An algorithm on simultaneous optimization of performance and mass parameters of open-cycle liquid-propellant engine of launch vehicles

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Abstract. In this paper, a new method for the determination of optimum parameters of open-cycle liquid-propellant engine of launch vehicles is introduced. The parameters affecting the objective function, which is the ratio of specific impulse to gross mass of the launch vehicle, are chosen to achieve maximum specific impulse as well as minimum mass for the structure of engine, tanks, etc. The proposed algorithm uses constant integration of thrust with respect to time for launch vehicle with specific diameter and length to calculate the optimum working condition. The results by this novel algorithm are compared to those obtained from using Genetic Algorithm method and they are also validated against the results of existing launch vehicle.

1. Introduction
The need to increase the specific impulse of rocket launcher engines has led design engineers to start the development of more complex engine cycles starting from simple gas pressure fed engines to the very complex staged combustion cycle engines. Currently, a wide range of different cycles exist and they are not only differ in terms of performance, but also in terms of mass, cost, reliability, etc., which in general makes it difficult to quickly determine the one that is best suited for certain mission or task. Ernst (2014) has applied the modular approach where engine components are sized using performance, dimension and mass model, making use of corrected ideal rocket theory and empirical relations [1]. His work focused mainly on the methodology and the construction of the models, and further includes the optimization of an upper stage and verification, validation and uncertainty and sensitivity analysis of the tool and the optimization to assess its accuracy, precision and applicability. Consequently, the research develops analysis modules for classifying components of liquid rocket engine and computing performance and weight of each component. Each module analyzes with combinations of input and output during system analysis. In order to increase its fidelity, the modules are based on the empirical formulas and practical data [2].

Furthermore, Lee and Roh (2013) presented system analysis program that can compute the weight and specific impulse used as performance indicator of the propulsion system, and the performance of
each component for given values of propellant, required thrust and combustion chamber pressure [2].

Analysis of today's liquid-propellant engine of launch vehicles is a challenging and complicated task, which requires advanced tools and algorithms. In general, an optimum design of engine itself will not result in the optimum operation of launch vehicle since the interaction between the rocket's subsystems must also be taken into consideration. There are several software packages that have been designed to determine the optimum performance parameters of the engine for specific inputs such as REDTOP2 [3]. Furthermore, there are also software packages like Fordu that are able to determine the optimum parameters of the propulsion subsystem including engine mass rate, pumps' input pressure and engine mass [4, 5]. However, it has been noted that the mass parameters are not typically considered together in the optimization process with the performance parameters.

It is believed in this study that simultaneous optimization of both performance and mass parameters of the rocket subsystems can lead to better results. Hence, a new algorithm is introduced in this study, which uses semi-empirical relations to calculate the mass of the propulsion subsystems such as engine, tanks and pressurizing subsystem, and this describes the propulsion subsystem with a better accuracy. The main advantage and novelty of this proposed algorithm is the coupling between the performance and mass parameters of rocket subsystems, which are optimized simultaneously instead of separately like in most existing algorithms.

2. Performance Characteristics Algorithm for Propulsion Subsystem

For the optimization of performance parameters, the following algorithm is proposed. In the first step, some constraints have to be defined as inputs. In this algorithm, the combustion chamber's pressure, burning time, thrust force, ratio of mass flow of oxidizer to the mass flow of fuel (O/F) and type of propellant are taken as inputs. Propulsion subsystem is considered open-cycled for this paper and its schematic is shown in Figure 1.

![Figure 1: Simplified schematics of open-cycle liquid-fueled rocket [6]](image)

According to the preliminary thrust force and specific impulse, the mass flow rate of the propulsion can be calculated from Equation 1.

\[
\dot{m}_{eng} = \frac{T_{eng}}{I_{sp_{eng}} g_0}
\]
Then, each component mass flow rate than can be driven from the following Equations 2 and 3:

\[ \dot{m}_{\text{eng,ox}} = \dot{m}_{\text{eng}} \frac{(O/F)_{\text{eng}}}{1+(O/F)_{\text{eng}}} \]  
(2)

\[ \dot{m}_{\text{eng,fe}} = \dot{m}_{\text{eng}} \frac{1}{1+(O/F)_{\text{eng}}} \]  
(3)

