AUTOMATIC CONTROL OF A SPACECRAFT TRANSFER TRAJECTORY FROM AN EARTH ORBIT TO A MOON ORBIT

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The paper addresses a Spacecraft transfer from an elliptical orbit around the Earth to a circular orbit around the Moon. The geometric parameters of the transfer trajectory are computed, consisting of two orbital arcs, one elliptical and one hyperbolic. Then the state equations are set, describing the dynamics of the three bodies (Spacecraft, Earth and Moon) relative to the Sun. A nonlinear control law (orbit controller) is designed and the parameters of the reference orbit are computed (for each orbital arc);The MATLAB/Simulink model of the automatic control system is designed and with this, by numerical simulation, the transfer path (composed of the two orbit arcs) of the Spacecraft is plotted relative to the Earth and relative to the Moon, the evolution of the Keplerian parameters, the components of the position and velocity vector errors relative to the reference trajectory, as well as the components of the command vector.

Keywords: elliptical orbit, nonlinear control, reference orbit.

1. Introduction

Considering that the Spacecraft (S) runs an elliptical orbit, with Earth (P) located in one of the ellipse foci, the S transfer over a circular orbit around the Moon (L) is done by traversing a trajectory composed of two orbital arcs, one elliptical and one hyperbolic.

From the multitude of papers studied on this topic in order to elaborate the present paper, we mention mainly the following [1] - [15].

In this paper one defines and calculates the parameters of the two orbits of the two components (arcs) of the trajectory of the transfer, considered as reference orbits. Starting from the equations describing the dynamics of the three bodies (S, P and L) relative to the Sun, an automatic control structure is designed for the

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transfer trajectory, comprising a reference trajectory modeling block; this block shapes successively the two reference orbits.

The control law is obtained from the stability condition of the system (using a positive defined Lyapunov function).

For the designed automatic control structure, the MATLAB/Simulink model is built and, by numerical simulation, the transfer trajectory of S is plotted relative to the Earth and the Moon, the evolution in time of the Keplerian parameters of the trajectory and the components of the error position and error velocity vectors of S relative to the reference trajectory components.

2. Computing the transfer trajectory of the spacecraft from an orbit around Earth to an orbit around the Moon

The transfer of a spacecraft S from an orbit around a planet (for example Earth) to an orbit around another celestial body (for example Moon) it can be done on an orbit modeled by various methods, from which are remembered: Hohman transfer, PCA (patched conic approximation), ballistic capture method [16].

The PCA method approximates the transfer trajectory with two orbit arcs (for instance elliptical), as it results from fig. 1; E_1 represents the orbit around Earth, from which the transfer is made in the injection point S_1; E_2 and E_3 are the orbits from the componence of the transfer trajectory; E_4 represents the orbit S around the Moon. The radius of the Moon sphere of influence is expressed by the formula [2]

$$\boldsymbol{r}_2 = \boldsymbol{r}_{LP}\left(\frac{m_L}{m_P}\right)^{2/5} \tag{1}$$

where $\boldsymbol{r}_{LP} = 384\,400\,\mathrm{km}$ is the distance between the center of mass of P and L, and m_P and m_L – the masses of these bodies; $\frac{m_L}{m_P} = \frac{1}{81.3}$; it results $\boldsymbol{r}_2 = 66183\,\mathrm{km}$.

If the injection is produced at the perigee of E_1, then $\Delta \boldsymbol{V}_1 = 0$ and $\boldsymbol{V}_1 = \boldsymbol{V}_0 = 10.88\,\mathrm{km/s}$ and $\gamma_0 = 0$ (the angle between velocity vector and the normal to vector $\vec{\boldsymbol{r}}_0$; $\boldsymbol{r}_0 = 6700\,\mathrm{km}$).

The eccentricity of orbit E_2 is [2]

$$e_2 = \sqrt{1 + 2E_2 \frac{K_2^2}{K_p^2}} \tag{2}$$

where E_2 is the specific energy, and K_2 - specific moment,

$$E_2 = \frac{\boldsymbol{V}_0^2}{2} - \frac{K_p}{\boldsymbol{r}_0}, \; K_2 = \boldsymbol{r}_0 \boldsymbol{V}_0 \cos\gamma_0; \tag{3}$$

it results $e_2 = 0.98973$.

The semi-major axis of E2 ellipse is expressed by the formula in [6], it results $a_2 = 652594\,\text{km}$.

The norm of the position vector $\vec{r}_1$(it represents the position of S) relative to Earth, according to fig. 1

$$\boldsymbol{r}_1 = \sqrt{\boldsymbol{r}_{LP}^2 + \boldsymbol{r}_2^2 - 2\boldsymbol{r}_{LP}\boldsymbol{r}_2\cos\lambda_1}; \tag{4}$$

Choosing $\lambda_1 = 60\,\text{grad}$, one obtains $\boldsymbol{r}_1 \approx 356000$ km.

