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Multi-Disciplinary System Design Optimization of a Launch Vehicle Upper-Stage

Mostafa Zakeri[1], Mehran Nosratollahi[1], Alireza Novinzade[2]

ABSTRACT: The design method presented in this paper is related to the upper-stage system and its instrumentation, expedition and facilitation so as to transfer the satellite from the destination orbit to the target orbit. We used an integrated design method with a structure based on multi-disciplinary system design optimization and developed a simple systematic interference method for designing aerospace products. The subsystems' convergence in an optimized environment, matrix relationship, and integration of the subsystems' parameters and presentation of design give results while meeting all requirements and considering the limitations of the design were the main aims of the research. Instead of a merely mathematical optimization design, in the present study a new design method with a systematic multipurpose optimization approach was designed. In this context, the optimization means the parameters are optimized as a result of the design convergence coefficients. Validation of the design method was not only obtained through comparison with a specific product but also with the systematic parameters of all upper-stage systems with a similar operation through the results of statistical design graphs. The approximate similarities of the results indicate an acceptable and genuine design with a quite systematic approach which is better than an unreal and merely optimized design.

KEYWORDS: Upper stage, Systems design, Multidisciplinary design optimization, Systems integration.

INTRODUCTION

In a article named "Technologies for future precision strike missile systems - missile design technology" (Fleeman 2001), there is a survey of missile technology concepts, influential parameters in design, and balance among subsystems, using new technologies with lower weight and cost communicating with the launcher. Overall configuration as well as missile simulation results from such a design method. The detailed explanations for the study are available in his book of Tactical Missiles Design. It needs to be mentioned that design depth is limited in the method yet the functional area is high while lack of integration and systemic communications implementation are the biggest weak points in this method which he explains and completes in his 2012 edition of the book. It is claimed in the study that all main parameters of a missile at conceptual design are taken into consideration while the missile has operational capability. Operational capability as well as systemic relation integration is the first act in the current study.

The history of transition from classic design to modern design is not accountable in this study, however, some developed countries have been able to take advantage of system design and multidisciplinary optimization methods to improve conceptual design process through considerable savings in design time and costs (Olds 1993).

Brown and Olds (2006) surveyed multi-objective optimization techniques of collaborative optimization (CO), modified collaborative optimization (MCO), bi-level integrated system synthesis (BLISS), and all at once (AAO) on a reusable satellite. The study claims that the best design method cannot be chosen since such activities are for research

1.Space Research Institute – System Engineering Department – Space System Design Laboratory – Tehran/Tehran – Iran. 2.Khaje Nasir Toosi University of Technology – Faculty of Aerospace – Systems Division – Tehran/Tehran – Iran.

Author for correspondence: Mostafa Zakeri | Space Research Institute – System Engineering Department – Space System Design Laboratory | Postal Address: 1316943551 – Tehran/Tehran – Iran | Email: mostafa.zakeri5@gmail.com

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purposes only and require various studies on the results. Systemic design activities are briefly mentioned in this study while most of the activities are focused on comparing 3 design techniques. In the final conclusion, a comparison is made among design methods based on running time as well as quality comparisons, which could be said that BLISS designed outputs have higher quality.

Balesdent *et al.* (2011) surveyed various multipurpose optimization methods in space systems design quantitatively. Various MDO methods to design satellite missiles are surveyed in the study and some features, such as strength, price calculations, flexibility and convergence speed and problem implementation are taken into consideration so as to select the most suitable design method in designing launch vehicle. Mathematical equations of optimization of every method as well as main profile of the algorithm with optimization activities of every method are mentioned briefly. Selection of the optimal method to design space system based on the mission and various situations such as implementation time, cost, complexity, etc. is the final conclusion of the study (Balesdent *et al.* 2011).

Riddle (1998) states that using MDO in designing complex systems comes with 2 obstacles. One of them originates from disconnected and nonlinear essence of design process most of mathematical optimization methods are facing. One other unattractive point of MDO methods is design teams' unwillingness to use it and similarity of automatic decision-making with creative process of innovative design.

Tsuchiya and Mori (2002) claimed that in spite of the higher speed of MDO with parametric methods, and based on which studies are reported to improve system and destination optimization for reusable launch vehicles (RLV), they are still recognized unsuitable for space systems design, especially satellite-carrying missiles which are essentially more complicated in configuration steps and trajectory design.

The need to design with a systematic approach in addition to design implementation based on physics of an aerospace product attracts some researchers to optimized systematic design. Aldheeb *et al.* (2012) tried to create an optimized design for a Micro Air Launch Vehicle.

In the present study, optimization and trajectory design are done through a design algorithm with systematic approach to reduce payload mass. The look on functional design physics in the main algorithm of the article and model of subsystems is clear. Villanueva *et al.* (2013) used a systematic approach in an article to design solid fuel engine. Conceptual design in the current study is optimized through a genetic algorithm.

It could be claimed that creation of a systematic approach and transforming MDO to multi-disciplinary system design optimization (MSDO) includes development of MDO methods which develops operational capability based on MDO. MSDO approach is available in a limited number of the articles.

According to Wronski and Gray (2004), one can verify a comprehensive MSDO implementation in a specific case in the Massachusetts Institute of Technology (MIT). The specific importance of the study is that it gives a true expression of MSDO systematic design, multipurpose optimization and an obvious process of the algorithm.

Additionally, design and multipurpose optimization of an aircraft was studied through implementation and integrated design tools so as to predict and optimize the implementation and related expenditures of commercial aircrafts design and production with the aim of reducing the noise through accurate selection of configuration and mission parameters (Diedrich *et al.* 2006). After surveying some design samples, a brief look to upper stage activities are made.

Engine and trajectory design in Casalino and Pastrone (2010) is optimized simultaneously. Systematic design is brief in the study and it could be included among the optimization articles with systematic approach for propulsion engine.

An upper stage activity is presented with a brief look at upper stage engine of solid fuel engine (Casalino and Pastrone 2010). Adami *et al.* (2015) designed an upper stage performed through three forms of MDO. Mathematically, a detailed comparison among the three design methods is presented. The optimal proof of choice is surveyed mathematically.

After considering the aforementioned articles in addition to their quantitative and qualitative analyses, Table 1 presents a summary of their features.

METHODOLOGY

Designing an optimized multidisciplinary system is a modern example of aerospace product design. MSDO can be complex product design process and multidisciplinary engineering systems. In this method, the subsystems are related to each other and to a system in an optimized and converging space. Also the main feature of this method is the

Table 1. Distribution of mass components.

