MICROGRAVITY RESEARCH ABOARD THE PROGRESS VEHICLE IN AUTONOMOUS FLIGHT

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ABSTRACT

Three modes of uncontrolled rotation of the Progress space vehicle are proposed for experiments to study microgravity environment. They are described in the paper: triaxial gravitational orientation, gravitational orientation of the rotating vehicle and rotation in the orbital plane around the axis of the maximal moment of inertia of the vehicle. The modes were tested from May 24 to June 1, 2004, on the Progress M1-11 vehicle. Real motion of the vehicle around its center of mass in these modes was determined on the base of telemetric data on electrical current from the solar arrays. Values of current obtained on several hours time interval were processed with the help of the least squares method and integration of the vehicle rotational motion equations. As a result of processing, initial conditions of the motion and parameters of the mathematical model used for experiment were estimated. For the motions investigated, the quasi-static component of the micro-acceleration was calculated for the point aboard the vehicle where research equipment can be mounted.

1. INTRODUCTION

Microgravity environment on Russian segment of the International Space Station (ISS) is rather disturbed. That is why the opportunities are studied to perform microgravity research aboard the Progress vehicles when their operations in the framework of the ISS program are completed [1]. These Progress vehicles are supposed to be equipped by research equipment preliminary delivered to station and to be sent to autonomous flight not far from the station. During several days of this autonomous mission when the vehicle is mostly in uncontrolled rotational motion, experiments will be performed aboard the vehicle in automated mode. After that, the vehicle will redock to the station, equipment will be taken back to the station and the vehicle can be prepared for further research or for deorbit.

To check merits of such mission the special flight tests were executed on the Progress M1-11 vehicle from May 24 to June 1, 2004. Purpose of the tests was to estimate microgravity environment in some modes of uncontrolled rotational motion and investigate stability of these modes. Three modes were tested: 1) triaxial gravitational orientation, 2) gravitational orientation of the rotating vehicle, 3) rotation in the orbital plane around the axis of the maximal moment of inertia of the vehicle.

The Progress M1-11 vehicle had not acceleration sensors to measure microgravity environment. So, onboard micro-accelerations were calculated on the base of real vehicle rotational motion reconstruction. Such approach permits to find only the quasi-static component of micro-acceleration. However, this component is the most important one for the most of microgravity experiments. The Progress M1-11 vehicle had not advanced facilities to determine its real uncontrolled rotational motion either. This task was solved by means of statistical processing of telemetric data on electrical current from the vehicle solar arrays. Description of such data processing method, results of real vehicle rotational motion reconstruction with the help of this method and results of the quasi-static component of micro-acceleration calculations are given below.

2. SPACECRAFT ROTATIONAL MOTION EQUATIONS

Two systems of such equations were used. They differ by the ways of approximation of aerodynamic torque affecting the spacecraft. The first system is traditionally used for the problems of determination of artificial Earth satellites uncontrolled rotational motion [2]. The spacecraft is assumed to be a rigid body and its center of mass geocentric motion is supposed to be Keplerian elliptic motion. Elements of this motion are determined from the spacecraft orbital tracking data. Let us introduce two right-hand Cartesian coordinate systems: orbital $OX_1X_2X_3$ system and the system formed by principal central axes of inertia of the spacecraft $Ox_1x_2x_3$. The point O is the spacecraft center of mass, axes OX_3 and OX_1 are directed respectively along geocentric radius vector of the point O and tangential to orbital trajectory at this point. To simplify the model we assume that axis Ox_1 is directed along the longitudinal axis of the spacecraft to its assembly module, axis Ox_2 is perpendicular to the solar arrays plane. Photosensitive surface of the solar arrays is directed into semi-space $x_2 > 0$.

