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System Analysis of a Gas Generator Cycle Rocket Engine

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Abstract

A system analysis program has been developed for a gas generator cycle liquid rocket engine of 30 ton class. Numerical models have been proposed for a combustor, a turbopump, a gas generator and pressure drop through a regenerative cooling system. Numerical algorithm has been validated by comparing with the published data of MC-1. The major source of error is not the numerical algorithm but the imperfect performance models of subsystems. So the precision of the program can be improved by revising the performance models using experimental data. The sea level specific impulse and vacuum specific impulse have been demonstrated for a 30 ton class gas generator engine. The optimal condition of combustor pressure and mixture ratio for specific impulse which is a typical characteristic of a gas generator cycle engine has been illustrated.

Key Words : Liquid rocket engine, Gas generator cycle, System Analysis, Optimal Condition, Specific Impulse

1. Introduction

A liquid rocket engine is defined as a machine generating thrust without propellant supplied from outside. The starting point of development is generating subsystem specifications satisfying the requirements from the higher level system. The objective of the present study is to develop a system analysis program of a gas generator cycle engine which is composed of a combustion chamber to generate thrust, a turbopump to pressurize propellant, a gas generator to drive a turbine, a regenerative cooling system and etc. When the required rocket engine is completely new, there is no available performance information of subsystems to analyze the engine system performance. In this case, mathematical models of subsystems performance are required. These models are mathematical relations between design variables and the subsystems performances which are theoretically calculated or experimentally estimated. The system analysis gives the system performance from the combination of lower level components performances. The major results are thrust, specific impulse, thrust to mass ratio, envelope and etc. The development periods or budget is sometimes predicted as a system analysis results. In the present study, the system analysis results are confined as purely technical performances. Though the system analysis of a liquid rocket engine is widely used, it is hard to find the analysis method which is applied to a real rocket engine (see chapter 3.1 for related contents). The present paper presents the verified analysis method and the application example for optimal operating condition for sea level specific impulse and vacuum specific impulse.

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2. Literature survey

A systematic analysis of liquid rocket engine performance has been reported by O'Brien and Ewen[1]. They estimated the system performance for various combinations of engine cycle, propellant and cooling concept. They compared the performance of gas generator cycle and staged combustion cycle. They considered RP-1 and CH3 as fuel and RP-1, LH2 and LCH4 as coolant. The optimal design of a rocket engine system is decided by comparing the single index defined by engine performance, engine mass, pump exit pressure, combustion pressure, mixture ratio, turbine temperature, coking, IPS (Inter-Propellant Seal), coolant requirements with respective weighting factor. NASDA reported the prediction method for LE-5[2]. The error of turbine exhaust mass flow rate is 3% and the errors for other parameters are less than 1%. McHugh[3] presented empirical correlations to estimate the engine envelop, engine mass, propellant flow rate, and turbopump head and power. The input variables are propellant combination and engine thrust to generate major specifications of a new engine. The basis of this method is the statistical correlation of previous engine design database, and the error of the results is 10% to 20%. The database is composed of 14 existing engines for around 30 parameters. The considered engines are HM7B, RL10A-3-3A, LE-5, S-4(MA-3), LR91, 5C, H-1, HM60, J-2, RS-27, LE-7, RL87, SSME and F-1. The major parameters are thrust, specific impulse, combustion pressure, area ratio, mixture ratio, engine length, exit diameter, dry mass, design (old/new), L*, contraction ratio, injector pressure drop, mass flow rate, pump pressure rise, number of pump stages, pump efficiency, number of turbine stages, pressure ratio, turbine efficiency, turbine power and gas generator mass flow rate/ temperature/combustion pressure/mixture ratio. SCORES (SpaceCraft Object-orient Rocket Engine Simulation)[4] is an analysis tool for a launcher and a spacecraft. Kauffmann et al.[5] presented engine system analysis for large a thrust Lox/kerosene engine. The subject of the study is MC-1 engine of gas generator cycle. Aerojet reported the integrated analysis tool to reduce computational cost[6]. This method spends only 5~6 minutes while the conventional method consumes 4 hours for the same problem.

