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Optimal design of four stage launch vehicle considering multi objective NSGA II algorithm and mass-energetic concepts

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A B S T R A C T

A solid fuel launch vehicle is a rocket with an engine that has been widely used in aerospace missions. Utilizing such launch vehicles depends on the simplicity of the manufacturing, maintenance, operation and development of the control systems. The purpose of optimization in solid fuel launch vehicles design is to find the best possible design for the mission with regard to the available equipment, constraints and infrastructures. Therefore, the main purpose of this research is to optimally design a launch vehicle for customized missions based on successful experiences, as well as technology, manufacturing capabilities and facilities. In this context, NSGA-II Intelligent Optimization Algorithm is considered based on multi-objective optimization principles and Mass-Energetic concepts. The optimal design of the launch vehicle is performed by applying intelligent algorithms and technological opportunities and limitations. The result showed that the present optimization method can design the launch vehicle based on technological limitations.

Nomenclature

$M_{o,i}$	Final mass of the stage i
$M_{fu_{,i}}$	Fuel weight of stage i
M_{st_i}	Structural weight of stage i
M_{w_i}	Initial weight of stage i
M_{paylod}	Payload mass
$m_{To,i}$	Weight of body structure block i
$m_{cy,i}$	Weight of navigation and control system block i
$m_{o,i}$	Final weight of the block i
$m_{w,i}$	Initial weight of the block i
$I_{sp,i}$	The specific impulse
no	Trust to weight ratio
μ_{pl}	Stage weight Ratio
L/D	Ratio of Length to diameter
Th_i	Trust for stage i
tb_i	Burning time for stage i
$\alpha_{\delta i}$	Connection coefficient for stage i
α_{Ti}	Structural coefficient for stage i

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α_{ci}	Control coefficient for stage i
γ_{gi}	Propulsion coefficient for stage i
Li	Length for stage i
Di	Diameter for stage i
V	Orbit velocity
μ_p	Weight Ratio
μ_k	Ratio of the final weight of the stage i
V_F	Require speed
N_{opt}	Number of stage
G	Gravitational constant
S	Cross section area of the stage i
α , β	Energy mass coefficient
L_p	Largest dimension of the payload

1. Introduction

The solid fuel launch vehicle is one of the systems used to carry cargo into space. Reduction in cost of manufacturing and maintenance processing, operational capability and also the improvement of control systems are the main factors in expanding use of the launch vehicles in the space missions. Best solid fuel launch vehicles ever known are the US-made Four-Stage launch vehicle Minotaur and the Russian-made Four-Stage launch vehicle Start1 (Norris & Kristensen, 2009). They were upgraded to space-based launch vehicles by adding a single stage to the Three-Stage Continental Ballistic Missiles Minuteman III and RS12 Topol (Woolf, 2009). New types of these launch vehicles such as Spyder are used for delivering a light 6U CubeSat into circular, low Earth orbits (Bennett, 2019). Hence, in a multi-objective design space, traditional single-objective optimization methods are not responsive (Roshanian & Keshavarz, 2007). Optimal design of launch vehicles is a complicated challenge that requires the application of special methods called Multidisciplinary Design Optimization (MDO) techniques. MDO methodologies are interesting strategies applied in various domains to solve optimization problems (Balesdent, Bérend, Dépincé, & Chriette, 2012). The use of multi-objective optimization tools for launch vehicle design has expanded significantly, and designers are using intelligent multi-objective optimization algorithms to achieve a more technical and economic design (Hammond, 2001). US space agency NASA used multi-purpose smart methods for several years in the design of the multiple-use launch vehicle with the main objective of reducing their dry weight, which requires parallel analysis of aerodynamics, structure, propulsion and cost (Bhatnagar, Rajan, & Saxena, 2012; Braun, Moore, & Kroo, 1996; Cormier et al., 2000; Tartabini, Wurster, Korte, Lepsch, & Rockets, 2002; Tsuchiya & Mori, 2002). The hybrid optimization method based on genetic algorithms and simulated annealing has been formulated and used to optimize a small four stage solid propellant launch vehicle. The results presented it as an economical and effective method to design and optimize launch vehicles (Villanueva & Abbas, 2015). The integrated environment Modefrontier optimization software is used to perform the optimization. This software, coupled with high accuracy and precision for design, has the capability of linking with other engineering designs and analytics software, causing automation of the modeling and decision-making process progress quick and easy(Da Cás, Vilanova, Barcelos Jr, Veras, & Management, 2012).The purpose of this research is to develop a method for optimizing a Four-Stage solid fuel launch vehicle with consideration of existing technology and infrastructure as well as considering the technological constraints and limitations ahead. Optimal design of the launch vehicle requires several simultaneous objectives to be fulfilled, such as reducing the total weight of the launch vehicle, increasing the load capacity, reducing the number of stages, reducing the length to the diameter ratio, reducing the weight of the launch vehicle structure, and so on. The other capabilities of the software are being able to define the exact values of the inputs; perform optimization based on different types of multi-objective optimization algorithms as well as the analysis of the outputs and defining them as a constraint or objective function in the optimization problems. Using these capabilities, a new software program for optimizing Four-Stage solid fuel launch vehicles is designed and developed which reduces costs and increases the speed and efficiency in optimum launch vehicles design. As a final point, the optimization results of the software are analyzed.

2. NSGAII Multi-Objective Optimization Algorithm

Non-dominant sorting genetic algorithm NSGA II is one of the well-known multi-objective algorithms widely used in multi-objective challenges. This algorithm was introduced by Deb et al. (Deb, Pratap, Agarwal, & Meyarivan, 2002) which presented a unique Pareto Front estimation method. In fact, the NSGAII algorithm is similar to a genetic algorithm with multi-objective discussions added to it. The NSGAII algorithm is an extraordinary, evolutionary and elitist algorithm designed based on two main concepts ``non-dominant sorting" and "crowding distance". This algorithm uses Non-dominant sorting for fronting the solutions according to their priority. The application of crowding distance is to find the best solutions that are in the same front. Fig. 1 shows NSGA II algorithm in sorting and finding the optimum answers according to Pareto front.

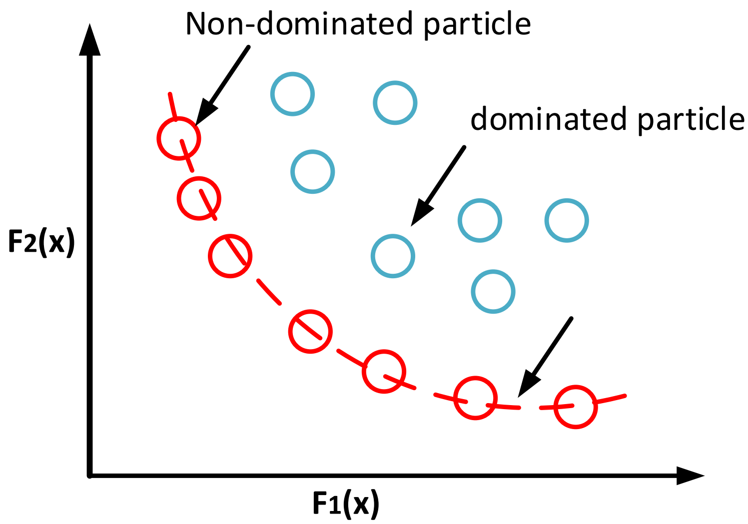

Fig. 1. Finding of the optimum answers by Pareto front of f1(x) and f2(x) objective functions using the NSGA-II algorithm.

The mechanism of this algorithm is that if the first ranking criterion (non-dominant sorting) fails to find all best answers, the second criterion is applied then, to find the remaining answers. In the NSGAII algorithm, when comparing all answers, the solution is situated by non-dominant sorts in different fronts. In this process, the best solutions are in the first front and if all of them sit in the same front, the better answers are selected according to the crowding distance criterion. This process is shown in Fig. 2.