Next, the calculation of pressure drop in the whole propulsion subsystem is done. To simplify these calculations, pressure drop including cooling channel, injectors, pipes and orifices can be considered as a factor of combustion chamber's pressure. In the designing process of engine, the effort is focused on minimizing these waste as much as possible. Usually, pressure drop in the pipes and control valves is between 35 to 50 percent of the combustion chamber's pressure. For regenerative cooling system, this drop is normally about 20 to 30 percent of the combustion chamber's pressure. The pressure drop in injectors depends on their shape and type. However, in this study, this drop percentage is taken to be about 30 percent of the combustion chamber's pressure. Also, pressure drop is usually about 6 to 10 percent in chamber's pressure for orifices [7]. Therefore, the discharge pressure of oxidizer and fuel pumps can be calculated using Equation 4 and Equation 5, respectively.

\[ P_{p_{\text{ox}}} = 1.5939P_{\text{cc}} \]  
(4)

\[ P_{p_{\text{fu}}} = 2.2248P_{\text{cc}} \]  
(5)

In order to decrease the pressure drop error, geometric factors have been introduced to optimize the discharge pressure of the pumps. This enables the geometry of the pipes to be taken as constant in the process of optimization. These geometric factors can be found from Equation 6 and Equation 7.

\[ K_{p_{\text{ox}}} = \frac{\rho_{\text{ox}}(P_{p_{\text{ox}}}-P_{\text{cc}})}{\dot{m}_{\text{ox,eng}}} \]  
(6)

\[ K_{p_{\text{fu}}} = \frac{\rho_{\text{fu}}(P_{p_{\text{fu}}}-P_{\text{cc}})}{\dot{m}_{\text{fu,eng}}} \]  
(7)

To obtain mass flow rate of the turbine, power of the pumps is calculated according to Equation 8 and Equation 9. In this study, the pumps' input pressure has been chosen as an algorithm input, where \( \nabla P_{p_{\text{ox}}} \) and \( \nabla P_{p_{\text{fu}}} \) defined by Equation 10 and Equation 11, respectively, are the pressure difference of the pumps. In the meantime, \( i \) and \( e \) indicate the input and exit section of the pumps, respectively.

\[ P_{reg_{\text{ox}}} = \frac{g_{\omega} \dot{m}_{\text{ox,eng}} \nabla P_{p_{\text{ox}}}}{\eta_{p_{\text{ox}}}} \]  
(8)

\[ P_{reg_{\text{fu}}} = \frac{g_{\omega} \dot{m}_{\text{fu,eng}} \nabla P_{p_{\text{fu}}}}{\eta_{p_{\text{fu}}}} \]  
(9)

\[ \nabla P_{p_{\text{ox}}} = P_{\text{poxe}} - P_{\text{poxi}} \]  
(10)

\[ \nabla P_{p_{\text{fu}}} = P_{\text{pfue}} - P_{\text{pfui}} \]  
(11)

Moreover, the efficiency of the pumps has a drastic effect in producing more power and consuming less
mass flow rate. This heavily depends on their manufacturing technology and these parameters are also considered as constant inputs. For this case, $\eta_p$ has been chosen equal to 0.65.

The adiabatic work of the turbine depends on the expansion ratio and temperature of gases leaving the gas generator. The expansion ratio for open-cycle engines typically ranges from 20 to 50 and the temperature is based on the tolerable temperature of the blades. The efficiency of turbine also depends on its manufacturing and in this work, it is taken as 0.4.

$$L_{ad} = RT_{gg} \left[ 1 - \left( \frac{1}{P_{tr}} \right) ^{\gamma_{gg}} \right]$$  \hspace{1cm} (12)

$$\dot{m}_g = \frac{P_{req,g} - P_{req,\mu}}{L_{ad} \eta_t}$$  \hspace{1cm} (13)

