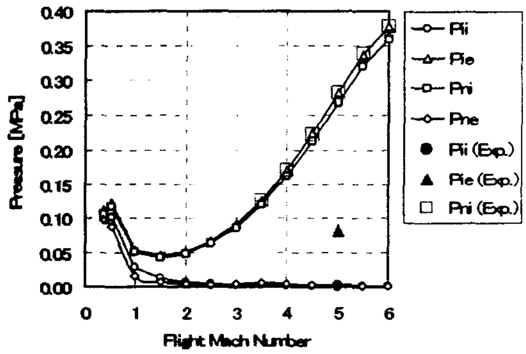

Fig. 20 Pressure Variation of HRE

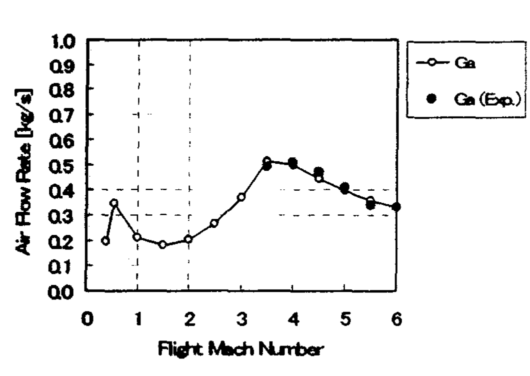

Fig. 21 Air Flow Rate of Exhaust Nozzle

Figure 22 shows designed temperature variation of the Hypersonic Ramjet Engine. Firing test results are also plotted in this graph. Firing test is conducted under Mach 3 condition. Intake exit temperature (Tie) is 600K and nozzle inlet temperature (Tni) is 2140K, on the test. Nozzle throat area (Ant) on the test is shown in Fig. 19. Heat resistant material of the ram combustor and cooling structure of the variable exhaust nozzle are proved under the designed temperature condition.

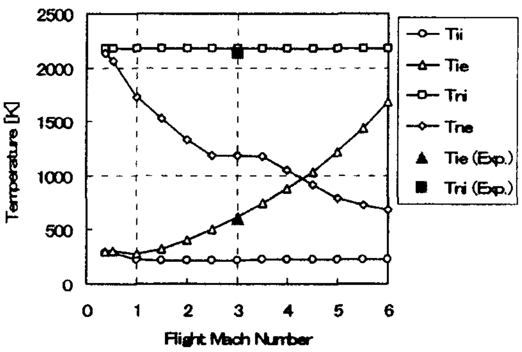

Fig. 22 Temperature Variation of HRE

Concluding Remarks

Variable air intake and variable exhaust nozzle for hypersonic engines are designed and tested in this study. As results, followings are obtained.

- Variable geometry shape which meets the requirements for pre-cooled turbojet engine is obtained with a design analysis.
- Hypersonic Ramjet Engine is designed as a scaled test bed for each component.
- An air intake model, a ram combustor model, and an exhaust nozzle model are fabricated based on the design study.
- Pressure recovery and mass capture ratio of the variable air intake at Mach 5 is obtained by a hypersonic wind tunnel test.
- Flow characteristics of the variable exhaust nozzle are obtained by a low temperature flow test.
- Wall temperature and heat flux of the variable exhaust nozzle at Mach 3 is obtained by a firing test.
- The intake and the nozzle are proved to be used at designed pressure and temperature environment.

Acknowledgements

The Authors acknowledge the contribution of Dr. Murakami (JAXA) and Mr. Enomoto (KHI) on the aerodynamic design of the variable air intake. The authors also acknowledge the contribution of Hypersonic Wind Tunnel Team and Continuous Ram Combustion Facility Team on the testing of the variable intake and the variable nozzle.

References

1) Taguchi, H. et. al., "Conceptual Study of Pre-Cooled Air Turbojet / Rocket Engine with Scramjet (PATRES)," ISABE 99-7024
2) Taguchi, H. et. al., "Airbreather / Rocket Combined Propulsion Systems Research for Japanese SSTO Spaceplane," AIAA 99-4811.
3) Taguchi, H. et. al., "Analytical Study of Pre-Cooled Turbojet Engine for TSTO Spaceplane," AIAA 2001-1838
4) Taguchi, H. and Futamura, H., "Engines for Hypersonic Airplanes," Japanese Patent 2003-107016, 2003.
5) Harloff, G. J. and Berkowitz, B. M., "HASA-Hypersonic Aerospace Sizing Analysis for the Preliminary Design of Aerospace Vehicles," NASA CR-182226, 1988.
6) Morimoto, T., "Design of CMC Inlet Vane for a Model Air Breathing Engine," Proceedings of the 27th Annual Cocoa Beach Conference on Advanced Ceramics and Composites, 2003
7) Anderson, Jr., J. D., "Fundamentals of Aerodynamics," McGraw-Hill International Editions, 2001.
8) Mattingly, J. D., "Elements of Gas Turbine Propulsion," McGraw-Hill International Editions, 1996.