ISSN 0038-0946, Solar System Research, 2021, Vol. 55, No. 7, pp. 634–641. © Pleiades Publishing, Inc., 2021.
Russian Text © The Author(s), 2020, published in Vestnik NPO imeni S.A. Lavochkina, 2020, No. 2, pp. 13–21.

Active Braking for Soft Landing on the Surface of Mars: Part 1: Braking Conditions Analysis and Sequence of Operations

A. E. Shirshakov[a], *, I. A. Grudev[a], **, V. N. Likhachev[a], *, and P. E. Rozin[a], ******

[a] *Lavochkin Association, Khimki, Moscow oblast, Russia*
**e-mail: shirshakov@laspace*
***e-mail: grudevia@laspace.ru*
****e-mail: likhachev@laspace.ru*
*****e-mail: rozinpe@rambler.ru*

Received July 10, 2020; revised July 10, 2020; accepted July 10, 2020

Abstract—Performing a soft landing of an unmanned spacecraft on Mars' surface requires implementation of several technically challenging flight phases. The final one is an active braking with the use of a steerable thrust jet engine. In this article we present an analysis of the flight environment before and during of active braking, specify the composition of technical aids for motion control, overview one of the possible braking profiles, sequence of the active braking modes, algorithms of guidance and control, and mathematical simulation results.

Keywords: gravitational acceleration, inertial navigation aids, attitude, guidance, control, thrust, braking engine

DOI: 10.1134/S0038094621070170

INTRODUCTION

The exploration of planets, their satellites and others celestial bodies of the solar system leads to the issue of delivering scientific equipment and mobile devices to their surface. As a rule, for a spacecraft (SC) performing a delivery mission, at the moment it reaches the surface of the explored celestial body, a rather low velocity of collision (contact) with the surface is required and, in addition, a certain orientation of the longitudinal axis of the spacecraft relative to the normal to the underlying surface of the relief (Artem'ev et al., 2019).

The relative closing speed of the spacecraft with the celestial body under study, as a rule, is 3–6 km/s, therefore, reducing this speed to the permissible contact velocity is a complex technical problem. For each celestial body, this problem is solved individually. The delivery method depends on the presence or absence of an atmosphere, the magnitude of the gravitational acceleration on its surface, the characteristics of the relative orbital motion, as well as the delivered mass. In the absence of an atmosphere (Moon, asteroids), a decrease in the relative speed can be achieved only through the use of jet braking engines, which is associated with significant fuel consumption (Likhachev et al., 2012; Zhukov et al., 2012). The presence of a dense atmosphere (Venus, Earth) makes it possible to reduce the relative velocity by aerodynamic braking aids by converting the kinetic energy of the spacecraft into thermal energy; this requires the use of a certain aerodynamic shape of the spacecraft body, its thermal protection and parachute system.

Mars has a rather rarefied atmosphere, which, however, allows the use of a combination of aerodynamic braking aids: a heat-protecting aerodynamic screen and a parachute system. But even at the expense of a large parachute area, a decrease in the relative speed to acceptable values cannot be ensured by the time the spacecraft contacts the surface. The technically realizable speed of the spacecraft relative to the surface during the descent by parachute can be reduced to 45–65 m/s. Therefore, to reduce the speed to acceptable limits on the final segment of the spacecraft descent, it is necessary to use jet braking engines.

In this article, we analyze the flight conditions before the parachute separation, substantiate the sequence of operations for preparing the start of active braking, consider the composition of the technical aids of controlled braking, and propose the logic and algorithms for guidance and control during the operation of the braking engine, which ensure a reliable decrease in its speed at the moment of contact with the surface.

