AJCPP 2008
March 6-8, 2008, Gyeongju, Korea

Firing Test of Core Engine for Pre-cooled Turbojet Engine

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Keywords: Propulsion, Hypersonic, TBCC

Abstract

A core engine for pre-cooled turbojet engines is designed and its component performances are examined both by CFD analyses and experiments. The engine is designed for a flight demonstration of pre-cooled turbojet engine cycle. The engine uses gas hydrogen as fuel. The external boundary including measurement devices is set within 23 cm x 23 cm of rectangular cross section, in order to install the engine downstream of the air intake. The rotation speed is 80000 rpm at design point.

Mixed flow compressor is selected to attain high pressure ratio and small diameter by single stage. Reverse type main combustor is selected to reduce the engine diameter and the rotating shaft length. The temperature at main combustor is determined by the temperature limit of non-cooled turbine. High loading turbine is designed to attain high pressure ratio by single stage.

The firing test of the core engine is conducted using components of small pre-cooled turbojet engine. Gas hydrogen is injected into the main burner and hot gas is generated to drive the turbine. Air flow rate of the compressor can be modulated by a variable geometry exhaust nozzle, which is connected downstream of the core engine. As a result, 75% rotation speed is attained without hazardous vibration and heat damage. Aerodynamic performances of both compressor and turbine are obtained and evaluated independently.

Introduction

High speed air transportation system is desirable to enhance growth of world economy and mutual understanding of different cultures. JAXA has established a long term vision toward 2025[1)]. The vision proposes a development of a hypersonic airplane, which has a capability to fly over the Pacific Ocean in 2 hours. The airplane also has a capability to launch small payloads into low earth orbit by using upper stage rocket. 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[2,3)]. As a result, pressure ratio and equivalence ratio was found to be principal parameters to determine thrust to mass ratio and specific impulse.

Several hypersonic engines have been tested in Japan. ATREX engine has demonstrated pre-cooled engine technologies using liquid hydrogen. HYPR engine has demonstrated turbo-ramjet combined cycle including mode transition method. Small pre-cooled turbojet engine (Fig. 1) is under development as a successor of ATREX and HYPR engine. The engine has variable intake and nozzle to adapt wide flight Mach number range. The engine size is determined by the size of Ramjet Engine Test Facility (RJTF) at JAXA Kakuda Research Center. The engine will be tested both in the test facility and real flight environment.

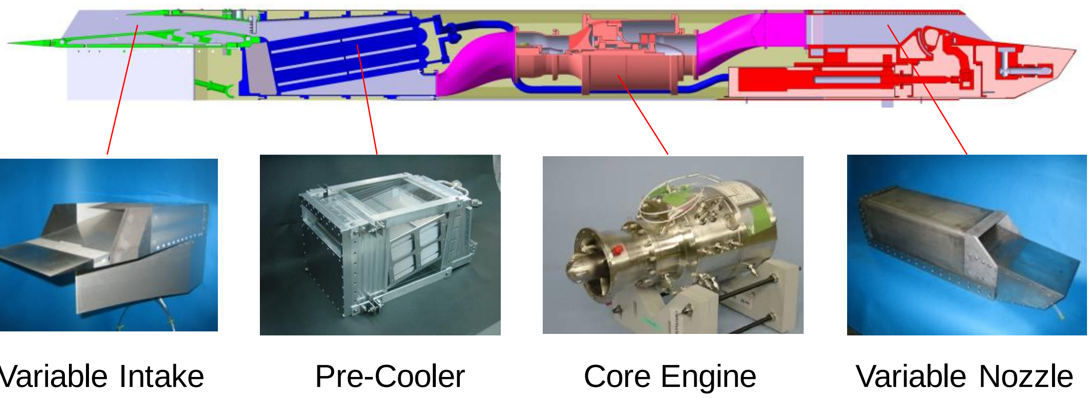

Fig. 1 Small Pre-Cooled Turbojet Engine

Components of a sub-scale pre-cooled turbojet engine for flight experiments are designed. Variable intake, pre-cooler, core engine, after burner, and variable nozzle are tested under simulated conditions. The engine is expected to operate from Mach 0 to Mach 5 by applying the pre-cooling system and using liquid hydrogen as fuel. Length of the engine is 2m. Mixed compression type compressor is designed by CFD analyses. Component test of the main combustor is performed using gas hydrogen. After the combustion tests of the engine components, total engine system is designed by 3 dimensional CAD. The weight of the engine satisfies the requirement of the flight test.