We will specify location of the system $Ox_1x_2x_3$ with respect to the system $OX_1X_2X_3$ by the angles γ, δ and β, which will be introduced by the following way. The system $OX_1X_2X_3$ can be transformed into the system $Ox_1x_2x_3$ by three consecutive turns: 1) through the angle $\delta + \pi/2$ around the axis OX_2, 2) through the angle β about formed new axis OX_3, 3) through the angle γ about formed new axis OX_1 coincident with the axis Ox_1. We designate the matrix of transition from the system $Ox_1x_2x_3$ to the system $OX_1X_2X_3$ as $\| a_{ij} \|_{i,\,j=1}^{3}$, where a_{ij} is cosine of the angle between the axes OX_i and Ox_j. Elements of this matrix are expressed in terms of the angles introduced with the help of simple trigonometric formulae.

Gravitational and restoring aerodynamic torques are taken into account in the equations of the spacecraft rotational motion. These equations take the form:

$$\dot{\omega}_1 = \mu(\omega_2\omega_3 - ha_{32}a_{33})$$

$$\dot{\omega}_2 = \frac{1-\lambda}{1+\lambda\mu}(\omega_1\omega_3 - ha_{31}a_{33}) + \frac{\lambda\kappa v_3}{1+\lambda\mu}$$

$$\dot{\omega}_3 = -(1-\lambda+\lambda\mu)(\omega_1\omega_2 - ha_{31}a_{32}) - \lambda\kappa v_2$$

$$\dot{a}_{11} = a_{12}\omega_3 - a_{13}\omega_2 - \omega_0 a_{31}$$

$$\dot{a}_{12} = a_{13}\omega_1 - a_{11}\omega_3 - \omega_0 a_{32}$$

$$\dot{a}_{13} = a_{11}\omega_2 - a_{12}\omega_1 - \omega_0 a_{33} \qquad (1)$$

$$\dot{a}_{31} = a_{32}\omega_3 - a_{33}\omega_2 + \omega_0 a_{11}$$

$$\dot{a}_{32} = a_{33}\omega_1 - a_{31}\omega_3 + \omega_0 a_{12}$$

$$\dot{a}_{33} = a_{31}\omega_2 - a_{32}\omega_1 + \omega_0 a_{13}$$

$$\lambda = \frac{I_1}{I_2}, \quad \mu = \frac{I_2 - I_3}{I_1}, \quad h = \frac{3\mu_e}{R^3}$$

$$\kappa = E\rho\left(q_1\sqrt{v_1^2 + v_2^2 + v_3^2} + q_2\,|\,v_2\,|\right)$$

Here, dots above the letters in the equations signify differentiation with respect to time t; ω_i и v_i $(i = 1, 2, 3)$ are the components of absolute angular velocity of the spacecraft and its velocity relative to the Earth's surface in the system $Ox_1x_2x_3$; parameters q_1 and q_2 show aerodynamic torque affecting the spacecraft; ω_0 is the modulus of absolute angular velocity of the orbital coordinate system; I_i – moments of inertia of the spacecraft relative to the axes Ox_i; μ_e is the gravitational parameter of the Earth; R is geocentric distance of the point O; ρ is the density of atmosphere at this point; E is the scale factor. When Eqns. 1 are numerically integrated the unit of time measurement is 1000 s, the unit of distance measurement is 1000 km, velocity is measured in km/s, the unit of angular rate measurement is 0.001 s^{-1}. Atmosphere density is calculated in kg/m^3 according to model [3], $E = 10^{10}$. Elements a_{2i} are calculated from formulae $a_{21} = a_{32}a_{13} - a_{33}a_{12}$, etc.

The variables a_{1i} and a_{3i} are not independent. They are related by conditions of the matrix $\| a_{ij} \|$ orthogonality. That is why initial conditions for a_{1i} and a_{3i} are expressed through the angles γ, δ and β.