3. Analysis Methods

3.1. Specific Impulse

The specific impulse is the indicator of efficient propellant consumption. The engine thrust of gas generator cycle is summation of combustion chamber thrust and turbine exhaust nozzle thrust. The mass flow rate of the turbine can be calculated from required pump power. The total mass flow rate of the engine is determined from turbine mass flow rate and combustion chamber mass flow rate. Fig. 1 is a schematic of a gas generator cycle engine composed of a combustion chamber, a gas generator, a turbine, pumps and etc. Fig. 2 describes the algorithm to obtain converged mass flow rate.

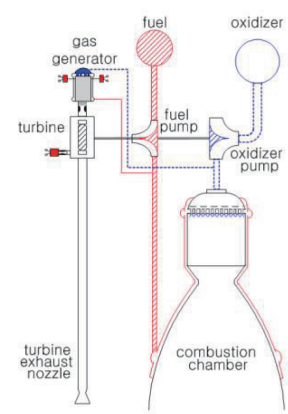

Fig. 1 Schematic of a gas generator cycle engine[7]

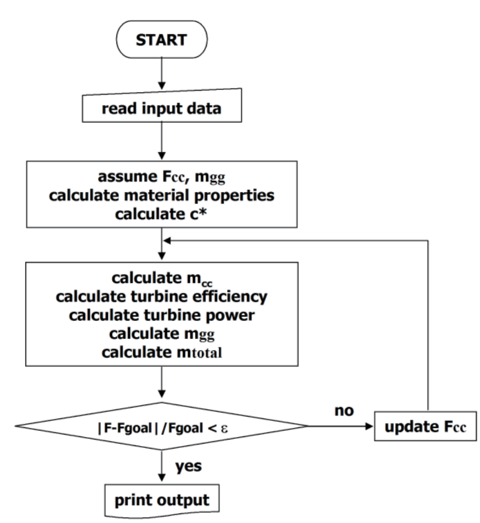

Fig. 2 Algorithm for converged mass flow rate[7,8]

The algorithm shown above has been applied to the liquid rocket engines of KSLV-II[7-9] and are verified by comparing the performance of MC-1[10]. The analysis error does not come from the numerical algorithm but from inaccurate performance models of subsystems[7,8].

3.2. Performance Model for Combustion Chamber

The combustion chamber generates thrust from chemical reaction of oxidizer and fuel. Propellants are mixed at combustion chamber and then burn to generate high pressure and temperature gas. Finally, the combustion gas accelerates to super-sonic flow through a converging-diverging nozzle. The performance model corrects the theoretical performance by considering combustion efficiency and nozzle efficiency. In the present study, nozzle efficiency is a function of expansion ratio and the combustion efficiency is a function of mixture ratio. The combustion chamber performance is given by Eq.(1)~Eq.(4).

$$I_{SP} = \frac{c^* C_f}{g} \quad (1)$$

$$c^* = \frac{(p_c)_{ns} A g}{\dot{m}_{tc}} \quad (2)$$

$$c^* = \eta_{c^*} c^*_{th} \quad (3)$$

$$C_f = \eta_{C_f} C_f^{th} \quad (4)$$

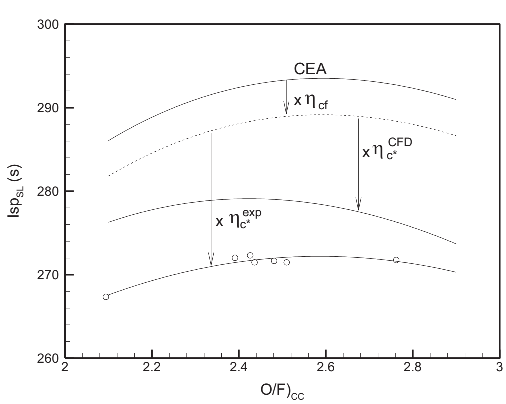

Fig. 3 Correction concept of combustor efficiency

Theoretical performance is calculated by using CEA with contraction ratio 3 for finite area condition. This value is higher than the real performance because no fluid friction is considered and the perfect mixing is assumed. The ideal performance is corrected by two steps. First flow loss is corrected by introducing η_{C_f}. Next incomplete combustion effect is considered by η_{c^*} which corrects the combustion loss to meet CFD value or experimental value according to the correcting efficiency. In this study correction is conducted to meet the experimental value[11,12]. Fig. 3 describes the combustor efficiency model. The hollow symbols in the figure present measured data.