Design methods	Physics-based methods	Mathematical-based methods	Full configuration	Balanced subsystems (converge)	Requirement meeting between subsystems	Subsystems design depth	Imbedded baseline statistics data
Conceptual design classic	×	_	×	_	_	×	×
System design optimization	×	×	_	×	_	×	×
MDO	_	×	×	_	×	_	×
MSDO (this study)	×	×	×	×	×	×	×

presence of human expert in the Designing tool environment and integration of all designing subdivisions. The aim of this approach is the creation of sophisticated and advanced engineering systems that are competitive not only in terms of performance but also in terms of value of life cycle.

The most important properties of MSDO method can be stated as follows:

- Deal with design models of realistic size and fidelity that will not lead to erroneous conclusions.
- Reduce the tedium of coupling variables and results from disciplinary models.
- Allow for creativity while leveraging rigorous, quantitative tools in the design process. Hand-shaking: qualitative *versus* quantitative.
- Data visualization in multiple dimensions.
- Incorporation of higher-level upstream and downstream system architecture aspects in early design: staged deployment, safety and security, environmental sustainability, platform design, etc.

In this procedure, design algorithm similar to other design algorithms is not of tree type or merely a mathematical optimization. The MSDO algorithm is designed to link all the subdivisions directly to each other and the best convergence is applied according to physics and subdivisions. In this way, the designer can easily put all the limitations and restrictions of design to work. In this paper, the concepts of design and multistep optimization are used in a strategic computing environment to design upper-stage which transfers from parking orbit to the target orbit. Presented parameters in this procedure are classified as:

- Design or independent variables: including fuel mass ratio, engine structural mass ratio to the whole engine, etc.
- Simple limitations: including mass ratios, I_{sp}, etc.
- Restrictions: diameter, orbital altitude, payload.
- Combined merit functions: aiming at minimizing the total mass and designing the best way of sending the satellite to the target orbit.

The most important specifications of the MSDO algorithm are as follows:

- Offering new approaches in systematic design derived from MSDO designing method.
- Using statistical processing in the design process (increasing accuracy and rapid convergence).
- Offering innovative convergence methods.
- Optimizing system and subsystems' design parameters by using communication matrixes.
- Convergence of designing upper stage with previous stages of the rocket.
- Integrating all the design parameters and meeting all the restrictions and requirements.
- Ability to enter any new special requirement in the way of designing.

Algorithm design of upper stage includes the following:

- Statistical designing and analysing statistical data.
- Designing the layout of subsystems and subdivisions.
- Dynamics and trajectory design.
- Propulsion system design and tank design.
- Feeding system design.
- Analysis of mass-dimensions and mass associated with the previous stage of launch vehicle design.
- Structural design and stiffener.
- Systematic analysis (configuration, integrating and optimization).

All the requirements and limitations of designing the upper stage is done based on the objectives, bottlenecks, and administrative constraints. These constraints are applied in all the phases with the presence of the designer in the design environment. All the requirements and restrictions can be classified as follows:

- The requirements of trajectory.
- The requirements of the launch vehicle and launch.
- The requirements of subsystems and subdivisions.
- The requirements of construction and assembly.
- The limitations in choosing the hardware.

Basic hypotheses regarding the design of upper stage are as follows:

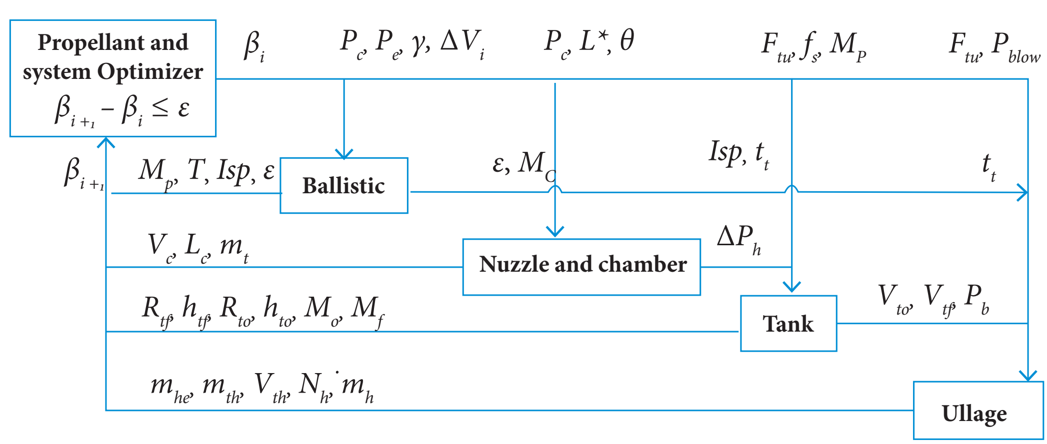
- Payload.
- Parking orbit and target.
- Mechanical properties.
- Helium mechanical properties.
- The characteristics of the chosen fuel.
- Safety factors.
- Temperature of tanks and the flame.

The main body of the communication among subsystems, within each other, and the system is created according to the design matrix. Design matrixes are the designer's guide for displaying design communications and the effects of the parameters on each other (Peoples and Schuman 2003). The communicative matrix between the components of propulsion system is shown in Fig. 1 due to the importance of the communication of the propulsion system components. The most important design matrix is the comprehensive design matrix which is shown in Fig. 2. Only the main parameters of design are mentioned in the matrixes.

In designing the MSDO algorithm, several optimization and convergences were used. The goal of optimization is to achieve the least amount of goal parameters; however, the goal of convergence is to converge all design parameters within each other as well as meeting all the requirements and limitations. Optimization includes:

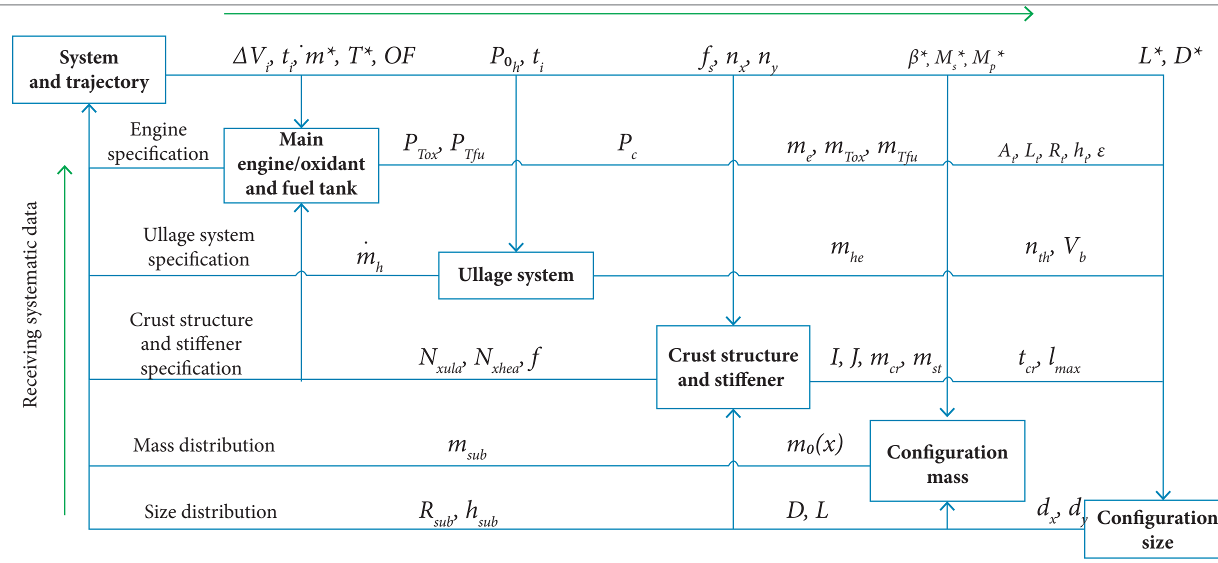
- Optimizing comprehensive design matrix based on the MSDO and through genetic algorithm.
- Optimizing trajectory through genetic algorithm.
- Optimizing propulsion system through genetic algorithm.
- Optimizing total mass through genetic algorithm.
- Optimizing the thickness of the crust and stiffener based on the buckling test and through genetic algorithm.

In Table 2, one can see the above-mentioned optimization properties. Design convergence items in MSDO algorithm include the acceleration of design process according to the statistical equations (reducing the time while increasing accuracy).



P_c and P_e: Chamber and exit pressure; P_{blow}: Blow tanks pressure; P_h: Distribution of the pressure gradient;
L^*: Combustion characteristic length; ΔV_i : Energy change in each burn t_t:
Burn time distribution; R_t and ht : Radius and height of the tank;
V_{th} and m_{th}: Volume and mass of the helium tanks'
m_h: Blow rate; Nh: Number of blowing tanks; Vto, Vtf and Vc: Fuel tank capacity oxidation and combustion;
ε: Expansion ratio; T and Isp: Thrust and specific impulse; L_c: Combustion chamber length;
F_{tu}: Yield stress; f_s: Reliability factor; t_s: Thickness of body structures; t_{sm}: Maximum thickness of body structures.

Figure 1. Propulsion system matrix.



OF: Oxide fuel ratio; R_{sub} and h_{sub}: Subsystem dimensions; A_t, L_t, R_t, and h_t: Engine size distribution;
n_x and n_y: Axial and lateral acceleration coefficients; d_x and d_y: Dimension distribution;
$m_0(x)$ and m_{sub}: Mass distribution and mass of the subsystems; m_e and m_T: Mass of the engine and of the tanks;
N_{equ}: Equivalent stress distribution; V_b: Volume of gas tanks.

Figure 2. Design correlations matrix.

Primary values for design are obtained with statistical equations. Propulsion system convergence through propulsion system mass factor is defined as (Motlagh *et al.* 2013):

$$\beta = M_s / (M_s + M_p) \tag{1}$$

$$\beta_n = \beta_{n-1} < \varepsilon \tag{2}$$

The convergence of upper stage compared with the previous stage launch vehicle is:

$$\alpha = M_{p1} / M_{p2} \tag{3}$$

$$\alpha_n - \alpha_{n-1} < \varepsilon \tag{4}$$

In Fig. 3, it is presented the interference between convergence

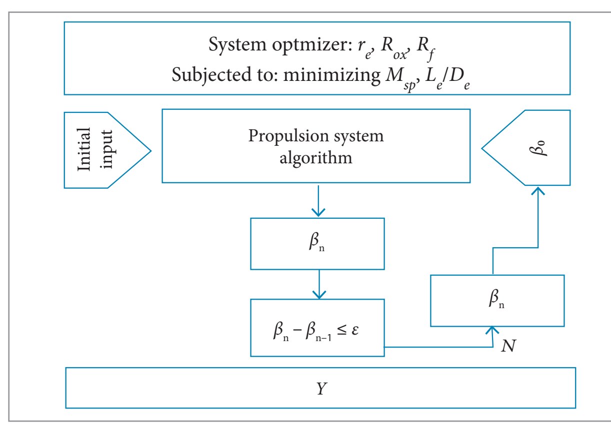


Figure 3. Interference between convergence and optimization.

Table 2. Optimization properties.

System optimizer	Subjected to	Generations	Population	Crossover	Mutation
MSDO optimization	*Min*: M_t and L/D	100	100	Scattered	Uniform (0.2)
Orbital optimization	*Min*: M_p	100	20	Scattered	Uniform (0.2)
Propulsion optimization	*Min*: M_{sp} and L_e/D_e	100	100	Scattered	Uniform (0.2)
Total mass optimization	*Min*: M_t	100	20	Scattered	Uniform (0.2)
Structure optimization	*Min*: M_{st}	100	20	Scattered	Uniform (0.2)

and optimization of the algorithm (Fig. 4). Design variables are described in Table 3.

Design output includes the following:

- Systematic parameters of upper stage.
- Subdivisions' mass-dimension distribution.
- Systematic data of subdivisions.

Table 3. Design variables.

System optimizer	Subjected to	Design variables	Name	Unit	Limitation	Description
MSDO Optimization	*Min*: M_t and L/D	P_c	Chamber pressure	bar	$5 < P_c < 15$	–
		OF	Oxide to fuel ratio	–	$3.5 < OF < 4.5$	According to fuel
		N_h	Number of helium tanks	–	$2 < N_h < 12$	
		N	Thrust to weight ratio	–	1.5< n <4.5	According to configuration
Orbital Optimization	*Min*: M_{sp}	r_{pt}	Transfer orbit Perigee	km	200 ...36000	According to orbit design
		r_{at}	Transfer orbit Apogee			
Propulsion Optimization	*Min*: M_{sp} and L_e/D_e	r_e	Nozzle exit	m	$r_e < D$	–
		R_{ox}	Oxidizer tank profile		$R_{ox} < D - t_s$	
		R_f	Fuel tank profile		$R_f < D - t_s$	
Total mass Optimization	*Min*: M_t	–	Selection of components	–	–	According to data feasibility
Structure Optimization	*Min*: M_{st}	t_s	Body thickness	m	$t_{sm} - 2 < t_s < t_{sm}$	–
		J	Stiffener rigidity	m⁴	–	Stiffener selection
		–	Structural Materials	–	–	–
Propulsion system convergence	$\beta_n - \beta_{n-1} < \varepsilon$	β	Propulsion Mass factor	–	$0.8 < \beta < 1.5$	According to statistical data
Stages fuel mass ratio convergence	$\alpha_n - \alpha_{n-1} < \varepsilon$	α	Stages mass factor	–	$0.08 < \alpha < 0.3$	According to statistical data