The scale of the engine is restricted by the vehicle size and the facility size. The engine length is limited to about 2.7m. The front area is set to 230mm in height and 230 mm in width. The weight is about 150kg. Variable intake is installed in front of the engine. Rectangular intake is selected to adapt the rectangular pre-cooler. The intake is designed to reduce the moving force by a pressure balance. Rectangular pre-cooler is used to obtain the same length of heat exchanger in the direction of airflow as the target engine. In the small engine, the cross sections of the pre-cooler should be larger than that of intake in order to attain enough thermodynamic performances. So, these are hidden downstream of an intake strut. The engine cycle is realized except for the external drag of the strut.

Mixed flow compressor is selected to attain high pressure ratio and small diameter by single stage. Reverse type main combustor is selected to reduce the engine diameter and the rotating shaft length. The temperature at main combustor is determined by the temperature limit of non-cooled turbine. High loading turbine is designed to attain high pressure ratio by single stage. Variable combustor nozzle with regenerative cooling wall is used as a principal thrust generator. After burner is integrated into the variable nozzle in order to reduce the engine length.

Performances of core engine components for a pre-cooled turbojet engine are evaluated. A mixed flow compressor and an axial turbine are designed and those performances are analyzed using CFD analyses. Performance maps including relations between pressure ratio and corrected mass flow rate are investigated through the rotational speed from 50% to 100%.

The propulsive performance of the pre-cooled turbojet has been shown by performance analyses. However, flight demonstration using actual engine has not been carried out. Then, a flight demonstration test is planned using a stratospheric balloon[4)]. In this plan, the flight experimental vehicle will be suspended under the stratospheric balloon. The vehicle realizes the supersonic flight as Mach 2 by falling freely. It acquires the performance of the engine in a real flight environment.

In this study, design and performance evaluation of core engine which is a component of a small pre-cooled turbojet engine are carried out. The small core engine composed of single stage mixed flow compressor, counter flow type hydrogen fueled combustor and single stage axial flow turbine is designed in order to perform flight demonstration of this engine cycle in the minimum scale. On the compressor, high load aerodynamic shape such as pressure ratio of 6 is assumed. The performance map is made by CFD analyses. On the turbine, high load aerodynamic shape with pressure ratio of 2.4 is assumed. The performance map is made by CFD analyses, as well as that of compressor. Those components are assembled into a core engine and tested using hydrogen fuel.

Design of Core Engine

The core engine of pre-cooled turbojet engine should operate in flight environment which greatly changes from low speed to hypersonic speed. There are some design requirements to be considered. Since inlet temperature increase with the rise in Mach number, the aerodynamic design of compressor has to include the effect of the change of tip clearance of rotor blades. To determine the tip clearance change, thermal expansion of the rotor has to be analyzed. In the hypersonic engine, it is necessary to minimize the front area of the engine in order to minimize the external drag. The precooler is installed in front of the compressor in order to extend the operation Mach number range by decreasing compressor inlet temperature. It is necessary to keep stable operation of the compressor with the circumferential change of the temperature formed at the exit of pre-cooler.

As a system for satisfying this required condition, the mixed flow compressor with the medium character of centrifugal compressor and axial flow compressor is adopted. The mixed flow compressor has high wing strength, relatively high air mass flow rate per front area. In case of the pre-cooled turbojet, the compressor entrance area can be decreased further than the intake entrance area, since the air density rises by the pre-cooler. Therefore, the largest diameter of compressor casing can be set larger than the compressor entrance diameter, in the restriction of intake front area.