3.3. Performance Model for Turbopumps

The pump efficiency is given as a function of specific speed[13] defined by Eq.(5).

$$\Omega_s = \frac{\Omega\, Q^{0.5}}{(g \times \Delta h / N_{stage})^{0.75}} \quad (5)$$

The inducer is considered in this paper to reduce cavitation. The oxidizer pump for liquid oxygen and the fuel pump for kerosene are considered as connected by a single shaft. Both pumps are single stage. However sometimes multi-stage pumps are required for engines operated under higher pressure. This can be implemented by upgrading subsystem performance models. The efficiency of turbine is given as a function of the ratio of isentropic spouting velocity and turbine blade velocity. Eq.(6) and Eq.(7) describe the turbine performance.

$$\eta_{TB} = f(u/C_0) \quad (6)$$

$$C_0 = \sqrt{2\frac{\gamma}{\gamma-1} R T_i \left[1 - \left(\frac{1}{\pi}\right)^{\frac{\gamma-1}{\gamma}}\right]} \quad (7)$$

3.4. Performance Model for Gas Generator

The gas generator produces working fluid to drive a turbine. The gas product temperature should be moderate not to damage the turbine material. Because the turbine for a rocket engine is operated usually without cooling. The typical combustion condition of the gas generator is extremely fuel rich to meet moderate combustion temperature. The equilibrium analysis for combustion is unsatisfactory for this condition. So experimental data is used for the performance model. The molecular weight, combustion temperature and specific heat ratio are modeled as a function of mixture ratio and combustion pressure. Eq(8)~Eq.(10) are given to model the gas generator performance.

$$M = f\left(\frac{O}{F}, p_{GG}\right) \tag{8}$$

$$T_{GG} = f\left(\frac{O}{F}, p_{GG}\right) \tag{9}$$

$$\gamma_{GG} = f\left(\frac{O}{F}, p_{GG}\right) \tag{10}$$

3.5. Required Pump Head Model

The objective of pumps of a liquid rocket engine is to pressurize the propellant. It should be delivered to a high pressure combustor overcoming pressure drop through the combustion chamber cooling passage, supply pipes and valves. The pressure drop through the cooling passage is the major part among pressure drop factors. So the required pressure difference along the regenerative cooling passage should be given for system analysis. The present study developed the pressure drop model considering the published data[1, 11]. The present model gives pressure drop through the cooling passage considering combustion pressure and mixture ratio. The thermal resistance of carbon deposit, liner material and coking temperature of coolant may be considered for more accurate prediction. Fig. 4 shows the required pressure drop of regenerative cooling. The parabolic increase of pressure drop through the regenerative cooling passage with respect to combustion pressure agrees qualitatively with the published data[1, 11]. The pressure drops more for higher mixture ratio because the combustion product temperature increases as the mixture ratio approaches to equivalent ratio 1 condition.

Figure 5 depicts the sea level specific impulse. The difference of each contour line is 1 s. Some levels are omitted in the figure for better readability. The vacuum thrust of the target engine is 30 ton class. The engine is considered as a booster stage so the area ratio is constrained to meet the minimum exit pressure 0.06 MPa. A typical range of exit pressure for booster stage engines is 0.04 MPa to 0.1 MPa. The combustor performance increases as the combustion pressure increases or mixture ratio increase. However more cooling capacity meaning cost in performance is required. So optimal condition exists for gas generator cycle engines while monotonic increase performance for staged cycle engines along combustion pressure. The pressure drop through the regenerative cooling passage is proportional to the square of the combustion pressure and the propellant consumption of the turbine is linearly proportional to the pump power. While the increase of performance of the combustor is less than linearly proportional to combustor pressure and soon saturated. After certain pressure the performance gain by increase of combustor pressure is negligible. This is the major rationale that optimal combustor pressure exists for gas generator cycle engines. However, the optimal combustor pressure depends on the performance of the subsystems. So optimal combustor pressure is not universal and changes to the individual design of the engine.