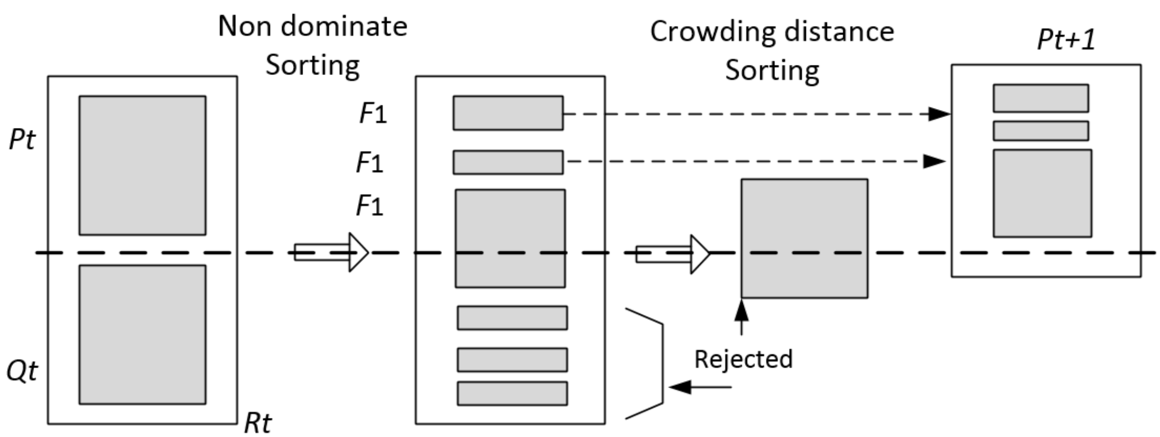

Fig. 2. NSGA-II procedure to find the best solutions

Fig. 2 indicates the NSGA II algorithm in non-dominate and crowding distance sorting for selection of the best solutions. As shown in Fig. 2, the aim is selecting eight answers on the answer set. After fronting the answers by non-dominate criterion, solution starts from the first front (F1) and three answers are selected. Then it goes to the second front (F2) and two answers are selected then it enters the third front (F3) and because there are five answers in third front and the target is selecting eight answers. Therefore the second criterion of order comes into play, and the answers that are higher in crowded distance criterion are selected in order to complete eight answers.

3. Systemic Design of Solid Fuel Launch Vehicles

The type of structure used in the process of construction of solid launch vehicles can cause technological constraints for manufacturers. For example, using titanium for a launch vehicle structure because of its light-weight is an advantage (Bruhn, 1967) but requires the provision of specific infrastructure for the proper welding of metal parts. Furthermore, using composite materials for the launch vehicle structure due to their light weight and high resistance to pressure has many advantages, but it requires high technology to provide accurate layer thickness (Fakoor & MEHRE, 2016; Fakoor, Sabour, & Khansari, 2014; Khansari, Farrokhi, & Mosavi, 2019) as well as resistance of the composite inner layer against severe corrosion of the launch vehicle. Recently, machine learning and deep learning methods were employed to enhance the mechanical properties of aerospace composite structures (Shamsirband & Mehri Khansari, 2021). In addition, clustering the engines and integrating launch vehicle structures is one of the most important complex tasks that any inattention may cause serious damage to the achievement of the mission objectives. One of the fundamental problems in systemic design of launch vehicle structure is extracting design parameters from statistical data without considering the technological capability of the manufacturer country in constructing the launch vehicles. Disregarding this issue often makes it impossible to construct a launch vehicle to carry out the mission. Energy-Mass coefficients are parameters that reflect technological ability of the manufacturer in fabricating the launch vehicles. Review of past space missions showed that launch vehicles that were designed without considering Mass-Energy coefficients could not reach the required speed during their missions, mainly because of the limitations and lack of technology required in the manufacturing process. For that reason, in designing new launch vehicles one must take into account the constraints of manufacturing technology and on the other hand meet the desired outcomes both economically and based on schedule. Multi-objective optimization techniques are one of the tools available to designers for the mentioned purpose. The main purpose of this research is the design of a launcher while considering the limitations of technology, manufacturing capabilities, manufacturing facilities, etc.

3.1. Mass-Energy equations for launch vehicle

In the process of designing launch vehicles mass and energy of each stage highly depend on technological capabilities. These capabilities will be reflected in the so-called Mass- Energy equations. The Mass- Energy equations establish the quantitative relationships between the mass of the structure and energy-related properties of the launch vehicle in each stage. These properties depend on the mass of the fuel, mass of the structure, manufacturing technology and the engine properties of each stage. The mass of structural parts depends on the capability of technology and the type of equipment used in constructing the structural parts. Furthermore, the mass of the engine (nozzle) depends on the energy properties of the engine and its fuel volume. The Mass-Structural (energy) equations express the relationship of these factors, are shown in following equations 1and 2.

$$M_{o,i}=m_{o,i}+M_{o,i+1} \tag{1}$$
$$M_{w,i}=m_{w,i}+M_{o,i+1} \tag{2}$$

In these equations $M_{w,i}$, and $M_{o,i}$ are initial and final weights of stage i of launch vehicle and $m_{w,i}$ and $m_{o,i}$ are initial and final weight of the block i of the launch vehicle. The difference between stages and blocks is illustrated in Fig. 3.

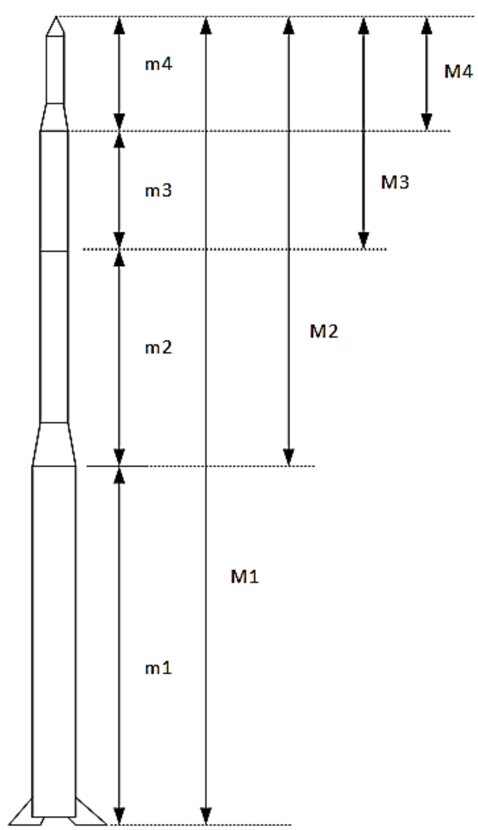

Fig. 3. The difference between stages (M) and blocks (m) on launch vehicle

The structure's weight of the stage *i* (M_{st_i}) can be obtained from the following Eq. (3).

$$M_{st_i} = m_{To,i+}m_{\delta,i+}m_{cy,i+}m_{gy,i} \tag{3}$$

where, $m_{To,i}$ is the weight of body structure, $m_{cy,}i$ is the weight of navigation and control system (structure, elements and equipment), $m_{gy,i}$ the engine weight (nozzle) and $m_{\delta,i}$ is the weight of the accessories (inter-stage structure, cabling, power supply system, connections, etc. excluding the mass of the fairing). The weight of body, control systems and accessories at each stage are defined as follows in Eq. (4) to Eq. (7).

$$m_{To,i=}\ \alpha_{To,i}M_{o,i}(1-\mu_{p,i}) \tag{4}$$
$$m_{cy,i=}\alpha_{cy,i}\ M_{o,i}(1-\mu_{p,i}) \tag{5}$$
$$m_{\delta,i=}\ \alpha_{\delta,i}\ M_{o,i}(1-\mu_{p,i}) \tag{6}$$
$$\mu_{p,i} = \frac{\left(\sum_i^4 M_{o,i+1} + \quad M_{paylod}\right)}{\left(\sum_i^4 M_{o,i} \ + \quad M_{paylod}\right)} \tag{7}$$

In these equations $\alpha_{To,i}$, $\alpha_{cy,i}$ and $\alpha_{\delta,i}$ are the mass coefficients of the launch vehicle in stage i which depends on the manufacturing technology, type of structural system, material type, and type of launch vehicles control system. Also, $\mu_{p,i}$ is relative weight of total load of the launch vehicle in stage of i. The initial weight of the engine structure (nozzle) of stage i is defined as a coefficient of the amount of thrust power (Bruhn, 1967) as follows in Eq. (8).

$$m_{gy,i} = \gamma_{gy,i}\, n_{o,i}\ M_{o,i} \tag{8}$$
$$n_{oi} = \frac{Th_{o,i}}{M_{o,i}}$$

In equation (8), $\gamma_{gy,i}$ is the energy coefficient of the nozzle stage which depends on the mass and thrust of the generating system (motor nozzle) and n_{oi}represents the thrust to weight ratio of the stage, which is one of the most important parameters of the launch vehicle design. By using the relationships (3) to (8) the initial weight of the structure of stage i is counted by the following Eq. (9):

$$M_{st_i} = (\alpha_{To,i+}\alpha_{cy,i+}\alpha_{\delta,i})\ M_{o,i}\ (1\text{-}\mu_{p,i})\ +\gamma_{gy,i}\ M_{o,i}n_{o,i} \tag{9}$$

With considering Eq. (9) and Eq. (4) to Eq. (8) in Eq. (10) is obtained:

$$\mu_k = \alpha_i + \mu_{p,i}(1-\ \alpha_i) + \beta_i\ n_{o,i} \tag{10}$$

where, α, β are called launch vehicle Mass-Energy coefficients and are defined by the following equation:

$$\alpha_i = \frac{\alpha_{To,i}+\ \alpha_{\delta,i}+\ \alpha\ _{cy,i}}{1+\alpha_{To,i}} \quad \beta_i = \frac{\gamma_{gy,i}}{1+\alpha_{To,i}} \tag{11}$$

From these relationships it can be concluded that the ratio of the final weight of the each stage is a function of the main design parameters ($\mu_{p,i}$and $n_{o,i}$) and the energy mass coefficient (α, β). In fact mass-energy coefficient controls that the specifications calculated for designed launch vehicle parameters are accessible by the technology capabilities of the manufacturer country. These relationships facilitate the evaluation of main design criteria with making changes in main parameters and on the other hand it enables the possibility of incorporating technological constraints in the manufacturing components, connections, and engine, navigation and control systems by using their mass-energy coefficients in the process of design and analyses.

