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A Multi-Step Sequential System Optimization Design Method for Upper Stages

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ABSTRACT

This paper proposes a new approach in multi-step sequential system optimization (MSSO) to implement a conceptual design for satellite upper stage with a maneuver in the conditions close to reality. In this method of design, there are two main cycles; trajectory optimization cycle and optimal design cycle, each one is correlated to each other in another cycle called configuration. In the trajectory optimization, the optimization objective is to place the upper stage in the destination orbit, using the minimum amount of fuel consumption. In this cycle, a new approach has been introduced for a three-dimensional trajectory using two genetic algorithms inside each other. In addition, selecting the suitable engine is carried out in this cycle. Convergence of design and exclusion of design are carried out in the configuration cycle. Convergence and optimization of subsystems design are carried out in the optimal design cycle. The innovations of this paper are implementation of the design according to multi-step sequential system design in which optimization is performed step by step, and orbital optimization is introduced according to a new approach. Choosing a desirable criterion for optimization process and proper coefficient for convergence in design, are among considerable characteristics of this paper. Validation has been performed using one of the upper stages in the world.

Keywords: Design optimization; Upper stage rocket engines; Multidisciplinary design optimization; Trajectory optimization.

INTRODUCTION

The upper stages are responsible for transporting the satellites to the final orbit. Their performance trajectory is out of the atmosphere. Today, obtaining the design methods for upper stages, which have an optimization process and are less dependent on statistical studies, is an attractive subject. Accordingly, this study proceeds the researches performed based on designing the upper stage, aiming to optimize the transfer trajectory and applying its effect on the system design. Orbital transfer is an important issue, which has been investigated for years. It should be noted that most of the efforts were done for optimization and achieving the minimum amount of fuel consumption in space missions; because reducing 1 kg payload saves a lot of money. For this reason, investigating this subject is very important. It should be noted that impulsive maneuvers are not totally momentary and if there is not a proper understanding of the satellite position at the time of the impulse, a large amount of fuel will be spent for correction of the orbital position, causing less accuracy.

In this paper, it has been tried to design an optimized trajectory from the engine start moment to the moment of placement in the transfer orbit, using one of the optimization methods. So far, the optimized trajectory design has been implemented in different ways. The optimization is performed in a non-linear method by Vincent and Grantham (1997) and Brinda and Dasgupta

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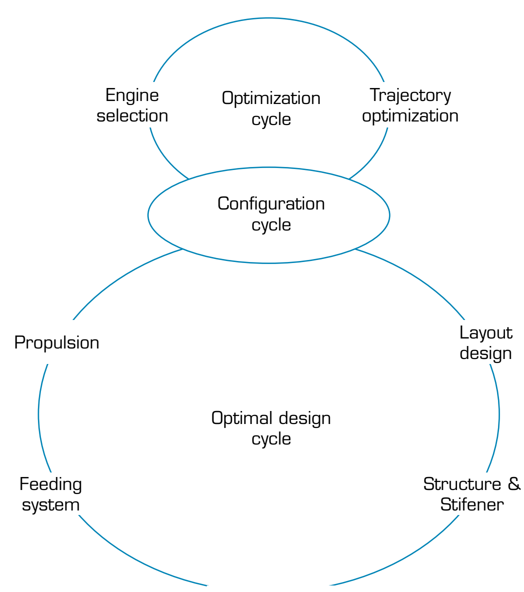
(2005). Control optimization is carried out by Lu *et al.* (2003) and Brinda and Punnoose (2007) and the trajectory design dynamic optimization has been carried out by Betts (1998), Betts and Huffman (1993) and Ross *et al.* (2003). In this paper, according to Maani *et al.* (2013; 2014), the orbital trajectory optimization has been carried out with three major differences; that are the solution method, reality of orbital mission and integration with the system design. In this paper, trajectory optimization has been performed inside an optimization cycle, which is a part of a comprehensive design approach called multi-step sequential system optimization design.

PROBLEM DESCRIPTION

The purpose of this paper is the implementation of the optimized design, including system design of all subsystems in order to pass the requirements for orbital mission in an optimal way. In many studies, the main objective of the optimization is reducing fuel consumption (Maani *et al.* 2013; 2014). Here a question arises: Is the optimization acceptable and implementable in practice?

If the mass of the fuel changes, the dry and total mass of the upper stage also changes. Changing the mass of the fuel, at first it changes the mass and volume of the fuel tanks and then it influences many relevant parameters. By changing the total mass, the trajectory optimization loop should be run again. The results in this paper could be considered valuable for considering the relationship and interaction between optimization cycle and design cycle. Innovations presented in this article include, first, presenting a new approach in design; the concepts of design and interactive optimization are applied in a strategic computing environment to design an upper stage for transferring the satellite from the parking orbit to the destination orbit in a new approach. Second, trajectory optimization; in the paper, the trajectory optimization is performed by two genetic algorithms for continued orbital maneuvers in a three-dimensional environment. Third, providing a system design in a collaborative design environment, which is implementable as primary design; collaborative design is one of the successful methods in system design in a software environment, where the intervention of the user during the run time is taken into account (Boy and Gruber 1990). The main characteristic in this method is the presence of the expert human in the design environment and in the integration of all subsystems. In this method, just like multi-disciplinary design optimization methods (MDO), no mathematical based design is presented. In MSSO design, reviewing and changing the design parameters according to constraints could be performed in every point of the algorithm. In this method, optimization is done step by step and in the development procedure of the design process. Fourth, the methodology of MSSO design process, which is based on Fig. 1, as shown below.



Source: Elaborated by the authors using data from Boy and Gruber (1990).

Figure 1. MSSO design process.