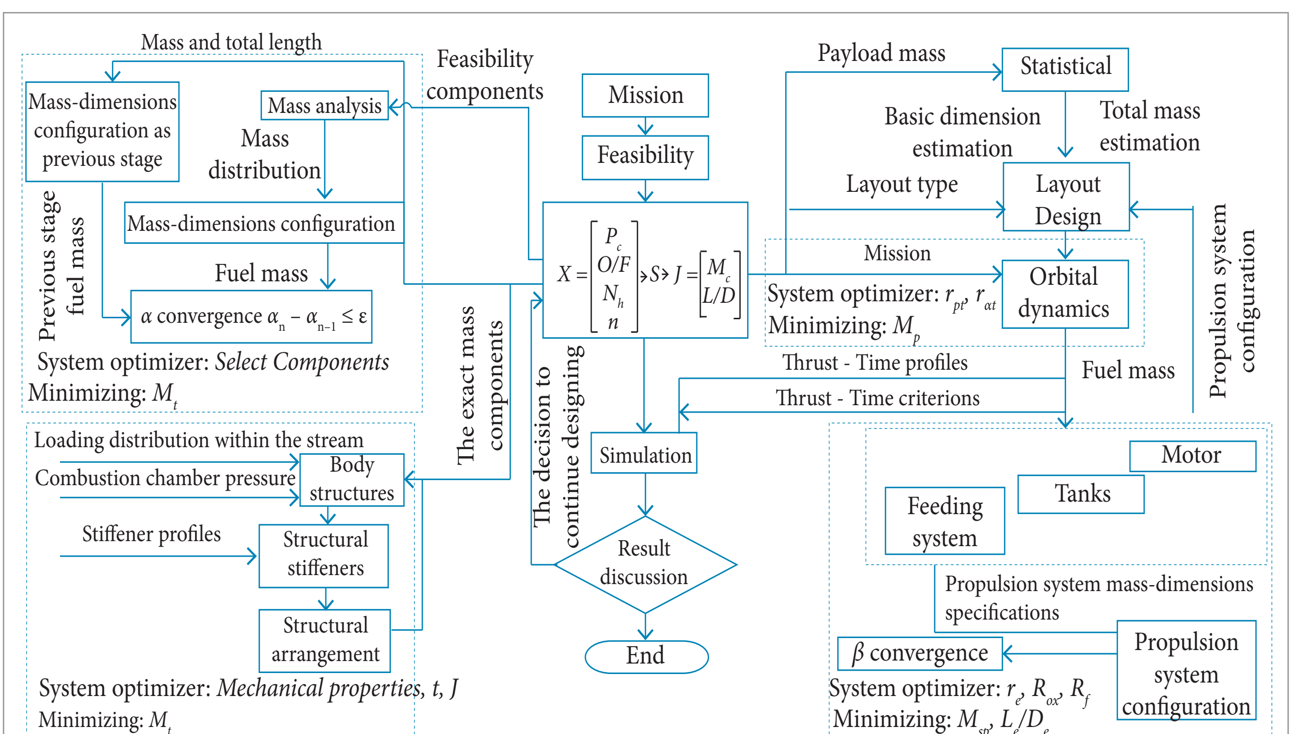


Figure 4. MSDO algorithm.

However, the most important outputs of the algorithms include the following:

- Multidisciplinary optimization system.
- Adaptation of all design parameters.
- Meeting all the requirements and limitations.

DESIGN METHODOLOGY OF SUBSYSTEMS

Statistical Design

Preliminary estimation of systematic parameters in upper stage design process is of utmost importance due to the following reasons: creating a basic configuration for upper stage and faster convergence and more optimal design.

In statistical design via obtained data, the created population and needed graphs were extracted to create initial input (Mirshams and Khaladjzadeh 2010). For instance, 2 sample graphs are given in Figs. 5 and 6. The payload mass is the first and the most important input in designing an upper stage.

$$M_k = 1.58 + 1.2 M_{pay} \qquad (R^2 = 0.96) \tag{5}$$

where: M_k and M_{pay} are the final and payload mass.

The thrust to weight relative to burn time is represented by:

$$n = 19.38 - 0.035t + 2.6\times10^{-5}t^2 - 6.88\times10^{-9}t^3 \quad (R^2 = 0.98) \tag{6}$$

Statistical equations are derived as follows: μ_p and μ_f payload mass ratio and dry mass ratio.

$$M_F = 0.026\, M_{pay}^2 + 0.799\, M_{pay} + 2.546 \tag{7}$$

$$M_0 = -0.066\, M_F^2 + 3.439\, M_F + 3.004 \tag{8}$$

$$M_P = 0.91\, {M_0}^{0.917} \tag{9}$$

$$\mu_p = 3.267\, \mu_f^2 - 0.559\, \mu_F + 0.273 \tag{10}$$

$$n = 2880\, t^{-0.94} \tag{11}$$

$$T = 3.668\, n^2 - 20.21\, n + 107.7 \tag{12}$$

$$\dot{m} = 0.525\, n^2 - 0.937 n + 14.45 \tag{13}$$

$$T = 4422\, \left(\dot{m} = \frac{M_P}{t}\right) - 0.497 \tag{14}$$

$$L = 109.5\, e^{-2.33D} \tag{15}$$

$$LD^2 = 0.323 M_0^2 - 7.997 M_0 + 116.6 \tag{16}$$

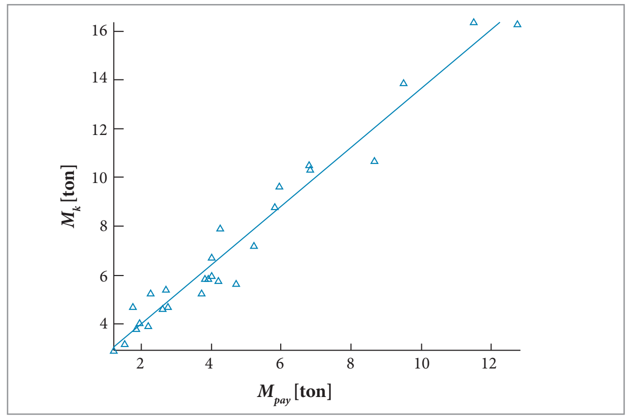


Figure 5. Dry mass *versus* payload mass.

