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Multi-objective Particle Swarm Optimization Design an Optimal Liqui-Rocket Engine

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Abstract: A liquid rocket engine has an open gas generator cycle that uses LOX/RP-1 (kerosene). The engine is designed to com-pute the size of the combustion chamber pressure, nozzle expansion ratio, and mixture ratio O/F. Therefore, this study proposes particle swarm optimization to determine the values of these three variables for an optimal engine. In this work, the engine's design was obtained through multi-objective optimization. Multiple targets presented the energy balance condition. The first function was the setting of the specific impulse (I_sp), and the second function was the thrust-to-weight ratio (T/W). The physical properties of the combustion chamber were modeled on the basis of typical parameter val-ues. The simulation results demonstrated that the algorithm gave a specific thrust increase of more than 3.5%, and the T/W increased by up to 20%. Furthermore, Pareto frontier lines were obtained for various thrust.

***Keywords**: Liquid Rocket Engine, Multi-objective Optimization Design, Particle Swarm Optimization*

1. Introduction

Liquid rocket engines in gas generation are related to projectile works [1]. A liquid rocket engine is a system com-posed of a combination of various parts, such as a combus-tion chamber, a turbine, and a turbo pump [2]. The design of each of these parts has complex variables interacting with one another. The performance of a liquid rocket engine is impulsive, and it can be expressed as a specific impulse (I_sp) and a thrust-to-weight ratio (T/W). The mixture ratio O/F, mass flow rate, and combustion chamber pressure significantly influence the particular thrust of a liquid rocket engine. Meanwhile, the combustion chamber pressure and nozzle expansion ratio affect the thrust-to-weight balance, thus considerably influencing the machine's weight. In liquid rocket engines, many design variables affect one another; as such, finding optimal design variables is challenging because each part can use various analysis methods [3]. Therefore, a technique for modularizing and analyzing each component, which is expressed as an input/output variable, is used. In the first stage of the liquid rocket engine design, appropriate de-sign parameters that meet the system requirements must be determined [4]. In the past, design variables were set by predicting performance on the basis of experimental data [5]. However, since the 1980s, various system design techniques and optimization techniques have been used. Then, since the 2000s, much research on program development using multidisciplinary optimization has been con-ducted. To identify the optimized design for the initial stages, several aerospace institutes in the United States or China, Korea, and Russia have been developing specific software programs that can provide parameters that produce the best performance, launch the vehicle of the upper sys-tem, conduct a reliability analysis, and predict the cost of production [6]. In general, although the ANSYS or SOLD-WORK programs have a simple entry that can design the variables of the fuel, cycle, propellant, required thrust, and design altitude, finding the optimal performance for each component of the engine is challenging [7].

Hargus et al. estimated the weight of a liquid rocket engine by presenting the weight of the components as a function of the combustion chamber pressure on the basis of the data of the actual liquid rocket engine [8]. In their study, Hargus's weight evaluation method was applied. The weight of the thrust chamber was estimated using the response plane method from the graph expressed as a function of nozzle expansion ratio, thrust, and combustion chamber pressure. Meanwhile, Gradl and Peter [9] demon-strated that the weight of the entire engine can be determined as a function of the nozzle expansion ratio, combustion pressure in the main thrust chamber, and required thrust. Recently, engineers have developed many projects and found the optimal design for gas generator cycles us-ing genetic algorithms using the MATLAB program. The engineering can quickly obtain the desired performance design variables [10]. Evolutionary techniques, genetic algorithms, and particle swarm optimization can find solutions only with function values suitable for system optimization programs composed of module programs [11]. In this study, each module was composed of a simple analysis program using particle swarm optimization. The program was configured for later application in module analysis programs. This study aimed to determine the optimal de-sign parameters with specific thrust and thrust-to-weight ratio as the multi-objective function for the gas generator cycle using particle swarm optimization.

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2. Liquid Rocket Engine

2.1 Combustion

In the system analysis process, the sub-modules of the liquid rocket engine were composed of a thrust module that includes a combustion chamber and a nozzle, a turbo pump, a turbine, a gas generator (pre-burner), a valve or pipe, and a supply system. The analysis program for the liquid rocket engine system was configured to calculate the fuel's required mass flow rate, pump power, turbine-generated power, and specific thrust by inputting the central thrust chamber combustion pressure, O/F ratio, nozzle extension ratio, and required thrust. Fig. 1 shows the schematic diagram of the main combustion chamber.