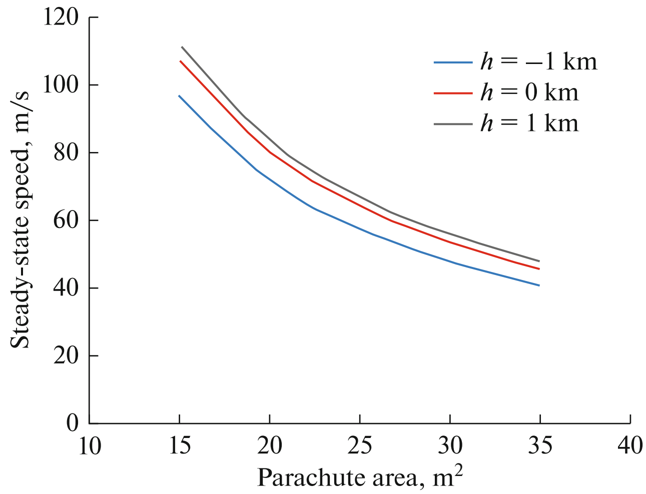

Fig. 1. Spacecraft descent speed by parachute.

1. ANALYSIS OF FLIGHT CONDITIONS

When operating the braking engine (BE), the formation of the descent profile is significantly influenced by the flight conditions preceding its activation.

As a rule, to reduce the speed of the spacecraft entry into the atmosphere, a simple method of ballistic aerodynamic braking of a closed capsule is used, inside which a lander is located with scientific instruments and other devices that will operate on the surface of Mars. The capsule consists of a frontal aerodynamic screen that with the front hemisphere protects the lander from the effects of high temperatures and pressure of the incoming flow, and a protective casing that protects it from the rear hemisphere. At the same time, the capsule has its own static aerodynamic stability and sufficient thermal protection to pass through the plasma region of the descent trajectory. In the segment of the aerodynamic braking, the speed of the capsule decreases from the speed of entry, reaching 5600 m/s, to the values acceptable for the operation of the parachute system.

The steady-state speed at which the parachute system can provide descent upon reaching the surface of Mars can be estimated based on the ratio of the equality of the parachute braking force and the attractive gravitational force of Mars

$$C_p S \frac{\rho V^2}{2} = m g_m,$$

where C_p is the parachute drag coefficient; S is the parachute area; ρ is the density of atmosphere; V is the platform descent velocity; $g_m = 3.723$ m/s^2 is the free fall acceleration on the surface of Mars; m is the landing platform mass including the parachute system mass. Correspondingly, the steady-state descent velocity before engaging the braking engine will be

$$V = \sqrt{\frac{2gm}{\rho C_p S}}. \quad (1)$$

Figure 1 shows the dependence of the steady-state speed on the diameter of the parachute used for the lander mass with a parachute system of the order of 1800 kg and a flight altitude relative to the average level of the Martian surface of 0 ± 1 km.

It follows from the Fig. 1 that when using a parachute with a diameter of 15 m, the descent speed is about 100 m/s and more. To reduce the steady descent speed, it is preferable to have a parachute diameter of 25–35 m, providing a steady descent speed within 65–45 m/s, respectively.

Aerodynamic braking in the rarefied atmosphere of Mars and a parachute descent to a steady descent speed is associated with a deficit in the altitude. This leads to the fact that by the time the braking engines are turned on, the steady descent speed by parachute is not always achieved. In this regard, for the payload delivery, it is recommended to choose landing sites where the surface of Mars is below the average level by 1.5–2.5 km, and where, accordingly, the density of the atmosphere increases by 20% relative to its values on the average Martian surface.

Descent by parachute has a number of features.

First, there is the effect of the wind on the movement of the lander. The parachute is decelerated relative to the air flow, and in the presence of a horizontal wind component, it is drawn into horizontal movement at the speed of the wind along with the oncoming flow. According to the Martian atmosphere model, the wind speed in the near-surface layers can reach 25 m/s. That is, along with the steady-state vertical speed of descent, the lander can have a horizontal speed of the above magnitude. In this case, at the time of the parachute separation, the longitudinal axis of the lander is nearly parallel to the gravitational acceleration, and the velocity vector deviates from this direction by 30°–40°.