After the mass flow rate and O/F ratio of the gas generator have been determined, the mass flow rate of each propellant component is calculated using Equation 14 to Equation 18. The O/F ratio of the gas generator is determined with regards to the thermal resistant limitation of its material. In this case study, the ratio is equal to 0.16 that shows a rich-fueled gas generator.

$$\dot{m}_{\mu,g} = \frac{\dot{m}_{gg}}{1 + (O/F)_{gg}}$$  \hspace{1cm} (14)

$$\dot{m}_{\mu,g} = \dot{m}_{gg} \frac{(O/F)_{gg}}{1 + (O/F)_{gg}}$$  \hspace{1cm} (15)

$$\dot{m}_{ox,cc} = \dot{m}_{ox,eng} - \dot{m}_{ox,gg}$$  \hspace{1cm} (16)

$$\dot{m}_{\mu,cc} = \dot{m}_{\mu,eng} - \dot{m}_{\mu,gg}$$  \hspace{1cm} (17)

$$\dot{m}_{cc} = \dot{m}_{\mu,cc} + \dot{m}_{ox,cc}$$  \hspace{1cm} (18)

Therefore, the ratio of oxidizer to fuel for combustion chamber can be calculated using Equation 19.

$$\left( \frac{O/F}_{cc} \right) = \frac{\dot{m}_{ox,cc}}{\dot{m}_{\mu,cc}}$$  \hspace{1cm} (19)

At this point, the necessary means to calculate specific heat ratio and produce the gas characteristic speed within thermal equilibrium has been made available. The Astra software for the computation of the produced gas is used. As a result, theoretical diameter of throat can be obtained as in Equation 20.

$$D_t = \sqrt{\frac{4c}{\pi P_{cc}}} \dot{m}_{cc}$$  \hspace{1cm} (20)

This is the first step for the determination of the throat diameter. After this, the throat diameter of the combustion chamber is kept constant. This leads to more accurate answers in the optimization space. In the next following steps, for each chamber’s pressure, for a constant O/F ratio and throat diameter, the performance and mass parameters are calculated. In the end, the one corresponding to the highest specific impulse to gross mass will be selected as the optimum point of design.

To find the mass of tanks, the amount of propellant in the launch vehicle is required. This amount depends on the burning duration of the propellant. To further simplify the calculations, the integral of thrust with respect to time is kept constant in all of the iterations. As a result, the duration of burning
will be known after determination of thrust. For specific amount of mass flow rate and chamber’s pressure, the specific heat ratio and velocity of produced gas will be calculated. Therefore, the total mass flow rate of chamber, specific impulse and the produced thrust of the chamber can be derived from Equation 21 and Equation 22 [8, 9], where $\lambda$ is the efficiency of nozzle that is usually taken to be around 0.9 to 0.98.

$$\dot{m}_{cc} = \frac{P_{cc} A_{cc}}{C_{cc}}$$  \hspace{1cm} (21)

$$I_{\dot{m}_{cc}} = \frac{C_{cc}^+ A_{cc}}{g_o} \left[ \gamma_{cc} \sqrt{\frac{2}{\gamma_{cc} - 1} \left( \frac{\gamma_{cc} + 1}{\gamma_{cc} - 1} \left( 1 - \frac{P_{cc}}{P_{cc}} \right)^{\gamma_{cc} - 1} \right) \right]$$  \hspace{1cm} (22)

$$T_{cc} = I_{\dot{m}_{cc}} \dot{m}_{cc} g_o$$  \hspace{1cm} (23)

The gas generator mass flow rate is to be defined next. Accordingly, specific amount for mass flow rate is initially chosen – usually about 4 percent of chamber’s mass flow rate – and check whether the produced power is equal to the consumed power by pumps. The process is then iterated by changing the amount of the mass flow rate until both the produced and consumed power are equal to each other. Using the same procedure, the mass flow of gas generator can be found [10].