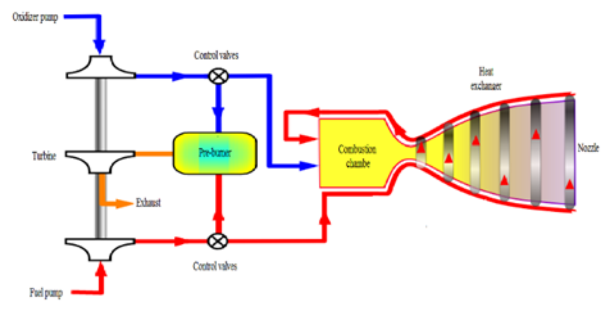

Fig. 1 Schematic diagram of Main Combustion Chamber.

2.2 Chamber and Nozzle

Inside the combustion chamber of a rocket engine, heat and pressure have been converted to kinetic energy. The optimal design performs as high as possible by managing the heat, temperature, and pressure. However, the challenge is how to contain and control the high temperature inside the combustion chamber due to the change in the shape of the nozzle (convergent-divergent). The nozzle has three regions (impinging, mixing, recovery of swirling, and re-separation), as shown in Fig. 2.

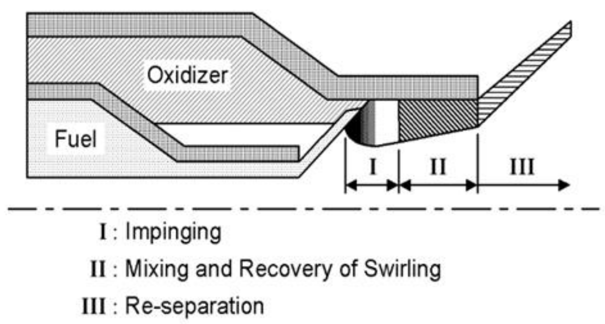

Fig. 2 Construction of Construction Chamber

2.3 Modeling of Liquid Rocket Engine

To determine the mathematical expansions of the nozzle device, we must define m'MC as the mass flow rate of the main thrust in the combustion chamber. m'MC can be obtained through the relational expression of the non-injection IspMC with its main thrust corresponding to the demand of the room FMC, as shown in Eq. 1. The value of the gravitational acceleration (g = 9.8 m/sec2) is fixed, and the equation is expressed as follows:

$$m_{MC} = \frac{F_{MC}}{Isp_{MC}\, g} \qquad (1)$$

For thrust coefficient Cf and characteristic speed c*, the IspMC can be expressed as shown in Eq. 2. The characteristic speed c* can be obtained from Eq. 3 and Eq. 4.

$$Isp_{MC} = \frac{c^*}{g} C_f \qquad (2)$$

$$c^* = \frac{\sqrt{\gamma R T}}{\gamma \sqrt{\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma-1}}}} \qquad (3)$$

$$C_f = \sqrt{\frac{2\gamma^2}{\gamma-1}\left[\frac{2}{\gamma+1}\right]^{\frac{\gamma+1}{\gamma-1}}\left[1-\left(\frac{p_e}{p_c}\right)^{\frac{\gamma-1}{\gamma}}\right]} + \varepsilon\left(\frac{p_e - p_a}{p_c}\right) \qquad (4)$$

where ἐ is the non-heat ratio, γ is the gas constant, and R is the pre-nozzle at temperature T. In addition, Pe is the outlet pressure, Pa is the external pressure, and pc is the combustion chamber pressure. Enthalpy is the multiplication of volume, pressure, and internal energy. As such, it is the sum of all the energy of that system. The higher the enthalpy in the system is, the more potential it has to perform work. e is the nozzle outlet area divided by the nozzle throat area. γ and R can be obtained using CEA2, NASA's propellant chemical reaction program [12]. If CEA2 is executed every time the calculation is performed, it takes much time. By making the interval of the input variables dense when calculating through linear interpolation, the relative error was locked with CEA2 and did not exceed 10 -4

Table 1. Properties of Gas-generator.