MSSO DESIGN PROCESS ALGORITHM

This algorithm consists of three basic cycles. Trajectory optimization cycle, configuration cycle, and optimal design cycle. Determining initial values and basic configuration of the upper stage is carried out using statistical Eq. 1–6. These equations are the result of data collection of upper stages from information websites (Brügge 2021; Mirshams *et al.* 2016; Wade 1996). The information required for determining initial configuration of design was obtained from Arianespace (2006) and Angelopoulos *et al.* (2014). The following process has been done to achieve these equations:

- Identify the parameters used in the design;
- Drawing different graphs to obtain the relationship between two parameters;
- Deriving equations from the behavior of graphs;
- Selecting equations with at least 95% accuracy.

$$M_F = 0.026M_{pay}^2 + 0.799M_{pay} + 2.546 \tag{1}$$

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

$$n = 2880\, t^{-0.94} \tag{3}$$

$$T = 3.668\, n^2 - 20.21\, n + 107.7 \tag{4}$$

$$L = 109.5\, e^{-2.33D} \tag{5}$$

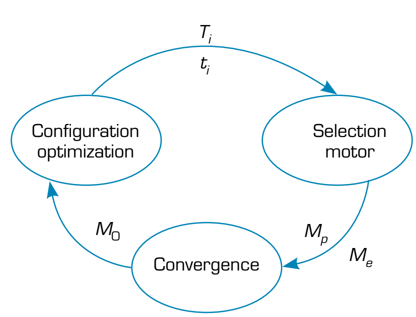
$$LD^2 = 0.323M_0^2 - 7.997M_0 + 116.6 \tag{6}$$

where M_0 is the total mass, M_f is the final mass, M_{pay} is the payload mass, T is the thrust, n is the thrust to weight ratio, L is the length, D is the diameter and LD^2 is volume.

After defining the mission and creating the primary configuration of the upper stage, the trajectory optimization cycle would be carried out. In this cycle, trajectory optimization and selection of a proper engine for the upper stage are defined. The characteristics of this cycle include the following:

- Choosing the engine by using the results of trajectory optimization and a list of available engines;
- The advantage of the engine selection and trajectory optimization being common in optimization cycle is minimizing the errors of ballistic parameters in the simulation with actual ballistic properties of the usable engines;
- Fuel mass and the selected engine are the most important outputs of this cycle.

System relations of trajectory optimization cycle is shown in Fig. 2. Mission and total mass are trajectory optimization cycle inputs. Fuel mass, thrust-time profile, and the characteristics of the selected engine are trajectory optimization cycle outputs.



Source: Elaborated by the authors using data from Pan *et al.* (2012).

Figure 2. Trajectory optimization cycle relations matrix.

By using spherical coordinates and the Eqs. 7–13, dynamical model of the upper stage movement (Pan *et al.* 2012) is:

$$\dot{r} = V\, sin\, \gamma \tag{7}$$

$$\dot{\theta} = \frac{V \cos\gamma\ \cos\omega}{r \cos\varphi} \tag{8}$$

$$\dot{\varphi} = \frac{V\ \cos\gamma \sin\omega}{r} \tag{9}$$

$$\dot{V} = \frac{T\ \cos\vartheta \sin\delta}{m} - \frac{\sin\gamma}{r^2} \tag{10}$$

$$\dot{\gamma} = \frac{T\ \cos\vartheta \sin\delta}{m\,V} + (\frac{V^2}{2} - \frac{1}{r^2})\frac{\cos\gamma}{r} \tag{11}$$

$$\dot{\omega} = \frac{T\ \sin\vartheta}{m\,V\ \cos\gamma} - \frac{V}{r}\cos\gamma \cos\omega \tan\varphi \tag{12}$$

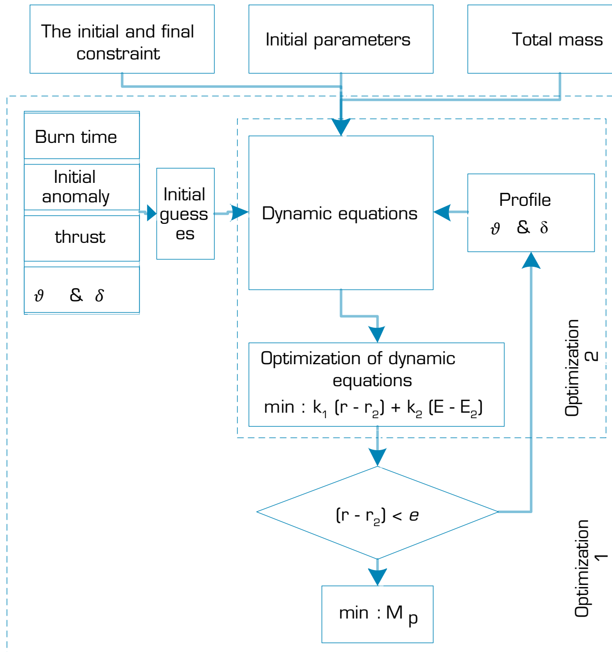
$$\dot{m} = -\frac{T}{c} \tag{13}$$

where r is distance between Earth center and upper stage center, and θ and φ are the longitude and latitude of the upper stage. V is the magnitude of the velocity vector, γ is the flight path angle, and ω is the velocity heading angle. ϑ and δ represent direction of thrust. ε is the out of plane thrust angle between thrust and the plane of position, and velocity vectors and δ is the in-plane thrust angle between velocity and the projection of thrust vectors in the vertical plane formed by position and velocity vectors of the upper stage. All lengths in equations are normalized by a reference length which is R_0 (radius of Earth), and all the times in equations are normalized by $\sqrt{}(R_0/g_0)$ where g_0 is gravitational acceleration. Therefore, velocity is normalized by the definitions above. M is normalized by initial mass and T is thrust to weight ratio and c is a constant which is $c=I_{sp}/\sqrt{}(R_0/g_0)$. (Pan *et al.* 2012). In a timed transfer, which is happening the system propulsion is turned on at a certain point of the first orbit, and is turned off after being placed in the second orbit. In this paper, orbital maneuver is considered in a real situation, meaning that both two orbits have no intersection and have rotation relative to each other at least in one direction. Therefore, the main objective is to solve orbital equations, flying from the first orbit, and placement in the second orbit with minimum errors. Among the answers, which have minimum errors, the transfer with the lowest fuel consumption would be the final choice of trajectory optimization cycle. In other words, the aim of optimizing the trajectory is to reach the second orbit with the least amount of fuel. That's why, in this method, two optimization genetic algorithms have been used together. The optimization objective is optimizing fuel consumption and this purpose requires the controlled work to be minimized (Eq. 14) (Kirk 2004; Motlagh *et al.* 2013):

$$J(U) = \int_{t_0}^{t_f} \frac{T(t)}{c}\, dt \tag{14}$$

Required orbital equations are obtained from Chobotov (1996) and Curtis (2005). Flight optimization algorithm is shown in Fig. 3. In this study, flight optimization algorithm includes using the genetic algorithm two times. The characteristics of the two methods are listed below:

Optimization 2 is performed inside optimization 1 and within each time step. The objective of optimization 2 is the minimum amount of $(r - r_2)$ at any moment and the objective of optimization 1 is the minimum amount of fuel consumption in transfer. After each use of optimization 2, the condition of the algorithm $(r - r_2) < \epsilon$ is checked. If the condition applies, the amount of fuel mass is calculated, and if the condition does not apply, the new range of ε and δ is obtained according to the Eq. 9 and 10 and optimization 2 is carried out again. The changes of δ and ε profile and the placement of upper stage in the second orbit according to Eqs. 15 and 16, is the result of intermittent performance of optimization 2.



Source: Elaborated by the authors.

Figure 3. Flight optimization algorithm for transmission.

$$\vartheta_i - k < \vartheta_{i+1} < \vartheta_i + k \tag{15}$$

$$\delta_i - k < \delta_{i+1} < \delta_i + k \tag{16}$$

The identification process of r and r_2 using the entries in Table 1 is explained as follows (Fig. 4.):

Table 1. Specifications of optimizations in Fig. 4.

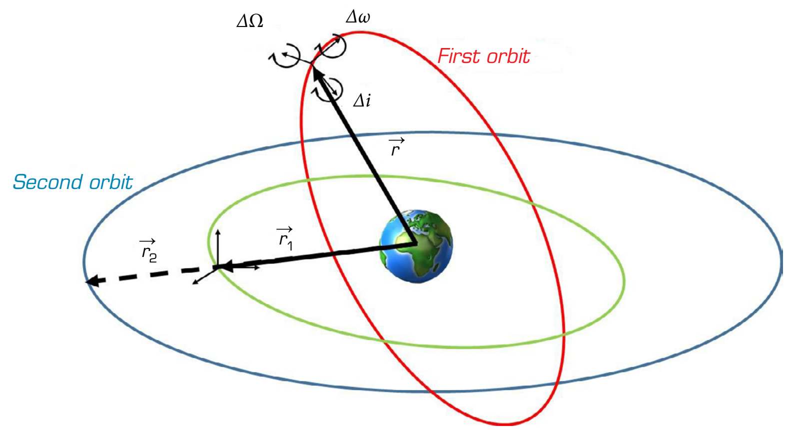
Subjected	Population number	Variable range	Variable name	Process
Minimum M_p	400	$t_{min} < t < t_{max}$	Burn time	Optimization 1
		$-30 < \theta < 0$	Initial anomaly	
		$T_{min} < T < T_{max}$	Thrust	
		$-15 < \vartheta < 15$	ε_0	
		$-15 < \delta < 15$	δ_0	
Minimum $(r - r_2)$	300	$\vartheta_{i-1} - 5 < \vartheta_i < \vartheta_{i+1} + 5$	ε	Optimization 2
		$\delta_{i-1} - 5 < \delta_i < \delta_{i+1} + 5$	δ	

Source: Elaborated by the authors.

- Calculating r, instantaneous position vector of upper stage (starting from parking orbit);
- Calculating the angle difference of two orbits in three directions (Δi, $\Delta\Omega$, $\Delta\omega$);
- Calculating the Anomaly difference between two orbits ($\Delta\theta$);
- r_1: Rotation of position r in three values of two orbits angle differences (Δi, $\Delta\Omega$, $\Delta\omega$);
- r_2: Identification of corresponding point of r_1 in target orbit with preserving the anomaly difference ($\Delta\theta$).

Algorithms shown in Fig. 4 should be performed for each orbital maneuver. The final fuel mass is the sum of fuel mass required for each orbital maneuver according to Eq. 17.

$$M_p = \sum_{i=1}^{n} M_{pi} \tag{17}$$



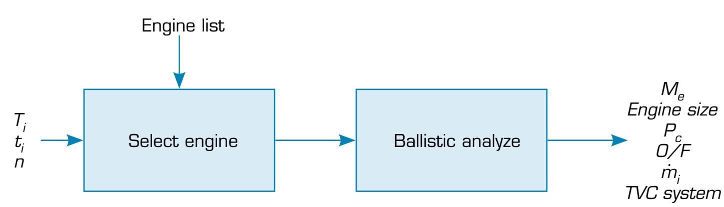
Source: Elaborated by the authors.

Figure 4. Spherical coordinates of upper stage trajectory.

The selection process and system performance analysis of the propulsion system in different space environments creates slightly different characteristics for a space propulsion (Kenny and Friedman 1965). The selection of the engine is completed according to the optimization cycle and other propulsion activities in the design cycle. Selecting a performance analysis of propulsion in this paper includes the following:

- Selection of the appropriate engine (among the available engines) using thrust-time profile obtained from the simulation;
- Receiving the nominal values of engine performance parameters such as thrust, pressure distribution of combustion chamber, fuel to oxidizer ratio;
- Calculation of engine ballistic performance during flight and combustion chamber pressure distribution;
- Calculation of fuel and oxidizer flow rate;
- Engine mass-dimension and performance specifications.

Propulsion system design equations are available in Sutton and Biblarz (2001), Darabi *et al.* (2015) and Humble *et al.* (1995). Figure 5 shows how to achieve propulsion engine specifications.



Source: Elaborated by the authors.

Figure 5. Selection and performance analysis of propulsion system. OF: oxide to fuel ratio; TVC: thrust vector control.