4. Launch Vehicle Components

Various topics are involved in optimum design of a launch vehicle. The most important topics are customer order, requirements and constraints. The arrangement of issues and their relationships, as well as the relation with the optimization algorithm, constitute the structure of the multi-objective design problem. In the present optimal design by considering the Mass-Energy relationships, objective functions, constraints, type of multi optimization algorithm and its settings the main design variables of the launch vehicle are specified. The following sections describe the design process of the launch vehicle step by step.

4.1. Basic Parameters of Launch Vehicle

The basic design parameters of launch vehicle are grouped into three main categories:

1. Mission Parameters: Including payload weight, launch site specifications, type and height of orbit and required speed (Tsuchiya, Mori, & Rockets, 2004).

2. Technical Parameters: At this section, the most important technical parameters of the launch vehicle and their ranges are determined using the statistical data of similar launch vehicles. For controlling available technological capabilities, these parameters must couple with Mass-Energy coefficients next.

3. Performance Parameters: At this section, Mass-Energy coefficients are considered to ensure that the mission is achieved with the available technological capabilities.

4.1.1. Definition of Mission Parameters

The details of a successful mission of the United States Minotaur launch vehicle, as described in Table 1, are available.

Table 1. Profile of simulated mission

Payload Weight	Payload	Latitude	Mission Date	The Height of The Circular Orbit	Orbital Inclination
0.374ton	**TacSat-2 + GeneSat-1**	**37.0°**	**2006**	**400 -400**	**40.0°**

Typical Minotaur launch vehicle mission profile is showed in Table 2. Fig. 4 shows typical Minotaur launch vehicle mission profile extracted from Orbital (2015).

Table 2. Typical Minotaur launch vehicle mission profile (Minotaur, I., User's Guide)

No	Event	Time (s)	Altitude (km)	Velocity (m/s)	Latitude(deg)	Longitude(deg)
1	Stage 1 ignition	0	0.11	0	34.576	120.632
2	S1 step/s2 ignition	61.3	31.69	1499.18	34.351	120.687
3	SR19 Skirt Separation	78.4	50.37	1769.56	34.171	120.732
4	Fairing separation	123.3	109.61	2870.05	33.431	120.918
5	Stage 2 separation	128.1	116.63	2899.22	33.327	120.944
6	Stage 3 ignition	130.3	119.80	2899.22	33.328	120.956
7	Stage 3 burnout	203.5	244.16	2888.89	30.865	121.556
8	Stage 3 separation	623.9	734.46	5099.42	13.026	125.695
9	Stage 4 ignition	634.9	736.83	5095.82	12.583	125.794
10	Stage 4burn out/orbit insertion	763.8	741.00	7621.09	5.5705	127.333

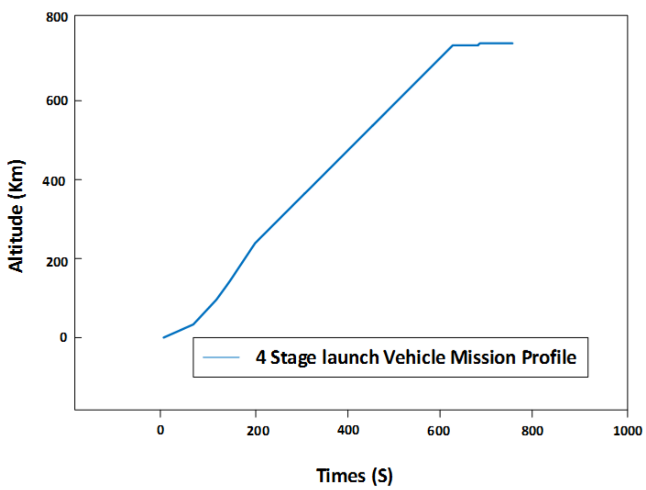

Fig. 4. Typical Minotaur launch vehicle mission profile

In the process of designing the orbital specification, the launch point and payload weight were considered as constant values. The required speed to reach the orbit must be calculated as well as the deceleration coefficients of speed. The most important factors for deceleration of launch vehicle speed are: gravity, thrust vector control force, aerodynamic drag force and loss due to non-uniform motor operation. The amount of loss of speed can be characterized by Loss Velocity Coefficient at each stage is calculated by the following Konstantin Tsiolkovsky relation.

$$\Delta V = -g * I_{sp,i} * ln\mu_{K}, \quad (12)$$

In this relation ΔV is the specific velocity of each stage, g is acceleration of gravity of the Earth, I_{SPi} is the specific impulse amounts of each stage, and μ_{Ki} is the ratio of the final weight of the stages which is calculated by energy-mass coefficients. The amount of calculated ΔV is not the final speed of stage. Factors such as control vector force, Engine performance, gravity and aerodynamic drag force of each stage and deceleration of speed of each stage should be deducted from the final speed of each stage. With the following equations, the amount of deceleration on each stage of the launch vehicle is calculated.

$$\Delta V_{control}=\rho_{\mathrm{control}} *\Delta \mathrm{V} \tag{13}$$
$$\Delta V_{Eng}=\rho_{\mathrm{Eng}} *\Delta \mathrm{V} \tag{14}$$
$$\Delta V_{Grav}=\rho_{\mathrm{Grav}} *tb \tag{15}$$
$$\Delta V_{Aero}=\rho_{\mathrm{Aero}}* \quad \Delta \mathrm{V}*S/\mathrm{Th} \tag{16}$$

where $\Delta V_{control}$ is the amount of speed loss due to control vector force, ΔV_{Eng}, is the amount of loss speed due to Engine performance, ΔV_{Grav} is the amount of speed loss due to gravity and ΔV_{Aero} is the amount of loss due to aerodynamic drag force. In these relationships, tb is the time of burning, T_h is thrust force, S is the launch vehicle cross –sectional area and ρ is the speed loss coefficient on each stage. The amount of speed loss coefficients can be obtained from experimental tests or using speed characteristics of similar launch vehicle such as velocity characteristic, the amount of thrust and the time of burning of each stage. Table 3 illustrates the values of speed loss coefficients calculated for design launch vehicle. The amounts given in Table 3 are calculated according to mission profile of launch vehicle. For example, ρ_{Grav} is the slope of Gravity loss-flight time graph of launch vehicle between two consecutive stages. Although the gravity loss coefficient is not constant for stage 3 as the slope changes it will be approximated for simplicity of calculations. Fig. 5 shows gravity loss changes over time during launch vehicle mission profile obtained from Mirshams (Mirshams, 2008).

Table 3. Velocity loss coefficient on the designed four stage launch vehicle

loss coefficient	Stage1	Stage2	Stage3	Stage4
ρ_{Aero}	1.655	0	0	0
ρ_{cont}	0	0.045	0.076	0.076
ρ_{Eng}	0.014	0	0	0
ρ_{Grav}	1.46	1.46	1.36	1.26

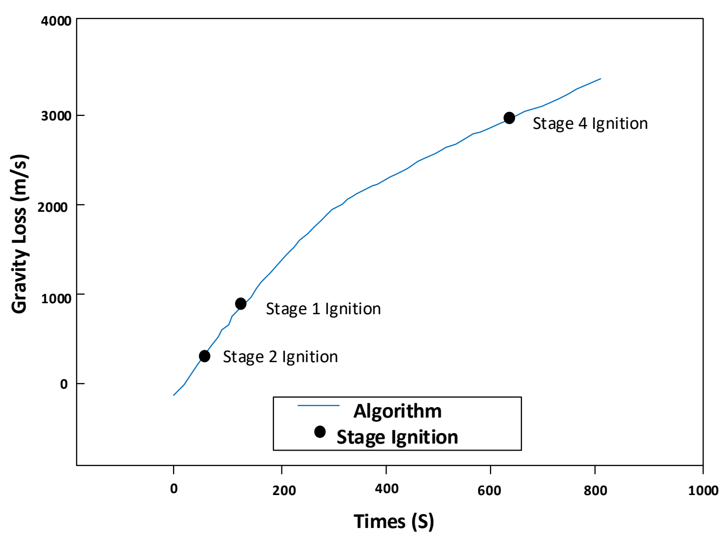

Fig. 5. Gravity loss changes over time

By using these coefficients, the required speed to reach the orbit is calculated. The final speed of the launch vehicle at each stage is obtained from the Eq. (17):

$$\mathrm{V}=\sum_{i=1}^{4} \Delta V-\sum_{i=1}^{4} \Delta V_{Aero}-\sum_{i=1}^{4} \Delta V_{control}-\sum_{i=1}^{4} \Delta V_{Eng}-\sum_{i=1}^{4} \Delta V_{Grav} \tag{17}$$

For calculating speed in orbit of the mission first the Earth's rotation speed at launch point is calculated by Eq. (18).

$$\mathrm{Vearth}=\omega_E R_E \cos(\varphi) \tag{18}$$

where, ω_E is the rotational speed of the earth and φ is latitude of the launch point. Then the speed mission orbit is calculated by Eq. (19).

$$V_c=\sqrt{\frac{\mu}{H_e+R_E}} \tag{19}$$

where μ is the gravity constant, H_e is the height of the orbit and R_E is the Earth's radius. Finally, required speed for the launch vehicle in orbit is calculated by equation (20).

$$V_F = \sqrt{|V_c|^2 + |V_{earth}|^2 - 2|V_c||V_{earth}| \cos i} \tag{20}$$

where, i is the inclination of the orbit. Table 4 shows the amount of parameters needed for calculating the required speed value.