The angle φ_1 is computed with

$$\varphi_1 = \arccos\frac{\boldsymbol{r}_1^2 + \boldsymbol{r}_{LP}^2 - \boldsymbol{r}_2^2}{2\boldsymbol{r}_1\boldsymbol{r}_{LP}} = \arccos\frac{\boldsymbol{r}_{LP} - \boldsymbol{r}_2\cos\lambda_1}{\boldsymbol{r}_1} \tag{5}$$

with the previous given values, one obtains φ_1=9.26 grad.

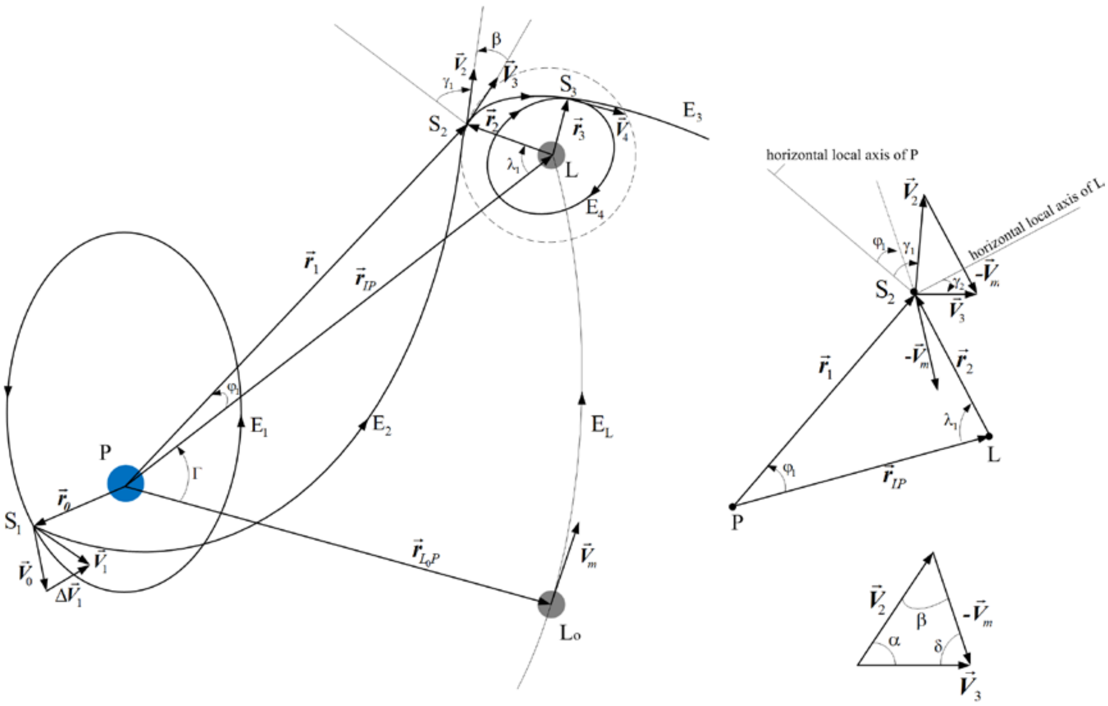

Fig. 1. The orbit of S and the position vectors of S, P and L

In fig. 1 are represented the local horizontal axis of P (perpendicular on the local vertical in the point S_2, which is the vector $\vec{r}_1$) and the horizontal axis of the Moon (perpendicular on the Moon vertical, which is the vector $\vec{r}_2$).

The norm of the velocity vector in the point S_2 is computed with [3]

$$\boldsymbol{V_2} = \sqrt{K_L\left(\frac{2}{\boldsymbol{r_1}} - \frac{1}{a_2}\right)}, \tag{6}$$

where K_L – gravitational constant of the Moon (K_L = 4902.8 km^3/s^2); one obtains $\boldsymbol{V_2}$ =1.276 km/s.

The slope angle γ_1 is computed with [6]

$$\gamma_1 = \text{arctg}\left(\frac{e_2 \sin \nu}{1 + e_2 \cos \nu}\right), \tag{7}$$

where ν is the true anomaly,

$$\nu = \arccos\left(\frac{\frac{\boldsymbol{r_0}}{\boldsymbol{r_1}}(1+e_2) - 1}{e_2}\right); \tag{8}$$

it results the values $\nu = 166.54$ grad and $\gamma_1 = 80.766$ grad .
The movement time between the points S_1 and S_2 is computed with [6]

$$t_{12} = (\varepsilon - e \sin \varepsilon)\left(\frac{a_2}{K_P}\right)^{-3/2}, \tag{9}$$

where ε is the eccentric anomaly,

$$\varepsilon = \arccos \frac{e_2 + \cos \nu}{1 + e \cos \nu}; \tag{10}$$

it results $t_{12} = 49{,}752$ h .