On the combustor, gas hydrogen is used as fuel. Large fuel injection manifold is required in order to supply low-density gas hydrogen. The fuel manifold is placed in the downstream of core engine in order to keep outer diameter smaller than the intake width. Counter flow type combustor is adopted and the fuel is injected in the direction of upstream. On this combustor, combustion test[5)] is carried out simulating the combustor inlet condition from takeoff to Mach 5. The combustion temperature of 1223 K is achieved with almost uniform temperature using gas hydrogen fuel.

Since the compressor exit air becomes high temperature in the hypersonic engine, and the cooling effect by the air cannot be expected, the turbine is designed without cooling. Therefore, the turbine inlet temperature is set as 1223 K with temperature limit of non-cooled turbine.

Table 1 shows core engine design specifications. Figure 2 shows external view of core engine. The design rotational speed was set at 80000 rpm referring to bearing DN value of liquid hydrogen turbo-pump for the LE-5B engine. By adopting the counter flow combustor, the distance of compressor and turbine is shortened. The shaft length is shortened and critical speed of rotating parts exceeds design rotational speed. The air mass flow of 1.0 kg/s is adopted in order to utilize element test facility in JAXA.

Table 1 Core engine design specifications

Rotational Speed	80000	rpm
Air Mass Flow Rate	1.0	kg/s
Compressor Inlet Total Pressure	100	kPa
Compressor Outlet Total Pressure	600	kPa
Turbine Inlet Total Pressure	540	kPa
Turbine Inlet Total Temperature	1223	K

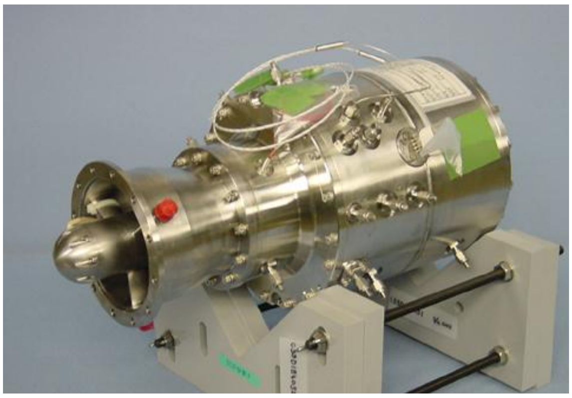

Fig. 2 Core engine

Design of Compressor

The mixed flow compressor with high load which satisfies the requirement of core engine is newly designed. In the basic design of the compressor, tip speed is set at about 550 m/s with the aim of getting 1.0 kg/s flow rate and 6.0 pressure ratio in single stage. In the design of the aerodynamic shape, basic form of stator vanes was set referring to the design technique of axial flow compressor. Tip speed of rotor blade was set using design technique of centrifugal compressors. Confirming the flow field by the CFD analysis, and repeatedly carrying out correction of the shape for avoiding the counter flow region, aerodynamic shape which satisfied the design requirement is determined. In the CFD analysis, turbo-machinery design software AxCent of Concepts NREC is used. Improved Daws Code is used as solver. Sparart-Almaras model is used as turbulence model.

Compressor shape which is obtained as a result of the design analysis is shown in fig. 3. The splitter is placed in the rotor in order to avoid the separation at rotor blade exit. Two stage stators are used so that velocity vector may become axial at the stator exit. The production shape was set by the execution of the structure analysis on the rotor disk. By using the heat-resistant titanium base alloy, it was confirmed that the structural strength could be ensured at the design rotational speed.

Fig. 3 Three-dimensional compressor shape

Design of High Load Turbine

Axial flow turbine with high load which satisfies requirement of core engine is newly designed. In the basic design of the turbine, 1.0 kg/s mass flow rate and about 2.4 pressure ratio are assumed as design target. And, the power over 250 kW which can drive the compressor in design point was considered to be a constraint condition. From the constraint of the layout of the core engine shown in fig. 2, the turbine is placed inside of combustor. Tip speed is set as 350 m/s at 80000 rpm design rotational speed. Since the temperature of compressor exit air becomes high at hypersonic flights, the structure is designed as non-cooled. Total temperature of 1223 K is used for entrance condition of design point. Assuming the pressure loss in the combustor as 10%, total pressure of 540 kPa is used.

