UDC 629.7.05+531.551

A.V. Golubek

A PRIORI ANALYSIS OF THE INJECTION ACCURACY OF A LAUNCH VEHICLE INTO EQUATORIAL ORBIT

Abstract: The article is devoted to a priori analysis of the accuracy of the satellite injection to the first ascending node of a 500 km high near-circular equatorial orbit with a small-lift launch vehicle under the influence of stochastic disturbances of a strapdown inertial navigation system, propulsion system characteristics, mass and aerodynamic characteristics.

Keywords: injection accuracy, error box, payload capability, factor by factor analysis, statistical modelling.

Problem statement

One of the most important tasks of preparing a modern small-lift launch vehicle (LV) of a light class with a strapdown inertial navigation system (SINS) for launch is an a priori analysis of the accuracy of the launch. Based on its results, satellite integration is carried out, flight mission assignment for launch is adjusted, and guaranteed fuel reserves are estimated to ensure the successful completion of the mission.

Due to their algorithmic complexity, SINS in LV guidance systems with terminal guidance have not been previously used. Only the emergence of sufficiently powerful calculators, built on the digital principle of information processing, together with the satellite navigation system (SNS) in recent decades has allowed the introduction of SINS into modern LV control systems of foreign production.

At the same time, a number of developments are underway in Ukraine to create both modern LVs with terminal guidance and SINS to them, which require an assessment of the accuracy of the derivation under the influence of stochastic perturbations of various nature [3, 4, 6, 7], as well as development ballistic design for them.

Analysis of recent research and publications

A new approach to the posterior evaluation of the vehicle injection accuracy according to the test results using the cast operator is proposed [1].

The work [2] is devoted to the methodology of a posteriori accuracy evaluation of the Zenit LV, including the determination of the experimental-theoretical characteristics of satellite accuracy.

The work on the possibility of building SINS together with the SNS for commercial LV of the light and middle classes based on sensitive elements made by MEMS technology is devoted to [3, 4]. A structural diagram of the navigation system is developed, a priori evaluation of the accuracy of satellite injection into a circular orbit by various LV are carried out.

A comparative analysis of various methods for a priori evaluation of the accuracy of LV injection [6]. A mathematical model of LV motion with terminal guidance that takes into account the influence of SINS errors on its trajectory has been developed. It is shown that the use of a factor by factor analysis of the injection accuracy for LV with terminal guidance can lead to sufficiently large methodological errors in determining the deflection of the apogee height, perigee height and perigee argument.

The work [7] is devoted to evaluation of the injection accuracy for a modern small-lift launch vehicle from SINS to various near-Earth orbits. The distributions of the kinematic and osculating elements of the satellite’s orbit in the first ascending node of autonomous flight are obtained. The influence of SINS errors on the trajectories error box and deviations of the residues of the fuel components was investigated.

In the article [9], a design of the motion control system architecture for Indian LV SLV-3, ASLV, and PSLV is presented.

The work [12] is devoted to the development of an inertial navigation system integrated with the astronautical system. The configuration of the navigation system is presented, its dynamic and stochastic models used in the filtering algorithm are developed. The feasibility of the proposed solutions was analysed.

The article [13] is devoted to the development and verification of the feasibility of the algorithm for improving the accuracy of the inertial navigation system of the LV by using SNS data using the Kalman filter.

Article [15] is devoted to the estimation of errors in determining the SINS angular velocities using the Kalman filter.

Monograph [17] is devoted to accuracy methodology for LV.

In [22] the solution to the problem of creating a model for assessing the accuracy of tactical missiles and testing its performance was considered.

The article [23] is devoted to the development of a conceptual design of a navigation system for LV with SINS and SNS, which displays small-sized satellites in low and medium Earth orbits. The analysis of possible motion models of LV and navigation measurements is carried out, the key potential difficulties of their creation are identified and the ways to resolve them using object-oriented software are identified.

Based on the analysis of available publications, it should be noted that work is being done to study the accuracy of small-lift launch vehicle with terminal guidance, but design and ballistic support for a priori accuracy evaluation is in the process of development and creation.

Task definition

The satellite injection into the first ascending node of a near circular equatorial orbit 500 km high of small-lift LV under the influence of stochastic perturbations of the propulsion system’s characteristics, mass and aerodynamic characteristics [11], as well as SINS [18].

Necessary:

1. To identify the determining disturbing factors affecting the accuracy of satellite injection in the first ascending node of autonomous flight and the deflection of the mass of residual fuel components at the time of satellite separation from the LV.

2. For two modes of operation of SINS without complexing the SNS (inertial mode) and (inertial-satellite mode) under the action of disturbances:

– determine the distribution of deflections of the osculating orbital elements in the first ascending node of autonomous flight;

– to investigate the change in the LV error box versus flight time;

– determine the distribution of mass deflections of the residual fuel components at the time of satellite separation from the LV.