The medium velocity of the Moon around the Earth is V_m = 1.023km/s. In the point S_2, on the satellite is acting the velocity $\vec{\boldsymbol{V}}_2$ and the velocity $\vec{\boldsymbol{V}}_m$ (equivalent to fixed Moon). From the triangle of velocities, the resultant velocity $\boldsymbol{V_3}$ is computed. Finally the following values are obtained $\gamma_2 = 57.05$ grad , $\boldsymbol{V_3}$ = 1.359 km/s.

The specific energy of S is computed for the point S_2

$$E_3 = \frac{\boldsymbol{V}_3^2}{2} - \frac{K_L}{\boldsymbol{r_2}}; \tag{11}$$

it results E_3 = 0.84936 km^2/s^2. The specific moment on this orbit is

$$K_3 = \boldsymbol{r}_2 \boldsymbol{V}_3 \cos \gamma_2; \tag{12}$$

it results $\boldsymbol{K_3}$ = 48 920 km^3/s.

With E_3 the semi-major axis of E_3 orbit is computed, [6]

$$a_3 = -\frac{K_L}{2E_3}; \tag{13}$$

one obtains $a_3 = -2886{,}2\,\text{km}$.

The eccentricity of the orbit E_3 is computed with equation [3]

$$e_3 = \sqrt{1 - \frac{K_3^2}{K_L a_3}}\,; \tag{14}$$

one obtains e_3 = 13.0432. The negative value of the semi-major axis (a_3) and the higher than one value of eccentricity (e_3) expresses the fact that the orbit E_3 is hyperbolical and not elliptical. E_4 is chosen as a circular orbit with $a = 3{,}113 \times 10^7\ \text{m}$.

3. Dynamics of the three-body relative to the Sun

The dynamics of S on its orbit around Earth is perturbed by the interactions generated by the other celestial bodies, mainly by the Sun and Moon, but also by the solar radiation pressure, Earth's atmosphere and unequal mass distribution. In fig. 2 the orbit of the spacecraft is represented (satellite, rocket), the position vectors $\vec{r}_S, \vec{r}_P, \vec{r}_L$ of the three bodies S, Earth and Moon relative to the Sun and the position vectors which express the position of S relative to the Moon ($\vec{r}_{SL}$), respectively the position of Moon relative to the Earth ($\vec{r}_{LP}$).

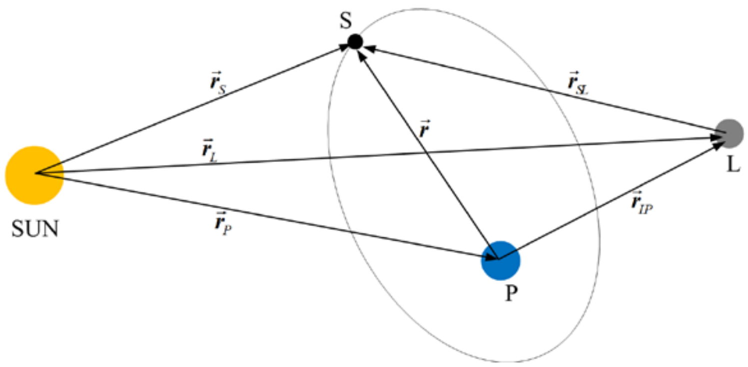

Fig. 2. The orbit of S and the position vectors of S, P and L

The force of attraction of the body *i* toward the body *j* is expressed with the equation [14]

$$\vec{F}_i = -\chi \sum_{j=1}^{4} \frac{m_i m_j}{r_{ij}^3} \vec{r}_{ij}\,, \tag{15}$$

where $\chi = 6.67259 \cdot 10^{-11}\,\text{m}^3\text{kg}^{-1}\text{s}^{-2}$ – gravitational constant and $\vec{r}_{ij} = \vec{r}_i - \vec{r}_j$ with $\vec{r}_i$ and $\vec{r}_j$ - position vectors of the body *i* and *j* relative to the Sun. From this it results the acceleration of the body *i* relative to body *j*

$$\ddot{\vec{r}}_{ij} = -\chi \sum_{j=1}^{4} \frac{m_j}{r_{ij}^3} \vec{r}_{ij} \,, r_{ij} = \left\| \vec{r}_{ij} \right\|, \tag{16}$$

relation which is equivalent with the state equations system

$$\begin{aligned} \dot{\vec{r}}_i &= \vec{V}_i \,, \\ \dot{\vec{V}}_i &= -\chi \sum_{j=1}^{4} \frac{m_j}{r_{ij}^3} \vec{r}_{ij} \,. \end{aligned} \tag{17}$$

