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Optimal Mission Design for Moon Exploration Expedition

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The main goal of this paper is to introduce the Moon exploration mission design based on existing technology.The Moon exploration mission design entailsoptimal maneuvering orbit, payload and launch vehicle design. Optimal maneuvering orbit is designed with respect to Circular Restricted Three Body Problem (CRTBP) to model the motion of a spacecraft in the Earth/Moon system. To this end, optimal maneuvering orbitadopted CRTBP as dynamical model and obtained three-dimensional Earth to Moon transfers with low cost. This method is more preferable and flexible than Hohmann transfer because of its lower cost and its access to various inclinations in departure and arrival.The optimal Launch Vehicle Conceptual Design (LVCD) algorithm is based on optimization of major design parameters. LVCD algorithm is coded in a software to let the design engineer explore the design space and to reduce the cost and time of the conceptual design phase that is developed by the authors.The optimization process is performed subject to the restrictions and the performance index is optimized in a mutual iteration mechanism. Consequently, the designed launch vehicle ability to satisfy the mission objectives and its requirements is evaluated.

Keywords: Mission design, Moon exploration, Circular restricted three body problem, Launch vehicle conceptual design

Nomenclature

x , y , z	position components in synodic frame
$\dot{x}$, $\dot{y}$, $\dot{z}$	velocity components in synodic frame
r_1	distance bewwen spacecraft andheavier primaries
r_2	distance bewwen spacecraft andlighter primaries
m_{Moon}	mass of the Moon
m_{Earth}	mass of the Earth
μ	mass ratio of the Moon and Earth
L_1	firstlagrangian point
Δv	velocity change

Introduction

The Moon is a readily accessible celestial object, and we can learn from its composition and interior structure much about planetary construction and how the Earth-Moon system is formed. The Moon is in some ways a Rosetta stone, providing a template for deciphering and understanding the history and evolutionary processes of the terrestrial planets.

The Moon may represent a potential resource for commercial exploitation. There have been many proposals to export lunar resources for their use on Earth, as well as proposals to use lunar-generated energy and to use the Moon for education, entertainment, or space tourism.

The Moon is a natural space station, providing a benign environment with one-sixth gravity for human utilization and exploration [1].

There are some countries that have jointed to the space science and technology community and want to expand its attendance in space effectively. For reaching to this goal and according to the long term plan for earth orbit and space exploration missions, one of the first steps will be robotic missions to lunar orbit for Moon exploration. Clearly the role of robotic missions as precursors to human exploration is well

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established. To access to this destination there will be a lot of important capabilities and requirements that must be meet. One of them is a launch vehicle that can carry out transportation phase to lunar orbit of such a mission. Today many developing countries have launch capability to LEO and can develop this potential up to systems with the capability of reaching further distances like Moon.

From 1990 up to now, a new era of lunar exploration missions has started that is different from such missions accomplished before 1976.

The first era of lunar exploration interest had a lot of political motives and began right after primary development in space technology. Another important difference between the new era and earlier missions is the number and variety of pioneering countries.Inthe first period,these developments werejust in the Soviet Union and the United States monopoly.

In the recent era,there are many scientific motives and goals in lunar exploration missions, especially when reaching the Moon is a stepping stone toward reaching Mars.

Some of these new scientific goals for lunar exploration are selection of safe landing sites, identification of lunar resources, study of how the lunar radiation environment will affect humans, upgrading and testing technological capabilities, obtaining data on elemental abundance, mineralogical composition, topography, geology, gravity, and the lunar and solar-terrestrial plasma environments, etc.

In this new era also, some newly developed countries in space science and technology have taken parts as active members in lunar exploration missions.

In the last two decades, almost every mission to the Moon has used methods and technics based on two body problem, such as flybyand Hohmann transfer. For example, KAGUYA (SELENE) mission was conducted to a lunar transfer trajectory from a 924 x 232,731 km earth orbit or Clementine mission had 2 flyby missions from LEO to lunar orbit. Asanother case, SMART 1 mission used geostationary transfer orbit, 742 x 36,016 km for traveling to the Moon. Also,Chandrayaan 1 started its travel to the Moon from 255 x 22860 km transfer orbit with an inclination of 17.9 degrees [1-4].