TIT (K)	C_P	γ	R (KJ/Kg)	O/F
900	2.679	1.101	244.87	0.324
950	2.705	1.111	261.54	0.354
1000	2.723	1.118	277.24	0.387
1050	2.739	1.126	294.26	0.418
1100	2.754	1.134	314.24	0.450
1150	2.764	1.141	328.47	0.482
1200	2.772	1.148	344.39	0.516

The propellant was calculated for a liquid oxygen oxidizer and RP-1 fuel, and the range of O/F ratio was 1.5 to 3.5. The content of the combustion chamber pressure was set to 5 MPa or more and 30 MPa or less.

$$[T\ \ \gamma\ \ R] = CEA2(p_c OF_{MC}) \tag{5}$$

Nozzle outlet pressure Pc is the external atmospheric pressure set by the user, and Pe is the nozzle expansion equipment combustion chamber pressure. It was received as a design variable in the optimization program and calculated through iterative calculation. The iterative formula in Eq. 6 was used.

$$\frac{p_e}{p_c} = \varepsilon^2\ \frac{\gamma+1}{\gamma-1}\left(\frac{2}{\gamma+1}\right)^{\frac{2}{1-\gamma}}\left(1-\left(\frac{p_e}{p_c}\right)^{\frac{\gamma-1}{\gamma}}\right)^{-\frac{\gamma}{2}} \tag{6}$$

Turbine-generated power and pump-generated power are presented in Eq. 7 and Eq. 8, respectively. The required outlet pressure of the pump was calculated as the sum of the pressure loss in the combustion chamber and the supply system parts. The loss pressure of supply system components, such as pipes and valves, was assumed to be the values in the reference literature. The values for the specific speed and RPM were used for the efficiency of turbines and pumps [13].

$$L_T = m_T c_{PT} \eta_T T_{Tin}\left(1-\left(\frac{1}{P_r}\right)^{\frac{\gamma-1}{\gamma}}\right)^{-\frac{\gamma}{2}} \tag{7}$$

$$L_P = \frac{m_P\ \Delta P_P}{\rho_P\ \eta_P} \tag{8}$$

$$\dot{m}_{O_{MC}} = \dot{m}_{MC} \times \frac{OF_{MC}}{OF_{MC}+1} \tag{9}$$

$$\dot{m}_{F_{MC}} = \dot{m}_{MC} \times \frac{1}{OF_{MC}+1} \tag{10}$$

$$\dot{m}_{O_{gg}} = \dot{m}_{gg} \times \frac{OF_{gg}}{OF_{gg}+1} \tag{11}$$

$$\dot{m}_{F_{gg}} = \dot{m}_{gg} \times \frac{1}{OF_{gg}+1} \tag{12}$$

where LT and Lp are the power of the turbine and pump, respectively; m is the mass flow rate, cp is the specific heat at constant pressure, (η) is the overall efficiency, TTin is the turbine inlet temperature, (ρ) is the density, subscript T is the turbine, and P represents the pump. The properties of the turbine were considered the same as those of the gas generator. Furthermore, the data for the turbine inlet temperature in Table 1 were used for the properties of the gas generator. The pump calculated the oxidant pump and the fuel pump. The mass flow rate in each pump could be obtained through the mass flow rate and O/F of the main thrust chamber and gas generator.

Table 2. Pressure Losses of Supply Components.

Oxidiser Parts	Pressure Loss (% of pc)	Fuel Parts
Injector	20.0	20.0
Pipe Lines	2.5	1.0
Main Valve	3.5	1.5
Torus Dome	15.0	
Cooling Jacket		27.0
Calibration Orifice		11.0

The pump's required power Lp and the turbine's generated power LT must be the same and T/W. This scenario is called the energy balance condition. These values include the gas generator mass flow rate m'gg as a variable. Therefore, an appropriate m'gg must be determined. In this study, the gas generator mass flow rate was determined using the bisection method, which is an explicit method for convergence.

Table 3. Result of Code Verification.