Customizing these relations for i = S, P, L, one obtains the following results, where $\vec{V}_S, \vec{V}_P$ and $\vec{V}_L$ are the velocities of the bodies S, P and L;

$$\begin{aligned} \dot{\vec{r}}_s &= \vec{V}_s \\ \dot{\vec{V}}_s &= -\chi \frac{m_P}{\left\| \vec{r}_s - \vec{r}_P \right\|^3} \left(\vec{r}_s - \vec{r}_P \right) - \chi \frac{m_S}{\left\| \vec{r}_s \right\|^3} \vec{r}_s - \chi \frac{m_L}{\left\| \vec{r}_s - \vec{r}_L \right\|^3} \left(\vec{r}_s - \vec{r}_L \right) \end{aligned} \tag{18}$$

$$\begin{aligned} \dot{\vec{r}}_P &= \vec{V}_P \\ \dot{\vec{V}}_P &= -\chi \frac{m}{\left\| \vec{r}_P - \vec{r}_s \right\|^3} \left(\vec{r}_P - \vec{r}_s \right) - \chi \frac{m_L}{\left\| \vec{r}_P - \vec{r}_L \right\|^3} \left(\vec{r}_P - \vec{r}_L \right) - \chi \frac{m_s}{\left\| \vec{r}_P \right\|^3} \vec{r}_P \end{aligned} \tag{19}$$

$$\begin{aligned} \dot{\vec{r}}_L &= \vec{V}_L \\ \dot{\vec{V}}_L &= -\chi \frac{m}{\left\| \vec{r}_L - \vec{r}_s \right\|^3} \left(\vec{r}_L - \vec{r}_s \right) - \chi \frac{m_P}{\left\| \vec{r}_L - \vec{r}_P \right\|^3} \left(\vec{r}_L - \vec{r}_P \right) - \chi \frac{m_S}{\left\| \vec{r}_L \right\|^3} \vec{r}_L \end{aligned} \tag{20}$$

m_S, m_P, m_L and m are the masses of the Sun, Earth, Moon and spacecraft. The state equations that express the dynamics of S on Earth's orbit (according to fig. 2), with $\vec{r} = \vec{r}_S - \vec{r}_P$;

$$\begin{aligned} \dot{\vec{r}} &= \vec{V}, \ \vec{r} = \vec{r}_S - \vec{r}_P \,, \\ \dot{\vec{V}} &= \dot{\vec{V}}_S - \dot{\vec{V}}_P \,. \end{aligned} \tag{21}$$

with $\dot{\vec{r}}_s$, $\dot{\vec{V}}_s$ and $\dot{\vec{r}}_P$, $\dot{\vec{V}}_P$ of form (18) and (19).

4. Automatic control of spacecraft orbit

In the time interval in which S evolves on the elliptical orbit around the Earth, the position vector $\vec{r}$ of S has the origin in the center of the Earth (P is in one of the ellipse foci).

The structure of the control system for S orbit is given in fig. 3.

The control law is of type non-linear, based on using a Lyapunov function of form

$$V_l = \frac{1}{2} k_1 \Delta \boldsymbol{r}^{\mathrm{T}} \Delta \boldsymbol{r} + \frac{1}{2} k_2 \Delta \boldsymbol{V}^{T} \Delta \boldsymbol{V}\,, \tag{22}$$

where k_1 and k_2 are positive constants. From the stability condition of the closed loop circuit system ($\Delta\boldsymbol{r}$ and $\Delta\boldsymbol{V}$ tend simultaneously to zero), that is from the

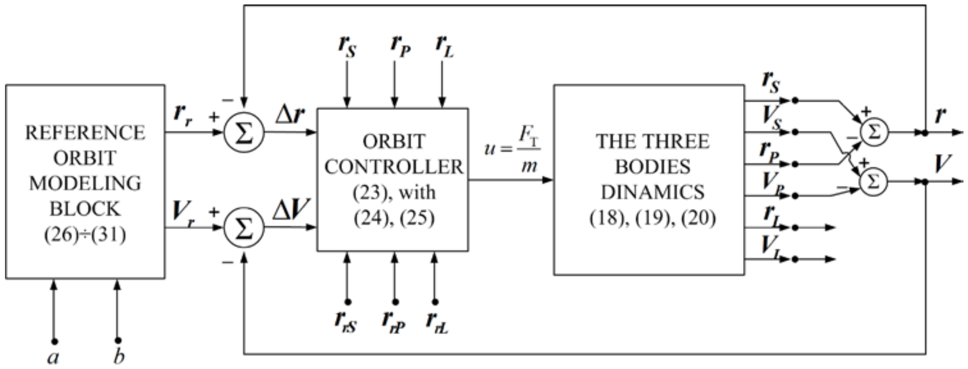

Fig. 3. The structure of automatic control system of Spacecraft orbit

condition $\dot{V}_l \le 0$, the control law is obtained [13]