Second, a parachute with a landing platform on slings represents a pendulum system, where the point of suspension is the focus of the parachute – the point of application of the braking force, the length of the lines is the length of the pendulum suspension, and the mass of the lander is the mass of the pendulum. The period of pendulum oscillations is determined by the known relation

$$T = 2\pi\sqrt{\frac{\ell}{g_m}},$$

where ℓ is the distance from the focus of the parachute to the center of mass of the lander. For the above range of parachute diameters, $\ell = 40$–50 m, one obtains $T = 20$–23 s. The amplitude of the pendulum oscillations

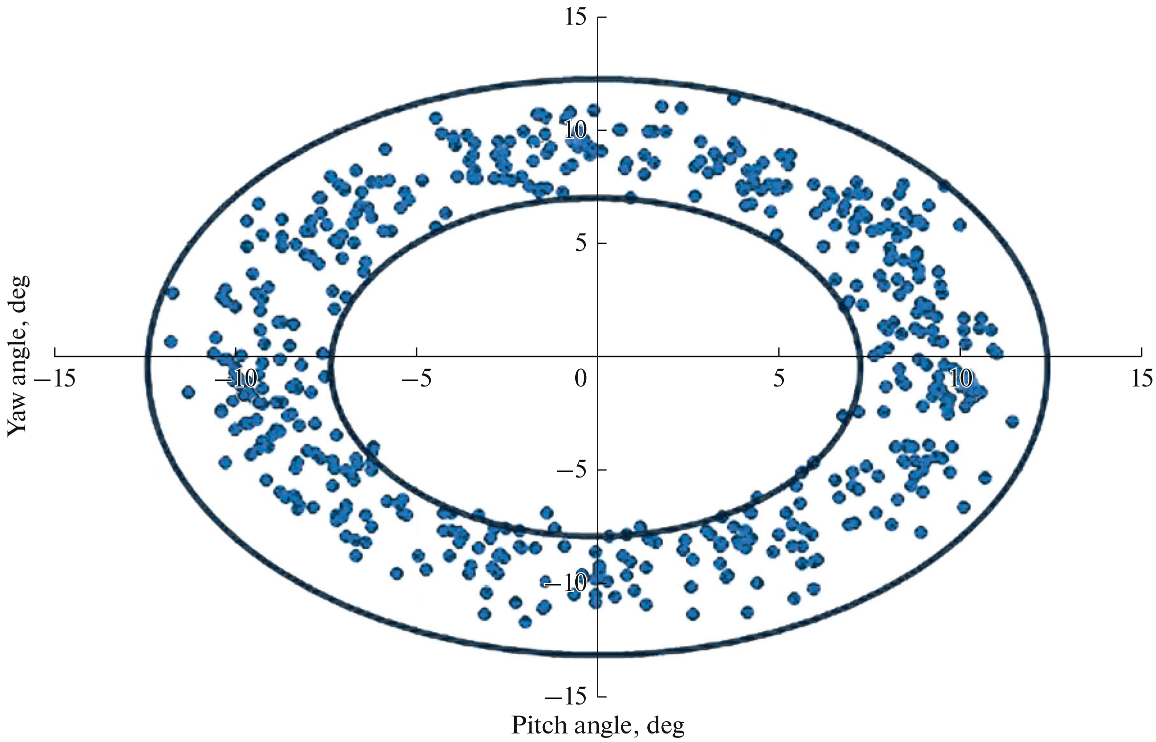

Fig. 2. The range of orientation angles of the longitudinal axis of the lander at the moment of separation of the parachute system with a protective casing.

depends on the disturbances during the opening of the parachute. Shown in Fig. 2 is a typical range of the deviation angles of the longitudinal axis of the lander from the vector of "air velocity" at the moment of parachute separation.

It follows from the figure that the lander under the parachute makes conical movements with an amplitude from 7 to 12 deg. Due to these vibrations, an additional harmonic component of the angular velocity arises with an amplitude of about 3.3°/s and the horizontal velocity with an amplitude of up to 2.6 m/s with the same period.

Thus, at the moment of the braking engine turns on, the longitudinal axis of the lander (braking engine thrust vector) can deviate from the velocity vector by an angle of up to 45°–50°, which will require a certain duration of lander reorientation in the process of active braking.