Parameter	Result	Ref.[12]	Unit	Diff.
Thrust	3,3	3,336	kN	0.00 %
Isp	275	262.4	S	+3.92 %
T/W	99	100.0		-1 %
Mainstream flow	1,190	1,255.5	Kg/s	-0.91 %
Gas generating flow rate	44	41.73	Kg/s	+9.4 %
Pump request power	19	19.86	MW	-4.36 %
Turbine creation power	19	19.86	MW	-4.36 %

The supply system parts used pressure loss as a performance factor, and the pressure loss was assumed at a constant rate of combustion pressure Pc in the main thrust room. On the basis of Reference [11], the pressure loss in each part was determined, as shown in Table 2. Eq. 13 is the conclusion of Reference [10]

$$w_{engine} = f(\varepsilon\ \ P_c\ \ T_{req}) \tag{13}$$

2.4 The System Model

For most actual engine data, detailed information other than significant performance, such as the performance of thrust, specific thrust, and nozzle expansion ratio, was generally treated as confidential. Therefore, to verify the code, a relatively detailed analysis data compared the results calculated in the condition of a sea-level standard, which had a combustion chamber pressure of 68.9 bar (1,000 psi), required thrust of 3.3 kN (750,000 lb), and external conditions [12]. Liquid oxygen (LO2) and RP-1 were used as fuel, and the Isp ratio was set at 275. As shown in Table 3, the combustion chamber pressure was fixed at 1,000, and the nozzle extension ratio was. Each performance was related to the other. When one value changed, different values also changed in the chain. Thus, the ac-curacy of the program was difficult to judge by looking at the size of the error in a specific part. Nevertheless, the interpretation method of each part was different, and an accuracy of 5% could be considered high. The different results were due to the variations in the method of calculating the combustion gas properties in the combustion chamber and the efficiency of the turbine and pump. The pressure of the combustion chamber was generally designed to be less than 27.5 MPa (about 4,000 psi) in the case of a gas generator. Moreover, the search range was set to search up to 30.0 MPa with a slightly larger value. The nozzle expansion ratio and O/F ratio were set between 5 and 35, thus reflecting the general design range when LO2 and RP-1 were used in a sea-level standard. The turbine inlet temperature was set to 1,000 Kilven, and the pump and turbine rotation speed was assumed to be 7,000 RPM bearing a single-axis system. The main performance variables of the rocket engine, a representative engine of the gas generator cycle, were entered, and the results were compared. In addition, the combustion pressure was 70 bar, the nozzle expansion ratio was 2.5%, the pump ratio O/F was 16, and the turbine speed was 5,000 RPM. The results were obtained and given an error difference of less than 3% of the thrust-to-weight ratio, similar to the error seen in [7]. The thrust-to-weight ratio showed a big difference because the material or structural reinforcements were not considered in the weight estimation. This point can be found in foreign programs.

3. Matlab Simulation Design

To analyze the system and find the optimum design, all components defined in section 2 were connected. The constants, gains, mathematical model, construction, and the block diagram could be simulated using MATLAB, as shown in Fig. 3

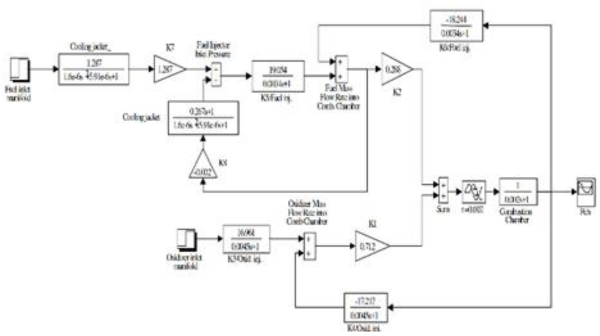

Fig.3 Thrust chamber block diagram representation.

4. Particle Swarm Optimization

In 1995, Kennedy and Eberhart proposed particle swarm optimization (PSO). It is an algorithm designed on the basis of biocommunity, which is the social behaviour of animals, such as birds and fish, and differs from natural selection evolution, including genetic algorithm. PSO has similar processing characteristics and performs optimization using the concepts of swarm and particle [14]. Each randomly formed particle in the initial swarm moves through the dimensional space up to the number of parameters that the user selects to find the optimal value. After the search of the initial swarm is over, from the next swarm, each particle remembers the value of the objective function for the position and finds the optimal solution by sharing the deal with other particles [15]. Given this feature, the initial global search changes to a local search as generations pass and finally converges to a single point to determine the optimal solution. Compared with other algorithms, the optimal convergence value can be found at a quick and stable rate with excellent computational efficiency. In the PSO algorithm, the swarm has n particles designated by the user, which are a set of parameter values selected by the user and are characterised by sharing parameters and objective function values among particles and moving individually. The equation of the PSO algorithm is as follows (Eq. 16, Eq. 17, Eq. 18, and Eq. 19).