The main research material

We introduce the following assumptions:

– LV is a solid body of variable mass, the main characteristics of which are given in [11];

– the starting point is given by coordinates 2,29 deg s. lat. and 44,38 deg w. lon.;

– deployment using a transfer orbit;

– separation of the steps and separation of the lead and satellite take place instantaneously;

– the LV thrust vector is controlled by a complex guidance system with functional guidance at the first stage [8, 14, 20] and terminal using the modified Chandler and Smith algorithm [8, 18] - at the second and third;

– the LV stabilization system is ideal;

– the inertial navigation system is strapdown and consists of three accelerometers and three gyroscopes, with the characteristics and errors given in [18];

– the Earth is WGS-84 standard;

– the gravitational potential of the Earth is a decomposition into a series, taking into account the influence of the second, third and fourth zonal harmonics;

– the atmosphere of the Earth is standard GOST 4401-81;

The composition of the disturbing factors.

1. SINS (values are given in [18]):

– accelerometer errors: zero drift run by run, zero drift on run, scale factor error, sensitivity axis setting error;

– gyroscope errors: zero drift run by run, zero drift on run, random drift, scale factor error, sensitivity axis setting error;

– other errors: error of the initial orientation process, quaternion drift, errors of SNS.

2. Errors of engine installation (values are given in [11]):

– deflection of the thrust of the mid-flight and steering engines of the first second and third stages;

– mid-flight and steering engines mass flow rate scatter of the first second and third stages;

– target section of jet pipe scatter of the mid-flight and steering engines of the first and second stages.

3. Errors of aerodynamic characteristics (values are given in [11]):

– deflection of the longitudinal, normal and cross-section aerodynamic force factors;

– deflection of characteristic area.

4. Mass characteristics errors:

– the deflection (3σ) of the dry mass of the first second and third stages is 0,1 %;

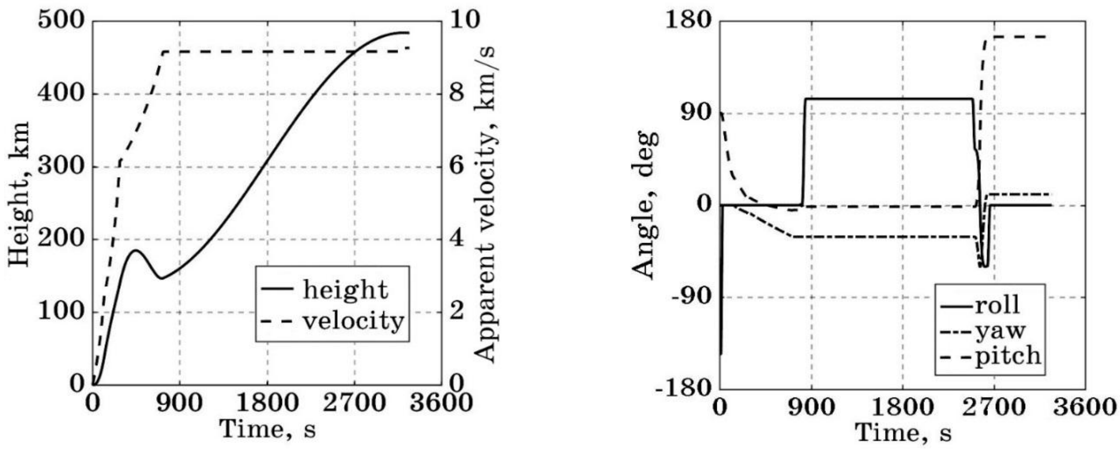
– the deflection (3σ) of the propellant load mass of the first second and third stages is 0,5 %;

– the deflection (3σ) of the payload fairing and satellite mass is 0,05%.

As parameters characterizing the accuracy of the injection, the osculating orbital parameters were adopted: semi-major axis, eccentricity, inclination, right ascension of ascending node (RAAN) and the argument of perigee at the moment the satellite passes the first ascending node of autonomous flight.

For modeling the flight of the LV, the mathematical model given in [6] is used. SINS, a complex SNS, uses a Kalman filter [16, 5, 10, 19, 21].

The nominal trajectory of the LV flight to a near circular equatorial orbit that satisfies the given initial conditions and the introduced assumptions is presented in Fig.1.



a) b)

Figure 1. Nominal satellite injection trajectory

As a result of a factor analysis of the accuracy of satellite injection in the first ascending node of an autonomous flight, the following determining disturbing factors were identified:

– apogee height - zero drift of the gyroscope Z run by run (about 47% of the total error), zero drift of the accelerometer X run by run (about 23%), error of the scale factor of the accelerometer X (about 15%) and the steering engine trust scatter of the third stage (about 8%);

– perigee height - zero drift of the gyroscope Z run by run (about 46%), zero drift of the accelerometer X run by run (about 11%) and the steering engine trust scatter of the third stage (порядка 7%);

– inclination - zero drift of the gyroscope Z run by run (about 50%), roll quaternion drift (about 14%), random drift of the gyroscope Z (about 8%) and random drift of the gyroscope X (about 7%);

– RAAN - zero drift of the gyroscope Z run by run (about 65%) and roll quaternion drift (about 10%);

– the argument of perigee is the zero drift of the gyroscope Z run by run (about 50%), the steering engine trust scatter of the third stage (about 23%), and the error of the scale coefficient of the accelerometer X (about 7%).