Configuration cycle

This cycle connects the two other cycles. In this cycle, the optimal fuel mass is received from the trajectory optimization cycle and the optimized dry mass is received from the optimization design cycle. Exit from the system design algorithm is carried out in this cycle, and the reciprocating behavior between the cycles will continue until the convergence of mass values in this cycle. Convergence of structure efficiency (β) (Motlagh and Novinzadeh 2012) is the exclusion criterion from the algorithm of Fig. 1 by using Eqs. 18 and 19.

$$\beta = \frac{M_F}{M_F+M_P} \tag{18}$$

$$\beta_n - \beta_{n-1} < \epsilon \tag{19}$$

Mass configuration is composed of three main mass parameters: fuel mass, subsystems mass and structure mass. Fuel mass has a direct relationship with the optimization cycle, the mass of subsystems and the mass structures has a direct relationship with the design cycle. Mass of the upper stage in an integrating process between the two cycles of trajectory optimization, and the optimal design is obtained from the Eq. 20 and the mass of subsystems is obtained from the total mass of subsystems according Eq. 21:

$$m_o^{(j+1)} = m_{pay} + m_{sub}^{(j)} + m_{str}^{(j)} + \sum_{i=1}^{n} \quad (m_{fuel_i}^{(j)}) \tag{20}$$

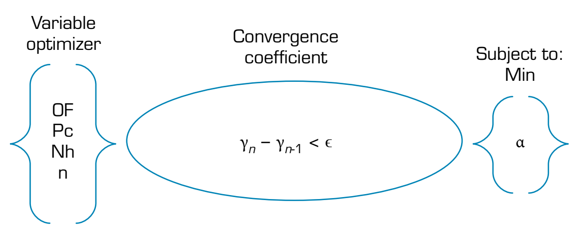
$$m_{sub}^{(j)} = \sum_{k=1}^{n} \quad (m_{sub_k}^{(j)}) \tag{21}$$

where i is the Number of the orbital maneuvers, j and k are the repeatable integers. The mass of each subsystem is converged in the design cycle process. Fuel mass, engine mass from the trajectory optimization cycle and mass of subsystems from the optimization design cycle, are convergence cycle inputs. Fuel mass, Thrust- time profile, and Specifications of the selected engine are Convergence cycle outputs.

Optimal design cycle

Upper stage subsystem design is performed in this cycle. This cycle has the following system specifications:

- Design cycle as shown in Fig. 1 includes six parts;
- All subsystems of upper stage are designed in this cycle;
- The output of this cycle is upper stage subsystem specifications;
- The process of design optimization and convergence in this cycle is shown in Fig. 6.



Source: Elaborated by the authors.

Figure 6. Optimization and convergence in optimization design cycle. OF: oxide to fuel ratio.

Figure 6 shows the convergence process in the optimal design cycle. In this cycle, the convergence factor is γ and the optimization criterion is minimum α, which is shown in Eq. 22–26.

$$\mu_f = \frac{M_F}{M_0} \tag{22}$$

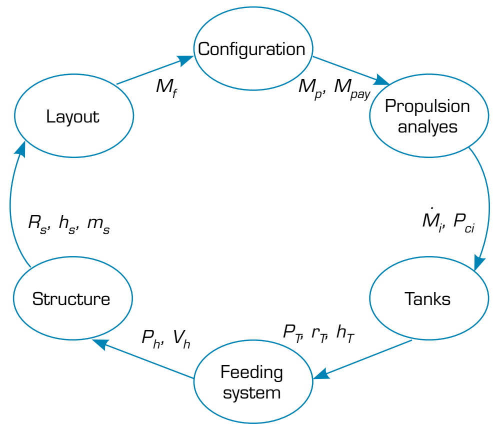
$$\mu_p = \frac{M_{pay}}{M_0} \tag{23}$$

$$\alpha = \frac{1-\mu_f}{\mu_p} \tag{24}$$

$$\gamma = \frac{M_S}{M_0} \tag{25}$$

$$\gamma_n - \gamma_{n-1} < \epsilon \tag{26}$$

Where μ_p is the fuel mass ratio and μ_F is the dry mass ratio. The reason for choosing α is to create optimized mass distribution for the upper stage. System relations of optimal design cycle is shown in Fig. 7:



Source: Elaborated by the authors.

Figure 7. System relations of optimal design cycle.

Optimal design cycle uses the trajectory optimization cycle data to determine the optimization parameters. This cycle automatically acts according to the trajectory optimization cycle as long as configuration parameters do not change. If the mass configuration in the configuration cycle is not acceptable, the trajectory optimization cycle will operate according to new information based on the updated configuration.

SUBSYSTEMS DESIGN

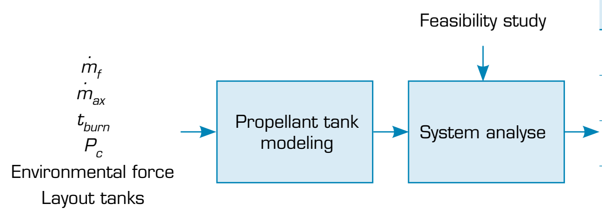
Optimal design cycle includes the design of all subsystems that affect the final configuration. Proper design is a collection of balanced characteristics of subsystems (ILS 2004). After propulsion analysis, which was described above in this section, the design of other subsystems will be reviewed.

The design process of tanks

The shape of a tank is a function of weight, rate of tank outflow, the tank volume and its layout limitations. Tanks with more sphericity have the best empty weight ratio to loaded weight (Hutchinson Junior and Olds 2004). Design process of tanks can be summarized as follows:

- Selection of the geometry and arrangement of tanks (layout limitations);
- Calculation of the required volume of tanks;
- Dimensional design of tanks (focus on sphericity with movement limitations);
- Calculation of pressure distribution of ullage during a flight (control volume between the tanks and the combustion chamber);
- Designing the tanks structure and layout of other components;
- Tanks mass-dimensional distribution.

Figure 8 shows the process of achieving the tanks design model. The calculation design process in propellant tank modeling and conceptual estimation process of other components are carried out using feasibility studies in tank system analysis.