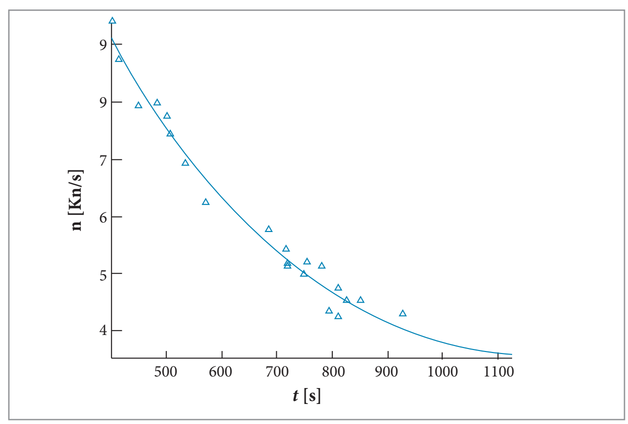


Figure 6. Thrust/weight *versus* burn time.

Trajectory Simulation and Design

In this study, trajectory design (Fig. 7) is done based on Hahman's approach as well as 2 references (Chobotv 1996; Curtis 2005).

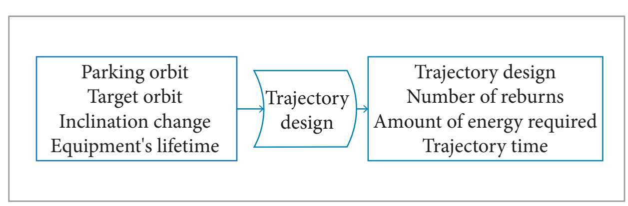


Figure 7. Trajectory design algorithm.

Mass Analysis

According to statistical studies, linear relationship between the dry mass of the rocket block with the portable mass of the same rocket block represents the similarity of mass ranges of the whole subsystems and their subcategories in each block with the same objectives. Thus, determining mass ranges of the main subcategories based on the main effective parameters is viable. In Table 4, all the objects and the way to achieve them are illustrated. Final mass of the upper stage is achieved via the table and considering all the mass parameters. The final mass of upper stage can be achieved with all mass parameters presented in Table 4.

Table 4. Distribution of the mass components.

Upper Stage		Estimated mass	Computational mass	Selective mass
Structure	Satellite installation stand	×		
	Body structure		×	
	Tank protector	×		
	Motor holder	×		
	Nacelle	×		
	Motor front and back flange		×	
	Separation equipment	×		
Feeding system	Helium		×	
	Helium tank		×	
	Other equipment	×		
Propulsion	Fuel tank		×	
	Oxidant tank		×	
	Connection tunes	×		
	Motor elements	×		
	Combustion chamber		×	
	Nozzle		×	
	Other components of motor	×		
Guidance and control hardware	Flight computer			×
	Guidance control block			×
	Inertia measurement block			×
	Valves	×		
	Accessories	×		
	Telemetric system	×		
Actuators	Electromechanical actuators			×
	Cables and electrical connections	×		
	Thrusters			×
	Braking motor			×
	Acceleration motor			×
Cases guidance	Central computer			×
	Sensors			×
	Gyro planes			×

Propulsion Subsystem Analysis

Different specifications of a space propulsion system with other propulsion systems of a rocket are summarized as (Friedman and Kenny 1965):

- Different outside conditions (space conditions).
- On-Off numbers (based on the trajectory design).
- Less thrust-to-weight ratio.
- The use of pressure feed system (high accuracy but low thrust) (Sutton and Biblarz 2001).

Figure 8 shows the propulsion analysis algorithm.

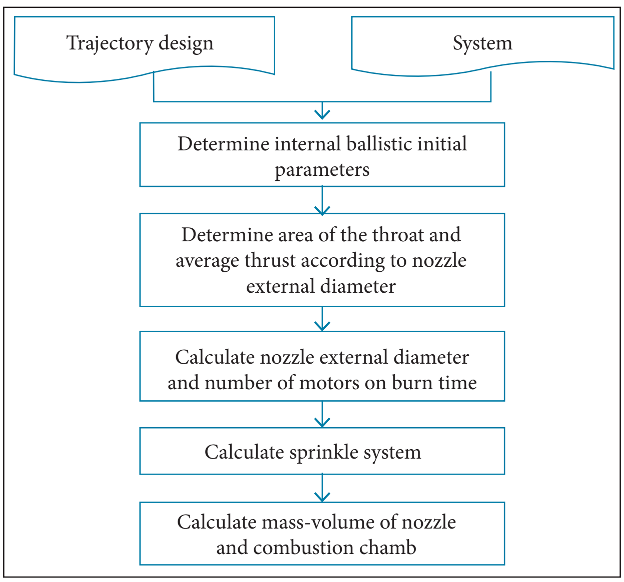


Figure 8. Propulsion sub-algorithm.

Propulsion System Convergence

In the first design loop, the amount of fuel mass and the final mass of upper stage in every burning is calculated through the following equations:

$$M_{f1} = M_0 e^{-\left(\Delta V_1 / I_{sp} g_0\right)} \tag{17}$$

$$M_{P1} = M_0 - M_{f1} \tag{18}$$

$$M_{fi} = \left(M_0 - \sum_{k=1}^{i-1} M_{pk}\right) e^{-\left(\Delta V_i / I_{sp} g_0\right)} \tag{19}$$

$$M_{Pi} = M_0 - \sum_{k=1}^{i-1} M_{pk} - M_{fi} \tag{20}$$

$$M_P = \sum_{k=1}^{i} M_{Pi} \tag{21}$$

After propulsion system design convergence, the amount of optimal fuel mass in every engine-on status is achieved. In the present study, the optimal propulsion system design is achieved through converging the structural convergence factor (β). Using upper stage mass in every design loop, all subsystems' propulsion design is achieved and then the new value for fuel mass and convergence factor is obtained.

$$M_{pj} = \frac{1-\beta}{\beta}\left(M_{sh\,j} + M_{se\,j} + M_{sT\,j}\right) \tag{22}$$

where: j is the integration loop counter design; M_{sh}, M_{se}, and M_{sT} are dry mass of subsystems. The equation for convergence coefficient (β) is defined as follows:

$$\beta = \frac{M_s}{M_s + M_p} \tag{23}$$

The internal relations of the propulsion system are shown in Fig. 1.

Feeding System Analysis

Controlling pressure in fuel tanks is easily possible using the pressure system feed. Furthermore, the simplicity of adjusting pressure in pressure system feed determines its high reliability, thus the process of switching and flow control is easily possible. Output flow could be controlled with installing a heater or pressure control valves. Generally, the concurrent process of disembarkation of capsules containing compressed gas and filling propellant tanks could be shown via Eqs. 24 and 25 (Huzel *et al.* 1992).