Table 4. Amounts of mission velocity

ω_E constant rotational	φ latitude	R_E is Earth's radius	**μ gravity constant**	H_e**The height of the circular orbit**	**i Orbital inclination**
2.5 ×10-3 (rad /sec)	37.0°	6,378 (km)	**3.986 ×1014 (m3 /sec2)**	**400 (km)**	**40.0°**

Table 5 shows the results of the speed calculations. Refer to (He, 2002) for more information on the computational relationships.

Table 5. Amounts of computed velocity

Required speed(V_F)	**Orbit speed (V_C)**	**Earth's speed at launch point(V_{earth})**
7.38 km/s	**7.66 km/s**	**0.41 km/s**

In order to make sure about success of mission the fallowing equation must be confirmed

$$V = V_F \pm 3\% V_F \tag{21}$$

4.1.2. Determination of Technical Parameters

The technical variables of the problem which have a significant impact on the amount of constraints and objective functions must be chosen. Initial data for launch vehicle design, such as the mass, structure and propulsion characteristics of launch vehicle blocks, are obtained from specification of existing statistical information or simple calculations. In this research, a group of Four-Stage launch vehicles with different classes of mass and payload are determined and their specifications are extracted. This initial data is optimized by multi-objective optimization methods based on the type of flight mission, constraints, available construction technology (Mass-Energy coefficients) and the specified target functions. It should be noted that these values can be changed in the optimization software. Table 6 shows the range of design variables at different stages of the launch vehicle. In Table 6, μp represents the weight ratio between two stages, $n_{o,i}$ represents the ratio of thrust to weight, D stage diameter, L stage Length and Isp is the specific impulse of each stage. These parameters are obtained from the specifications of past successes of solid four stage launch vehicles such as Minotaur, Start 1, etc. All of these parameters are defined as input variables in the multi-objective optimization software tool.

Table 6. Range of technical variables from statistical data (Braun et al., 1996; Bruhn, 1967)

Parameter	**Stage1**	**Stage2**	**Stage3**	**Stage4**
μ_p	0.35-0.44	**0.37-0.45**	0.2-0.25	0.21-0.27
n_{oi}	3.44-3.5	**3.44-3.94**	3.56-3.77	0.76-2.78
D	1.61-1.65	**1.32-1.60**	1.27-1.50	1.27-1.40
L	7.35-8.5	**5.2-6**	2.2-3	1.34-2.5
I_{sp}	232-263	**280-291**	280-285	283-295

4.1.3. Determining Performance Parameters

The limits of the launch vehicle and Mass-Energy coefficients depend upon the capabilities of available manufacturing technology, and they are determined by technical measurements of past constructed launch vehicles according to standards of manufacturing. For example, according to construction standards, axis of gravity must coincide along the long axis of the launch vehicle but weakness in technology of construction causes non-compliance of the axis of gravity along the long axis of the launch vehicle. The purpose of these coefficients is to examine the feasibility of completing a mission with existing capabilities. The range of these coefficients for the considered design launch vehicle is shown in Table 7 (Mirshams, 2008). These Mass-Energy coefficients are the linking factors between technical parameters (historical data) and technological challenges. As mentioned before in Eq. (4) to Eq. (11) these coefficients affect the weight of the stages also according to equation 26 these coefficients are effective in burning time of the stages. These amounts may be dissimilar in different countries. All of these parameters are defined as input variables in the multi-objective optimization software tool. According to Table 7, $\alpha_{To,i}$, $\alpha_{cy,i}$ and $\alpha_{\delta,i}$ are the Mass coefficients of the launch vehicle at stage i which depend on the manufacturing technology ($\alpha_{\delta,i}$), type of structural system and material type ($\alpha_{To,i}$), and type of launch vehicles control system ($\alpha_{cy,i}$). Also the coefficient $\gamma_{gy,i}$ is the Energy coefficient of the stage nozzle which is dependent on the mass and thrust of the generating system (motor nozzle).

Table 7. Range of design Mass-Energetic coefficients

Parameter	Stage1	Stage2	Stage3	Stage4
$\alpha_{To,i}$	0.05-0.103	**0.09-0.15**	0.05-0.16	0.05-0.16
$\alpha_{cy,i}$	1E-4 - 1E-3	**1E-5 -1E-4**	1E-5 -1E-4	1E-5 -1E-4
$\alpha_{\delta,i}$	3E-5 - 3E-4	**1E-6 -1E-5**	1E-6 -1E-5	1E-6 -1E-5
$\gamma_{gy,i}$	0.01-0.022	**0.01-0.028**	0.01-0.029	0.01-0.029

4.1.4 Computational Relationships Required for Design

During the design process, it is necessary to calculate technical specifications of the design launch vehicle. In this section, the most important of these computational relationships are given.

Weight of stages for i=1, 2, 3:

$$M_{o_i} = \frac{\left(\sum_{i+1}^{4} M_{o,i+1} + \quad M_{paylod} \quad\right) * \ (1-\mu_{pi})}{\mu_{pi}} \tag{22}$$

Weight of stages for i = 4:

$$M_{o_{i=4}} = \frac{\left(\ M_{paylod} \quad\right) * \ (1-\mu_{pi})}{\mu_{pi}} \tag{23}$$

Trust of stages:

$$Th_{\mathrm{i}} = n_{oi} * M_{\mathrm{o_i}} \tag{24}$$

Fuel Weight stages:

$$M_{fu_i} = M_{o_i} - \ M_{st_i} \tag{25}$$

Burning time stages:

$$t_{b_i} = \frac{g_o I_{SPi}(1-\mu_{k_i})}{no_i} \tag{26}$$

4.2 The Objective Functions of Launch Vehicle

Different optimization functions can be considered for such optimization problems. The main goal in the process of designing launch vehicles is decreasing the total weight. Minimizing the weight of stages, minimizing the number of stages, minimizing the total weight of launch vehicle and each stage structure weight, maximizing the fitness ratios (Length to diameter ratio), decreasing the burning time in early stages and increasing it at higher stages are the most important objective functions in the process of optimizing launch vehicle designs. Optimal weight distribution is an important factor because it can increase speed with lower fuel consumption. The objective functions for the optimization problem of this study are defined in Table 8.

Table 8. The definition of the objective function

parameter	objective function	Type of objective function
M_{st_i}	Structure weight	**Minimize**
M_{o_i}	stages weight	**Minimize**
l_i/D_i	fitness ratio	**Minimize**
$t_3\,,t_4$	final two stages burning time	**Minimize**
t_1, t_2	first two stages burning time	**Maximize**

4.3. Definition of launch vehicle Constraints

After defining the objective functions, the constraints of the optimization problem must be determined. The control of the descending trend of the length and diameter of the stages as well as control of the launch vehicle speed in the mission trajectory are the most important issues to be considered in launch vehicle design. Table 9 shows these constraints and allowable range according to Table 6, 7 and Eq. (21) where Di is the stages' diameter, Li is the stages length and V_F is the required speed.

According to Table 9 length and diameter of each stage must be larger than the next one to keep the launch vehicle in its logical shape and structure. Furthermore, payload size must be smaller than the diameter of stage 4 to be easily installed on.

Therefore, the largest dimension of the payload is conservatively considered to be 10% smaller than the minimum size of the diameter in the 4th stage.