Subdivision	Design	Estimation
Tank dimension	•	
Tank structure	•	
Insulation and cooling		•
Measuring equipment and discharge system		•
Other components		•

Source: Elaborated by the authors.

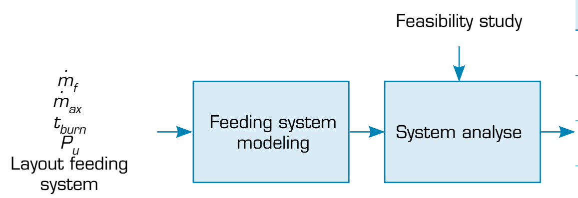
Figure 8. Tank design matrix.

Feeding system design process

Simplicity of pressure regulation in the feeding system determines its high reliability (Huzel *et al.* 1992), thus on and off switching and flow control is easily possible. Helium as a gas or liquid goes under pressure in helium tanks and with the use of heater and pressure control valves, the expected flow rate output is obtained. Feeding system design process is as follows:

- Calculation of the required volume for helium tanks;
- Choosing the number of tanks (layout limitation);
- Arrangement and layout of the tanks (Tanks are spherical);
- Calculation of required pressure during the flight (control volume between the fuel and oxidant tanks with the helium tanks);
- Tanks mass-dimensional distribution.

In Fig. 9, the method to achieve the feeding system design structure is presented.



Subdivision	Design	Estimated
Helium tanks structure	•	
Helium mass	•	
Helium tank dimension	•	
Valves and heaters		•
Other components		•

Source: Elaborated by the authors.

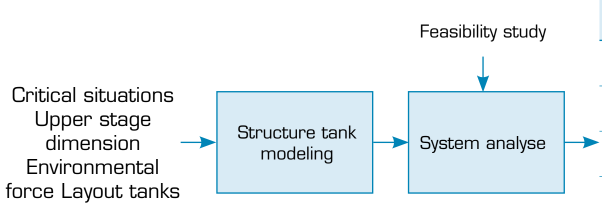
Figure 9. Feeding system design matrix.

The design process of structure

To determine the mass of the structure, the loads applied on each section of the structure should be determined during the different stages, from preparation to the end of the flight. After identifying critical situations, loading in these situations is used in structural analysis. According to the above, the structural process and stiffener design can be summarized as follows:

- Identifying the critical situations;
- Calculation of transverse and longitudinal loads in critical situations;
- Calculation of equivalent stresses on the structures;
- Calculation of body structures thickness;
- Calculating the number, distance and rigidity of stiffener;
- Selecting the stiffener profile after buckling test;
- Structure mass-dimensional distribution.

In Fig. 10, the method to achieve the structure design model is presented.



Subdivision	Design	Estimated
Body structure thickness	•	
Stiffener rigidity	•	
Stiffeners number	•	
Separation equipment		•
Other components		•

Source: Elaborated by the authors.

Figure 10. Structure and stiffener design matrix.

Other divisions

A good mass-dimensional distribution estimation of other components of upper stages can be achieved using statistical and feasibility studies. Table 2 shows how to access other components such as actuators and case guides.

Table 2. Mass-dimension classification of other upper stage divisions.

			estimated	calculated	selected
Other components of upper stage	Guidance and control hardware	Flight computer			•
		Guidance and control block			•
		INS (Inertial Navigation System)			•
		Valves	•		
		Other components	•		
		Telemetry system	•		
	actuators	Electromechanical actuators			•
		Cabling	•		
		thrusters			•
		Braking engines			•
		Accelerating engines			•

Source: Elaborated by the authors.

Equations 27 and 28 show how to calculate the total length and diameter of the upper stage.

$$L_t = L_f + L_o + L_{of} + L_e + L_s \tag{27}$$

$$D_t = D_p + 2 \times (t_s + t_t + t_r) \tag{28}$$

MSSO PERFORMANCE ANALYSIS

In this paper, a mission similar to Cent. D-5 SEC upper stage has been selected. Table 3 shows the input parameters of this mission.

Table 3. Upper stage design inputs.

Parameter name	Initial height	Initial velocity	Target orbit	Payload mass	Inclination change
value	134	5700	GTO	4750	27°

Source: Adapted from Martin (1999).

Initial design configuration is shown in Tab. 4 using statistical equations.

Table 4. Statistical design outputs.

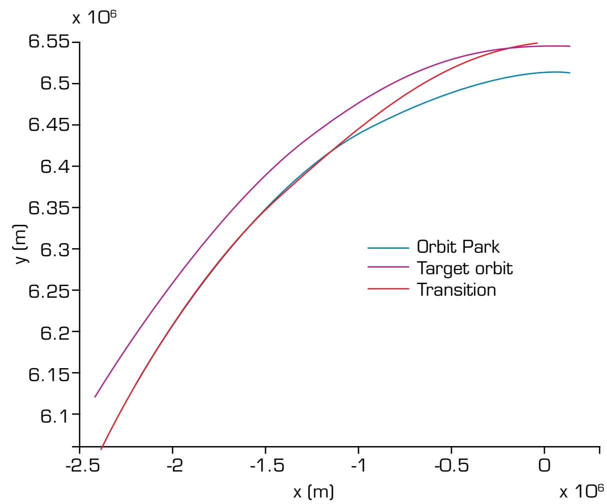
Fuel mass (ton)	Total mass (ton)	Dry mass ratio	Payload ratio	S tructure ratio	Thrust to weight ratio	Thrust (kN)
19.99	28	0.266	0.176	0.109	1.61	97.9

Source: Adapted from Martin (1999).

In this case, the code obtained from the algorithm of Fig. 1 was converged for 11 cycle consecutive times. In the following, the results of the last design loop are described for all three design cycles.