With regard to the mass deflection of residual fuel components, the determining factors are: mid-flight engine trust scatter of the third stage (about 47%), mid-flight engine trust scatter of the second stage (about 17%), the error of mid-flight engine mass flow rate of the third stage (about 11%), the error thrust of mid-flight engine of the first stage (about 9%) and the error of mid-flight engine mass flow rate of the third stage (about 7%).

From the results obtained it follows:

– the determining errors affecting the accuracy of satellite injection are the zero drift of the gyroscope Z run by run, the zero drift of the accelerometer X run by run, the error of the scale coefficient of the accelerometer X, the roll quaternion drift, the random drift of the gyroscopes X and Z and the steering engine trust scatter of the third stage;

– the determining errors affecting the mass deflection of residual fuel components at the time of separation of the satellite from the LV are the deviation of the thrust of the mid-flight engines of the first, second and third stages and the mass flow rate scatter of the second and third stages;

– the remaining errors for the orbit under consideration can be ignored in view of their smallness.

For the considered trajectory and errors, an a priori analysis of accuracy using statistical modeling is performed. As a result, for two modes of operation of the SINS inertial and inertial-satellite, the distribution of deflections of the osculating parameters of the satellite orbit in the first ascending node of autonomous flight is obtained, shown in Fig. 2-6 (a - inertial mode, b - inertial-satellite mode) and in table. 1 and 2. Designations of deviations of the table. 1 и 2: □ha – apogee height, Δhp – perigee height, Δi – inclination, □□ – RAAN; □□ – argument of perigee. Abbreviations tab. 1 and 2: MV – mean value; SD – standard deviation.

Table 1.

Parameters of distribution of deflections of the osculating parameters of motion in inertial mode

	Δ ha, km	Δ hp, km	Δ i, deg.	$\Delta\Omega$, deg.	$\Delta\omega$, deg.
MV	-0,154	0,692	-0,001	2,924	-6,598
SD	1,830	1,790	0,043	14,146	10,374
Excess	26,890	25,142	-0,365	66,895	14,964
Asymmetry	-4,310	4,129	-0,115	7,695	-0,525
Minimum (3σ)	-14,782	-2,173	-0,131	-2,762	-78,049
Maximum (3σ)	2,965	15,333	0,092	160,836	86,203

Table 2.

Parameters of distribution of deflections of the osculating parameters of motion in inertial-satellite mode

	Δ ha, km	Δ hp, km	Δ i, deg.	$\Delta\Omega$, deg.	$\Delta\omega$, deg.
MV	0,511	0,571	0,000	-0,018	-5,798
SD	0,474	0,510	0,001	0,538	5,315
Excess	0,165	0,894	0,188	0,422	0,772
Asymmetry	0,688	0,973	-0,239	0,324	-1,153
Minimum (3σ)	-0,554	-0,386	-0,002	-1,610	-27,364
Maximum (3σ)	2,160	2,773	0,002	2,216	1,909