Here, at first, we present a review of the recent Moon exploration missions and then optimal maneuvering orbit is designed with respect to Circular Restricted Three Body Problem (CRTBP) to model the motion of a spacecraft in Earth/Moon system; and finally, launch vehicle conceptual design (LVCD) software used to design an optimal launch vehicle for Moon exploration mission sispresented.

Circular Restricted Three Body Problem (CRTBP)

The traditional approaches to transfer from the Earth to the Moon, such as Hohmann transfer, required muchbudgetandwere undesirable. Many efforts have been made to reduce the cost of a lunar transfer. For example, in 1968, Charles Conley [8] constructed a low-energy lunar transfer based on the three body dynamics. In 1990, the Japanese Hiten mission [9] used a low-energy lunar transfer based on the Weak Stability Boundary Theory. In 2000, Koon et al. [10] showed how invariant manifold can be utilizedto reproduce a Hiten like mission by using numerical computations. The most attractive case for a lunar transfer is thoroughly discussed in [8]. The basic idea, similar to the work of Koon et al. but in 3D, is to use two different three body systems: Sun/Earth and Earth/Moon. By combining and patching the manifold trajectories of these systems, the spacecraft would be ableto fly from the Earth to the Moon in a low-energy trajectory.

We used Circular Restricted Three Body Problem (CRTBP) to model the motion of a spacecraft in Earth/Moon system. In this model, two main masses (the Earth and the Moon in our case) generate a complex gravity field in which the spacecraft is moving without being influenced. This model became very popularin the recent years due to some special features specific tothis problem, such as Halo and Lissajous orbits, their stable and unstable manifolds, etc. Studies [11] have demonstrated that by using and combining these features, the Moon can be reached saving propellant mass with respect to the usual trajectories based on two body problem, i.e. Hohmann transfer.

The non-dimensional equations of motion for the spacecraft in the rotating frame are [8]:

$$
\begin{aligned}
\ddot{x} &= 2\dot{y} + x - \left(1-\mu\right)\frac{x+\mu}{r_1^3} - \mu\frac{x-1+\mu}{r_2^3} \\
\ddot{y} &= -2\dot{x} + y - \left(1-\mu\right)\frac{y}{r_1^3} - \mu\frac{y}{r_2^3} \\
\ddot{z} &= -\left(1-\mu\right)\frac{z}{r_1^3} - \mu\frac{z}{r_2^3}
\end{aligned} \tag{1}
$$

Where

$$
\begin{aligned}
r_1 &= \sqrt{\left(x+\mu\right)^2 + y^2 + z^2} \\
r_2 &= \sqrt{\left(x-1+\mu\right)^2 + y^2 + z^2} \\
\mu &= \frac{m_{Moon}}{m_{Moon} + m_{Earth}} = 0.0121506
\end{aligned}
$$

and x, y, z are position components, $\dot{x}$, $\dot{y}$, $\dot{z}$ are velocity components and $\ddot{x}$, $\ddot{y}$, $\ddot{z}$ are acceleration components of the spacecraft in the synodic coordinate system. For the above system of equations exist five equilibrium points referred to as the Lagrange points. In this study, we focus on L_1 that lies between the

Earth and the Moon. Because of the hyperbolic character, the dynamics close to L_1 arethat of an unstable equilibrium.

We use a Halo orbit around L_1 and its stable and unstable invariant manifold to find the transfer trajectory between the Earth and the Moon. We utilize the stable manifold for transferring from a 200 km LEO parking orbit to the halo orbit and the unstable manifold for transferring to a 100 km lunar polar orbit. Theses transfers require a smaller Δv than Hohmann transfer [11]. Note that very few of the manifolds pass close to the Earth and leave Earth orbit. That is the manifold must approach Earth at a distance equivalent to the altitude of a parking orbit (200 km in our case). Therefore,we should solve two points boundary value problems, one for transferring from the Earth to a specific Halo orbit and the other for transferring from Halo orbit to the Moon. Thus, we turn to the method of differential correction [12] and multiple shooting method in order to optimize the transfer trajectory. We use the stable or unstable manifold as an initial guess in a differential correction scheme.