Trajectory optimization cycle

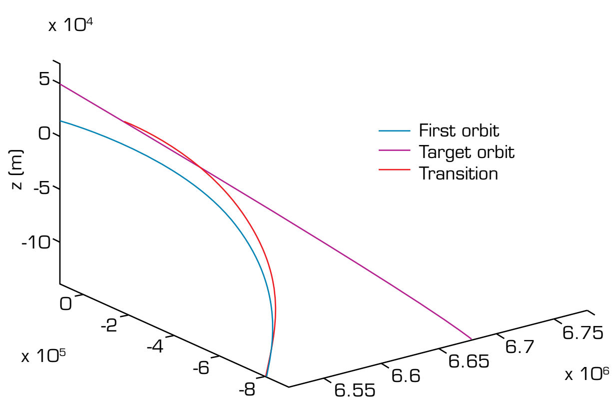
For this mission, according to the mission of Cent. D-5 SEC upper stage (Martin 1999), two orbital maneuvers (two phases) are considered. In both orbital maneuvers, orbits do not intersect with each other, and a change in the inclination is considered for the second maneuver. In the first phase, transfer from an altitude of 134 km with a specific speed is performed to 167 * 1418 km orbit. The optimal choice among different sets of solutions is achieved by the best placement of the upper stage in second orbit. The optimal trajectory of the first phase maneuver is shown in Fig. 11.



Source: Elaborated by the authors.

Figure 11. Flight path in the first phase of transition.

In the first orbital maneuver, the engine is turned on in true anomaly of -22.97° in the parking orbit direction and turned off in the true anomaly of -8.22° of second orbit. The average value of the thrust force in this transfer is 92 kN, thrust vector angle changes is between 0.41° to 2.5° and upper stage final mass is 7004 kg. In the second phase , the transfer is carried out from 167 * 1418 km orbit to GTO orbit with an inclination change of 27°. The three-dimensional indication of transfer trajectory is shown in Fig. 12.



Source: Elaborated by the authors.

Figure 12. Trajectory in the second phase of transition.

In the second phase of the transfer maneuver, the engine is turned on in the true anomaly of 172.8° of first orbit and turned off in true anomaly of 188.31° of the destination orbit. The average value of the thrust force in this transfer is 87.7 kN, ε is in the range of -45° to -12° and δ is between -4° and 2°. The final mass is 3000 kg. Table 5 shows the results of comparison between the second phase of Atlas V (401) upper stage with the results of simulation presented in trajectory optimization cycle without using multi-step system design (OMO) and Hohmann transfer.

Table 5. Comparison between three orbital maneuvers.

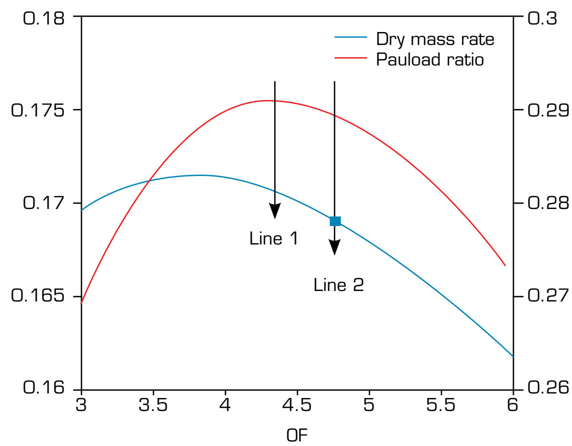
	Thrust (kN)	Burn time (second)	Fuel mass (ton)	Final mass (ton)	ΔV (km/second)
Atlas V (401)	2142.15	97.9	216	4.83	7.65
OMO	2135.9				
Hohmann	4476.4				

Source: Elaborated by the authors.

Table 5 shows orbital maneuver optimization with a new and simple method. Besides, this results proves that in such special launches, Hohmann transfer is not a good estimation for the design process.

Optimal design cycle

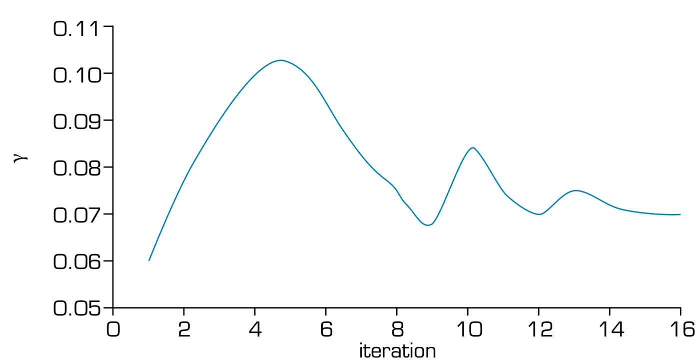
According to Fig. 13, the minimum value of α is equal to line 1, but in the design process, line 2 is calculated for α, which is a fuel to oxidizer ratio of 4.8. The calculated value of α represents applying constraints and limitations in MSSO design.



Source: Elaborated by the authors.

Figure 13. Changes of μ_f and μ_p due to OF. OF: oxide to fuel ratio.

Convergence process for γ in the optimal design cycle, for optimal value of α and in the last loop of the MSSO design is shown in Fig. 14:

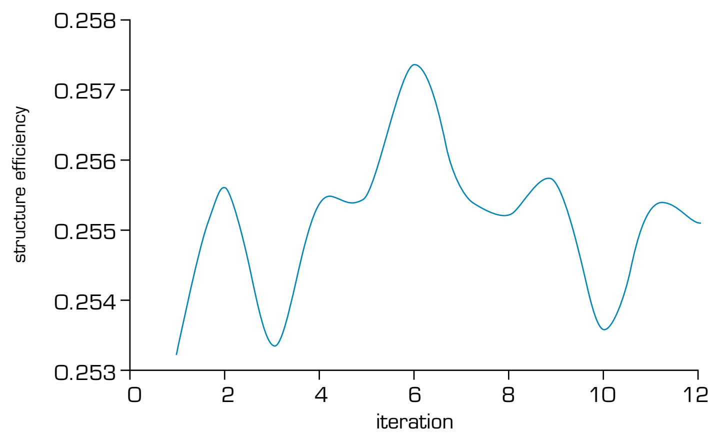


Source: Elaborated by the authors.

Figure 14. Convergence of γ in optimal design cycle of MSSO design.