3. METHOD OF SPACECRAFT ROTATIONAL MOTION DETERMINATION

We will approximate the real spacecraft motion with respect to its center of mass by solutions of Eqns. 1 selecting those solutions, which provide the best smoothing of telemetric data on current from the solar arrays. Electric current from the arrays is approximately proportional to cosine of angle of solar rays incidence to photosensitive surface of the solar arrays. Let the vector unit of the direction Earth-Sun to have the components $A_i(t)$ $(i = 1, 2, 3)$ in the orbital coordinate system. These components are found with the help of approximate formulae [4] and elements of the spacecraft Keplerian orbit. Cosine mentioned above and current from the solar arrays are specified by the formulae

$$\eta = A_1a_{12} + A_2a_{22} + A_3a_{32}\,, \quad I = I_0 \max(\eta, 0)$$

where I_0 is the maximal current received from the solar arrays in Earth orbit when solar arrays come to their surface perpendicularly, $I_0 \approx 29$ A. Actually, current calculations are more complicated and require to know more additional data, which are hardly available. However, simplified formula given here permits to obtain acceptable results. This formula can be even more simplified if to take into account that for the times with telemetric values of current exceeding some positive limit $I_{\min}$ the condition $\eta > 0$ is wittingly met. For such times current value can be calculated by formula $I = I_0 \eta$. In data processing we assumed that depending on the Sun height above the spacecraft orbital plane $I_{\min} = 0 \div 3$ A.

Telemetric data on current from the solar arrays present the sequences of numbers

$$t_n,\ I_n \quad (n = 1, 2, \ldots, N) \tag{2}$$

Here, I_n is approximate value of current at instant of time t_n, $t_1 < t_2 < \ldots < t_N$. As a rule, differences $t_{n+1} - t_n$ do not exceed several minutes. Processing is done for sequences of Eqn. 2 covering time intervals with $t_N - t_1$ equal to several hours.

Data of Eqn. 2 are processed with the help of the least squares method. Let the errors in values of I_n be independent and have equal normal distribution with zero average and standard deviation σ. Value of σ is unknown. On the base of solutions of Eqns. 1 taken within the interval $t_1 \le t \le t_N$ we will determine the functional

$$\Phi = \sum_{n=1}^{N} [I_n - I_0 \eta(t_n)]^2 . \tag{3}$$

We assume that the solution minimizing this functional is the approximation of real spacecraft motion on this interval. Minimization of Φ is executed with respest to initial conditions of solution at the point t_1 and parameters q_1, q_2, I_0. Values of parameters λ and μ are considered to be known: $\lambda = 0.16$, $\mu = 0.19$. To simplify description we combine all considered values in one vector, which is denoted by z, $\dim z = 10$. Then $\Phi = \Phi(z)$ and $z_* = \arg\min \Phi(z)$ is sought estimation of vector z.

Minimization of functional of Eqn. 3 (in this case, function $\Phi(z)$) was done by Levenberg-Marquardt method, which is one of modifications of Gauss-Newton method. Use of this method for the problems of satellites rotational motion determination on the base of onboard sensors measurements is described in [5]. Following the least squares method, we will characterize the accuracy of approximation of data in Eqn. 2 and spread in components z_* determination by appropriate standard deviations. Let C be the matrix of normal equations system computed in point z_*, which arises during minimization of Φ by Gauss-Newton method, $2C \approx \partial^2 \Phi(z_*) / \partial z^2$. Then standard deviation of errors in values I_n will be found by formula

$$\sigma = \sqrt{\frac{\Phi(z_*)}{N - 10}}$$

standard deviations of components z_* are equal to square roots from corresponding diagonal elements of the matrix $\sigma^2 C^{-1}$.

In order to support values of parameters q_1, q_2, I_0, which minimize functional of Eqn. 3, within physically reasonable margins, additional components were included into this functional

$$\varepsilon_1 \left(q_1^2 + q_2^2 \right) + \varepsilon_2 (I_0 - I_0^*)^2$$

Here, $I_0^* = 29$ A – nominal value of parameter I_0, $\varepsilon_1 = 10$, $\varepsilon_2 = 1$. The units of measurement of ε_1 and ε_2 are coordinated with the units, in which Eqns. 1 are integrated. Such substitution of functional takes into account a priori information on parameters to be corrected and regularizes the problem of minimum $\Phi(z)$ searching. We will keep previous designation for new functional. To calculate standard deviations we used new expression for Φ and corresponding matrix of normal equations.