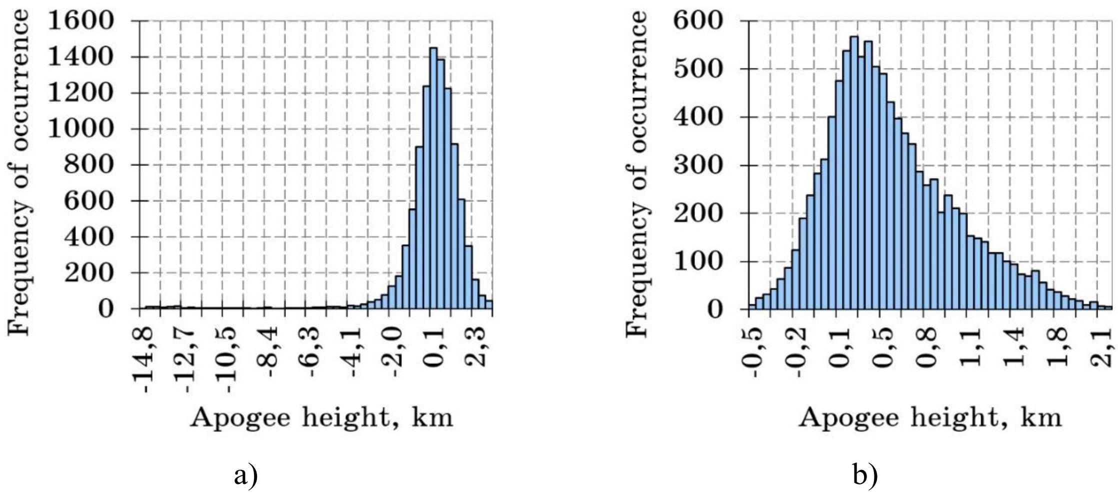
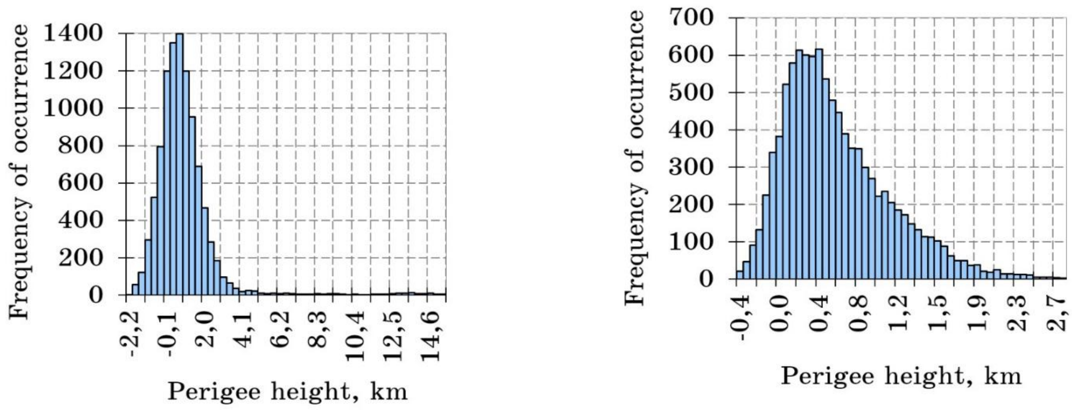
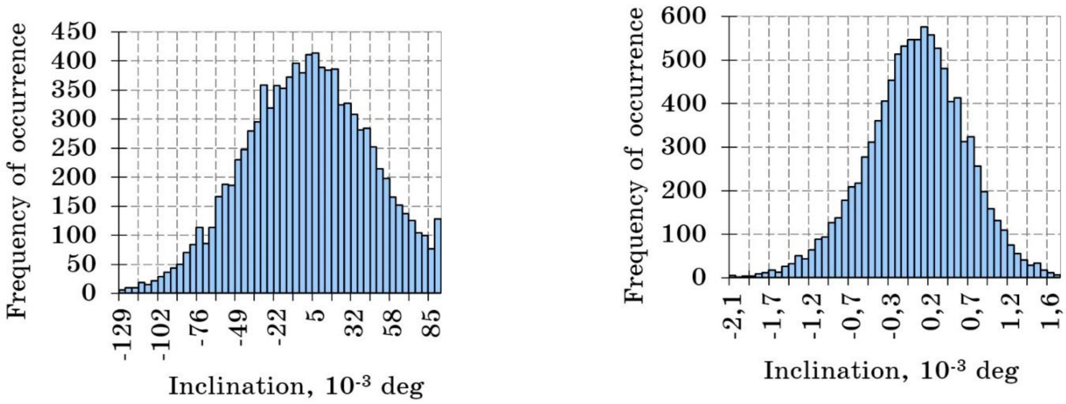


Figure 2. The histogram of deviation distribution satellite perigee height



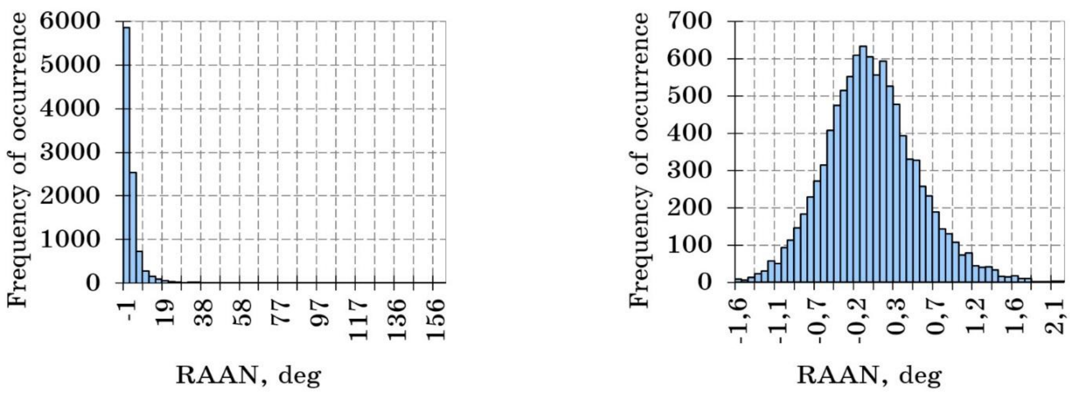
a) b)

Figure 3. The histogram of the distribution of the satellite apogee height deflection