High-pressure tanks (capsules):

$$I\begin{cases} \dfrac{dp}{dt} = \dfrac{(k-1)Z}{V}\left(\dfrac{dQ}{dt} - \dot{m}_d h_d - \dfrac{PV}{(k-1)Z^2}\dfrac{dZ}{dt}\right) \\ \dfrac{d\rho}{dt} = -\dfrac{\dot{m}_d}{V} \\ T = \dfrac{P}{R\rho} \end{cases} \tag{24}$$

Propellant tanks:

$$II\begin{cases} \dfrac{dp}{dt} = \dfrac{(k-1)}{V(t)}\left(\dfrac{dQ}{dt} - \dot{m}_i h_i - \dfrac{k}{(k-1)}P\dfrac{dV(t)}{dt}\right) \\ \dfrac{d\rho}{dt} = \dfrac{1}{V(t)}\left(\dot{m}_i - \rho\dfrac{dV(t)}{dt}\right) \\ T = \dfrac{P}{R\rho} \end{cases} \tag{25}$$

where: Q is heat transfer between the gas blowing and its environment; Z is the gas compressibility factor; V is the volume control; T is the temperature; P is the pressure; $\dot{m}_i$. $\dot{m}_d$ are blowing gas mass flow rate input and output volume control; h_i and h_d are blowing gas enthalpy entry and exit control volume.

The sub-algorithm of feeding system is shown in Fig. 9.

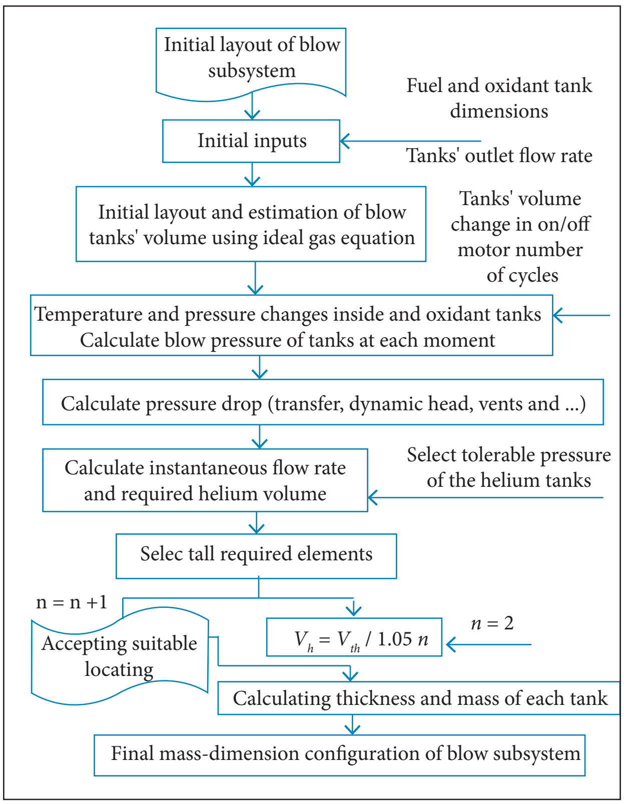


Figure 9. Blow system sub-algorithm.

Volume, thickness, and the mass of every helium tank is obtained from the following equations (Humble *et al.* 1995):

$$V_h = \frac{\gamma P_{po}}{P_{boo} - P_{boe}} V_{To} + \frac{\gamma P_{pf}}{P_{bfo} - P_{bfe}} V_{Tf} \tag{26}$$

$$t_G = \frac{P_{ho}}{2\sigma_{hw}} R_h \tag{27}$$

$$M_{Th} = \frac{3}{2}\frac{\rho_{hw}}{\sigma_{hw}} \frac{2\gamma V_b}{1 - \left(\frac{P_{Ge}}{P_{Go}}\right)} \tag{28}$$

where: V_h, t_G and M_{Th} are volume, thickness, and mass of blowing tanks; σ_{hw} and ρ_{hw} are the mechanical properties.

According to blow subsystem sub-algorithm, numbers of blow tanks are selected considering configuration and layout. The final mass of feeding system is calculated by the following equation:

$$m_{hellium} = m_h + m_{line} + m_{he} + m_{valve} + m_{heater} \tag{29}$$

Tanks Analysis

The shape of the tank is a function of weight, leakage rate, tank volume, and locating restrictions. Spherical tanks have the best empty weight-to-loaded weight ratio (Hutchinson and Olds 2004). Tank design sub-algorithm is shown in Fig. 10. Other elements such as control valves are selected according to input pressure and flow rate.

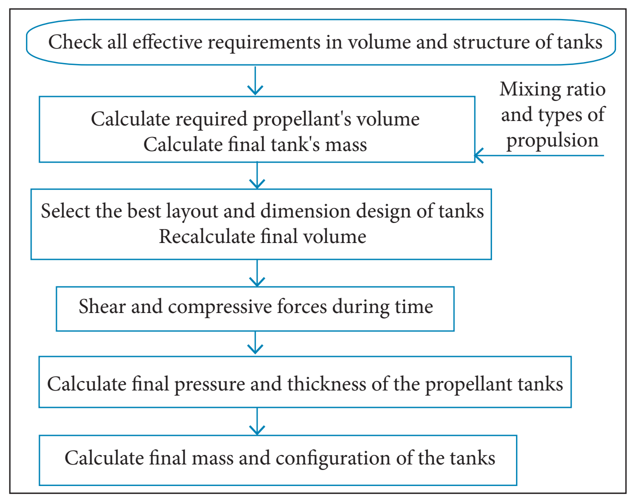


Figure 10. Tank design sub-algorithm.

Structure Analysis

In structural analysis, providing stability and structural strength to deal with all external pressures is the main goal. In order to determine the mass of the structure, at first, the loads on each section of the structure shall be determined via different stages of preparation to the end of the flight. Loads on each section mean axial force, shear force and bending torque which is applied under external loads during structure mission. Critical load for each section is occurred based on existing experiences in any of the selected above stages. Loading sub-algorithm and the thickness of the body structure is shown in Fig. 11.

After calculating longitudinal and lateral load flow, equivalent stresses are obtained and exerted to the structure which determines the thickness of the body (Ardema *et al.* 1996; Crawford and Burns 1963).

$$N_x = N_{xbend} + N_{xaxial} + N_{xullage} + N_{xhead} \quad (30)$$

$$\sigma_n = \frac{1}{\pi D t}\left(N + \frac{4M}{D}\right) \quad (31)$$

$$N_{equ}(x) = N(x) + \frac{4M(x)}{D(x)} \quad (32)$$

$$t = \frac{N_{equ}}{\pi D \sigma_n} \quad (33)$$

where: N is the axial force; M is the bending moment.