In the design of the aerodynamic shape, reaction turbine, in which the flow is accelerated in both rotor blades and stator vanes, is adopted. Reaction turbine can obtain both high load and high efficiency. Performance and flow field is confirmed by CFD analyses. By correcting the shape to avoid strong shock waves, aerodynamic shape which satisfies the design requirement is obtained. As well as the analysis of the compressor, turbo-machinery design software AxCent is used. Improved Daws Code and Spalart-Almaras turbulence model is used.

Three dimensional view of the turbine obtained as a result of the design analyses is shown in fig. 4. Long code length is used in order to obtain high load with pressure ratio of 2.5 in the single stage. Small aspect ratio is used for the airfoil shape. Small bore ratio is used to obtain design mass flow rate with the small outer diameter, by the constraint of the shape of counter flow combustor. Tip speed and centrifugal force are small, since the outer diameter is smaller than the compressor. Then the stress level of turbine blades and disk is relatively low. The shape is set by the execution of structure analysis on the turbine disk. By using the heat-resistant nickel base alloy, it was confirmed that the structure with the enough strength margin at the design rotational speed can be realized.

Fig. 4 Three-dimensional turbine shape

Firing Test

Those engine components are assembled to a core engine and tested. The setup of the firing test is shown in fig. 5. Firing tests were carried out in both Combustion Test Facility and Noshiro Multi-Purpose Test Center in JAXA. Gas hydrogen is used for the core engine. Liquid hydrogen is used for the pre-cooler and after burner.

Air intake with pitot tubes is connected in the upstream of the core engine. Variable geometry exhaust nozzle is connected downstream of the core engine. The air flow rate is modulated with changing the throat area of exhaust nozzle. The initial rotation was made by a brushless motor with high-speed rotation type. In the experiment, the followings were measured: Rotational speed, outer shell vibration, total pressure distribution, static pressure distribution, temperature distribution, air mass flow rate and fuel flow rate.

Fig. 5 Set-up for Firing Test

Test Sequence and Conditions

Table 2 shows nominal test sequence. Rotation of core engine is initiated by an electric motor. Rotation speed of 5% (4000rpm) is sustained until -5 sec. Then, rotation speed is increased to 20% (16000rpm) until 0 sec. Injection of main fuel to the core engine is started at 0 sec. Fuel flow rate is gradually increased to keep the main burner temperature below the design limit of 1223K. At 90sec, liquid hydrogen is supplied to the pre-cooler and after burner. Pre-Cooler fuel is stopped at 110sec, and main burner fuel is stopped at 120sec.

Table 2 Nominal Test Sequence

Time	Event
	Starter Motor Start (5% Rotation)
-20 sec	Count Start
-5 sec	20% Rotation
0 sec	Main Fuel Supply (Gas Hydrogen)
20 sec	Starter Cut-off
90 sec	Pre-Cooler Fuel Supply (Liquid Hydrogen)
110 sec	Pre-Cooler Fuel Cut off
120 sec	Main Fuel Cut-off

Table 3 shows test cases described in this paper. Mechanical rotation speed of the core engine is about 50% of design speed for all cases. In Case 1, only core engine is operated. In Case 2, core engine and pre-cooler are operated. Combustion in the after burner is not happened. In Case 3, all the engine parts are operated. However, the test is stopped at 95sec because of a temperature limit at downstream of the engine. In all cases, exhaust nozzle is fully opened.

Table 3 Test Cases

	Rotation Speed	Core Engine	Pre-Cooler	After Burner
Case 1	50%	ON	OFF	OFF
Case 2	50%	ON	ON	OFF
Case 3	50%	ON	ON	ON

Rotation Speed

Figure 6 shows time history of rotation speed and corrected rotation speed for compressor in Case 2. Rotation speed is measured using gap sensors installed at the compressor rotor blades and main shaft. At -20sec, rotation speed is about 4000rpm. From -5sec, rotation speed is rapidly increased by electric motor. From 0sec to 20sec, rotation speed is increased by the power of electric motor and main burner. From 20sec to 80sec, rotation speed is increased by the power of main burner. At around 20sec, rotation speed exceeds 30% and started acceleration without help of electric motor. From 90sec to 110sec, liquid hydrogen is supplied to pre-cooler and after burner. At 120sec, supply of main burner fuel is stopped.