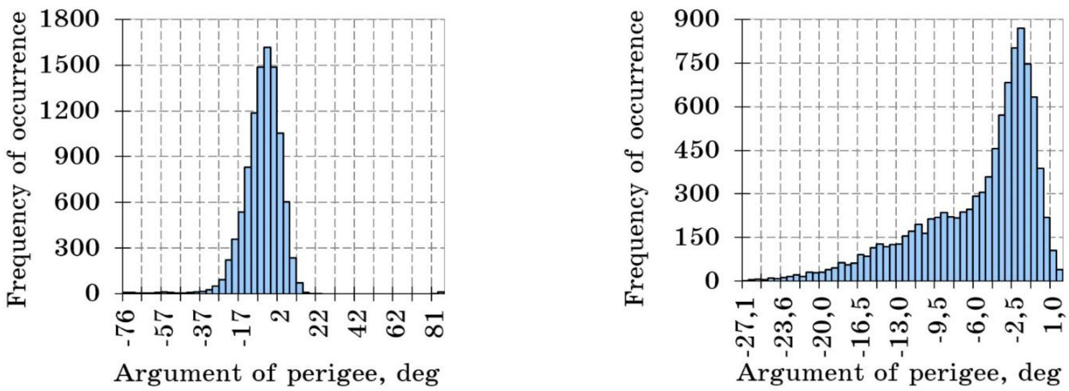
a) b)

Figure 4. The histogram of the satellite inclination deflection



a) b)

Figure 5. The histogram of the distribution of the RAAN deflection



a) b)

Figure 6. The histogram of the distribution of the argument of perigee deflection

From fig. 2-6, it follows that the distribution of deviations of the osculating orbital parameters in the first ascending node of the autonomous flight of the satellite has a complex form. The inclination and RAAN in the inertial-satellite mode (Fig. 4, b and 5, b) are close to normal, but in accordance with Pearson's criterion χ^2, they are not. Using the SNS significantly reduces the range of distributions of the osculating orbit parameters by more than five times (for the inclination and RAAN by almost two orders of magnitude). This suggests a significant increase in the accuracy of satellite injection into a given orbit.

The correlation analysis of deviations in the inertial mode (Table 3) showed that deflections of the apogee height and RAAN have a high direct correlation (correlation of more than 0.75). There is also a high inverse relationship between the height of the perigee and RAAN. In the inertial-satellite mode (Table 4), there is a high correlation between the apogee height and the perigee height. In addition, they are inversely related to the perigee argument. There is a high inverse relationship between the inclination and longitude of the ascending node.

Table 3.

Correlation analysis of deflections of the osculating parameters of the orbit in the inertial mode

	Δ ha	**Δ hp**	**Δ i**	**ΔΩ**	**Δ ω**
Δ ha	1,00				
Δ hp	-0,65	1,00			
Δ i	-0,47	0,42	1,00		
ΔΩ	-0,88	0,86	0,44	1,00	
Δ ω	0,29	-0,23	-0,29	-0,25	1,00

Table 4.

Correlation analysis of deflections of the osculating parameters of the orbit in the inertial-satellite mode

	Δ ha	**Δ hp**	**Δ i**	$\Delta\Omega$	$\Delta\omega$
Δ ha	1,00				
Δ hp	0,81	1,00			
Δ i	0,11	0,14	1,00		
$\Delta\Omega$	-0,17	-0,11	-0,79	1,00	
$\Delta\omega$	-0,89	-0,94	-0,06	0,05	1,00

From the data of the tables 3 and 4 it follows that the use of the SNS leads not only to a change in the distribution of deflections of the osculating orbital parameters, but also to the degree of dependence between them.

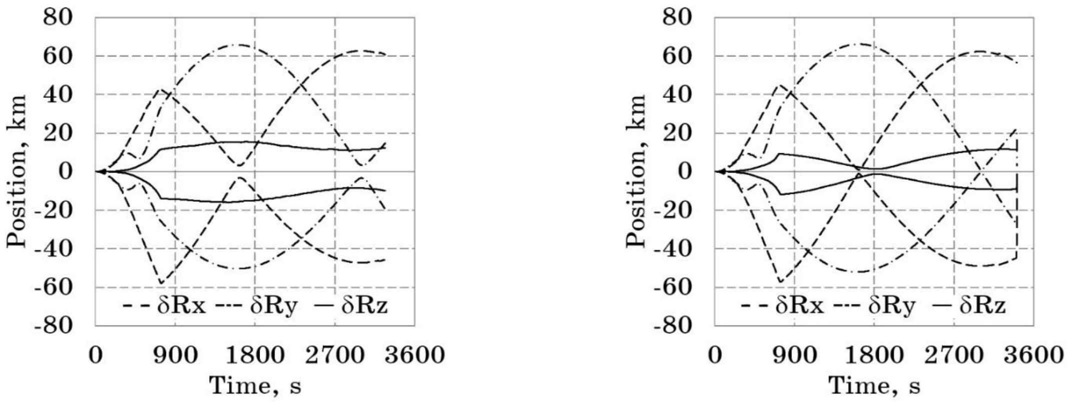
Consider the effect of LV error box for various SINS modes (Fig. 7-10, a - inertial mode, b – inertial-satellite mode). Designations fig. 7-10: δRx, δRy и δRz – projections of the vector of maximum deflections of the current position of the LV center of mass on the axis of the initial starting geocentric coordinate system; δVx, δVy и δVz – projections of the vector of maximum deflections of the absolute velocity of the LV center of mass on the axis of the initial starting geocentric coordinate system; δφ, δψ, δϑ – maximum deflections of the angular orientation in the channels of roll, yaw and pitch.