$$\nabla P_{\text{pox}} = \frac{k_{\text{pox}} \dot{m}_{\text{ox}}^{\gamma_{\text{eng}}}}{\rho_{\text{ox}}} + P_{\text{cc}} - P_{\text{poxi}}$$  \hspace{1cm} (24)

$$\nabla P_{\text{p}} = \frac{k_{\text{p}} \dot{m}_{\text{p}}^{\gamma_{\text{eng}}}}{\rho_{\text{p}}} + P_{\text{cc}} - P_{\text{p}}$$  \hspace{1cm} (25)

$$P_{\text{reqox}} + P_{\text{reqf}} = g_o \dot{m}_p \nabla P_p = \eta_p \dot{m}_{\text{gg}} c_p T_{\text{eq}} \left[ 1 - \left( \frac{1}{P_{\text{r, ratio}}} \right)^{\gamma_{\text{gg}} - 1} \right]$$  \hspace{1cm} (26)

Using both previous Equations 14 and 15, the mass flow rate of oxidizer and fuel in gas generator can now be determined. Subsequently, the total mass flow rate of fuel and oxidizer can be calculated as in Equation 27 to Equation 30.

$$\dot{m}_{\text{fueq}} = \dot{m}_{\text{fuc}} + \dot{m}_{\text{flu}}$$  \hspace{1cm} (27)

$$\dot{m}_{\text{oxeq}} = \dot{m}_{\text{oxc}} + \dot{m}_{\text{oxw}}$$  \hspace{1cm} (28)

$$\left( \frac{O}{F} \right)_{\text{eng}} = \frac{\dot{m}_{\text{oxeq}}}{\dot{m}_{\text{fueq}}}$$  \hspace{1cm} (29)

$$\dot{m}_{\text{eng}} = \dot{m}_{\text{oxeq}} + \dot{m}_{\text{fueq}}$$  \hspace{1cm} (30)

Finally, the total thrust force and specific impulse are calculated using Equation 31 to Equation 33. It should be noted that the specific impulse of gas generator depends on the type of the propellant.

$$T_{\text{gg}} = I_{\dot{m}_{\text{gg}}} \dot{m}_{\text{gg}} g_0$$  \hspace{1cm} (31)

$$T_{\text{eng}} = T_{\text{cc}} + T_{\text{gg}}$$  \hspace{1cm} (32)
\[ I_{p_{in}} = \frac{T_{eng}}{m_{eng} g_0} \]  

(33)

3. Mass Characteristics Algorithm of Propulsion Subsystem

The mass characteristics of propulsion system mainly depend on the regime parameters. Hence mass of the different parts of the propulsion system can be described as a function of the regime parameters. The propulsion subsystem’s mass characteristics primarily consist of the mass of the chamber, turbo-pump, engine and tank. For each of these characteristics, there are already some empirical functions that depend on the performance parameters of subsystems. They can be used to compute the total mass of propulsion subsystem. The mass of the turbo-pump can be obtained from the following Equation 34 and Equation 35.

\[
D = \sum_{i=\text{aux, pu}} \dot{m}_i \left( \frac{P_{p_{i,j}} - P_{p_{i,j}}}{\rho_i} \right)^{1.5} 
\]  

(34)

\[
m_{r,p} = \begin{cases} 
6.29 + 0.981 \times 10^{-3} D & 1170 < D < 3.22 \times 10^4 \\
21 + 0.54 \times 10^{-3} D & 3.22 \times 10^4 < D < 7.52 \times 10^5 
\end{cases}
\]  

(35)

The number of pump’s revolution, \( \omega \) can be calculated from Equation 36 and Equation 37, where \( C_{\text{cv, max}} \) is the cavitation maximum characteristic velocity that ranges from 3000 to 5000m/s while \( \Delta h_{\text{cv, res}} \) is the cavitation reserve factor of pumps that ranges between 10 to 30 J/kg.

\[
\Delta h_{\text{cv}} = \frac{P_m - P_t - \rho \Delta h_{\text{cv, res}}}{\rho} 
\]  

(36)

\[
\omega = C_{\text{cv, max}} \frac{\Delta h_{\text{cv}}^{0.75}}{248.9 \Delta h_{\text{cv}}^{0.5}} 
\]  

(37)