Table 9. The definition of constraints

Range	Control Parameters	Type of constraint
$1.65 \geq D1 \geq 1.61$ $1.60 \geq D2 \geq 1.32$ $1.50 \geq D3 \geq 1.27$ $1.50 \geq D4 \geq 1.27$	$D1 \geq D2 \geq D3 \geq D4$	Control the size of the stages' diameter
$8.50 \geq L_1 \geq 7.35$ $6.00 \geq L_2 \geq 5.20$ $3.00 \geq L_3 \geq 2.20$ $2.50 \geq L_4 \geq 1.34$	$L_1 > L_2 > L_3 > L_4$	Control the size of the stages' length
$1.2 > L_p$	$D4 > L_p$	Control the largest size of the payload
$7.60 \geq V \geq 7.16$	$V = V_F \pm 3\% V_F$	Control the speed of launch vehicle

4.4. Multi-objective Optimization Algorithm Selection

When a multi-objective optimization method and its algorithm of choice is selected, different aspects such as the ability to quickly respond to convergence, execution efficiency, generality and simplicity of use should be considered. In addition, the ability of the optimization algorithm to consider the constraints of the problem and convert it to the nonlinear optimization problem is one of the most important factors for a proper choice. In this research, there are 24 output variables, 3 constraints and 16 objective functions, which need to be minimized or maximized. Fig. 7 shows the overall structure of the model on Modefrontier software with objective functions, constraints and output parameters. For optimization purposes, NSGA II Multi-Objective Algorithm in Modefrontier optimization software is selected. Table 10 shows the parameters' specification of the NSGA II algorithm in the optimization software.

Table 10. NSGA-II set parameters

Algorithm type	NSGAII
Number of design	20
Number of Generation	50
Maximum Number of Evolutions	1000
Crossover Probability	0.9
Mutation probability for Real-Coded Vectors	1.0
Mutation probability for Binary Strings	1.0

5. Designing Four-Stage solid-fuel launch vehicle

In order to design launch vehicle in Modefrontier software, initially the range of design variables (technical and performance parameters) of launch vehicle, including diameter, length to the diameter ratio, thrust to weight ratio, relative payload ratio, specific impulse and mass-energy coefficients are entered for each stage of launch vehicle in the software. Then, the initial population (random method) is determined according to input data range, the NSGA-II multi-objective algorithm is selected then to perform the optimization in an integrated environment and find the best solutions by coupling the input data with the optimization algorithm method. In this process with an Excel file, fixed design parameters (mission parameters) including orbital characteristics, orbital velocities, launch coordinates, velocity drop coefficients, etc. are entered in optimization software. With the integrated environment of Modefrontier software, all design parameters (fixed and variable) are automatically entered into the MATLAB computing environment, and related launch vehicle parameters are computed. The objective functions and constraints are also defined in the Modefrontier software. The integrated environment of Modefrontier software at the same time coupling NSGA-II optimization algorithm with MATLAB's computational code and being able to find the best optimized design for Four-Stage launch vehicle by considering constraints and objective functions. Figure 6 shows Flowchart of finding best optimized designs for Four-Stage solid launch vehicles. Fig. 7 shows the model and connection between the components of designing launch vehicles in Modefrontier software.

According to Fig. 6 input variables of each stage such as $I_{SP,i}$, $n_{o,i}$, D_i , L_i , etc. are defined according to Table 6 and Table 7 as design parameters in separate blocks. The parameters such as velocity loss coefficients, $M_{Payload}$, amounts of mission velocity and required speed (V_F) are determined as fixed parameters. After determining the fixed and variable parameters, the multi-objective optimization algorithm is determined then both of design variables and fix parameters are transferred to the code for calculating Four-Stages launch vehicle design specifications. For each iteration, after calculating launch vehicle design specifications, three constraints are controlled using the code and if all of them are satisfied then the results are reported in separate blocks for each stage and the objective function optimized by multi multi-objective algorithm. This process is continued to find the best solution and the objective functions converge for Four-Stage solid launch vehicles by multi-objective algorithm.

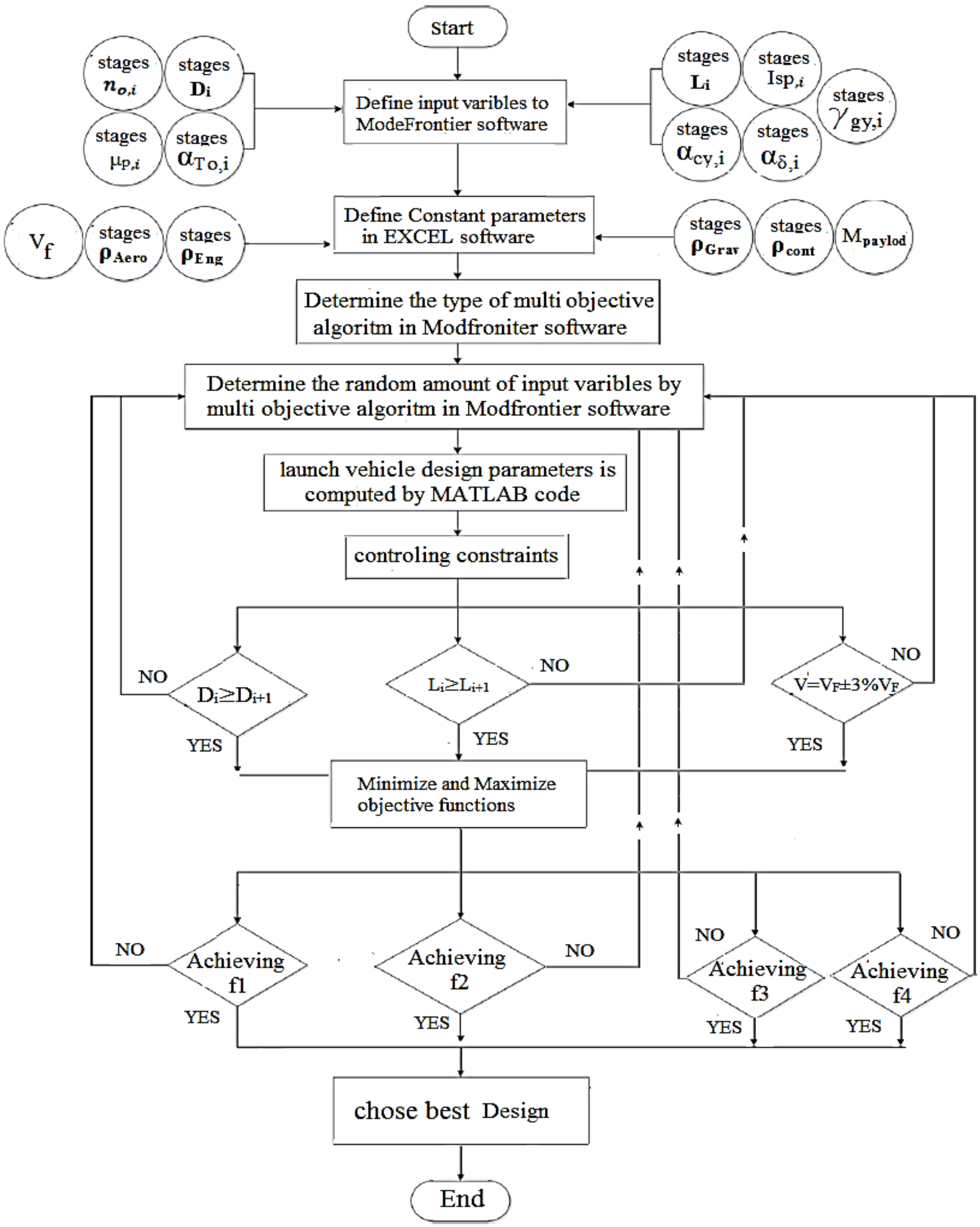

Fig. 6. Flowchart of finding best optimizes designs for four Stage solid launch vehicles

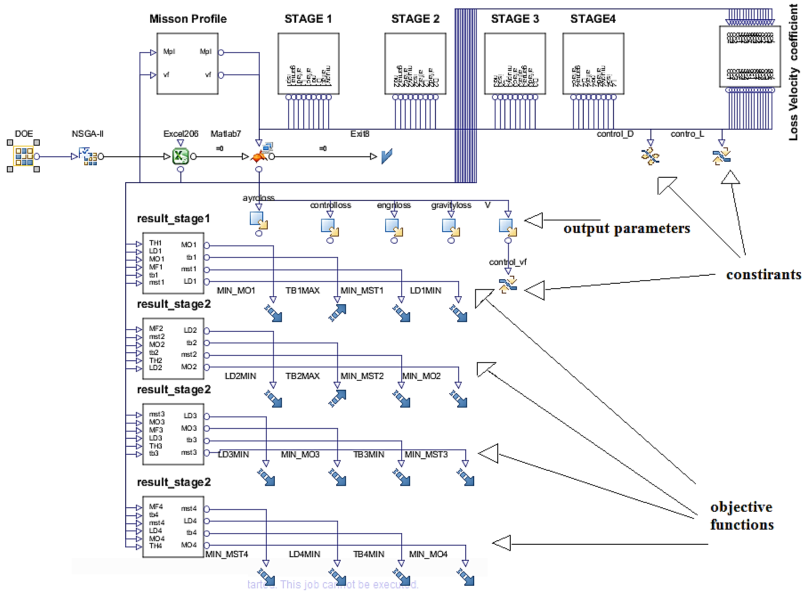

Fig. 7. The overall structure of model on Modefrontier software with objective functions, constraints and output parameters