To turn on the braking engine and further active control of the lander motion, information about the flight altitude and the velocity vector of the lander movement relative to the surface of Mars is required. These parameters can be monitored by an inertial system based on the information from gyroscopes and accelerometers and the initial state vector when entering the atmosphere. However, by the time the braking engine is turned on, their values contain significant errors that reach kilometers in height and tens of meters per second in speed. With such errors, the landing of a lander with the required parameters is almost impossible. Therefore, along with the instruments of inertial navigation onboard the lander, it is necessary to use a radar meter based on the Doppler effect—a Doppler velocity and distance meter (DVDM). A DVDM forms four narrowly directed radar beams, which are used to estimate the Doppler shifts of the emitted and received radar signal. Based on these shifts and delays of the wavefronts of the received signals, the DVDM extracts information about the projections of the velocity and distance to the surface in the direction of each beam. The specified information is converted by an onboard computer using parameters of the current orientation into vertical and horizontal projections of the velocity relative to the surface, as well as into the altitude of the spacecraft in flight. The DVDM should start operating even during parachute descent and supply this information to the control system to determine the moment of braking engine start and separation of the parachute system.

To save mass during the flight and when entering the atmosphere of Mars, the spacecraft can be stabilized by rotation with a certain angular velocity. By the time the parachute system is put into operation, the angular rotation speed can vary over a wide range of 10°–25°/s due to the aerodynamic asymmetry of the aerodynamic screen. After deployment, the parachute stops rotating very quickly due to the damping of the large area of the canopy, while in the absence of its own damping, the lander continues to rotate around the longitudinal axis. This rotation makes the DVDM operation more difficult. Therefore, after separation of the aerodynamic screen, it is necessary to use stabilization engines to reduce the angular velocity of the lander to small values, of the order of 0.5°/s, which enables the DVDM to start measuring the height and projections of the spacecraft velocity relative to the surface of Mars.

It is known (Likhachev et al., 2013) that controlled braking with the use of jet engines requires knowledge of the direction of gravitational acceleration on the planet's surface. The construction of the landing coordinate system is planned before the separation of the landing capsule from the flight module. By the time of separation, the flight module control system provides information on the direction to the center of gravity of Mars. The descent of the spacecraft in the extra-atmospheric trajectory segment with a duration of about 30 min and in the segment of aerodynamic descent with a duration of up to 6 min is performed with the spacecraft rotation. For the time indicated, the angle of proper rotation can be about 40000°. Calculation of the current spacecraft orientation in the landing coordinate system is performed on these segments using the information of strapdown (cardan-less) gyroscopes. However, due to the error of the scale factor of the gyroscope, even reduced due to calibrations to a value of 80 ppt by the time of separation, and the error in knowing the initial orientation relative to the gravitation center and the surface of Mars, the error in knowing the angle of proper rotation in the landing coordinate system will be on the order of 2.2°–3.6°. Since the longitudinal axis of the spacecraft is in an almost horizontal position throughout the entire length of aerodynamic deceleration, the error in knowing the orientation of the plane containing the gravitational acceleration of Mars accumulates by almost the same amount. On the one hand, this error completely turns into the error of knowing the orientation of the associated axes of the landing platform relative to the true direction of the gravitational vertical, and on the other hand, it gives a systematic component of the error in estimating the magnitude of the gravitational acceleration projection of the order of 0.16–0.2 m/s^2.

In addition, at an altitude of less than 5–15 m, the DVDM loses its operability, and it is no longer possible to use its information on the height and projections of the velocity. This information can only be obtained based on the inertial navigation system. With the descent duration from this height to contact with the surface of the order of 5–7 s, the inertial computing system can accumulate an error in the value of the navigation horizontal speed from 1 to 1.5 m/s due to the error in determining the direction of gravitational acceleration.

To improve the accuracy of determining the direction of the gravitational vertical before the start of controlled deceleration, one can use the method for specifying the gravitational acceleration vector described in (Likhachev and Fedotov, 2016). In this case, a computational procedure is implemented to clarify the direction of gravitational acceleration according to the information of the DVDM and the inertial system of computation. According to this procedure and the information from the DVDM, at time t, the velocity vector $\tilde{V}1(t)$ of the lander motion is fixed, while after the time interval Δt a new vector $\tilde{V}2(t + \Delta t)$ is fixed. The change in the velocity vector is related to the work of external forces and the influence of gravitational acceleration. The work of external forces is manifested in a change in the apparent velocity vector, which can be calculated onboard the spacecraft using information from inertial aids $\Delta\tilde{V}xar = \int_0^{\Delta t} \breve{A}dt$ where $\breve{A}$ is the vector of acceleration produced by external forces (except for gravitational forces).