$$x_{\_k} = \left(x_{\_k.\ 1}\ x_{\_k.2}\, x_{\_k.3} \ldots \ldots\ x_{\_k.\ n}\right) \qquad (16)$$

$$P_{\_best\,k} = \left(P_{\_best\_\,k.1}\ P_{\_best\_k.2}\ \ P_{\_best\_\,k.3} \ldots P_{\_best\_\,k.n}\right)(17)$$

Several researchers consider the velocity equation as an essential part of how the PSO works [16] and can be calculated using Eq. 18. This equation clarifies that adjusted speed depends on numbers that the calculator selects randomly and the distance between ($P_{\_\ best\_\,(k,\,d)}$)) and ($g_{\_\,(best\_d)}$)). The current velocity can adjust the position of each particle using Eq. 18.

$$V_{\_k.m}^{(Iter.+1)} = W\, V_{\_k.m}^{\ (Iter.)} + c_1\, r\left(P_{\ \_best\ \_k.m} - x_{\ \_k.m}^{\ (Iter.)}\right) + c_2\, r\left(g_{\ \_best\,m} - x_{\ \_k,m}^{\ (Iter.)}\right) \qquad (18)$$

$$x_{\_k.m}^{(Iter.+1)} = x_{\_k.m}^{(Iter.)} + v_{\ \_k.m}^{\ (Iter.)} \qquad (19)$$

where (k) is the number of particles, and K is {.1, .2, .3 ….,}. The space of dimensions is m, and m is {.1, .2, .3 …}. {Iter.} is the iteration of the pointer in Eq. 18 and Eq. 19. {C1, C2} are the acceleration constants. (r1 and r2) are arbitrary ran-dom variables whose values are taken from (0) to (1). (w) is a factor referring to inertia weight. The kth particle at itera-tion (Iter.) has a velocity of (Vk,mIert) and a position of (Xk,mIter). The previous best position of the kth particle is (P_best.). All the swarms have only one (G_best), which is the global best position that refers to the desired solution. Each module program receives input variables from the system analysis module, performs calculations, and delivers output variables that must meet the optimized values. Fig. 4a consists of the system analysis process, and Fig. 4b explains the optimization process. This process is repeated to search for design variables, as shown in Fig. 4c. These processes are used to find the optimal value of the desired objective function.

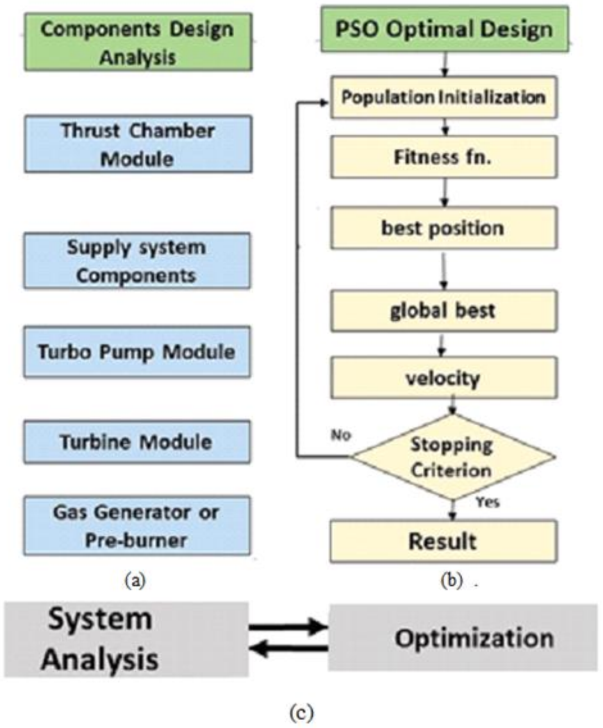

Fig 4. Program Structure

As an optimization method, an actual code genetic algorithm was used as a PSO. In addition, a tournament selection method was used as a reproduction operator, and search efficiency was increased by using an elite strategy that preserves a group close to optimal during a search. The initial group was set to 20 particles, and the maximum number of generations was set to 100 iterations.