According to the critical situation exerted on the upper stage structure, lateral stiffeners with low number could be used to strengthen body structure. Design algorithm for strengthening stiffeners is shown in Fig. 12.

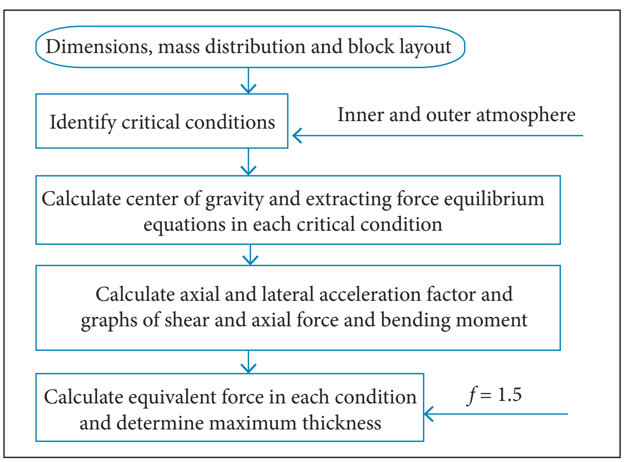


Figure 11. Body structure sub-algorithm.

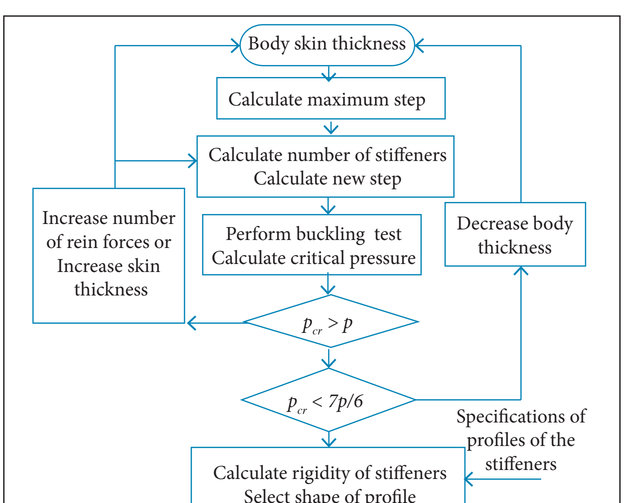


Figure 12. Stiffener structure sub-algorithm.

Stiffener rigidity and then the shape of the stiffener profile can be selected through the following equation:

$$J = \frac{(N+1)^{4/3} - 1}{(N+1)} \frac{t^3}{12(1-\mu^2)} \quad (34)$$

where: J is the rigidity stiffener.

Dimension Design

Dimension design is achieved with 2 different assumptions which, in one, upper stage diameter was determined and, in the other, it was calculated in the output. The calculation of the length and diameter can be achieved through the following equation:

$$L_t = L_f + L_o + L_{of} + L_e + L_s \quad (35)$$

$$D_t = D_p + 2(t_s + t_t + t_r) \quad (36)$$

RESULTS

SAMPLE SOLUTION

A sample solution for upper stage design is presented in this section to generally introduce the design method:

1. Mission definition
   - Payload mass is 1.5 tones with parking orbit which is 200 KMs and destination orbit is 36000 KMs with inclination of 45 degrees change.
   - Other design inputs are based on requirements and limitations.
2. Results of supplying design initial inputs are shown in Table 5.

Table 5. Initial input.

Input variable	Values
β	0.162
T (kn)	48.36
μ_p	0.75
M_p (ton)	10.23
M_F (ton)	3.47
M_0 (ton)	13.70

3. Trajectory design:
   - Hohmann's transfer (3 times Re burn).
   - Determining transfer orbit (according to energy limitation and optimized mode).

4. Determining material and construction inputs of the subsystem:
   - Used mechanical properties, environmental conditions, used fluid properties.
   - Determining fuel and oxidizer (hydrogen/oxygen).
5. Initial feasibility based on ability to transport about 13 tones to the parking orbit through stages of launcher rocket.
6. System design:
   - Determining configuration parameters according to Table 4.
   - Selecting initial mass factor.
7. Propulsion design parameters (Table 6).

Table 6. Propulsion engine parameters.

Parameter name	Input variable	Values
Thrust factor	C_f	1.68
Nozzle expansion ratio	ε	37
Specific impulse	Isp	326
Burning time	t_b (s)	680
Thrust	T (kn)	60
Propulsion flow rate	$\dot{m}$(kg/s)	20.52
Motor mass	m_m (m)	497
Motor diameter	d_m (m)	0.852
Motor length	L_m (m)	2.04
Nozzle length	L_n (m)	1.14
Nozzle outlet diameter	d_e (m)	0.78

8. Feeding system design (Table 7):

Table 7. Feeding system parameters.

Parameter name	Input variable	Values
Blowing system mass	m_h (kg)	66
Cases blowing mass	M_{Th} (kg)	34.5
Helium mass	m_{he} (kg)	20.6
Thickness of helium tanks	t_h (mm)	4.45
Radius of helium tanks	r_h (mm)	257
Number of helium tank	N_h	6

a. Selecting pressure-feed system.
b. Calculating pressure drop in fuel flow which is 3.2 bars and for oxidant is 4 bars.
c. Fuel tank pressure 11.2 bars and oxidant 12 bars.

9. Design of tanks: mass and volume of tanks (Table 8).

Table 8. Tank parameters.

Parameter name	Values
Oxidant volume	7.050 m^2
Oxidant mass	9.669 ton
Fuel volume	4.2827 m^2
Fuel mass	3.581 ton
Oxidant tank radius	1.1895 m
Fuel tank radius	1.1895 m
Oxidant tank pressure	12 bar
Fuel tank pressure	11.2 bar
Fuel tank thickness	0.72257 m
Oxidant tank thickness	3.801 mm
Fuel tank thickness	3.5485 mm

10. Body structure: determining body thickness and stiffener which requires determining critical modes properties of previous steps, fairing and types of stiffener.
11. Mass distribution:
    - Determining mass distribution (Table 9).
    - Upper stage primary masses.

Table 9. Other components mass distribution.

Parameter name	Values (kg)
Control block	245.88
Actuators	94.56
Cables	113.48
Body	189.1397
Telemetry	151
Stand	150
Disposal system	60
Engine maintenance	28.28
Control segments	18.91
Long tube	31
Flanges	28.28
Separation	75
Gyro planes and IMU	35

12. Upper stage mass and dimensions:
    - Upper stage dimensions (Table 10).
    - Upper stage primary masses (Table 11).

Table 10. Diameter and total length.