4. RESULTS OF SPACECRAFT ROTATIONAL MOTION DETERMINATION

Determination of the real spacecraft motion with respect to its center of mass on the base of Eqn. 2 was executed on 21 time intervals. On 9 intervals the motion was supported in the mode of triaxial gravitational orientation. 6 intervals were taken for both the mode of gravitational orientation of the rotating vehicle and the mode of rotation in the orbital plane. Examples of the found spacecraft motion are presented in Figs. 1-3. All figures are arranged in the same manner and divided into three parts – left, middle and right part. In the left part of the figures the charts of functions $\gamma(t)$, $\delta(t)$, $\beta(t)$ and $I_0 \eta(t)$ are presented. They are built within the interval $t_1 \le t \le t_N$ and the origin of coordinates on the axis t is located at the point t_1. Angles γ and δ are adjusted to interval $[0, 2\pi]$. Points (t_n, I_n) $(n = 1, 2, \ldots, N)$ illustrating smoothed data of Eqn. 2 are indicated at the chart of function $I_0 \eta(t)$ by markers. In the middle part of the figures the charts of the spacecraft angular rate components $\omega_i(t)$ are presented, in the right part – the charts of micro-acceleration components $\mathbf{b} = (b_1, b_2, b_3)$ and its modulus $|\mathbf{b}|$ at specified fixed point of the vehicle.

Micro-acceleration components are given in the system $Ox_1x_2x_3$ and were computed by formula from [6]

$$\mathbf{b} = \mathbf{r} \times \dot{\boldsymbol{\omega}} + (\boldsymbol{\omega} \times \mathbf{r}) \times \boldsymbol{\omega} + \frac{\mu_e}{R^3}\left[3(\mathbf{E}_3 \cdot \mathbf{r})\mathbf{E}_3 - \mathbf{r}\right] + c\rho\,|\mathbf{v}|\,\mathbf{v}$$

Here, $\mathbf{r}$ is the radius-vector of the point, in which micro-acceleration is calculated, with respect to the point O; $\boldsymbol{\omega}$ is absolute angular velocity of the spacecraft; $\mathbf{v}$ is the spacecraft velocity relative to the Earth's surface; $\mathbf{E}_3$ is the vector unit of the axis OX_3; c is ballistic coefficient of the spacecraft. In the coordinate system $Ox_1x_2x_3$ components of the vectors $\boldsymbol{\omega}$, $\dot{\boldsymbol{\omega}}$, $\mathbf{E}_3$ and $\mathbf{v}$ are equal to ω_i, $\dot{\omega}_i$, a_{3i} и v_i $(i = 1, 2, 3)$ respectively (see above). Micro-acceleration was computed for the point with coordinates $(-3.5\text{ м}, 0.5\text{ м}, 0.5\text{ м})$ in the system $Ox_1x_2x_3$. Value of ballistic coefficient for these calculations was taken from the orbital tracking data.

The following is indicated in figure captions: date and Moscow (winter) standard time (MST) of the point t_1; number N of the data used for processing; angle φ between the plane OX_1X_3 (orbital plane) and the vector unit of the direction Earth-Sun at instant of time $(t_1 + t_N)/2$. This angle is the positive one if the Sun locates in semi-space $X_2 > 0$, and it is the negative one if the Sun lies in semi-space $X_2 < 0$. In addition, there are also indicated the standard deviation of errors in the data of measurements and the standard deviations of estimations of the spacecraft motion initial conditions made on the base of angular variables.

Analysis of the obtained results allows to make a conclusion about overall successful reconstruction of the spacecraft rotational motion made on the base of telemetric data of current from the solar arrays. This reconstruction was the most successful for the case of gravitational orientation of the rotating vehicle and the least successful for the case of rotation in the orbital plane. These facts can be easily explained based on the elementary kinematics and location of the Sun relative to the orbital plane. For example, the vehicle rotation was performed at the time when the Sun was located essentially in the orbital plane. Angle of solar rays incidence to the solar arrays surface was at that time about $90°$. In such case, formula $I = I_0 \max(\eta, 0)$ contains a great error.