5. Multi-Objective Optimization

Particle swarm optimization effectively searches for global solutions. It can be applied even if the module analysis process changes because it requires only one output value information from many input values. However, given the disadvantage of repeatedly performing many calculations, each module program uses an approximation technique based on data to reduce the calculation time. The fitness function FF can be written as shown in Eq. 20.

$$FF = \alpha I_{sp} + (1-\alpha) T_{req/W} \qquad (20)$$

Optimizing two or more objective functions in an optimization problem is called multi-objective optimization. The specific impulse Isp and thrust-to-weight ratio Treq/W, which are representative performance indicators of liquid rocket engines, were selected as multi-objectives. In Eq. 20, the optimized values of the multi-objective optimization problem are represented by the Pareto frontier line. In the actual preliminary design stage, considering various situations, such as the target altitude of the launch vehicle, launch vehicle weight, total fuel weight, engine clustering, and price, the weight value (α) was adjusted from 0 to 1. This adjustment gave added weight to high specific thrust and obtained weight for the thrust-to-weight ratio. This design is called the Pareto frontier line. When the weight (α) value was 1, regardless of Tref/W, the design variables that maximized the specific thrust were found. When the value was 0, the design variables with the maximum Tref/W were found. Values are typical empirically for each term.

6. Optimization of Design Variables and Results

This first part of the study determined the fuel speed's effect on the efficiency η (Eq. 8). Fig. 5 shows the over-all efficiency with the two sizes of pipe.

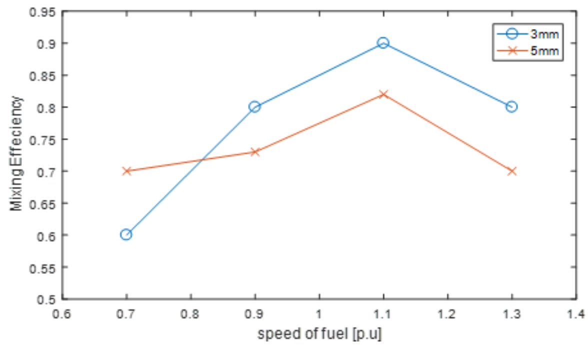

Fig 5 The maximum efficiency with a different size of pipe

The second part of this work designed the optimization. The objective function was not a complex multi-modal function and the risk of mis-application was low. However, several execution results were checked to confirm the con-vergence stability of the program. Fig. 6 shows the conver-gence process of each design variable every time the pro-gram is executed. The values are presented per unit, where (Pc/Pa (PU) = 0.00000015), (Ae/At (PU) = 15), and (F/O ratio = 2.5). It converged stably before about 60 generations. A sufficiently converged value was always obtained in 100 iterations, and the exact value weight was obtained at each trial, thus confirming that applying the code to this study did not pose any problems.

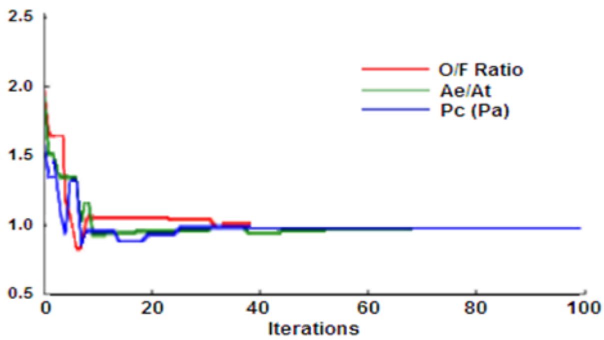

Fig. 6. Finding Optimal Design Variables during 100 Iterations

Problem-based PSO is easy to implement in any engineering because the theory is simple. PSO theory depends on the best value and the global best value, which will be the optimum solution at the last itera-tion (Eq. 18 and Eq. 19). The results of multi-objective optimization using PSO were obtained by setting the design variable α. First, it was de-rived to give the maximum specific thrust regardless of the weight. The yaw had the maximum thrust-to-weight ratio. The convergence of the nozzle extension ratio to the minimum boundary value of 5 re-flected that the nozzle extension ratio, along with the pressure in the combustion chamber, took up a large part in the weight evaluation. Results were obtained by setting the weight (α) to different values from 0 to 1. As shown in Fig. 7, the Pareto frontier line for the specific thrust and thrust weight ratios of the gas generator cycle of 3.5 kN was obtained. As is generally known, the shape of this graph shows a qualitative characteristic, that is, the thrust-to-weight ratio decreased as the specific thrust increased when the thrust was the same.