Corrected rotation speed is almost the same as mechanical rotation speed before 90sec. After the supply of liquid hydrogen, corrected rotation speed is increased in spite of almost constant mechanical rotation speed. This phenomenon is happened by the effect of pre-cooling. When the liquid hydrogen is supplied to the pre-cooler, air temperature at the compressor inlet is decreased and the corrected rotation speed is increased. In this experiment, maximum corrected rotation speed is about 75% (60000rpm).

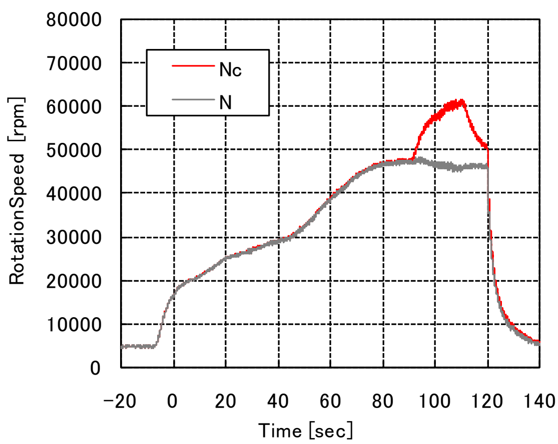

Fig. 6 Rotation Speed (Case 2)

Air Flow Rate, Fuel Flow Rate

Figure 7 shows time history of air flow and fuel flow in Case 2. Air flow rate is calculated using measured pressure data of pitot tubes installed at the exit of air intake. The accuracy of pressure sensor is not enough for low flow rate region, because the measured pressure level is very low comparing to the maximum range of the sensor. Tendency of air flow variation is similar to that of corrected rotation speed.

Fuel flow rate is measured by an orifice flow meter installed upstream of main fuel valve. There is a spike at about -3sec. This is caused by an initial leakage of flow control valve. Then, main valve is opened after the initial leakage is settled. From -3sec to 0sec, the fuel is supplied to a vent line.

At 0sec, fuel flow rate for ignition is set. Then, from 0sec to 20sec, fuel flow rate is increased to attain self acceleration condition. From 20sec to 40sec, fuel flow rate is set constant in order to make stable condition. In this period, rotation speed is increased by excessive power of turbine. From 40sec to 90sec, the fuel flow rate is increased again, to reach 50% mechanical rotation speed.

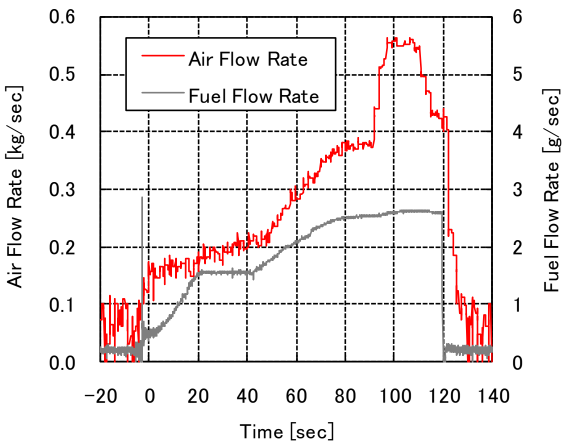

Fig. 7 Air Flow and Fuel Flow (Case 2)

Main Burner Temperature

Figure 8 shows time history of main burner temperature in Case 2. The temperature is largely increased from 0sec to 20sec. At 18sec, temperature increase is stopped in spite of increasing fuel flow. This may indicate the transition point to self accelerating condition. After 20sec, temperature is increased again in spite of constant fuel flow. At this time, power supply to the electric motor is stopped. Possible reason of the temperature increase is combustion efficiency is changing. After 40sec, temperature is increased as well as the increase of fuel flow and become stable condition.