Besides, reconstruction of the spacecraft rotational motion by the way described above for the modes of triaxial gravitational orientation and rotation in the orbital plane is possible only if the motion is disturbed. In case of perfect execution of these modes the relationship $\eta = \text{const}$ takes place (we ignore precession of the orbital plane) and the modes cannot be distinguished. Calculations of internal numbers of the matrix C introduced in section 3 demonstrated that for the real spacecraft motion in indicated modes this matrix is poorly determined and sufficiently high accuracy of reconstruction is impossible for any situation. Admittedly, presented standard deviations of estimations of the spacecraft motion initial conditions had to be regarded as very optimistic.

The level of micro-accelerations in the found motions appeared to be as it was predicted in [1]. Micro-accelerations detected in the mode of triaxial gravitational orientation were somewhat disappointing. For precise execution of this mode the components of micro-acceleration b_i must vary within narrow limits [1]. However, because of inexact initial conditions provision both times of the mode execution were very disturbed and the components b_i significantly varied. As a whole, quasi-static micro-accelerations on the Progress M1-11 vehicle appeared to be distinctly less than on the Foton satellites [7,8].

5. CONCLUSION

Performed investigation showed that it is worthwhile to continue experiments with the Progress vehicle. Onboard micro-accelerations were sufficiently low and they can be decreased further if to provide motion initial conditions more precisely. This to a great extent relates to the mode of triaxial gravitational orientation. For reliable determination of the spacecraft rotational motion and quasi-static micro-accelerations aboard the spacecraft, experiments with the rotational motion modes should be performed in the period when the Sun is at least $10°$ out of the orbital plane. To solve such tasks in future, it is better to equip the spacecraft with triaxial magnetometer.

6. REFERENCES

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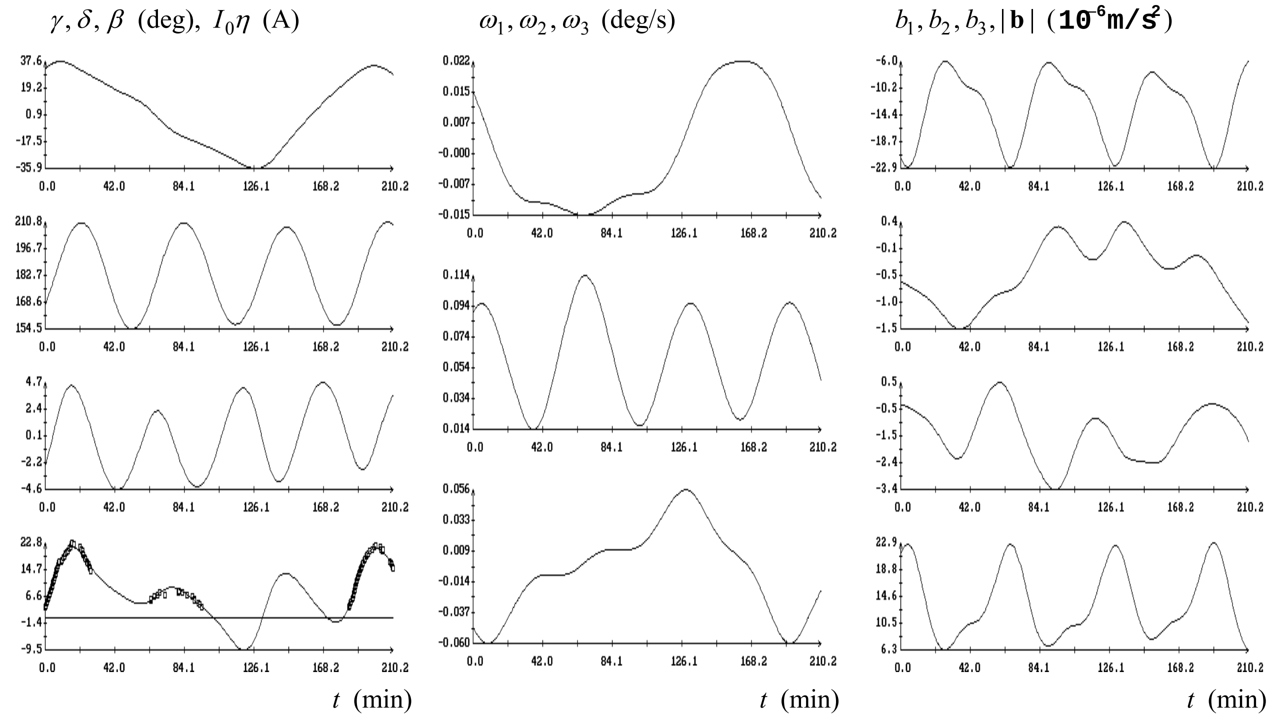