Configuration cycle

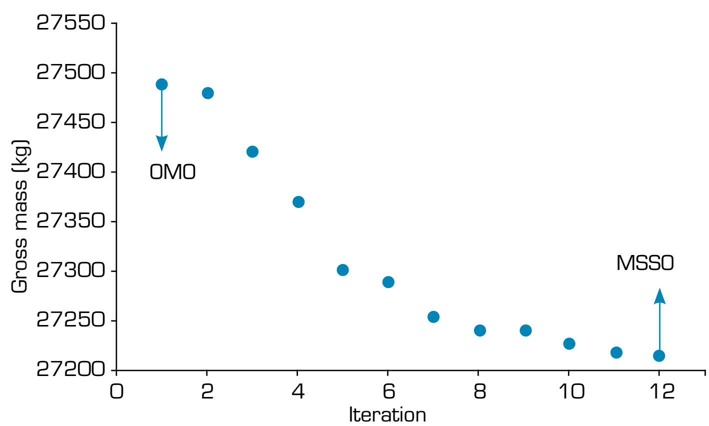
Convergence and exit from the design process of algorithm in Fig. 1 are performed in this cycle. The convergence criterion in this cycle is β. The changes of β is shown in Fig. 15.



Source: Elaborated by the authors.

Figure 15. Convergence of β in configuration cycle of MSSO design. MSSO.

The gradual process of upper stage total mass optimization in MSSO design is shown in Fig. 16.



Source: Elaborated by the authors.

Figure 16. Gross mass optimization.

Although mass coefficient (β) varies in the process of configuration cycle convergence, the gradual process of upper stage mass optimization is observable in Fig. 15. According to Figs. 15 and 16, it can be seen that if just orbital maneuver optimization (OMO) is used, the optimized mass will be obtained in point 1but the MSSO design will generate the optimum value of mass in point 12.

In this section, a comparison is provided between the obtained results of MSSO design, and the fuel usage consumption optimization process by Cent. D-5 SEC upper stage is shown in Table 6.

Table 6. Final results.

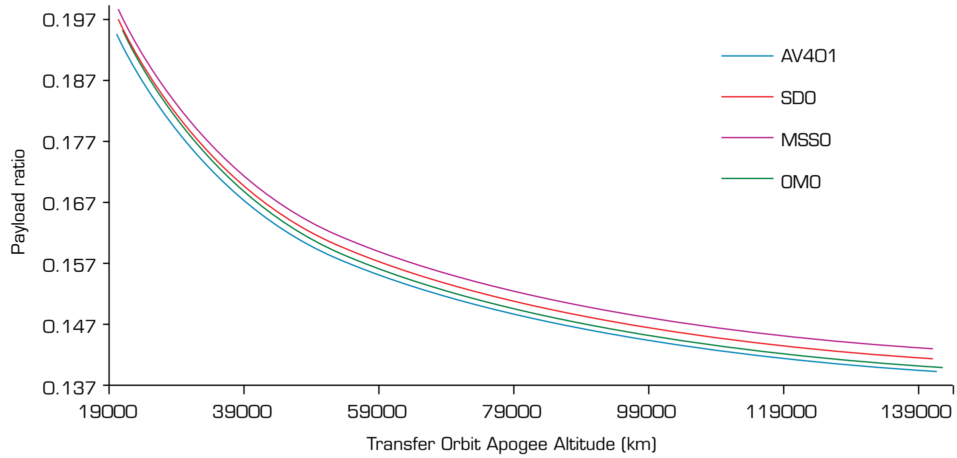
Approach name	Fuel mass	Total mass	Dry mass ratio	Payload mass ratio	Structure expansion ratio	Thrust to weight ratio	Thrust
Cent. D-5 SEC	19.65	26.78	0.266	0.18	0.10	1.68	97.9
OMO	19.45	26.58	0.266	0.163	0.123	1.49	95.9
MSSO	18.85	25	0.266	0.163	0.123	1.49	94.7

Source: Elaborated by the authors.

Figure 17 shows four situations of upper stages launching to transfer orbit, including:

- Actual flight of Atlas V (401) upper stage;
- Orbital maneuver optimization presented in this paper (OMO);

- Design method presented in paper SDO (system design optimization) without convergence in configuration loop;
- Multi-step collaborative system optimization presented in this paper (MSSO).



Source: Elaborated by the authors.

Figure 17. Comparison performance to elliptical transfer orbit.

According to the results obtained from the graph in Fig. 17, the amount of optimization in reducing the mass of the upper stage and the amount of optimization in the ability to carry the mass of the cargo is between 2 and 5 percent.

CONCLUSION

In this paper, a multi-stage optimal sequential design model is introduced based on the development of the MDO method for an upper stage. In this research, with a systemic approach, the relationships of the design parameters of an upper stage were extracted and considered. The model of this developed method includes inputs, optimization cycle, configuration cycle, optimal design cycle and criteria creation for design convergence and exit from the design process. The design is performed for an upper stage with basic system parameters according to the results of statistical design and for the assumed conditions of orbital transfer to the GEO orbit. The most important achievement of this study is to improve the MDO method and create an optimal design model for the upper stage in which all activities and parameters involved in the design are placed next to each other with the optimization step by step in convergence of all effective parameters in processes. Collective convergence of parameters involved in the design and optimization process, as opposed to point convergence (resulting from trajectory optimization, etc.) and local (subsystems separately) has been one of the most important results of this study. In this paper, specifically in the Cent D-5 SEC upper stage example, the design parameters are simultaneously converged and integrated.

CONFLICT OF INTEREST

Nothing to declare.

AUTHOR CONTRIBUTIONS

Conceptualization: Mehran N and Mostafa Z; **Data curation:** Mostafa Z and Vahid B; **Formal analysis:** Mehran N and Mostafa Z; **Acquisition of funding:** Mehran N and Mostafa Z; **Research:** Mostafa Z; **Methodology:** Mehran N and Mostafa Z; **Project administration:** Mostafa Z and Vahid B; **Resources:** Mehran N; **Software:** Mostafa Z and Vahid B; **Supervision:** Mehran N; **Validation:** Mostafa Z and Vahid B; **Visualization:** Mostafa Z and Vahid B; **Writing - Preparation of original draft:** Mostafa Z and Vahid B; **Writing - Proofreading and editing:** Mostafa Z and Vahid B;

DATA AVAILABILITY STATEMENT

All data sets were generated or analyzed in the current study.