Furthermore, the mass of the chamber can be found using Equation 38 to Equation 47, in which \( \beta_a \) is the nozzle wall’s imperfection at the exit and it is normally taken to be between 2 to 6 degrees.

\[
\gamma_{\text{T}_c} = 3.03 \frac{P_c \times 10^{-6}}{A_t} 1.58 \times 10^6 \leq \frac{P_c}{\sqrt{d_i}} \leq 5.86 \times 10^6 
\]  

(38)

\[
\gamma_a = -23.58 + 5.894 \times 10^{-2} \left( \frac{P_c}{\sqrt{\varepsilon D_i}} \right)^{0.475} 0.0632 \times 10^6 \leq \frac{P_c}{\sqrt{\varepsilon D_i}} \leq 0.316 \times 10^6 
\]  

(39)

\[
\bar{x}_c = s_c \frac{A_t}{A_c} = 3.54 \sqrt{\frac{P_c q_c}{m_{cc}}} - 2 \frac{1}{\sqrt{q_c c^*}} - \sqrt{q_c c^*} + 1 
\]  

(40)

\[
\bar{x}_{\text{sub}} = s_{\text{sub}} \frac{A_c}{A_t} = 2 \frac{1}{\sqrt{q_c c^*}} + 0.818 - 0.974 
\]  

(41)

\[
\bar{q} = \frac{\dot{m}_{cc}}{P_c A_{cc}} 
\]  

(42)
z = 1 - \left[ \frac{\sin(\beta_0)}{0.6 - (0.018\gamma - 0.0175)(\sqrt{A} + 24)} \right]^{\frac{4}{3}} \tag{43}

\bar{s}_{sup} = \frac{s_{sup}}{A_y} = \bar{s}_0 \left[ 1 - \left( 1.415 - \frac{0.27}{\sqrt{A}} \right) f(z) \right] \tag{44}

f(z) = 1 - e^{-(1-z)^4} \tag{45}

\bar{s}_0 = (32 - 10\gamma)(\sqrt{A} - 1) + (2.1 + 1.6\gamma^4)(\sqrt{A} - 1)^{2.25} \tag{46}

m_{Te} = \frac{mc^*}{P_{cc}} \left[ \gamma_{Te} (\bar{s}_c + \bar{s}_{sub}) + \gamma_{n}^* \bar{s}_{sup} + \frac{163}{(P_{cc}^* m)^{0.25}} \frac{8.5g}{P_{cc}} \right] \tag{47}

Now, by already knowing the mass of turbo-pump and combustion chamber, the total mass of the engine can be derived using Equation 48.

m_{eng} = \begin{cases} 
N \times m_{Te} + \sum m_{T_p} + 0.217T_g + 57.5 & 14.7 \leq \frac{T}{g} \leq 981 \\
N \times m_{Te} + \sum m_{T_p} + 0.377T_g + 93.1 & 982 \leq \frac{T}{g} \leq 1680 
\end{cases} \tag{48}

On the other hand, the mass of the tanks can be calculated using Equation 49 and Equation 50. It can be noted that this mass depends on type, pressure and mass of propellant.

m_{Tank,\text{fu}} = \frac{m_{prop,\text{fu}}}{(O/F) + 1} \rho_{fu} \left[ \frac{2.1 f_{s,\text{fu}} \rho_{Tank,\text{fu}} P_{Tank,\text{fu}}}{\sigma_{y,\text{Tank,\text{fu}}}} + \frac{4.2 \delta_{iso,\text{fu}} \rho_{iso,\text{fu}}}{D_{Tank,\text{fu}}} \right] + 0.649 \frac{\rho_{Tank,\text{fu}} D_{Tank,\text{fu}}^3}{\sigma_{y,B,\text{fu}}} f_{s,\text{fu}} P_{Tank,\text{fu}} \tag{49}

m_{Tank,\text{ox}} = \frac{m_{prop,\text{ox}}}{(O/F) + 1} \rho_{ox} \left[ \frac{2.1 f_{s,\text{ox}} \rho_{Tank,\text{ox}} P_{Tank,\text{ox}}}{\sigma_{y,\text{Tank,\text{ox}}}} + \frac{4.2 \delta_{iso,\text{ox}} \rho_{iso,\text{ox}}}{D_{Tank,\text{ox}}} \right] + 0.649 \frac{\rho_{Tank,\text{ox}} D_{Tank,\text{ox}}^3}{\sigma_{y,B,\text{ox}}} f_{s,\text{ox}} P_{Tank,\text{ox}} \tag{50}