From Fig. 7-10 follows. The error box does not exceed ±65 km on the projections of the vector of the current position of the center of mass on the axis of the initial starting geocentric coordinate system, ±160 m/s on the projections of the vector of the absolute velocity of the center of mass on the axis of the initial starting geocentric coordinate system, ±180 deg in the roll and pitch channels and ±90 degrees in the yaw channel. The time dependence of the error box is periodic with clearly defined parts of the propulsion system (up to 900 s and about 3200 s), where the tube either increases monotonously or sharply, and angular reorientation also occurs. The points of minima and maxima in the flight section above 900 s coincide with the passage of the ascending node about 3000 s of the flight and the latitude argument 270 deg about 1600 s of the flight. For angular orientation, the maxima points are observed in the reorientation areas (about 900 s and 2600 s of the flight).

In addition, it should be noted that the error boxes for the inertial and inertial-satellite modes vary slightly. It follows that the determining factors affecting the dimensions of the error box are the deflections of the LV parameters. The influence of SINS errors can be neglected.

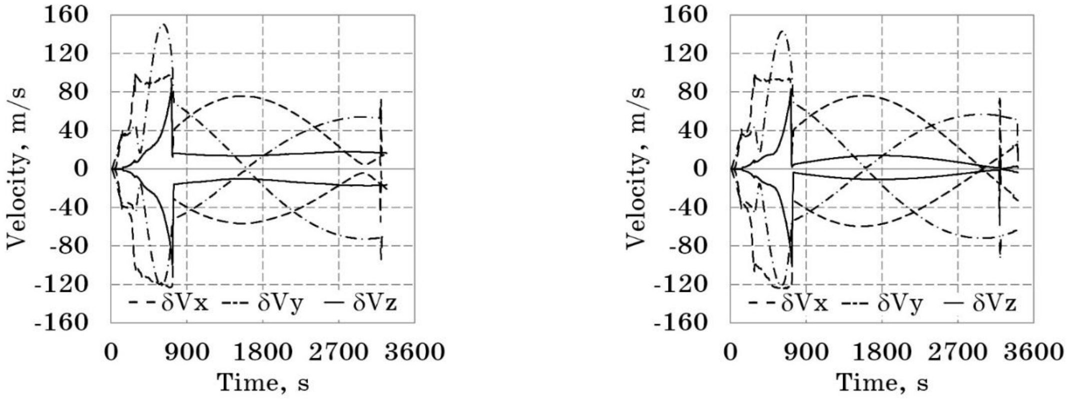
Consider the distribution of the mass deflection of residual fuel components at the time of separation of the satellite from the LV for various SINS operation modes (Fig. 10 and Tab. 5).

 ISSN 1560-8956



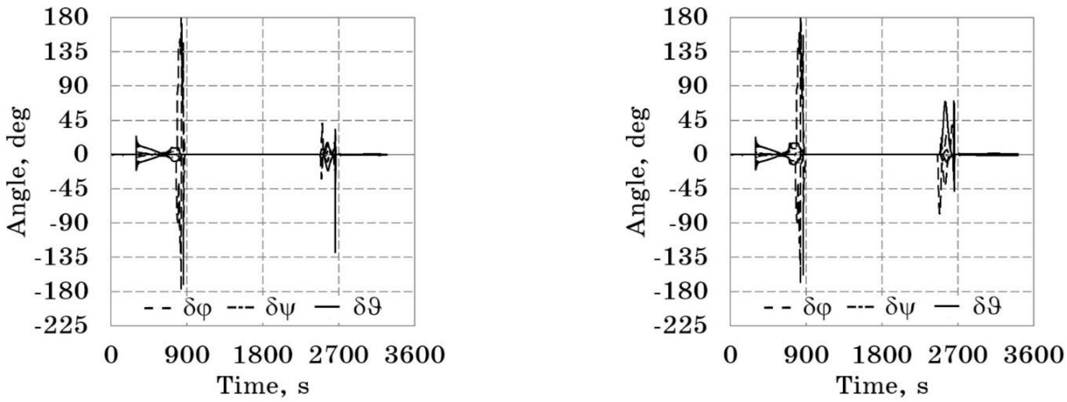
a) b)