$$u = -\frac{k_1}{k_2} \Delta \boldsymbol{r} - k_3 \Delta \boldsymbol{V} - (a_A + a_B), \tag{23}$$

with $k_3 > 0$ and

$$a_A = \dot{\boldsymbol{V}}_P - \dot{\boldsymbol{V}}_S = \chi \frac{m_S}{\|\boldsymbol{r_S}\|^3} \boldsymbol{r_S} + \chi \frac{m_L}{\|\boldsymbol{r_S} - \boldsymbol{r}_L\|^3} (\boldsymbol{r}_S - \boldsymbol{r}_L) - \chi \frac{m_L}{\|\boldsymbol{r_P} - \boldsymbol{r}_L\|^3} (\boldsymbol{r_P} - \boldsymbol{r}_L) - \chi \frac{m_S}{\|\boldsymbol{r_P}\|^3} \boldsymbol{r_P} \tag{24}$$

$$a_B = \dot{\boldsymbol{V}}_{rS} - \dot{\boldsymbol{V}}_{rP} = \chi \frac{m_L}{\|\boldsymbol{r_{rP}} - \boldsymbol{r}_{rL}\|^3} (\boldsymbol{r_{rP}} - \boldsymbol{r_{rL}}) + \chi \frac{m_S}{\|\boldsymbol{r_{rP}}\|^3} \boldsymbol{r_{rP}} - \chi \frac{m_S}{\|\boldsymbol{r_{rS}}\|^3} \boldsymbol{r_{rS}} - \chi \frac{m_L}{\|\boldsymbol{r_{rS}} - \boldsymbol{r}_{rL}\|^3} (\boldsymbol{r}_{rS} - \boldsymbol{r}_{rL}) \tag{25}$$

a_A and a_B are perturbations, having the signification of accelerations, with $\dot{\boldsymbol{V}}_{rS}$ and $\dot{\boldsymbol{V}}_{rP}$ - reference accelerations of S and P, and $\boldsymbol{r_{rS}}$, $\boldsymbol{r_{rP}}$ and $\boldsymbol{r_{rL}}$ - position vectors of S, P and L relative to the Sun (see fig. 2) corresponding to the reference orbit around the Earth. The control law (23) assures the fulfillment of the condition $\dot{V}_l = -k_2 k_3 \Delta \boldsymbol{V}^{\mathrm{T}} \Delta \boldsymbol{V}$.

5. Reference orbit model

For the computation of reference orbit, the unperturbed orbit is considered, defined by the Keplerian elements *p* and *e* (the movement is assumed in plane) with the initial position of the space vehicle at perigee.

The focal parameter p and the eccentricity of the elliptical orbit are computed from the semi-major and semi-minor axis a and b of the ellipse;

$$p=\frac{b^2}{a},\ e=\sqrt{1-\frac{b^2}{a^2}}, \tag{26}$$

The equation of the reference orbit in polar coordinates has the form:

$$\|\boldsymbol{r}_r\|=\frac{p}{1+e\cos(\varphi-\varphi_0)} \tag{27}$$

with φ - true anomaly, and the formula to compute the reference vector is [16]

$$\boldsymbol{r}_r=[\cos(\varphi-\varphi_0)\quad \sin(\varphi-\varphi_0)\quad 0]^{\mathrm{T}}\cdot\|\boldsymbol{r}_r\| \tag{28}$$

For the computation of the true anomaly, the medium anomaly is computed first $M=(2\pi/T)t$, where T is the revolution period (in which a full orbit is traveled), and t - the current time. Then the eccentric anomaly E is computed, numerically solving the equation $E-e\sin E=M$ through Newton iterative method. The angle φ is given by the relation [3], [16]

$$\varphi=2\arctan\left(\sqrt{\frac{(1+e)\tan\frac{E}{2}}{1-e}}\right) \tag{29}$$

The absolute value of satellite velocity is computed with the formula [3]

$$\|\boldsymbol{V}_r\|=\sqrt{K_p\left(\frac{2}{\|\boldsymbol{r}_r\|}-\frac{1}{a}\right)} \tag{30}$$

K_p being the gravitational constant of Earth, $K_p=\chi m_P$, and the reference velocity vector of S given by [6]

$$\boldsymbol{V}_r=[\cos\theta\quad \sin\theta\quad 0]^{\mathrm{T}}\|V_r\| \tag{31}$$

with θ, expressed with formula [6], [13]

$$\theta=\varphi+\frac{1}{2}\left[\pi-\arccos\left(\frac{2-2e^2}{\frac{r}{a}\left(2-\frac{r}{a}\right)}-1\right)\right] \tag{32}$$