For transferring from low Earth parking orbit to halo orbit, we use a stable manifold, as said before. The criterion for selecting a stable manifold is proximityto the Earth. In other words, selected stable manifold begins inhalo orbit and ends inthe nearest distance to the Earth. It is important to note that for generating the trajectory using stable manifold, the equations of motion are integrated backward in time and the end point of the trajectory should be on LEO parking orbit. Summation of differences between velocities at the beginning point of trajectory and a point of halo orbit, and also differences between velocities at the end point of trajectory and a point ofLEO is the required maneuver Δv for transferring from LEO to halo orbit, i.e. Δv_1 for insertioninto unstable manifold trajectory like from LEO parking orbit and $\Delta v_1'$ for insertioninto Halo orbit at the insertion point.

In some references, the concept requires only one satellite by exploiting the characteristics ofthree-dimensional halo orbits in the vicinity of the Earth-Moon L2 (EML2)liberationpoint [33]. Furthermore, this plan hasneverbeen implemented due to short time of the Apollo program. However, interest in the exploration of the far sideof the Moon hasrecently increased, particularly in the aftermath of the successful Artemis mission [34, 35].

The process for finding transfer trajectory to the Moon is the same, that is, we use an unstable manifold which passesthrough the target altitude, i.e. 100 km. Therefore, we need Δv_2 for perturbing spacecraft in order to be insertedintothe unstable manifold trajectory and Δv_3 in order for theinsertion of spacecraft to lunar polar orbit.

For implementing, we use the multiple shooting method with five patch points. The multiple shooting algorithm converges in ten iterates with 10e-9 desirable accuracy. We use the difference between weighted norm of i and $(i+1)^{th}$ for convergence condition. In Fig.1, you can see an artistic image of this transfer. The result of differential correction method is summarized in Table (1).

Table 1. Result of differential correction method

$\Delta v_1 + \Delta v_1'$	3.318
Δv_2	0.01
Δv_3	0.344
Δv_T (km/sec)	3.663

Table (2) shows a comparison of the Δv required for flying from the Earth to the Moon, i.e. from a 200 km LEO parking orbit to a 100 km lunar polar orbit.

Table 2. Δv required for this mission

Parameters	Hohmann	CRTBP	2D Shooting the Moon	3D Shooting the Moon
Δv (km/sec)	3.95	3.663	3.27	3.26
Transfer duration (days)	3.9	6	105	105

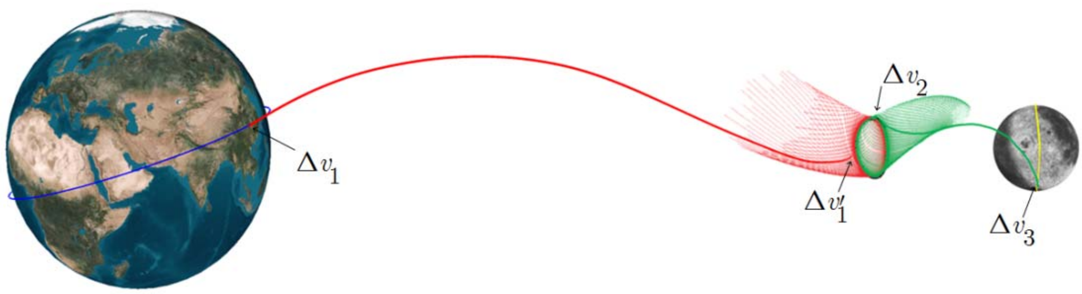

Fig. 1. Artistic image of the Earth to the Moon transfer

Optimal Launch Vehicle Conceptual Design (LVCD) Algorithm

Here, we present some launch vehicle conceptual design examples performed during the early design of launch vehicles. Several papers [13-16] have considered and evaluated the various algorithms of launch vehicle mass distribution in staging for reaching the minimum initial weight. In these references, thelevelof accuracy and rate of convergence in any algorithm was considered. Also, several papers [17-19] have considered and evaluated the various algorithms of launch vehicle pitch programs, and in [20-21], the optimum launch vehicle pitch program for reaching the maximal final velocity and minimal staging acceleration was suggested.

In [22], university of Maryland designed and builtalaunch vehicle which could deliver a 100 kg payload to 200 km orbit. In this reference, the principle of design wasstatistical modeling and this work used no optimization in design parameters.