Lastly, the final gross mass of the propulsion subsystem consisting of the mass of engine, tanks and propellant can be written as the following Equation 51 and Equation 52.

m_{inert} = m_{eng} + \sum m_{Tank,i} + m_{other} \tag{51}

m_{gross} = m_{prop} + m_{inert} \tag{52}
4. Case Study for Validation of the Proposed Algorithm

In the process of designing a liquid fuel engine, one of the most important issues is the achievement of basic knowledge regarding the mass and functional aspects of the engine and the optimal parameters of it. In this paper, various massive and functional dimensions are tried and found before designing a liquid fuel engine. This will be a great assistance to the designers of the propulsion system in order to select appropriate design parameters like thrust, specific impulse, etc. To see whether this algorithm is effective, its results for the sample case study are compared to that obtained using Genetic Algorithm method. In general, the Genetic Algorithm has been designed to maximize the objective function and search for the optimized working condition. The main inspiration behind Genetic Algorithm comes from Darwin’s evolution theory [11, 12]. This optimization technique is largely based on the idea of natural selection and reproduction.

The case study that is used to validate and demonstrate the effectiveness of the proposed algorithm is the optimization of third stage of Tsyklon-3 launch vehicle shown in Figure 2. The input variables for the procedure are tabulated in Table 1.

![Figure 2: Third stage of Tsyklon-3 with RD-861 engine as its engine](image)

Table 1: Input variables from Tsyklon-3 third stage engine

| Variable               | Value          |
|------------------------|----------------|
| Engine Type            | Open-Cycle     |
| Propellant Type        | UDMH-N2O4      |
| Chamber’s Pressure     | 88.8 bar       |
| Burning Time           | 125 s          |
| O/F Ratio              | 2.1            |
| Engine’s Exit Pressure | 0.2 bar        |
| Specific Impulse       | 317 s          |
| Thrust                 | 78.708 kN      |

The process flowchart of the proposed algorithm is illustrated in Figure 3. The results obtained are presented in Table 2 and Table 3. From Table 2, it can be observed that the performance results for specific impulse and thrust force are fairly acceptable. The largest error recorded is for the engine O/F ratio, which can be contributed to the difference between the gas generators defined in the algorithm and the real model. Meanwhile, in Table 3, one of the largest errors is corresponding to empty mass of the rocket. The primary reason for this condition is the fact that some components’ mass are neglected in the modelling.

Table 2: Results for the performance parameters by the proposed algorithm

| Parameters | Actual Value for Tsyklon-3 Third Stage | Results from the Proposed Algorithm | Percentage Error |
|------------|----------------------------------------|-------------------------------------|------------------|
| Specific Impulse | 317 s | 317.82 s | 0.26% |
| Thrust     | 78.708 kN | 78.74 kN | 0.04% |
| (O/F)_{eng} | 2.1 | 2.18 | 3.80% |
Figure 3: Simple flowchart of the proposed algorithm

Table 3: Results for the mass parameters by the proposed algorithm

| Parameters                   | Actual Value for Tsyklon-3 Third Stage | Results from the Proposed Algorithm | Percentage Error |
|------------------------------|----------------------------------------|-------------------------------------|------------------|
| Propellant Mass              | 3139 kg                                | 3297.3 kg                           | 5.04%            |
| Engine Mass                  | 120 kg                                 | 118.8 kg                            | 0.97%            |
| Empty Mass                   | 1407 kg                                | 1344.3 kg                           | 4.45%            |
| Gross Mass                   | 4666 kg                                | 4760.0 kg                           | 2.01%            |

Figure 4 depicts the relation between the ratio of specific impulse to the subsystem’s mass and both the mass flow rate and the chamber’s pressure. It can be observed that there exists a peak for specific impulse to mass ratio, which is the optimum operation point for the given data inputs.