Parameter name	Values
Dimensional ratio	2.7812
Diameter (m)	2.4921
Total length (m)	6.9314

Table 11. Mass parameters.

Parameter name	Values
Fuel mass	13.249
Final mass	3.391
Structure mass	1.891
Total mass	16.64

DIFFERENT VARIANTS AND VALIDATION

Technology for constructing a launch vehicle in countries varies from each other, however, the statistical graphs indicate the approximate similarity of systematic parameters for upper-stage systems. For instance, Fig. 13 shows the relationship of payload mass and dry mass in the upper-stage system. The real samples are circular and the samples derived from this paper for UDMH/N204 fuel are square-shaped. The comparative curve of burn time due to thrust/weight is presented in Fig. 14. Convergence factors (α and β) are illustrated in Figs. 15 and 16.

The close similarity of these graphs is the main reason to validate the results derived from the study. At this point another validation was compared with Cent.D-5 upper-stage of Atlas V (401) which is presented in Table 12. Problem inputs: M_{pay} = 4.75 ton; D = 3.05m and I_{sp} = 4378 N*s/kg.

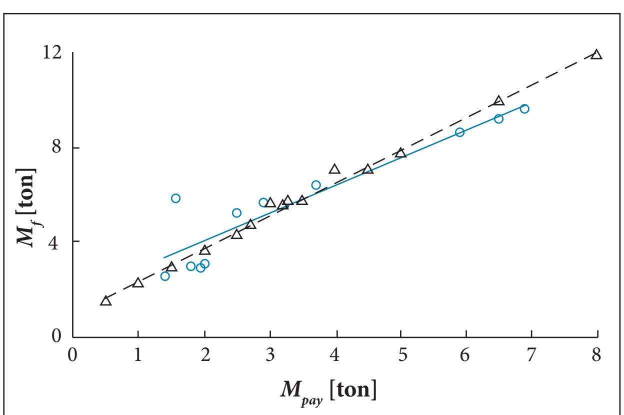


Figure 13. Final mass *versus* payload mass.

In Table 13, the errors in the statistical design methods and MSDO compared to the systematic data of Cent. D-5 upper stage system. As shown in Table 5, errors of fuel mass, dry mass and total mass in MSDO design are decreased by 9 to 16% compared with Cent. D-5. Furthermore, the accuracy of primary data derived from the statistical design is obvious in this table.

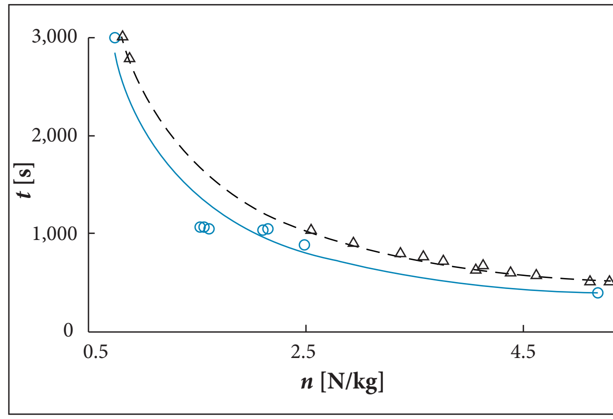


Figure 14. Comparative curve of burn time due to T/W.

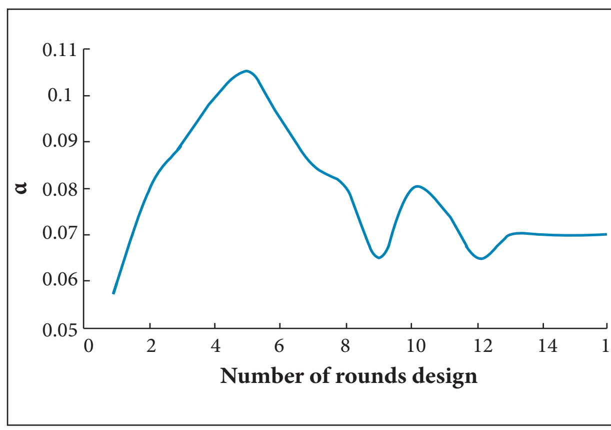


Figure 15. Convergence factor α.

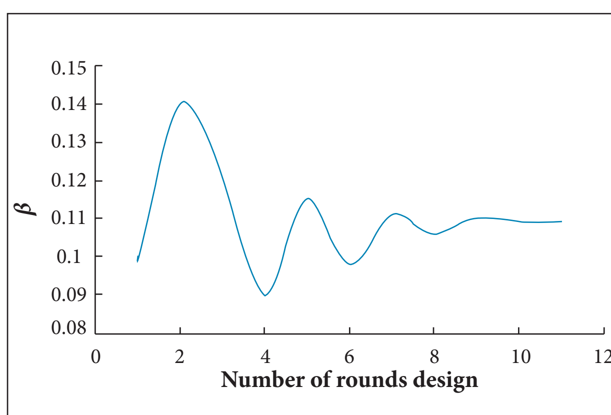


Figure 16. Convergence factor β.

Table 12. Comparation of validation to Cent D-5 upper-stage.

Output data	*Mp* (ton)	M_O (ton)	μ_f	μ_p	β	*n* (N/kg)
System parameters Cent. D-5	19.65	26.78	0.266	0.18	0.10	3.64
Statistical design	16.8	23.66	0.29	0.2	0.11	3.88
MSDO	18.93	26.01	0.27	0.182	0.109	3.61

Table 13. Methods errors compared to the systematic data of Cent. D-5 upper stage system

Design type method	Propellant weight [%]	Dry weight [%]	Lift off weight [%]
Statistical	11.6	11.3	14.5
MSDO	0.2	2.1	3.66
0.27	0.182	0.109	3.61

CONCLUSION

Design processes of all elements are done simultaneously in this paper. In each design loop, calculations become more accurate and results of each section, more suitable according to other systems. In this design method, a mistake can be found by system (because it affects all elements) and it would be easier to fix it. The main results are:

- Proportional with technology and ability for construction.
- Logical relation of all segments and subsystems.
- Developing a national design and method.
- Presenting a sample upper stage according to different inputs.
- Logical convergence of parameters to logical values.
- Ability of performing general feasibility.
- Extensibility to accurate initial design.
- Ability of networking the design under supervision of the system.
- Designing systematic sample of upper stage.
- General guideline based on implementation of detail design (limitations of methods and range of parameters).
- All the important design points are specified in order to divide the algorithm for performing different projects.

AUTHOR'S CONTRIBUTION

Zakeri M has provided the article's text; Nosratollahi M and Novinzade A carried out the guidance and control of the paper; the final set was conducted by Zakeri M and Nosratollahi M; the idea was developed by all authors.

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