In [23], the small launch vehicle for carrying satellite to LEO is developed by the ScorpiusCompany. The main goal of this projectis to achieve the minimum cost forlaunch. In the reference, the design is performed based on statistical methodology.Also, conceptual design of a small satellite launch vehicle was to place 10 lb university or research payloads in low Earth orbit in order to make use of the already existing rocket test facilities at Purdue and to keep test costs low. Also, in this reference no optimization was used in design parameters.

In [24], the small launch vehicle technologies demonstrationsbyusing sounding rockets are described.Aroundthe world, the sounding rockets areused to this end.

In [25], the launch vehicle conceptual design methodology has been developed. In the paper, disciplines (mass distribution and pitch program) are optimized to achieve the maximum final velocity and minimum lift off weight.

In [26], thealgorithm obtained in the previous reference is developed as software and madeuser-friendly to apply. Also, in [27], the effect of vortex breakdown in the tank beneath the ballistic parameters is considered.

In the present study,the results of the above references have been employed and developed in LVCD based on combinational optimization of major design parameters. Future research of this paper will concentrate on mission reliability analysis and risk management [28].

Generally, there are three techniques for conceptual design in the related literature. In the first technique, the design process is accomplished based on statistical data. The foregoing data including major design parameters and mass/energy coefficients are utilized to generate a statistical model [36]. These coefficients are representative ofthe technology level and energetic capabilities of designed launch vehicles. Although this technique has a low computational cost, the accuracy is not significant atthe conceptual design level. The stated method is dependent onthe statistical database. Based on the precision of the collected data, the method may have up to 20% error.

Another method which is conducted on the basis of mathematical models is applied for designing the launch vehicle parameters. This class of techniques is based on the classical optimization methods in calculus.

The mathematical models in this method are solely exploited for launch vehicleparameters determination such as optimum mass distribution, optimized pitchprogram, initial thrust to weight ratio and final pitch angle for various stages.

This method enjoysahigheraccuracythanthe former one due to the mathematical principlesemployed. Yet, the obtained results are not typically applicable due to technological constraints and existing facilities. Moreover, these methods have not been adequately developed for all design parameters.

Finally, the last method is the design of launch vehicle founded on the existing and pre-produced subsystems.

In this paper, the design is based on the foundation of the existing propulsion, fuel and oxidizer type, guidance and control subsystems. Since this method is subsystem-based, the use of a specific subsystem affects the optimization of other parameters. Hence, the local optimality of the whole launch vehicle is found.

In this research, a novel approach is presented to provide a method enjoyingthe advantages of the aforementioned triad approaches and being devoid oftheir drawbacks. To this end, a combination of statistical and mathematical models along with theexisting subsystems are proposed. The essence of combination process is rational and here is called LVCD Algorithm, Fig. 2.

This LVCD algorithm, provides a method containing the advantages utilizing ten sub-algorithms and removing the drawbacks of the afore-mentioned methods. With sub-algorithm 1, design begins based on a statistical model (mass/energy coefficients are selected), and optimization design parameters are performed in third, fourth and seventh sub-algorithms, thenalsothe existing technological capabilities and constraints are evaluated and considered in fifth sub-algorithm.

Therefore,LV with optimized major design parameters would be the outcome of this developed algorithm.

Launch Vehicle Conceptual Design (LVCD) code was employed to illustrate the capability of existingtechnologyfor lunar exploration.

LVCD Algorithm Results and Evaluation

The LVCD algorithm is applied to lunar launch vehicle conceptual design. The lunar launch vehicle is named SADRA-1 (in this paper) that is under developing. It has almost 125 ton and 175 ton force lift off mass and thrust, respectively. SADRA-1 is based onexisting technology [29-32].

The specifications of lunar exploration (the payload mass and park orbit) were obtained from the previous two sections (orbit and payload design). For evaluation of LVCD software, the above specificationsare used as seen in Table (3).

The LVCD software output is gave the ninety parameters the launch vehicle, but herein just a brief review of the most fundamental parameters are presented in table (4). This table shows the design major parameters regarding to mass, energetic and ballistic specifications of the designed launch vehicle of lunar exploration mission.