Figure 4: Ratio of specific impulse to gross mass against mass flow rate and chamber’s pressure
Comparison of the optimization results from the proposed algorithm and that of Genetic Algorithm is tabulated in Table 4. As can be seen, the ratio of specific impulse to gross mass is increased, which shows an optimized situation for the same family of launch vehicles. In addition, the obtained results from the proposed algorithm are fairly close to the Genetic Algorithm. The percentage differences in the values of the parameters between the optimized results from the proposed algorithm and the actual values in the Tsyklon-3 third stage engine are tabulated in Table 5, which highlights the improvements that have been made by using the proposed algorithm.

Table 4: Comparison of performance and mass parameters

| Parameters                  | Tsyklon-3 Third Stage | Optimized Model by the Proposed Algorithm | Optimized Model by Genetic Algorithm | Percentage Error (%) |
|-----------------------------|-----------------------|-------------------------------------------|-------------------------------------|----------------------|
| Rocket Type                 | Open-Cycle            | Open-Cycle                                | Open-Cycle                          |                      |
| Propellant Type             | UDMH-N2O4             | UDMH-N2O4                                  | UDMH-N2O4                           |                      |
| Chamber’s Pressure          | 88.80 bar             | 104 bar                                   | 103.93 bar                          | 0.07                 |
| Burning Time                | 125 s                 | 110.26 s                                  | 110.35 s                            | 0.08                 |
| Specific Impulse            | 317 s                 | 318.23 s                                  | 318.30 s                            | 0.02                 |
| Thrust                      | 78.71 kN              | 93.067 kN                                 | 92.995 kN                           | 0.08                 |
| O/F Ratio                   | 2.1000                | 2.0784                                    | 2.1541                              | 3.64                 |
| Propellant Mass             | 3139 kg               | 3287 kg                                   | 3286.42 kg                          | 0.02                 |
| Engine Mass                 | 120 kg                | 129.07 kg                                 | 129.02 kg                           | 0.04                 |
| Empty Mass                  | 1407 kg               | 1354.40 kg                                | 1353.81 kg                          | 0.04                 |
| Gross Mass                  | 4666 kg               | 4641.40 kg                                | 4640.24 kg                          | 0.02                 |
| Specific Impulse to Gross Mass | 0.0675              | 0.0686                                    | 0.068594                            | 0.01                 |

Table 5: Improvements by the proposed algorithm from the actual Tsyklon-3 third stage engine

| Parameters                  | Increment / Decrement | % Changes |
|-----------------------------|-----------------------|-----------|
| Chamber’s Pressure          | Increase              | 11.25     |
| Burning Time                | Decrease              | 1.16      |
| Specific Impulse            | Increase              | 3.78      |
| Thrust                      | Increase              | 1.60      |
| O/F Ratio                   | Decrease              | 1.02      |
| Propellant Mass             | Increase              | 2.42      |
| Engine Mass                 | Increase              | 14.02     |
| Empty Mass                  | Decrease              | 19.04     |
| Gross Mass                  | Decrease              | 3.10      |
| Specific Impulse to Gross Mass | Increase          | 0.14      |

5. Conclusion
Finding the optimum operation point in designing launch vehicles is a real challenge. As mentioned before, the energetic optimization of the propulsion subsystem such as specific impulse and thrust is not enough. The mass of the system also has to be considered for optimum condition as well. The most prominent feature of this presented work is to include the mass of the system into consideration during the optimization process through the new proposed algorithm. The results from the proposed algorithm have been shown to be good and fairly in line with other established optimization procedures such as Genetic Algorithm. The obtained ratio of specific impulse to the subsystem’s gross mass ratio with
constant integral of thrust to time has been discussed and better results have been obtained. Moreover, this viewpoint for optimization can also be extended to other liquid-propellant launch vehicles such as multiple gas generators and closed-cycle systems. This algorithm can be applied in preliminary and conceptual design of liquid-propellant launch vehicles to reduce the cost of experiment and time.

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