Fig. 7 shows the first input variables of each stage such as $I_{SP,i,}$, $n_{o,ihich}$, D_i, L_i , etc. which were defined in Modefrontier software as input variables in separate blocks according to Table 6 and Table 7. The constant parameters such as velocity loss coefficients, $M_{Payload}$, amounts of mission velocity and required speed (V_F) are inputted into an excel file as fix parameters. Furthermore, the multi-objective optimization algorithm NSGA-II is defined as a multi-objective optimization algorithm in the integrated environment of Modfrontier software. Each time that the program is run, the values of design parameters are determined by the multi-objective optimization algorithm, and then both design parameter values and fixed parameter values from Excel are entered into MATLAB software. MATLAB code calculates Four-Stage launch vehicle design specifications. For each iteration, after calculating the launch vehicle design specifications, three constraints are controlled by Modefrontier if satisfied, then the results are reported in separate blocks for each stage, and the objective function will be optimized by Modefrontier. This process is persistent to find the best solution and the objective functions converge for Four-Stage solid launch vehicles by Modefrontier, and this process continues until the best answer is selected. Fig. 8 illustrates solutions for satisfying velocity constraints in the Modefrontier software. The horizontal axis of the diagram shows survey number and the vertical axis shows the calculated velocity values with light color and the required orbital speed with dark color as the design constraint.

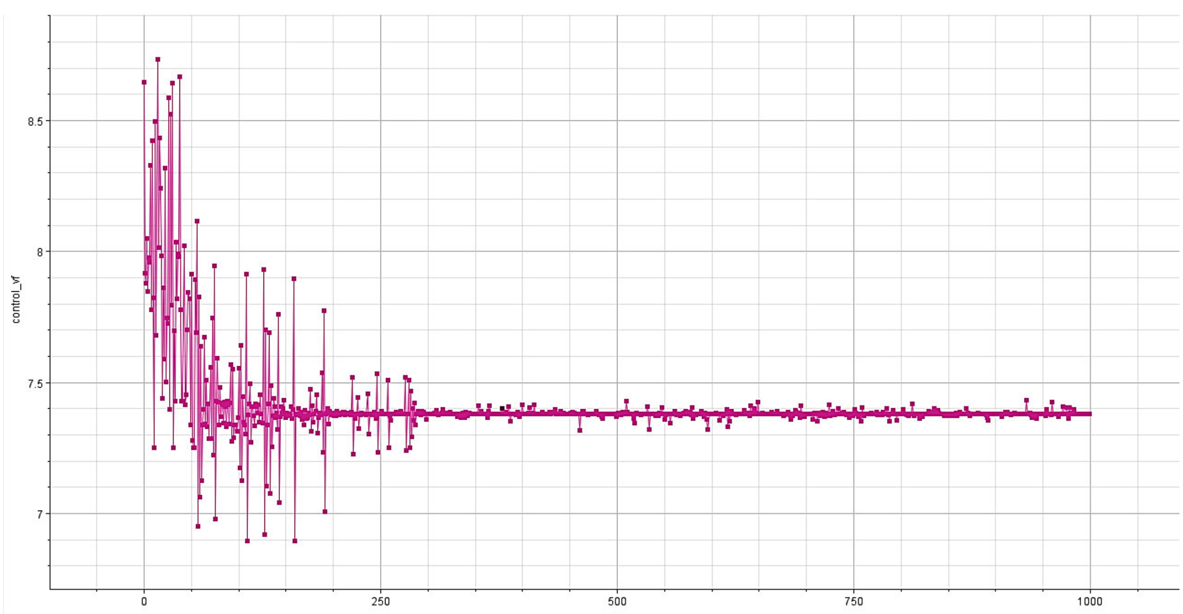

Fig. 8. The graph of convergence solutions and satisfaction of velocity constraint on Modefrontier software

6. Results and Analysis

The problem was set for 1000 evaluations. Fig. 9 shows chart of possible, impossible and the error cases on Modefrontier software.

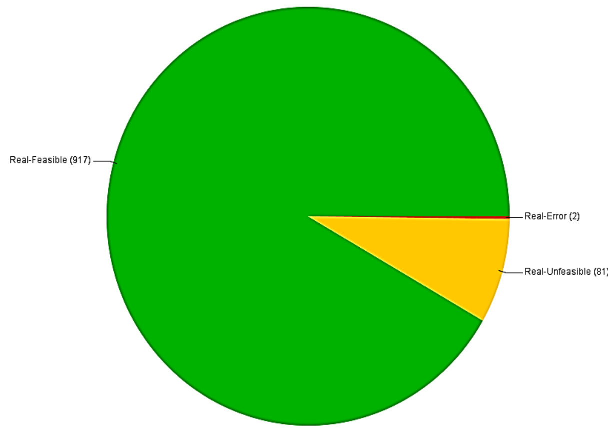

Fig. 9. Chart of possible, impossible and the error cases on Modefrontier software

According to the results, 91.7% of the candidate solutions (i.e. 917) were possible solutions. In other words, the constraints of the problem were satisfied. Also, 8.1% (81) were cases where the constraints were not satisfied and 0.2% of the responses were software errors in evaluation. The results show that it is possible to achieve the designed mission with the available facilities and technology. Also, these results are fully consistent with the permissible range of design parameters of the Four-Stage launch vehicle specifications. It is obvious that in multi objective problems the best solution is the solution that receives better rank in different Pareto fronts. Fig. 10 indicates the results and optimum point of MO1 and MO2 parameters and Fig. 11 indicates the results and optimum point of Mst3 and Mst4 parameters by using the Modefrontier software. According to Figs. 10 and 11 green points are possible solutions, yellow points are the solutions that constraints are not satisfied and black points show solutions that are chosen as the best design values.

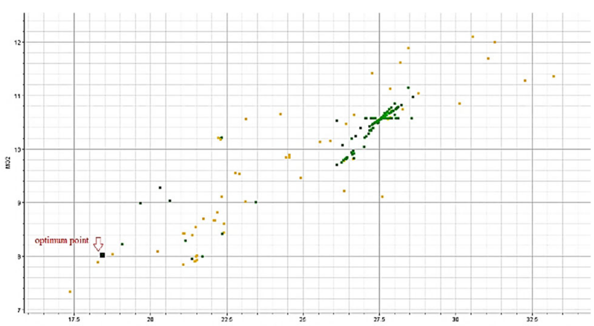

Fig. 10. The results and optimum point of $M_{o,1}$ (ton) and $M_{o,2}$ (ton) parameters

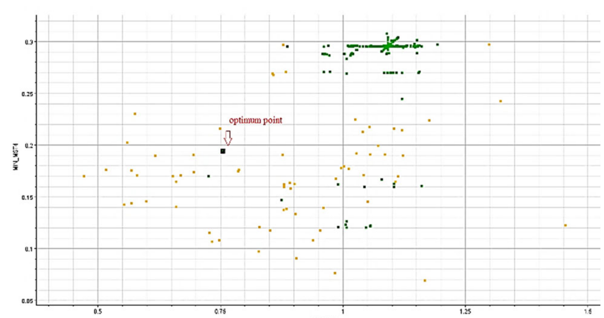

Fig. 11. The results and optimum point of $\boldsymbol{M_{st_3}}$ (ton) and $\boldsymbol{M_{st_4}}$ (ton) parameters.

6.1. Comparison to existing launch vehicles

Table 11 compares the launch vehicle specifications developed by the software to the specifications of US Four-Stage solid fuel launch vehicle Minotaur. The increase in the fitness coefficient denotes technological limitations and causes problems in manufacturing process and movement of the launch vehicle and also led to increase its fuel consumption. Table 11 shows that although the designed launch vehicle has greater length and diameter than the Minotaur launch vehicle, the fitness coefficient (length to diameter ratio) of the designed launch vehicle is lower than Minotaur at least in Stages 2 and 3. In terms of mass characteristics such as total mass, fuel mass and structure mass, the design launch vehicle is lighter compared to the Minotaur launch vehicle. These results show that the weight of the design launch vehicle is decreased by objective functions. In terms of energy properties, the amount of burning time for the first two stages of the design is significantly lower and in 4 Stages is greater than the Minotaur launch vehicle. This issue increases the difficulty to control the launch vehicle, especially in the torsion. Comparing impulse between two launch vehicles shows that the design launch vehicle has a better impulse in stages 2 and 3 but in stages 1 and 4 Minotaur has a better impulse. Fig. 12 compares the structural specifications of the designed launch vehicle to Minotaur launch vehicle and Fig. 13 shows the contrast of the energy specifications of the designed launch vehicle with Minotaur launch vehicle.