The same calculation was repeated for the required thrust (1 kN, 3 kN, and 4 kN). In Fig. 8, the same result is given. The thrust weight ratio T/W for each thrust ranged from about 4 to 7, and the specific thrust ranged from about 15 for each thrust. Meanwhile, the specific thrust and the thrust-to-weight ratio T/W increased as the thrust increased. Table 4 and Table 5 explain the results of optimization for Max. Isp and Max. T/W, respectively. respectively.

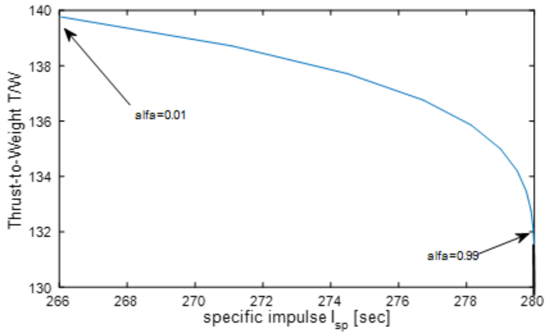

Fig. 7. Pareto Frontier Line of Gas-generator Cycle Rocket Engine (3.5kN).

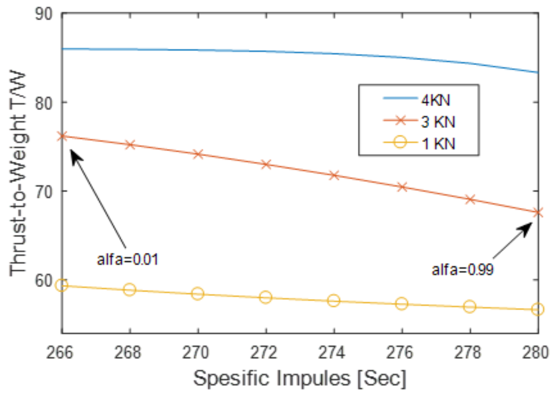

Fig. 8. Pareto Frontier Lines for Various Thrust.

Table 4. Result of Optimization for Max. Isp.

Design Variables	Result	Unit
combustion chamber pressure	130	Bar
nozzle extension ratio	16	
O/F	2.5	
Parameters	Result	
Thrust	3,5	kN
I_sp	280	Sec
T/W	115	
Main chamber actual flow rate	1,118	Kg/sec
Gas-generating substrate flow rate	68	Kg/sec
Required pump power	30	MW
Turbine generated power	30	MW

Table 5. Result of Optimization for Max. T/W.

Design Variables	Result	Unit
combustion chamber pressure	138.9	Bar
nozzle extension ratio	5.00 (Lower boundary)	
O/F	2.5	
Parameters	Result	
Thrust	3,5	kN
I_sp	271.2	Sec
T/W	123.3	
Main chamber actual flow rate	1,175.6	Kg/sec
Gas-generating substrate flow rate	78.12	Kg/sec
Required pump power	34.79	MW
Turbine generated power	34.79	MW

7. Conclusion

The conclusions can be summarized as follows:

1- PSO is a way to search random numbers of particles. If it is misapplied to a problem, different results are obtained each time it is run.

2- An analysis program for the gas generator cycle of a liquid rocket engine and a design variable optimization program using PSO were developed.

3- The objective function provides the optimized O/F ratio and the specific impulse and thrust-to-weight ratio. The optimum design can be obtained at the global best posi-tion of the particle variables in terms of the combustion chamber pressure and nozzle extension ratio.

4- The cycle analysis program showed a difference of less than 5% compared with the reference literature and a difference of less than 3% from the actual engine data.

5- As a result of optimizing for specific thrust using PSO, a design variable combination with a higher specific im-pulse Isp of 250 sec was found. The maximum thrust-to-weight ratio was obtained.

6- The relationship between the optimized system's spe-cific impulse and thrust-to-weight ratio obtained a factually similar relationship. The optimal design variable combina-tions for each thrust were determined by performing the same calculation for various thrust. This result can be used for rough performance analysis in the initial design of liq-uid rocket engines with gas generators.

7- The results can be used as data when designing a launch vehicle. For a quantitatively more accurate calcula-tion, additional research is needed on how to calculate the weight by reflecting the material on the basis of the shape of each part rather than how to infer the approximate weight only with the pressure in the combustion chamber. The accuracy of the program can be further improved by developing a module for modeling accurate parts.

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Statements and Declarations

☐ The authors declare that they have no known competing financial interests or personal relationships that could have appeared to influence the work reported in this paper.

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