6. Numerical simulations

The following values are used as initial vectors: $\boldsymbol{r}_s(0)=[1{,}460067\times10^{11}\quad 0\quad 0]^{\mathrm{T}}$ m, $\boldsymbol{r}_L(0)=[1{,}46356\times10^{11}\quad 0\quad 0]^{\mathrm{T}}$ m, $\boldsymbol{r}_P(0)=[1{,}46\times10^{11}\quad 0\quad 0]^{\mathrm{T}}$ m, $\boldsymbol{V}_s(0)=$

$=\begin{bmatrix}0 & 37392.9 & 0\end{bmatrix}^T$ m/s, $\boldsymbol{V}_L(0)=\begin{bmatrix}0 & 30772{,}9 & 0\end{bmatrix}^T$ m/s, $\boldsymbol{V}_P(0)=\begin{bmatrix}0 & 29680 & 0\end{bmatrix}^T$ m/s. The command $u = F_T / m$ (F_T - the resultant thrust force of the Spacecraft engines, m - mass of the spacecraft); $m_S = 1{,}9891 \times 10^{30}$ kg, $m_P = 5{,}972 \times 10^{24}$ kg, $m_L = 7{,}35 \times 10^{22}$ and $m = 300$ kg.

The values used for the computation of the reference orbits are: for the Earth orbit around the Sun $a = 1{,}496 \times 10^{11}$ m, $e = 0{,}0167086$, $T = 365{,}256363004$ days; for the orbit of the Moon around the Earth $a = 3{,}84748 \times 10^{8}$ m, $e = 0{,}0549006$, $T = 27.554550$ days; for the spacecraft orbit around Earth $a = 6{,}7 \times 10^{6}$ m, $e = 0$.

For the control of the spacecraft trajectory the system with the structure in fig. 3 is modeled in Matlab/Simulink, in which the reference block generates the corresponding arcs for the orbits E_2, E_3, E_4. The block contains a Matlab function that computes the orbital state vectors (position and velocity) for the Earth, the Moon and the Spacecraft given the orbital parameters.