Table 11. Comparison of designed launch vehicle to Minotaur launch vehicle

Specification	UNIT	Launch Vehicle	Stage1	Stage2	Stage3	Stage4
Length	m	Minotaur	7.353	5.214	2.187	1.34
		Design case	**8.174**	**5.392**	**2.234**	**1.808**
Diameter	m	Minotaur	1.66	1.32	1.27	1.27
		Design case	**1.628**	**1.517**	**1.353**	**1.305**
Length To Dimeter Ration(L/D)	-	Minotaur	4.42	3.95	1.72	1.05
		Design case	**5.020**	**3.554**	**1.650**	**1.385**
Total Weight	Mg	Minotaur	23.081	7.033	4.342	1.18
		Design case	**18.438**	**8.012**	**4.761**	**1.086**
Fuel Weight	Mg	Minotaur	20.785	6.253	3.924	0. 771
		Design case	**15.949**	**6.580**	**4.005**	**0.892**
Structure Weight	Mg	Minotaur	2.296	0.78	0.418	0.409
		Design case	**2.276**	**1.408**	**0.756**	**0.120**
Thrust	kN	Minotaur	791.8	272.0	160.6	32.2
		Design case	**750.1**	**309.7**	**167.7**	**20.8**
Specific Impulse	s	Minotaur	232.4	291.6	285.5	283.4
		Design case	**241.7**	**283.3**	**282.2**	**287.6**
Burning Time	s	Minotaur	61	67	69.7	67.7
		Design case	**35**	**30**	**43**	**86**

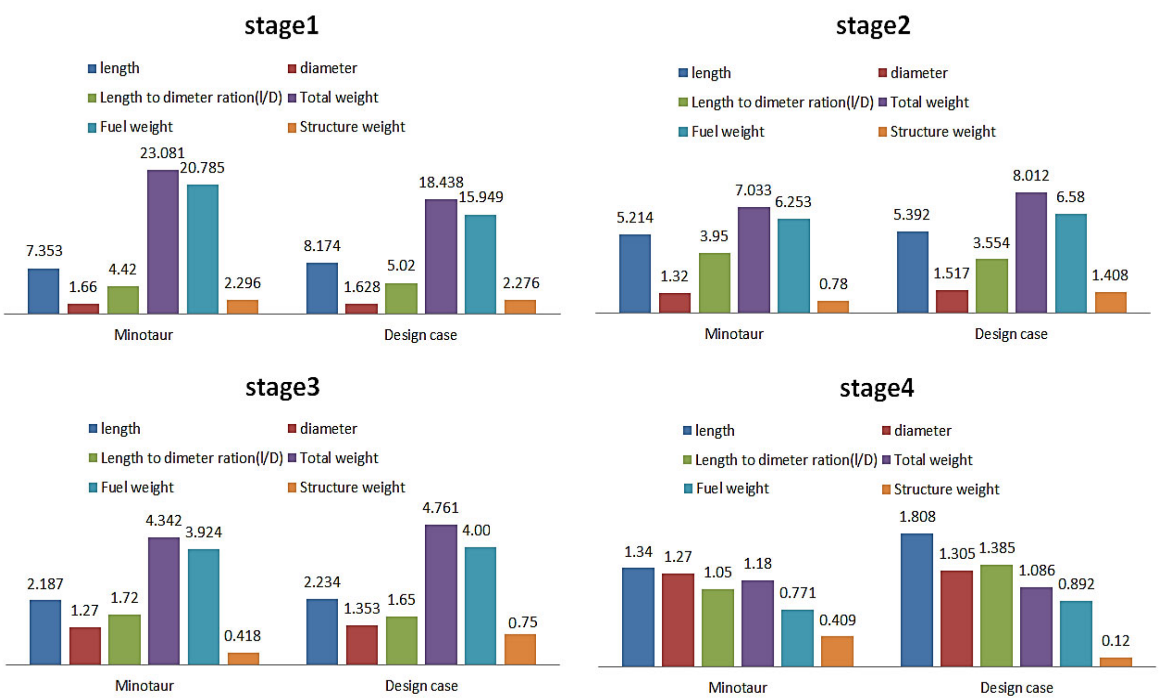

Fig. 12. Comparison of the structural specifications designed launch vehicle to Minotaur launch vehicle

According to Fig. 12 although the structure weight of two launch vehicles is identical in stage 1, the fuel weight of the Minotaur launch vehicle is heavier than the design case launch vehicle. In result, the total weight of the Minotaur launch vehicle is heavier than the design case launch vehicle. Also the length and diameter of the design launch vehicle are greater than that one in the Minotaur launch vehicle but its l/D ratio is lower than the Minotaur launch vehicle. In stages 2 and 3 although the fuel weight of two launch vehicles is approximately the same, the structure weight of design case launch vehicle is approximately twice heavier than the Minotaur launch vehicle in result total weight of design case launch vehicle is heavier than the Minotaur launch vehicle also the length and diameter of the design launch vehicle are greater than that one in the Minotaur launch vehicle but its l/D ratio is lower than the Minotaur launch vehicle. In stage 4 although the fuel weight of design case launch vehicles is approximately heavier than the Minotaur launch vehicle, the structure weight of the Minotaur launch vehicle is approximately four times heavier than design case launch vehicle in result total weight of design case launch vehicle is lighter than the Minotaur launch vehicle also the l/D ratio, length and diameter of the design launch vehicle are greater than the Minotaur launch vehicle.

According to Fig. 13 in stage 1, the thrust force of the Minotaur launch vehicle is greater than the design case launch vehicle. In contrast, the impulse of the Minotaur launch vehicle is lower than the design case launch vehicle and also the burning time of the design launch vehicle is almost twice lower than the Minotaur launch vehicle. In stages 2 and 3, the thrust of the Minotaur launch vehicle is lower than the design case launch vehicle in contrast the impulse of the Minotaur launch vehicle is greater than the design case launch vehicle. Also the burning time of the design launch vehicle is lower than the Minotaur launch vehicle. The thrust of the Minotaur launch vehicle is greater than the design case launch vehicle in contrast the impulse of the Minotaur launch vehicle is lower than the design case launch vehicle. Also the burning time of the design launch vehicle is lower than that one in the Minotaur launch vehicle. In stage 4, the thrust of the Minotaur launch vehicle is greater than the design case launch vehicle in contrast the impulse of the Minotaur launch vehicle is lower than the design case launch vehicle. Also the burning time of the design launch vehicle is greater than the Minotaur launch vehicle. The main reason for the difference between results of Figs. 12 and 13 is due to the difference in the Mass-Energy coefficients that reflect the technology status in two countries. Table 12 compares the energy mass coefficient that reflects the difference of technology level between two launch vehicles. These technologies include the type of control system, the throat and nozzle structural material and connections material as previously mentioned.

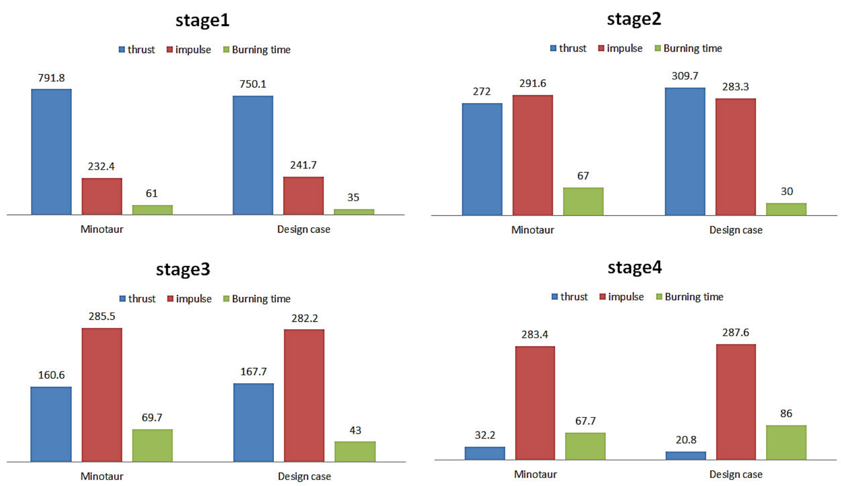

Fig. 13. Comparing the energetic specifications of designed launch vehicle to Minotaur launch vehicle.

Table 12. Comparison of designed launch vehicle Mass – Energetic coefficients to Minotaur

Coefficients	Launch Vehicle	Stage1	Stage2	Stage3	Stage4
$\alpha_{To,i}$	Minotaur	0.1036	0.114	0.163	0.163
	Design case	**0.0918**	**0.1312**	**0.1042**	**0.150**
$\alpha_{cy,i}$	Minotaur	0.00001	0.00001	0.00001	0.00001
	Design case	**0.01064**	**0.01192**	**0.012460175**	**0.0123**
$\alpha_{\delta,i}$	Minotaur	3.8 E-05	1 E-06	1 E-06	1 E-06
	Design case	**0.00913**	**7.34E-04**	**0.008058**	**0.0108**
$\gamma_{gy,i}$	Minotaur	0.022	0.0286	0.0295	0.0295
	Design case	**0.0206**	**0.0250**	**0.017801033**	**0.0200**

Fig. 14 shows the comparison of designed launch vehicle Mass – Energetic coefficients to Minotaur launch vehicle.