Conclusion

In this paper, we have presented moon exploration mission design based on existing technology.This paper is divided to three sections whicharethestatistical study of recent mission and primary estimation, Circular Restricted Three Body Problem (CRTBP),andoptimal Launch Vehicle Conceptual Design (LVCD) Algorithm.Thus, in the first section, we calculated the required payload mass based on the statistical data. Payload mass wasestimated approximately 2.0 tons according to the statistical study and Δv requirements computed in mission orbit design section thatrelated to computethe propellant mass. Then, in the second section, weadopted the CRTBP as the dynamical model and obtained three-dimensional Earth to Moon transfers with low cost. This method is more preferable and flexible than Hohmann transfer because of its lower cost and its access to various inclinations in departure and arrival. Although 3D shooting the Moon method provides smaller Δv , transfer duration is very long. The present case is the result of a trade-off between time and energy. Neither Δv is as big as Hohmann transfer nor time isas long as 3D shooting the Moon.It is important to mention thatour orbital mission is different from other similar missions in parking orbit specifications.In all of the recent missions, the parking orbitshave used GEO Transfer Orbit (GTO), but inour mission, we consideredLow Earth Orbit (LEO) as a parking orbit due to restricted launch vehicle technology.Withtheaboveexplanation,orbital analysis resulted in a Δv equal to 3.663 km/sec and an orbital maneuvering duration of 6 days.Finally, launch vehicle wasdesigned based onthe optimization of major design parameters. These optimized parameters are mass distribution of different stages to launch maximum payload mass to the orbit, launch vehicle pitch program to get to the maximum final velocity, minimum velocity loss due to gravity, and also minimum axial acceleration of various stages of launch vehicle.LVCD algorithm is coded in a software to let the design engineer explore the design space and also to reduce the cost and time of conceptual design phase by authors. By applyingthissoftwareto launch vehicle conceptual design, SADRA-1 almost obtained 125 and 175 tons at gross lift off mass and thrust force, respectively.

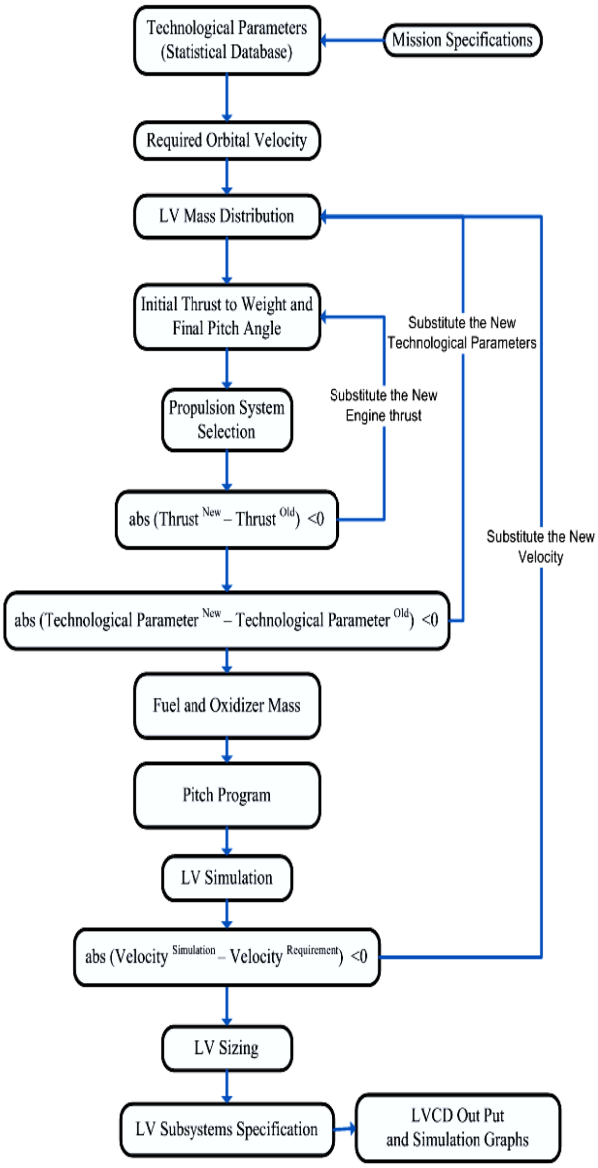

Fig. 2. LVCD Flowchart

Table 3. Mission definition of lunar exploration

Input Parameter		Value	Unit
1	Spacecraft Mass	2.0	Ton
2	Perigee Altitude	200	Km
3	Orbit Inclination	33	Deg
4	Launch Point Specification	$31^0 N, 53^0 E$	