Figure. 7. The error box of the current position of the LV center of mass



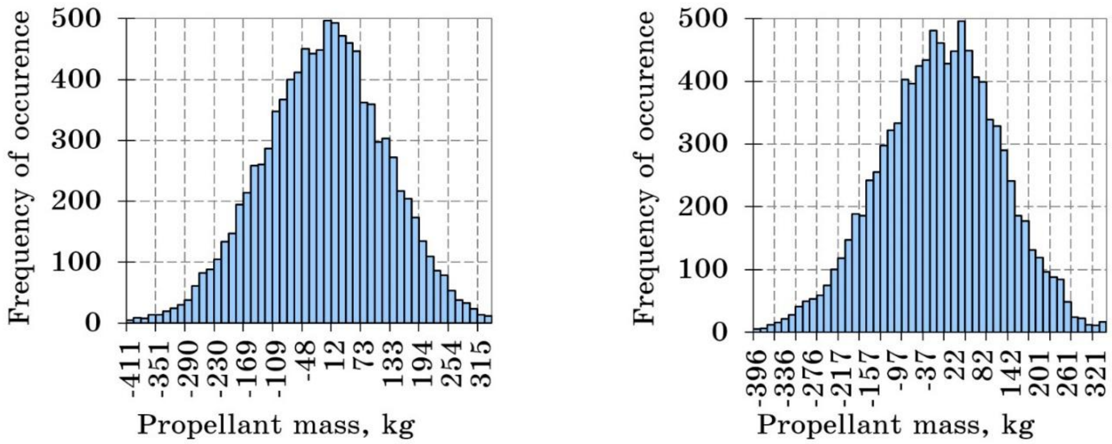
a) b)

Figure. 8. The error box of the absolute velocity of the LV center of mass



a) b)

Figure 9. The error box of the LV angular orientation



a) b)

Figure 10. The histogram of the distribution
of the mass deflection of residual fuel components of the LV

The obtained histograms of the distributions for the two SINS operation modes are similar to each other and close to normal, but according to Pearson's criterion χ^2, they are not. The similarity of two histograms is explained by the results of a factor by factor analysis, where it was determined that the influence of SINS errors on the mass deflection of residual fuel components due to their smallness in comparison with deflections of the LV parameters can be neglected.

Table 5.

Distribution parameters of the mass deflection of residual fuel components at the time of separation of the satellite from the LV

Parameter	SINS operating mode	
	Inert.	Inert.-satel.
MV, kg	-9,776	-8,084
SD, kg	125,957	125,346
Excess	-0,164	-0,178
Asymmetry	-0,110	-0,121
Minimum (3σ), kg	-418,524	-403,087
Maximum (3σ), kg	337,204	343,138

Conclusions

An a priori study of the injection accuracy of small-lift LV into a near circular equatorial orbit 500 km high was carried out and the following results were obtained:

– the composition of the determining disturbing factors was revealed that affect the accuracy of satellite injection into the first ascending node of the autonomous flight orbit and the mass deflection of residual fuel components at the time of satellite separation from the LV;

– multivariate distributions of the osculating parameters of the satellite orbit in the first ascending node of autonomous flight are obtained for two SINS operation modes (without SNS and with);

– it is shown that the use of SNS significantly reduces the range of distributions of osculating orbit parameters by more than five times;

– dependences of the LV error box on the flight time and two SINS operation modes are obtained and the features of their change are revealed;

– mass distributions of residual fuel components at the time of satellite separation from the LV for two modes of SINS operation were obtained;

– it is shown that the determinants of disturbance that influence the accuracy of satellite injection are SINS errors, and for the error box and mass deflections of residual fuel components - deflections of the LV parameters.

REFERENCES

1. Arsenyev V.N., Labetskiy P.N. Otsenivaniye harakteristik tochnosty systemy upravleniya rakety-nositelya po rezultatam puskov v razlichnyh usloviyah. Izvestiya VUZov. Priborostroyeniye. 2015. Vol. 58, N 1. P. 27-32. (in Russian).

2. Briker V.V., Litvinov V.S., Negoda A.A., Novikov A.V. Metodologiya i rezultaty analiza tochnosti vyvedeniya kosmicheskih apparatov raketoy-nositelem «Zenit» / Kosmichna nauka I tehnologiya. – 1996. Vol. 2, N 3-4. – P. 66-69. (in Russian).

3. Chucha Y., Tikhonov V., Degtyareva E., Sidorenko S., Matviyenko E., Lapko A., Laylyuk S. Performance evaluation of gins built by mems technology, without and with the use of integration with SNS UE // The 7-th International conference “Space technologies: present and future”. 21-24 may 2019, Dnipro, Ukraine. P. 132.