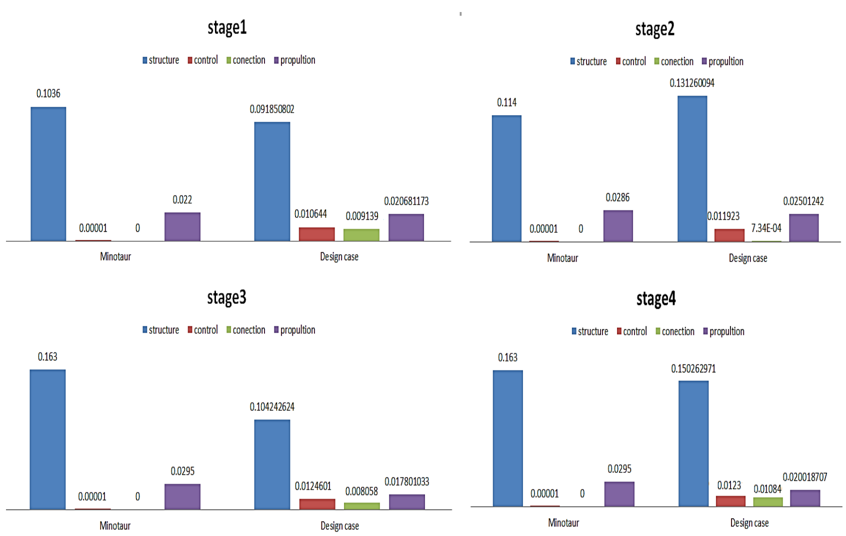

Fig. 14. Comparison of designed launch vehicle Mass – Energetic coefficients to Minotaur launch vehicle.

According to Fig. 14 and Table 12, although there is no obvious difference between the design coefficients of the structure and population (throat and nozzle material) in two launch vehicles, but the control system and connection coefficients in the design launch vehicle are greater and diverge with the US Minotaur launch vehicle.

6.2. Sensitivity Analysis

With the following equations, the amount of sensitivity on each stage of the launch vehicle is calculated. In this analysis, speed parameters are considered as the main parameters and their sensitivity to structural weight, specific impulse and thrust parameters of each stage is investigated. More information on the technical relationships and how to extract sensitivity formulation for the launch vehicle design can be found in the (He, 2002).The sensitivity of velocity to structural weight for Stage1:

$$\frac{\partial V_F}{\partial M_{st,1}} = \left[1 - \rho_{\mathrm{Eng},1} - \frac{\rho_{\mathrm{Aero},1} \mathrm{S}_1}{T_{h,1}}\right] (g_0) I_{SP,1} \frac{M_{fu,1}}{M_{O,1} M_{st,1}} \tag{27}$$

The sensitivity of velocity to structural weight for Stages 2, 3 and 4:

$$\frac{\partial V_F}{\partial M_{st,i}} = \frac{\partial V_F}{\partial M_{st,i-1}} - \left[1 - \rho_{\mathrm{cont},i}\right] (g_0) I_{SP,i} \frac{M_{fu,i}}{M_{O,i} M_{st,i}} \tag{28}$$

The sensitivity of velocity to specific impulse for stage1:

$$\frac{\partial V_F}{\partial I_{sp,1}} = (g_0) ln \frac{M_{O,1}}{M_{Fu,1}} - \frac{1}{I_{sp,1}} \left(\Delta V_{Aero,1} + \Delta V_{Grav,1} + \Delta V_{Eng,1}\right) \tag{29}$$

The sensitivity of velocity to specific impulse for stage 2, 3 and 4:

$$\frac{\partial V_F}{\partial I_{sp,i}} = (g_0) ln \frac{M_{O,i}}{M_{Fu,i}} - \frac{1}{I_{sp,i}} \left(\Delta V_{Grav,i} + \Delta V_{control,i}\right) \tag{30}$$

The sensitivity of velocity to thrust power for stage1:

$$\frac{\partial V_F}{\partial T_{h,j}} = \frac{1}{T_{h,j}} \left(\Delta V_{Aero,1} + \Delta V_{Grav,i}\right) \tag{31}$$

The sensitivity of velocity to thrust power for stage 2, 3 and 4:

$$\frac{\partial V_F}{\partial T_{h,j}} = \frac{\Delta V_{Grav,i}}{T_{h,j}} \tag{32}$$

Table 13 shows the sensitivity analysis of the results obtained for the design launch vehicle

Table 13. the results of the sensitivity analysis for designed launch vehicle

Sensitivity Ratio	Parameter	Unit	Stage1	Stage2	Stage3	Stage4
velocity to structural mass	$\frac{\partial V_f}{\partial m_{st}}$	$\frac{\mathrm{m}}{\mathrm{s}}/\mathrm{kg}$	0.88	-0.67	-3.53	-21.4
velocity to Impulse	$\frac{\partial V_f}{\partial I_{sp,j}}$	$\frac{\mathrm{m/s}}{\mathrm{s}}$	1.42	1.92	1.7	1.89
velocity to thrust	$\frac{\partial V_F}{T_{h,1}}$	$\frac{\mathrm{m}}{\mathrm{s}}/kn$	0.08	0.14	0.34	5.41

According to Table 13, the sensitivity of velocity to structural weight at first two stages is low but it is increased in the final stage. The sensitivity of the launch vehicle speed to the specific impulse parameter is similar in different stages. Furthermore, the sensitivity of the launch vehicle speed to the thrust power is higher in upper stages. Tables 14 and 15 illustrate the detailed specifications for the best solution among the acceptable and converged answers.

Table 14. The final results of launch vehicle conceptual designed that was suggested by Modefrontier software

Parameter	sign	unit	Stage1	Stage2	Stage3	Stage4
Total Weight	$M_{o,i}$	ton	18.438	8.012	4.761	1.086
Fuel Weight	$M_{fu_{,i}}$	ton	15.949	6.580	4.005	0.892
Structural Weight	M_{st_i}	ton	2.276	1.408	0.756	0.120
Impulse	$I_{sp,i}$	s	241.7	283.3	282.2	287.6
Trust To Weight Ratio	no	-	3.48	3.90	3.56	2.46
Stage Weight Ratio	μ_{pl}	-	0.435	0.437	0.234	0.256
Ratio of Length To Diameter	*L/D*	-	5.020	3.554	1.650	1.385
Trust	Th_i	kN	750.1	309.7	167.7	20.8
Burning Time	tb_i	s	35	30	43	86
Connection Coefficient	$\alpha_{\delta i}$	-	0.00913	7.3E-04	0.00805	0.0108
Structural Coefficient	α_{Ti}	-	0.0918	0.131	0.1042	0.150
Control Coefficient	αci	-	0.0106	0.011	0.0124	0.0123
Propulsion Coefficient	γ_{gi}	-	0.0206	0.0250	0.017801	0.0200
Length	Li	m	8.174	5.392	2.234	1.808
Diameter	Di	m	1.628	1.517	1.353	1.305

Table 15. The final results of conceptual designed launch vehicle obtained via Modefrontier

Parameter	Sign	Unit	Result
Orbit Velocity	V	km/s	7.38
Require speed	VF	km/s	7.25

The results showed that the designed space vehicle based on available technological limitations is able to meet the mission requirements, predicted orbital altitude and the required speed to reach the specific orbit. Also, all the constraints of geometric design in terms of height and diameter have been satisfied for different stages and possibly optimized the mass parameters (e.g. structure and stage weight), fitness coefficient (e.g. length ratio to diameter) and energy (e.g. burning time of stages).

7. Conclusion

This research presents a method for the conceptual design of a Four-Stage solid fuel launch vehicle by considering the technical and technological constraints of the manufacturer. This method is based on finding the optimum mass and energy values of the launch vehicle by making an allowance for the constraints in an iterative process. Moreover, the impacts of the technology and other capabilities are considered in this method. For this purpose, NSGAII Multi-Objective Optimization Algorithm is combined with MATLAB software in the integrated environment of Modefrontier software. In this method, the Modefrontier optimization software finds the best possible answers for the mission with respect to the objective functions and mission constraints. Then, by performing a sensitivity analysis and comparing it with existing launch vehicles, the design of the launch vehicle with current features and limitations is investigated. The results showed that the designed space vehicle is able to meet the mission requirements, predicted orbital altitude and the required speed to reach the specific orbit considering the available technological limitations in comparison with the Modefrontier. In addition, sensitivity analysis showed that the sensitivity of designed launch vehicle velocity with respect to the structural weight was low in the first two stages. However, in upper stages (i.e. 3rd and 4th), such sensitivity was increased with respect to the thrust.

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