From 90sec to 110sec, temperature is largely decreased because of pre-cooling. At this period, air flow rate is increased and fuel flow is constant. Then, equivalence ratio is reduced and combustion temperature is decreased. Air temperature at the inlet of compressor is largely reduced by pre-cooling. This also affect the decrease of main burner temperature.

From 110sec to 120sec, temperature is increased again. This is because the pre-cooling is stopped.

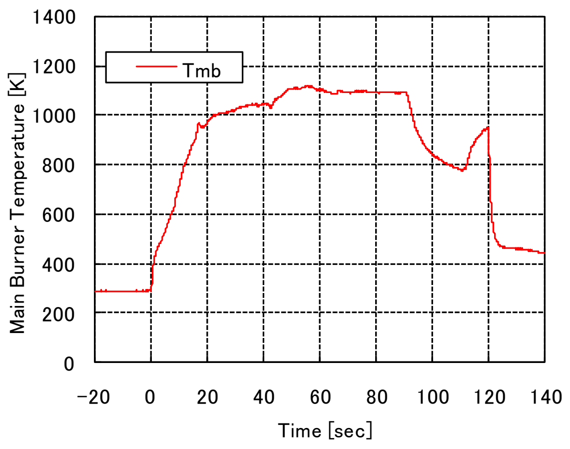

Fig. 8 Main Burner Temperature (Case 2)

Pressure Ratios

Figure 9 shows time history of pressure ratios for compressor and turbine in Case 2. Tendency of pressure ratio is similar to that of corrected rotation speed. Pressure ratios are gradually increased from -5sec to 50sec. Then, it is largely increased from 50sec to 90sec. This is because the effect of both acceleration rate of rotation speed and aerodynamic characteristics of compressor and turbine.

At 90sec, pressure ratios are rapidly increased by the effect of pre-cooing. This is also by the effect of aerodynamic characteristics at high corrected rotation speed. Maximum compressor pressure ratio is about 2.5 in this experiment.

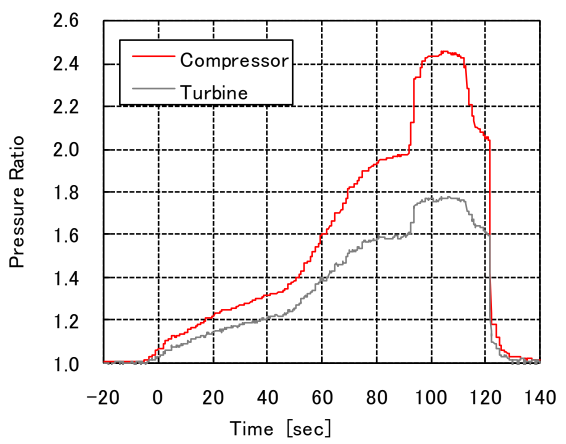

Fig. 9 Pressure Ratio (Case 2)

Aerodynamic Performances

Compressor Map

The relationship between compressor pressure ratio and corrected mass flow obtained by the CFD analyses is shown in fig. 10. At 100% rotational speed, the pressure ratio becomes over 6.0. On the assumption of the thermal expansion of the rotor blades, tip clearance is set as 0.4 mm in the analyses. The CFD analyses of 100%, 90%, 80%, 70%, 60% 50% and 30% are carried out in order to guess the performance map of the compressor. At 50% rotational speed, the pressure ratio is about 1.7. In either cases, there are small pressure ratio change as corrected mass flow changed, in comparison with axial flow compressors. This characteristic is close to that of centrifugal compressors. It seems to obtain stabilized operation, when corrected mass flow fluctuated in the rotational direction by temperature distributions of the pre-cooler exit.

Experimental results are shown with open marks. In Case 1, compressor operating line is in the middle of predicted operating range. When the corrected rotating speed is 50%, the operating point is below the line of predicted 50% line. In Case 2, corrected rotation speed reaches to 75% as shown in fig 6. Then, pressure ratio should be over 3.0 in thin case. However, the experimental value was about 2.5 and it was lower than the predicted value. It seems that pressure loss or surging has happened in the compressor. There was a temperature distortion in front of the compressor. It may cause the pressure loss or surging. In Case 3, experiment was stopped around 95sec, because the temperature around exhaust gas exceeded the safety limit. Then, corrected rotation speed is not so increased comparing to Case 2.