FUNDING

Not applicable.

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REFERENCES

[ILS] International Launch Service (2004) Proton launch system mission planer's guide. Reston: ILS. [accessed 2014 July 20]. https://www.ilslaunch.com/wp-content/uploads/pdf/PMPG%20Section%200.pdf

Angelopoulos V, Ashour MA, Lyons LR (2014) Space Physics Center. IGPP. [accessed 2014 July 20]. http://www.igpp.ucla.edu/spc/

Arianespace (2006) Soyuz CSG User's Manual. Évry-Courcouronnes: Arianespace.

Betts JT (1998) Survey of Numerical Methods for Trajectory Optimization. J Guid Contr Dyn 21(2):193-207. https://doi.org/10.2514/2.4231

Betts JT, Huffman WP (1993) Path constrained trajectory optimization using sparse sequential quadratic programming. J Guid Contr Dyn 16(1):59-68. https://doi.org/10.2514/3.11428

Boy G, Gruber TR (1990) Intelligent Assistant Systems: Support for Integrated Human-Machine Systems. Paper presented at: Proceedings of 1990 AAAI Spring Symposium on Knowledge-Based Human-Computer Communication; Stanford, United States.

Brinda V, Dasgupta S (2005) Nonlinear tracking guidance algorithm for an air-breathing launch vehicle. Paper presented at: AIAA Guidance, Navigation, and Control Conference and Exhibit; San Francisco, United States. https://doi.org/10.2514/6.2005-6189

Brinda V, Punnoose LS (2007) On-line trajectory optimization of a typical air-breathing launch vehicle using energy state approximation approach. Paper presented at: Proceedings of The Automatic Control & Dynamic Optimization Society; 296-302.

Brügge N (2021) Space Launch Vehicles. [accessed 2021 July 20]. http://www.b14643.de/Spacerockets_1/index.htm

Chobotov VA (1996) Orbital mechanics. New York: AIAA Education Series.

Curtis HD (2005) Orbital mechanics for engineering students. Oxford: Elsevier Butterworth Heinemann.

Darabi H, Roshanian J, Zare H (2015) Design of liquid propellant engine using collaborative optimization and evolutionary algorithms. Proc Inst Mech Eng G J Aerosp Eng 229(2):232-242. https://doi.org/10.1177/0954410014529423

Humble RW, Henry GN, Larson WJ (1995) Space propulsion analysis and design. New York: McGraw-Hill.

Hutchinson Junior V, Olds J (2004) Estimation of launch vehicle propellant tank structural weight using simplified beam approximation. Paper presented at: 40th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit; Fort Lauderdale, United States. https://doi.org/10.2514/6.2004-3661

Huzel DK, Huang DH, Arbit H (1992) Modern Engineering for Design of Liquid Propellant Rocket Engine. New York: AIAA.

Kenny RJ, Friedman PA (1965) Chemical pressurization of hypergolic liquid propellants. J Spacecraft 2(5):746-753. https://doi.org/10.2514/3.28273

Kirk DE (2004) Optimal control theory an introduction. dover books on electrical engineering. Mineola: Dover publications.

Lu P, Sun H, Tsai B (2003) Closed-Loop Endoatmospheric Ascent Guidance. J Guid Contr Dyn 26(2):283-294. https://doi.org/10.2514/2.5045

Maani E, Kosari AR, Fakoor M (2013) Two-objective Optimization of GEO Communication Satellite Trajectory Considering Continuous Orbital Maneuver. Modares Mech Eng 13(7):142-152. [accessed 2021 July 20]. https://mme.modares.ac.ir/article-15-10557-en.html

Maani E, Kosari AR, Fakoor M (2014) Two-objective Optimization of GEO Communication Satellite Trajectory Considering Impulsive Orbital Maneuver. Modares Mech Eng 14(5):121-128. [accessed 2021 July 20]. http://mme.modares.ac.ir/article-15-8011-en.html

Martin L (1999) Atlas Launch System Mission Planner's Guide, Atlas V Addendum. San Diego: International Launch Services.

Mirshams M, Roshanian J, Dehkoedi SY (2016) Launch vehicle collaborative robust optimal design with multiobjective aspect and uncertainties. Modares Mech Eng 15(11):339-350. [accessed 2021 July 20]. https://mme.modares.ac.ir/article-15-3789-en.html

Motlagh JA, Novinzadeh AB (2012) Solid upper stage design process using finite burn maneuvers for low Earth orbit–geosynchronous Earth orbit transfer phase. Proc Inst Mech Eng G J Aerosp Eng 227(6):966-976. https://doi.org/10.1177/0954410012445953

Motlagh JA, Novinzadeh AB, Zakeri M (2013) New approach in designing solid upper stage for interplanetary missions using finite burn assumption. IEEE Aerosp Electron Syst Mag 28(10):36-43. https://doi.org/10.1109/MAES.2013.6642830

Pan B, Lu P, Chen Z (2012) Three-dimensional closed-form costate solutions in optimal coast. Acta Astronaut 77:156-166. https://doi.org/10.1016/j.actaastro.2012.04.009

Ross IM, Souza CD, Fahroo F, Ross J (2003) A Fast Approach to Multi-Stage Launch Vehicle Trajectory Optimization. Paper presented at: AIAA Guidance, Navigation, and Control Conference and Exhibit; Austin, United States. https://doi.org/10.2514/6.2003-5639

Sutton GP, Biblarz O (2001) Rocket propulsion elements. Lewiston: John Wiley & Sons.

Vincent TL, Grantham WJ (1997) Nonlinear and optimal control systems. New York: Wiley & Sons.

Wade M (1996) Encyclopedia astronautica. [accessed 2011 June 04]. http://www.astronautix.com