Table 4. LVCD Results

LV Mass-Dimensional and ballistic specifications		LVCD	Unit
1	Initial Mass of LV (lift off mass)	125.384	ton
2	Block mass of 1st stage	106.239	ton
3	Block mass of 2nd stage	16.145	ton
4	Propellant mass of 1st stage	101.239	ton
5	Propellant mass of 2nd stage	14.245	ton
6	Thrust to weight of 1st stage	1.4	--
7	Thrust to weight of 2nd stage	0.81	--
8	Maximum achievable altitude	202	km
9	1st stage separation altitude	63.84	km
10	Time of maximum dynamic pressure	68	Sec
11	Altitude of maximum dynamic pressure	12.65	Km
12	1st stage final pitch angle regardingthe start coordinate	32.7	deg
13	LV total length	24.0	m
14	LV diameter	2.5	m

References

[1] Available, [on line]: http://www.jaxa.jp/projects/sat/selene
[2] Available, [on line]: http://nssdc.gsfc.nasa.gov
[3] Available, [on line]: http://sci.esa.int/science-e
[4] Available, [on line]: http://www.isro.gov.in/ Chandrayaan
[5] Frank A. Slazer, James C. Harvey, Robert J. Sirko, Jonathan D. Barr, David K. Bonnar, Scott C. Ward, Delta IV Launch Vehicle Growth Options to Support NASA'S Space Exploration Vision, 55th International Astronautical Congress, Vancouver, Canada, 2004.
[6] Charania, A.C., Andrew, M., Crocker, E. Bradford, J. andR. Olds, J.,"A Method for Strategic Technology Investment Prioritization for Advanced Space Transportation Systems,"*52nd International Astronautical Congress,* Toulouse, France, 1-5 Oct 2001.
[7] Hofeller, J.Bjelde, B. andVozoff, M., "Responsiveness of the Falcon 1 Launch Vehicle, Associated Challenges, and Projected Improvements," 6th Responsive Space Conference, Los Angeles, CA, April 28–May 1, 2008.
[8] Conley, C., "Low Energy Transit Orbits in the Restricted Three-Body Problem," *SIAM Journal of Applied Mathematics*, Vol. 16, 1968, pp. 732-746.
[9] Belbruno, E. and Miller, J., A Ballistic Lunar Capture Trajectory for the Japanese Spacecraft Hiten, JPL IOM 312/90.4-1731-EAB, June 15, 1990.
[10] Koon, W.S., Lo, M.W., Marsden, J.E., & Ross, S.D., "Shoot the Moon, Spaceflight Mechanics 2000,"*AAS*, Vol. 105, Part II, 2000, pp. 1017-1030.
[11] Parker, J. S. and Lo, M.W., "Shoot the Moon 3D,"*Advances in the Astronautical Sciences, AAS 05-383*, 2006.
[12] Koon,W.S.,Lo, M.W.,Marsden, J.E. andRoss, S.D.,"Dynamical Systems, the Three-Body Problem and Space Mission Design," word scientific, Equdiff 99, 2000, pp. 1167-1181.
[13] Howard D. Curtis, Orbital Mechanics for Engineering Students, Embry Riddle Aeronautical University Dayton Beach, Florida, 2005, pp. 570-574.
[14] Martin J. L. Turner, Rocket and Spacecraft Propulsion, Principles, Practice and New Developments (Second Edition), 2005, pp. 25-30.
[15] John R. Beisel, Martin Baltimore, "A Weight Engineer's Viewof The Space Age Vehicle,"*The Seventeenth Conference of S.A.W.E*,The Belmont plaza,New York, 1958.
[16] R. Pence, D.,"Preliminary Weight Estimation of Liquid Propellant Stages,"*26th Annual Conference at Boston*, Massachusetts, May 1-4, 1967.
[17] R. Reitz, G., Denver Colorado, "A Methodof Weight Estimation for Advanced Missile Design,"*23rd National Conference, SAWE Dalastexas,* May 18-21, 1964.
[18] Mirshams, M.,Karimi, H. andNaseh, H.,"Algorithm for Optimization of Mass Distribution in Multi-stage Liquid Propellant Launch Vehicle,"*6thIranian Society of Aerospace Conference, K.N. Toosi University of Technology*, 2007 (in Persian).
[19] Malyshev, V.V.,Krasilshikov, M.N.,Bobronikov, V.T. andDishel, V.D.,*Aerospace Vehicle Control, Modern Theory and Applications*, Institute aeronautics and Space, Brazil, IAE, 1996.
[20] E. Kirk, D.,*Optimal Control Theory An Introduction*, Monterey, California, 1970.
[21] Mirshams,M.,Karimi, H. andNaseh,H.,"Suggestion Algorithm for Determination of Pitch program in Multi-stage Launch Vehicle in Conceptual Design Phase,"*6th Iranian Society of Aerospace Conference,* K.N. Toosi University of Technology, 2007 (in Persian).
[22] Lewis, J. andM.Rice,Th.,"Design of University Launch Vehicle System,"*Journal of AIAA, Department of Aerospace Engineering,* University of Maryland at College Park, 1992.
[23] Chakroborty, Sh. R. Wertz, J. andConger,R., "The Scorpius Expendable Launch Vehicle Family and Status of the Sprite Small Launch Vehicle," *1stResponsive Space Conference*, April 1-3, 2003.
[24] Tsohas,J. andJ. Droppers, L.,"Sounding Rocket Technology Demonstration for Small Satellite Launch Vehicle Project,"*AIAA*, 24-27 April 2006.
[25] Mirshams, M.,Karimi, H. andNaseh, H.,"Multi-stage liquid propellant launch vehicle conceptual design based on combinatorial optimization of major design parameters,"*Journal of Space Science & Technology (JSST)*, Vol. 1, No. 1, 2008, pp. 21-36 (in Persian).
[26] Mirshams, M.,Karimi, H. andNaseh,H.,"Launch Vehicle Conceptual Design (LVCD) Software,"*7th International Conference Society of Aerospace*, Sharif University of Technology, 2009(in Persian).