As a result, one can compose a vector equation, where the unknown vector of gravitational acceleration is $\mathbf{g_m} = \frac{1}{\Delta t}[\tilde{V}1(t) + \Delta\tilde{V}xar - \tilde{V}2(t + \Delta t)]$. All components of the vector equation are calculated in a single coordinate system, so the gravitational acceleration vector will be determined absolutely exactly in this coordinate system, since the angular orientation error for each element of the vector equation is the same. The only reason for wrongly determining the gravitational acceleration vector orientation is the DVDM error in measuring the initial and current components of the velocity. The longer the time interval Δt, the more accurately the gravitational acceleration vector direction will be determined.

A further landing feature of spacecraft on the surface of Mars is that it is necessary to avoid covering the lander with the canopy of the parachute after its soft landing on the surface. The maximum covering probability corresponds to a purely vertical descent without wind. Covering can be avoided by performing a maneuver to withdraw the lander in the horizontal plane from the point of disconnection with the parachute system. The method of separating the parachute from under the lander significantly affects the scheme of further controlled braking. Let us consider the case when the parachute system remains fixed on a protective casing, and the casing itself is separated from the lander almost at the moment of the onset of the active phase of jet engine braking. Due to sharp decrease in the mass attached to the parachute, the speed of the casing decreases rapidly and it lands much later than the lander. After the landing of the casing, the parachute canopy can fold along the surface of Mars in the direction of the surface wind.

Analysis of the orientation angles of the landing platform when simulating its descent by parachute shows that the lander, before starting the braking engine, descends in an almost vertical position. The presence of a horizontal speed indicates the presence of a corresponding wind in the atmosphere. Obviously, the most preferred distance from the point of fall of the protective casing is the direction perpendicular to the direction of the wind speed, that is, the direction perpendicular to the measured horizontal velocity vector of the lander. The lateral deviation distance must be greater than the length of the parachute and its attachment to the protective casing, since with a possible change in the wind direction, the parachute system with the protective casing may start moving in the direction of the lander, shortening the distance. Therefore, the deviation segment of the trajectory must be executed with the maximum possible horizontal deviation speed. It is possible to create a deflection speed only when the braking engines are running by deflecting the thrust vector from the gravitational vertical by the maximum possible angle in the time interval allotted for this operation. For this purpose, the orientation control algorithm should be built on the principle of maximum operation rate. That is, the lander reorientation after the separation of the parachute system should be carried out with the maximum angular velocity in order to achieve the maximum permissible deviation angle of the longitudinal axis (vector thrust of the braking engine) in the shortest possible time.

2. LANDER MOTION CONTROLS

Taking into account the conditions of the flight until the moment of parachute separation, one needs the following:

— To have inertial aids of calculating the parameters of movement and the DVDM onboard the lander.

— To reduce the angular velocity of the lander to the values acceptable for the operation of the DVDM after separation of the aerodynamic screen.

— To predict the moment of braking engine start during the descent by parachute.

— To reorient the longitudinal axis of the lander (braking engine thrust vector) by an angle of up to 35° after braking engine start.

— To perform a withdrawal maneuver from under a separated parachute with a protective casing.

— To finish the braking process on the descent segment at a constant speed, completing the transient processes of attitude control, horizontal speed and braking engine thrust control.

Controlled braking using a jet engine for performing soft landing on the surface of Mars has much in common with performing precision braking on the surface of the Moon (Likhachev et al., 2013). Therefore, the technical means that ensure the control process on this segment may be identical.

The following means are used to control the lander movement when operating the braking engine:

— The onboard computer system (BCS).

— A four-beam Doppler radar speed and distance meter.

— Two cardanless inertial units (CLIUs), each of which contains three fiber optic gyroscopes and three accelerometers with mutually perpendicular identical axes of sensitivity.

— Eight stabilization engines capable of creating control torques along three axes the of the base coordinate system of the lander.

— A braking engine with adjustable thrust.

— A drive control unit (DCU) for the lander thrust control.