Fig. 1. Triaxial gravitational orientation. Time $t = 0$ corresponds to 10:29:18 MST 28.05.2004, $\varphi = 13.9°$, $N = 107$, $\sigma = 0.62$ A, $\sigma_\gamma = 1.7°$, $\sigma_\delta = 0.56°$, $\sigma_\beta = 0.74°$.

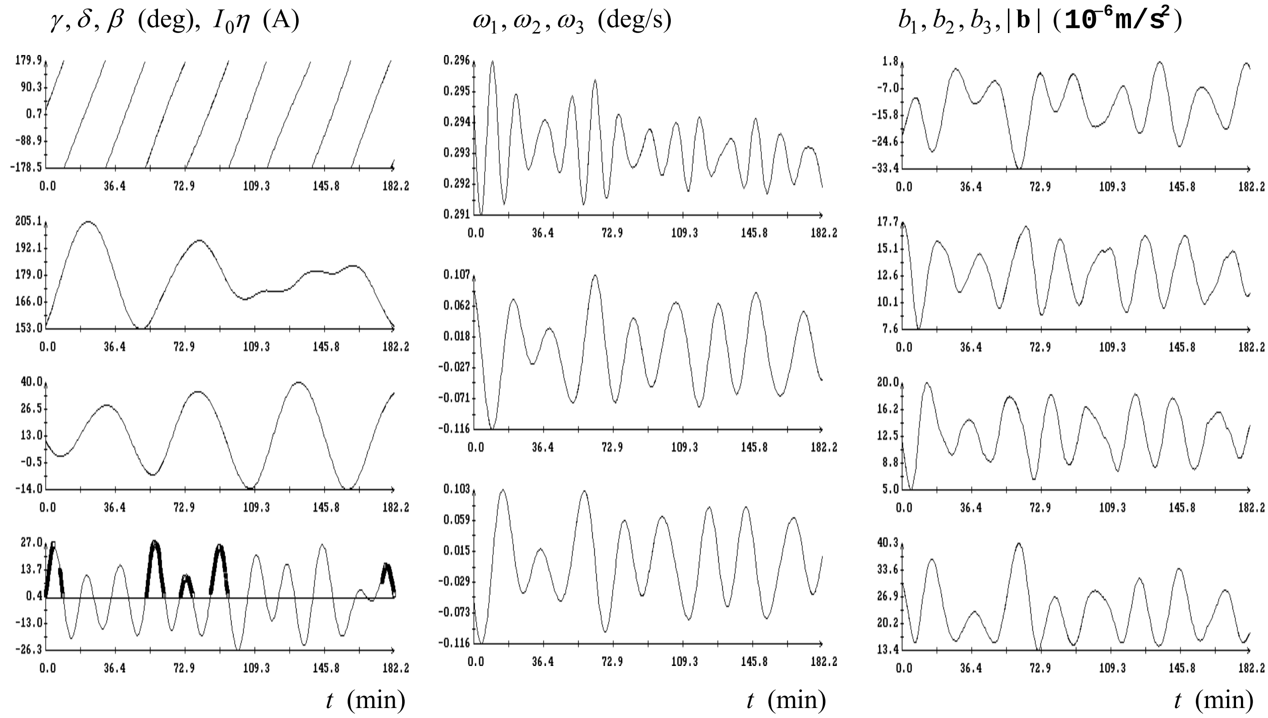

Fig. 2. Gravitational orientation of the rotating vehicle. Time $t = 0$ corresponds to 13:00:53 MST 30.05.2004, $\varphi = 4.6°$, $N = 295$, $\sigma = 1.4$ A, $\sigma_\gamma = 0.92°$, $\sigma_\delta = 1.1°$, $\sigma_\beta = 1.2°$.

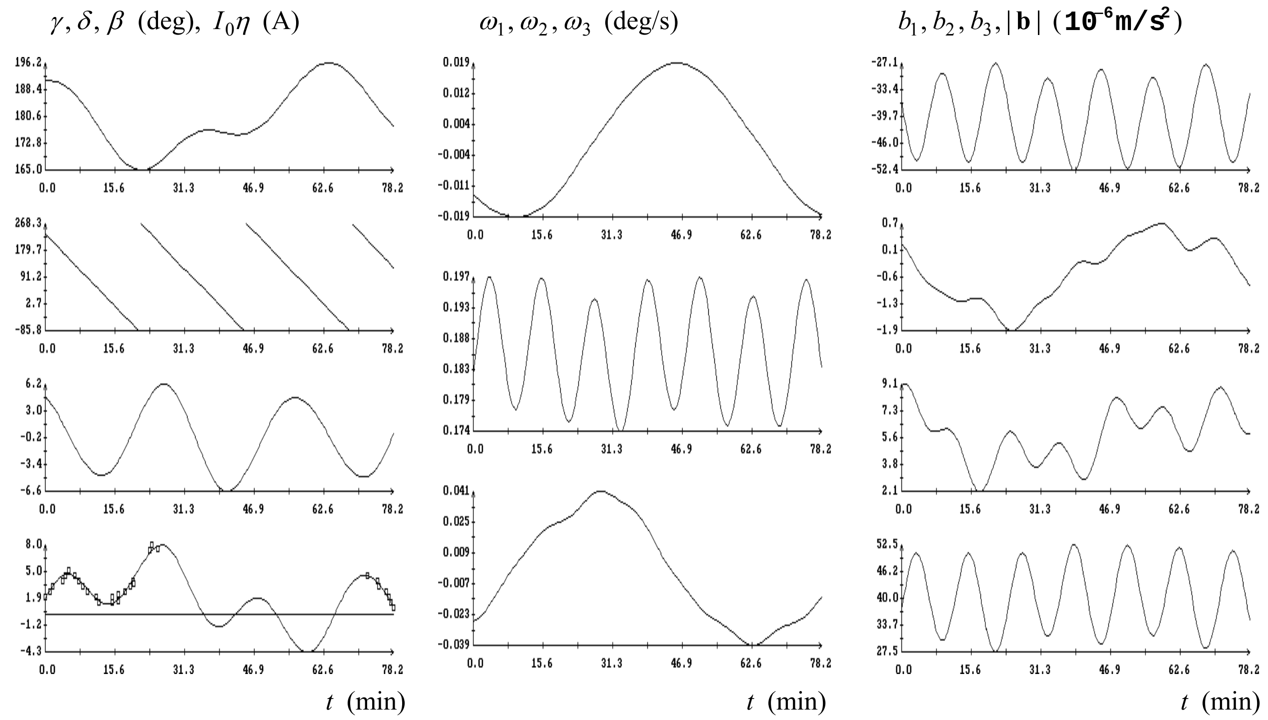

Fig. 3. Rotation in the orbital plane. Time $t = 0$ corresponds to 14:52:38 MST 01.06.2004, $\varphi = -3.7°$, $N = 30$, $\sigma = 0.40$ A, $\sigma_\gamma = 3.2°$, $\sigma_\delta = 9.7°$, $\sigma_\beta = 1.0°$.