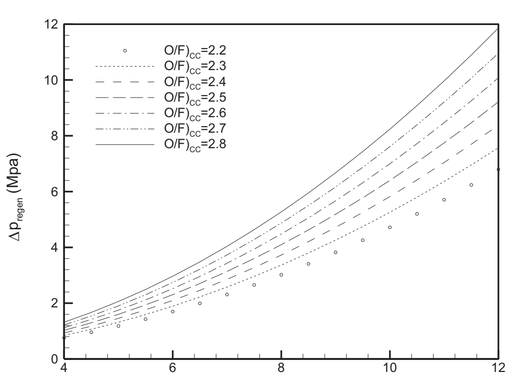

Fig. 4 Pressure drop of regenerative cooling

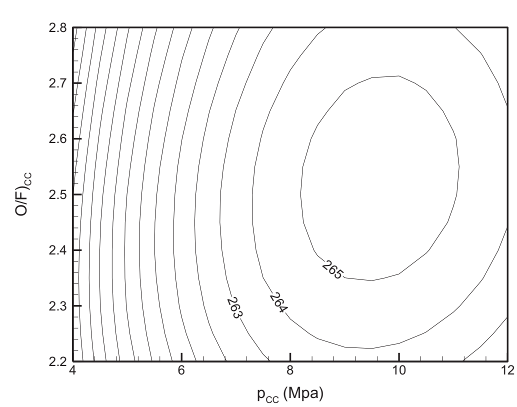

Fig. 5 Sea level specific impulse (unit: s)

4. Results and Discussions

Figure 6 shows the vacuum specific impulse distribution. Vacuum specific impulse increases as the area ratio increases. The vacuum specific impulse increases slowly for high combustor pressure condition because of the confined area ratio condition for booster stage engines. The total impulse which is proportional to the combination of sea level specific impulse and vacuum specific impulse during operating time determines the optimal condition of a booster stage engine. So the vacuum specific impulse should be given for stage design. In Fig. 6 the maximum vacuum specific impulse condition locates at a moderate combustor pressure because the present area ratio is limited to meet the exit pressure constraints for booster stage application as explained in the previous paragraph. For an upper stage application, the exit pressure constraint is meaningless and the optimal pressure will be higher than that is given in Fig. 6. On the other hand, the exit diameter can be a constraint to meet the stage envelop instead of exit pressure.

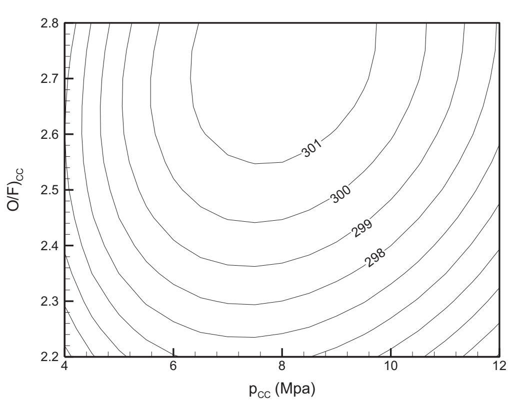

Fig. 6 Vacuum specific impulse (unit: s)

5. Conclusions

A system analysis method has been given for a 30 ton class gas generator cycle engine. Numerical models are described for a combustor, turbopumps, a gas generator and the pressure drop through a regenerative cooling passage. As a sample result, sea level specific impulse and vacuum specific impulse has been presented. The combination of the benefit of combustion performance and the cost of cooling with respect to combustion pressure and mixture ratio gives the optimal condition for sea level or vacuum specific impulse.

Nomenclature

A	area
I_{SP}	specific impulse
C_0	isentropic spouting velocity
C_f	nozzle coefficient t
N	number of stages
M	molecular weight
p_c	chamber pressure
Q	volume flow rate
R	gas constant
T	temperature
c^*	combustion characteristic velocity
g	gravitational acceleration
$\dot{m}$	mass flow rate
Ω	rotating speed
Ω_s	specific speed
γ	specific heat ratio
π	pressure ratio
η	efficiency

Superscript/subscript

h	head
ns	nozzle stagnation
C_f	nozzle coefficient
c^*	combustion characteristic velocity
tc	thrust chamber
th	theoretical value

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