— A control unit (CU) for the stabilization engines.

— Sensors for the landing supports contact with the surface.

When performing motion control tasks, an onboard digital computer performs the following operations:

— Forms a cyclogram of basic operations, generating appropriate commands to the system of the spacecraft electroautomatics.

— Provides reception and processing of information from the CLIU and DVDM.

— Performs the execution of algorithms and formation of navigation information for motion control of the lander.

— Executes the algorithms for guidance and control of the angular movement and movement of the lander center of mass.

— Generates initialization flags for each characteristic control segment.

— Issues command flags for the operation of pyrotechnic devices of the lander.

— Generates flags for control commands for the electropneumatic valves of the stabilization engines and drives for controlling the thrust of the braking engine.

2.1. Doppler Velocity and Distance Meter

The radar DVDM is the main tool for measuring the speed and altitude of the lander. The calculation of the trajectory according to the CLIU information by integrating the mathematical model of movement starts at a distance from the surface of the order of 100 km and is associated with large overloads in the aerodynamic braking segment. Even with nearly minimal measurement errors of the CLIU, this leads to the accuracy of knowing the flight altitude to the moment of braking engine start at several kilometers and a speed of 10–20 m/s. It is quite natural that active surface location by means of the DVDM can significantly increase the accuracy of knowing the trajectory parameters.

The DVDM is designed to obtain information about the distance to the surface along four narrow beams of its antenna system and about the speed of the lander relative to the surface in projection on the axis of these beams. To obtain information about distance, the device measures the time interval from the moment of emission by the radar transmitter impulse until the moment of arrival at the receiver of the impulse reflected from the surface. Velocity projection measurement is based on the determination of the Doppler frequency shift of the radio signal emitted and reflected from the surface.

2.2. Propulsion System

The propulsion system is intended:

— For damping the angular velocity of the lander under the parachute.

— To create control torques in three axes of the lander and the controlled braking force after parachute separation.

2.2.1. Choice of engines. To ensure the operating conditions of the DVDM to control the trajectory and angular motion on the lander, only low-temperature hydrazine engines can be used, which in the rarefied atmosphere of Mars do not interfere with the passage of the radar signal through the exhaust flares of the spacecraft engines. In addition, in the passive part of the trajectory, it is necessary to reduce the speed of rotation of the spacecraft under the parachute to 0.5°–2°/s. This task and the tasks of angular motion control in the pitch, yaw and rotation channels can be performed by at least eight hydrazine stabilization jet engines – four in the roll channel and two each in the pitch and yaw channels. According to the conditions of location on the lander, these engines can be placed on the periphery in the lower part of the lander. To eliminate the ingress of the exhaust jets of these engines on the structural elements of the lander and to maintain its controllability, stabilization engines can be mounted so that the axis of the nozzle of each engine should be turned outside the lander contour at an angle of 15°–25°, and in this tangential plane, the nozzle axis should be turned relative to the longitudinal axis of the lander at an angle of up to 55°–60°. These engines can be located on the lander in groups of two engines on each of the semiaxes of the base coordinate system of the lander. Therefore, to create a control torque in the rotation channel, one can turn on two or four engines, one from each group using 80% of their thrust. And to create a control torque in the pitch or yaw channels, one can turn on two engines of the same group, using 50% of their thrust. From the series of engines available, only hydrazine ones can be chosen. The thrust of the engines must be chosen from the condition of the speed of operation, taking into account the values of the moments of inertia of the lander.

A hydrazine engine should be used as the braking engine. To regulate the velocity decrease of the lander in the final segment of braking, its minimum thrust should be no more than 75% of the "Martian weight." For intensive deceleration of the lander after the separation of the parachute system, the braking engine thrust level must exceed the "Martian weight" of the lander by more than 2.5 times. That is, the depth of regulation of the braking engine thrust (the ratio of the maximum to the minimum thrust) should be at least 3.5 and about 5, taking into account possible deviations from nominal values. If n is the ratio of the programmed acceleration during intense deceleration to the free fall acceleration on the surface of Mars, then the gravitational losses with respect to the speed of switching on the braking engine will be $Kg = 1/(n - 1)$, reaching 100% for $n = 2$, 67% for $n = 2.5$, and 50% for $n = 3$.