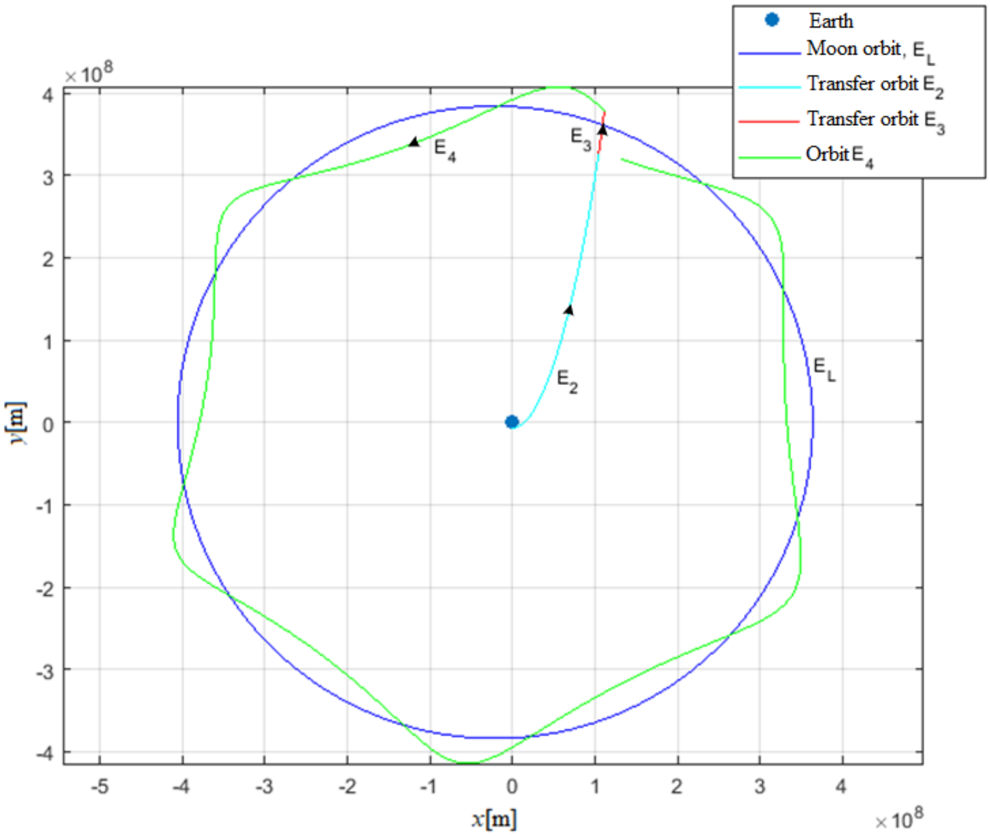

Fig. 4. Spacecraft trajectory relative to the Earth

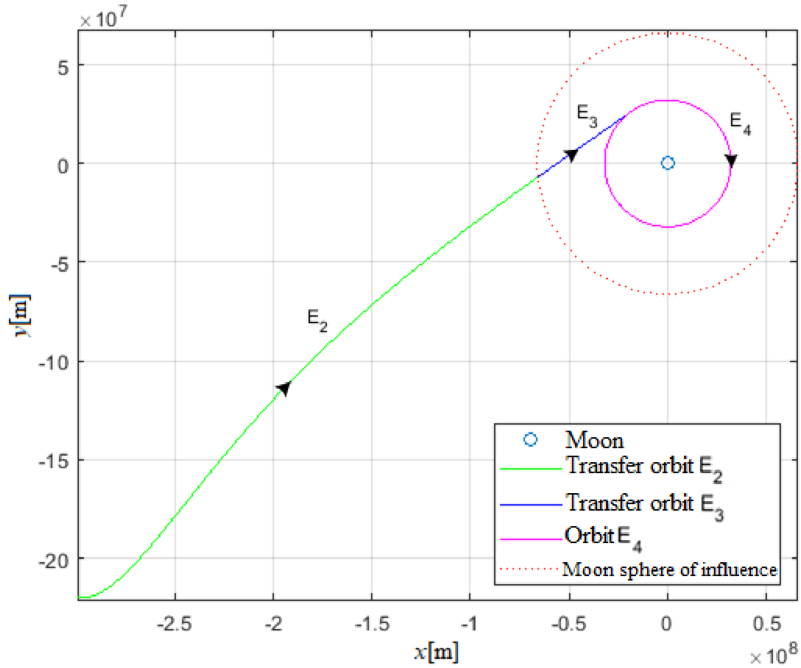

Fig. 5. Spacecraft trajectory relative to the Moon

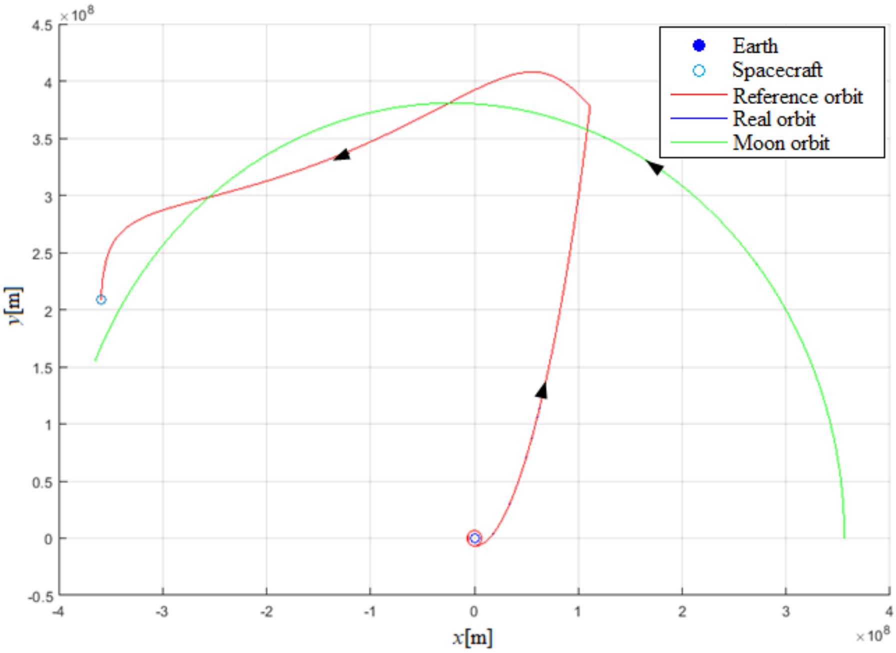

Fig. 6. Spacecraft real trajectory a reference trajectory relative to the Earth

In fig. 4 the spacecraft reference trajectory is presented for a transfer on an orbit around the Moon in the Earth centered reference frame, and in fig. 5 the same trajectory relative to the Moon.

Fig. 6 represents the reference trajectory (with red line) and real trajectory (with blue line); the two trajectories are overlapped.

In fig. 7 are presented the following time characteristics: a) the time evolution of some of the Keplerian parameters; b) the components of vector $\Delta \boldsymbol{r}$; c) the components of vector $\Delta \boldsymbol{V}$; d) the components of the command vector u. The moment of time when the transfer maneuver begins (the spacecraft is in S_1) is marked as t_1. The representation of the Keplerian parameters is done relative to the central body: Earth, before reaching S_2 and Moon, after S_2 point when the spacecraft enters the Moon sphere of influence.