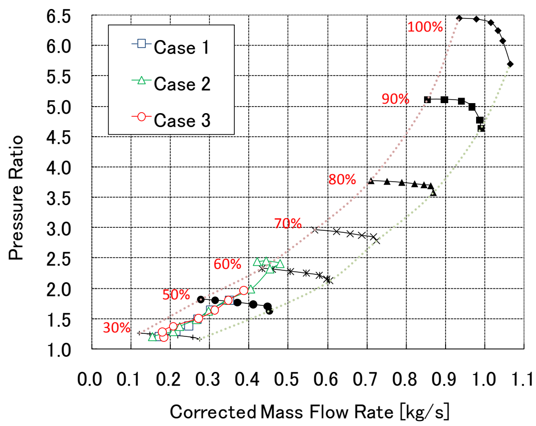

Fig. 10 Compressor map

Turbine Map

The relationship between turbine pressure ratio and corrected mass flow obtained in the CFD analyses is shown in fig. 11. At 100% rotational speed, excessive power of 250 kW is obtained at pressure ratio of 2.4.

The CFD analyses were carried out by changing the pressure ratio to make turbine performance map. Pressure ratio is changed by raising exit total pressure with fixed inlet condition. The flow in turbine stator vanes seemed to have reached the sound velocity when pressure ratio is over 1.8. There was no change of the corrected mass flow, even if the pressure ratio changed. In the meantime, mass flow rate decreases with the decreasing of pressure ratio under 1.8. The performance map was acquired at the partial load of 90%, 80%, 70%, 60% and 50% rotational speeds. At the low rotational speeds, the maximum pressure ratio is low. Tip speed of rotor blades is low and the relative speed of rotor entrance is high. Then, it seems that acceleration margin in the rotor blade exit decreases.

Experimental results are shown with open marks. In all cases, experimental values are a little smaller than predicted values. There is some fluctuation of experimental values. It is because the accuracy of pressure measurement is not enough for small pressure level of pitot tubes which is used to calculate mass flow rate. Maximum pressure ratio in this experiment was about 1.8. It will be increased by operating with higher rotating speed and compressor pressure ratio.

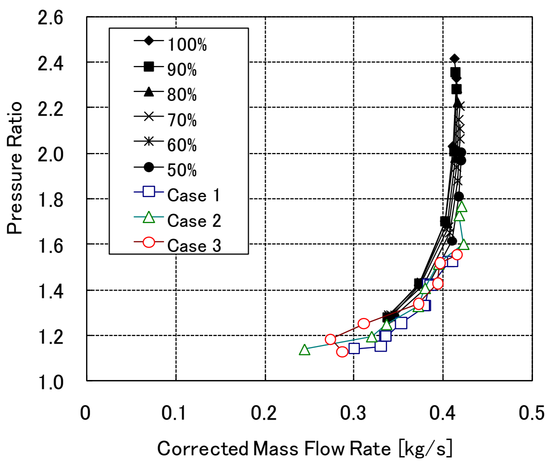

Fig. 11 Turbine map

Conclusion

Design and experiment of the core engine is carried out as a component of small pre-cooled turbojet engine for hypersonic flights. As a result, followings are obtained.

- The core engine can be accelerated by combustion gas when the rotation speed is over 30%.
- High load mixed flow compressor with pressure ratio of 6 is obtained by a CFD-base design method.
- Pressure ratio of compressor at 50% rotation speed is smaller than predicted value.
- High load axial flow turbine with pressure ratio of 2.4 is obtained by a CFD-base design method.
- Pressure ratio of turbine is a little smaller than predicted map.

Acknowledgements

The authors acknowledge the contribution of following researchers and organizations: Mr. Shimodaira of JAXA. Mr. Ohta, Mr. Okayasu, Mr. Fukutomi of ASIRI. Concepts NREC, INC engineering, University of Tokyo, Tokyo University of Science, Waseda University, Tamagawa University, Muroran Institute of Technology.

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