[27] Mahyari, M.N.,Karimi, H.,Naseh, H. andMirshams, M.,“Numerical and Experimental investigation of vortex breaker effectiveness on the improvement in launch vehicle ballistic parameters,”*Journal of Mechanical, Science and Technology*, Vol. 24, No. 10, 2010, pp. 1997-2006.

[28] Akhlaghi, A.M., Naseh, H., Mirshams, M., Irani, S.,“A Bayesian Networks Approach to Reliability Analysis of a Launch Vehicle Liquid Propellant Engine,”*Journal of Aerospace, Science and Technology (JAST),* 2012.

[29] Rothschild,W.,Boyd, D. andHenderson,E., “Shuttle Derived Launch VehicleConcepts,”*AIAA Conference, Long Beach,* California, 30 August -1 September 2005.

[30] Berglund, M.,Marin, D. andWilkins, M., “The Next-Generation Heavy-Lift Vehicle - The Inaugural Flight of the EELV Delta IV Heavy,”*AIAA Conference, Long Beach,* California, 30 August -1 September 2005.

[31] Ishimoto, Sh.,FujiiI,K..andShimura, K.,“A Design Study of a Next Generation Launch System,” ISTS 2008-g-9, 26th ISTS, Hamamatsu, Japan,June 2008.

[32] Joveini, H., “Safireomid,” Azarioon Book Company (published in persian), 2009.

[33] Kakoi, M., Design of Transfers from Earth –Moon L1/L2Libration Point, Orbits toa Destination Object, [Thesis PhD],Purdue University, 2015.

[34] Soo No, T.,Lee, J.,Eon Jeon, G. andLee, D.,“A Study on Earth-Moon Transfer Orbit Design,”*Int’l J. of Aeronautical & Space Sci.*Vol. 13, No. 1, 2012, pp. 106–116.

[35] I.Abouelmagd, E.,Asiri, H.M. andSharaf, M.A.,“The effect of oblateness in the perturbed restricted three-body problem,”MeccanicaVol. 48, No. 10, 2013, pp 2479–2490.

[36] L. Bradley, D.,M. Sharp, A. and Kumar, R.,“Student-Designed Liquid Rocket Engine From Concept to Completion,”*Joint Propulsion Conference, AIAA Propulsion and Energy Forum,* 4865, 2018.