4. Degtyarev A.V., Degtyarev M.A., Davydenko S.A., Makarov A.L., Snegirev M.G., Sirenko V.N., Tikhonov V.L., Shekhovtsov V.S. On the feasible development of gimballess control systems for launch vehicles using GPS satellite navigation equipment / Space Science and Technology. 2015. Vol. 21. N 6, P. 3-12. (in Russian).

5. Farrell J.A. Aided navigation GPS with High Rate Sensors. – The McGraw-Hill Companies, 2008. – 530 p.

6. Golubek A.V. Comparison of procedures for assessment of effect of controller complex errors on injection accuracy of launch vehicle with iterative guidance / Aerospace Engineering and Technology. – 2015, N 2(119). – P. 45-51. (in Russian).

7. Golubek A., Filippenko I., Tetarevskii K. Analysis of influence of flight director package errors on small-lift launch vehicle trajectory // The 6-th International conference “Space technologies: present and future”. 23-26 may 2017, Dnipro, Ukraine. P. 117.

8. Golubek A.V., Novikov A.V., Sheptun A.D. Osnovy navigatsii i navedeniya raket-nositeley: uchebnik / Pod red. akad. NANU Degtyaryova A.V. – Dnepr: Lira, 2019. – 236 p. (in Russian).

9. Gupta S.C., Suresh B.N. Development of navigation guidance and control technology for Indian launch vehicles. Sadhana, Vol. 12, Part 3, March 1988, pp. 235-249.

10. Hofmann-Wellenhof B., Lichtenegger H, Wasle E. GNSS – Global Navigation Satellite Systems GPS, GLONASS, Galileo, and more. – Springer-Verlag Wien, 2008. – 516 p.

11. Igdalov I.M., Kuchma L.D., Poliakov N.V., Sheptun Yu.D. Dynamic problems of rockets and spaces stages: monograph. Under the editorships of Academician S.N. Konyukhov. – Dniepropetrovsk: Lira, 2013. – 280 p.

12. Jamshaid Ali J., Jiancheng F. Realization of an autonomous integrated suite of strapdown astro-inertial navigation systems using unscented particle filtering / Computers and mathematics with applications. – 2009, Vol. 57. – P. 169-183.

13. Liu J., Zhu Y., Lai J., Yu Y. Inner attitude integration algorithm based on fault detection for strapdown inertial attitude and heading reference system / Chinese journal of aeronautics. – 2010, Vol. 23. – P. 68-74.

14. Lysenko L.N. Navedeniye i navigatsiya ballisticheskih raket: ucheb. posobiye. – M.: Izd-vo MGTU im. N.E. Baumana, 2007. – 672 p. (in Russian).

15. Manoylenko A.A., Tikhonov V.L. Raschot pogreshnostey otsenivaniya uglovyh skorostey rakety-nositelya / Visnyk Dnpipropetrovskogo universitetu. Seriya: Raketno-kosmichna tehnika. – 2012, Vol. 20, Iss. 15(1). – P. 93-99. (in Russian).

16. Matveev V.V., Raspopov V.Ya. Osnovy postroyeniya besplatformennyh inertsialnyh navigatsionnyh sistem. Pod obshch. red. d.t.n. V.Ya. Raspopova. – SPb.: GNTS RF OAO «Kontsern «TSNII «Elektropribor», 2009. – 280 p. (in Russian).

17. Novykov O., Tikhonov V., Litvinov V. Methods of analysis for launch vehicle injection accuracy: monograph. – Vilnius: VGTU Press Technika, 2015. – 256 p.

18. Proyektirovaniye sistem upravleniya obyectov raketno-Kosmicheskoi tehniki. T. 1. Proyektirovaniye sistem upravleniya raket-nositeley: uchebnik / Yu.S. Alekseev, Yu.Ye. Balabey, T.A. Baryshnikova i dr.; Pod obshch. red. Yu.S. Alekseeva, Yu.M. Zlatkina, V.S. Krivtsova, A.S. Kulika, V.I. Chumachenko. – Kh.: NAU «HAI», NPP «Khartron-ARKOS», 2012. – 578 p. (in Russian).

19. Rogers R.M. Applied mathematics in integrated navigation systems. – 2nd ed. – Reston: American institute of aeronautics and astronautics, 2003. – 330 p.

20. Sikharulidze Iu.G. Ballistika i navedenie letatel'nykh apparatov. Moscow: Binom. Laboratoriia znanii, 2011. 407 p. (in Russian).

21. Titterton D.H., Weston J.L. Strapdown inertial navigation technology. – 2-nd ed. – London: The Institution of electrical engineers, 2004. – 581 p.

22. Zhang L.-J., Xie X.-F., Li D.-D., Wang Y. Error analysis of strapdown inertia navigation system in tactical missiles / Advanced in control engineering and information science. – 2011, Vol. 15. – P. 1456-1460.

23. Zosimovych N. Modeling the integrated guidance system of a commercial launch vehicle international refereed / Journal of engineering and science (IRJES). – 2014, Vol. 3, Issue 6. – P. 39-54.