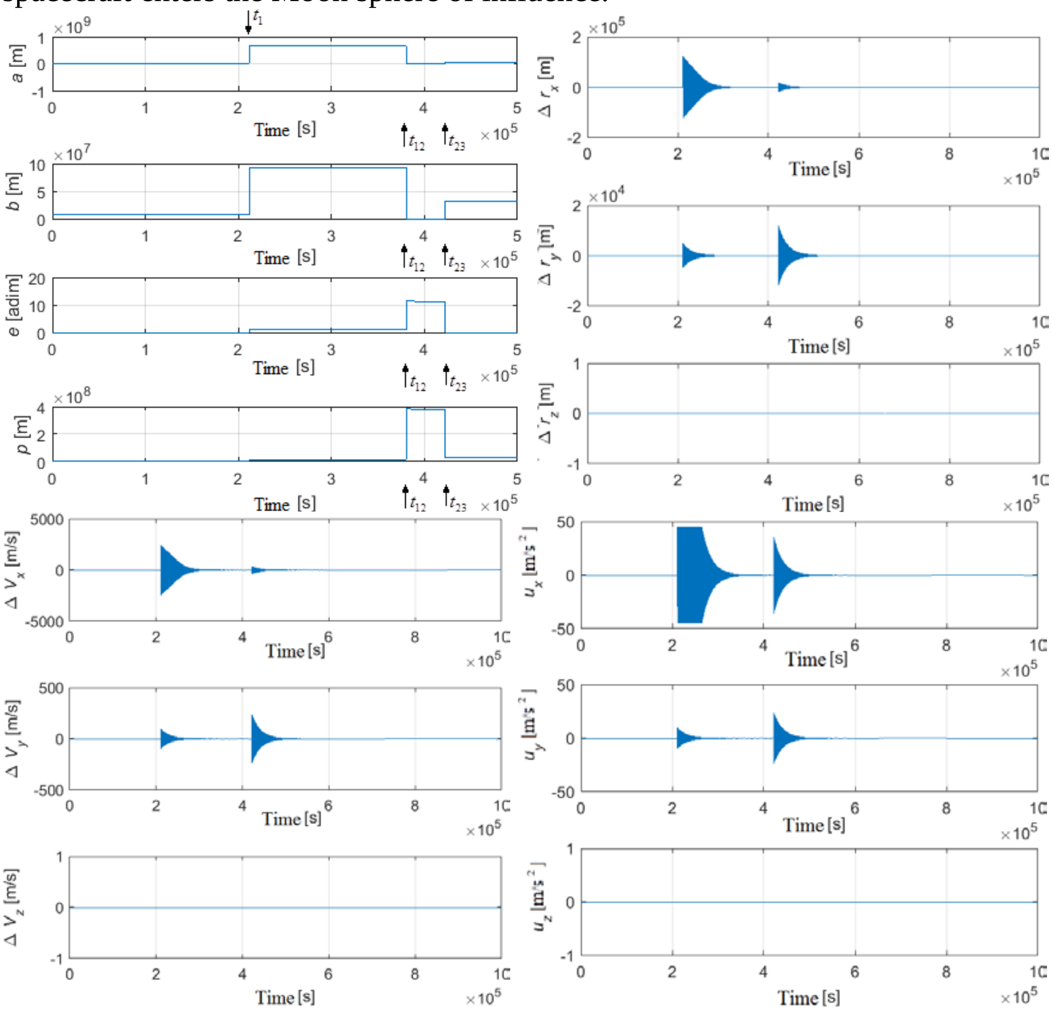

Fig. 7. The evolution in time of orbital parameters (semi-major axis, semi-minor axis, eccentricity, focal parameter), of the position and velocity deviation ($\Delta \boldsymbol{r}$, $\Delta \boldsymbol{V}$) and of the command u

7. Conclusions

A nonlinear control system is designed for the transfer trajectory of a spacecraft (S) from an elliptical orbit around the Earth to a circular orbit around the Moon, starting from the dynamics equations of the three celestial bodies (S, P, L) relative to the Sun. The orbital parameters are computed for each of the two orbit arcs (E_2 and E_3) which compose the reference transfer trajectory. The Matlab/Simulink model of the control system for the transfer trajectory is designed and with this are plotted the transfer trajectory of S relative to Earth and to Moon, the time evolution of the Keplerian parameters for the transfer trajectory, the components of position and velocity vector errors of S relative to the reference trajectory components, and also the components of the command vectors. The time intervals in which the two orbit arcs are traveled are delimited (t_{12} for the elliptical orbit E_2, between S_1 and S_2, and t_{23} for the hyperbolic E_3, between S_2 and S_3).

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