Since the range of a steady descent speed by parachute is 40–65 m/s, the duration of intensive deceleration will be 11–18 s at $n = 2$ and 6–9 s at $n = 3$. But by the time of the intensive braking, it is necessary to orient the longitudinal axis of the lander (the braking engine thrust vector) opposite to the velocity vector. However, at the time of the parachute detachment, as mentioned earlier, the deviation from the required orientation may be 35°. It is obvious that the control loop of the lander angular motion should have the maximal rapidity and produce significant control torques. Therefore, the braking engine, in addition to the thrust, should also generate control torques. To this end, it should have at least four chambers or four independent engines spaced from the longitudinal axis by a certain distance. To create control torques in the pitch or yaw channels, the thrust of diametrically opposed chambers/engines should be regulated within certain limits.

In the case of a four-chamber engine, a scheme can be used when one of the drives regulates the total fuel consumption (the total thrust of the four chambers), while the other two drives differentially change the flow through diametrically opposite chambers, creating a difference in the thrust of these chambers.

To control the rate of descent, the control commands enter the engine thrust control drive (TCD), and the commands to control the angular movement in the pitch and yaw channels go to two thrust control drives of diametrically opposite chambers (TCCs), changing their thrust from the level set by the TCD. When the TCD is set into a new mode of engine thrust, the perturbing torque created by the engine itself changes proportionally to the thrust. But the control torques also change in the same ratio while maintaining the TCC angle. That is, changes in the center of mass control channel do not change the operating mode of the angular motion control channels. Deviations in the ratio of the perturbing and control torque can occur only due to technological errors in the manufacture and adjustment of the chambers. In turn, changes in the TCC position in the angular motion control channels do not change the resulting thrust of the engine and thus do not affect the operation of the channel control of the lander center of mass.

When four independent engines are used, and the command to control the movement of the center of mass changes, the perturbing torques from the engines also alter. This is due to the fact that individual engines will have an individual dependence of the engine thrust on the TCD steering angle. In this case, with the unchanged command of angular motion control, the ratio between perturbing and control torques alters. This will undoubtedly cause dynamic angular deflections and changes in the commands for angular movement. In this case, the use of one executive device to control two channels always creates the possibility of a loss of efficiency of one of the channels due to limitations within the limits of the drive's operability. When four independent braking engines are used, both the command to control the TCD, and for the movement of the center of mass to control the pitch/yaw channels should be used. Therefore, to maintain the operability of the angular motion control channel, it is necessary to limit the required rate of change of the commands for controlling the movement of the center of mass. For example, it is necessary to limit the rate of change of the angle of rotation of the thrust regulator in the control channel of the center of mass movement at a level of 65–75% of the maximum angular speed of the TCD. This is necessary so that from the available range of the maximum speed of the TCD for the angular movement control, it is always possible to adjust the position of the actuator with an angular speed of 25–35% of the maximum available.

The actual control by vertical and horizontal speed of the spacecraft starts after separation of the parachute system and protective casing. To brake the vertical speed, the thrust of the braking engine is controlled, while to control the horizontal speed, the angle of deviation of the longitudinal axis from the gravitational vertical is controlled. The required turning speed of the order of 10°/s should be achieved within the interval of 1–1.5 s. Therefore, the angular movement control requires a maximum angular acceleration from the executive devices of the order of $10°/s^2$. For transverse inertia moments of the order of 1000 kg m^2, the required value of the control torques along the pitch and yaw channels should be 150–175 Nm. Since diametrically opposite braking engine chambers can be located at a distance of 0.5 m from the longitudinal axis, a decrease in the thrust of one chamber and an increase in the thrust of the other relative to their average level by 100–150 N is sufficient to create torques of this level.

Thus, to compensate for the perturbing torque from the eccentricity of the braking engine thrust and create the necessary control torque for controlling the lander orientation, it is necessary that the control range of the thrust difference between two diametrically opposite chambers account for 10–15% of the sum of the thrust of these chambers.

If four independent engines with a thrust of 750–3500 N are used, they will provide both vertical speed control and the creation of control torques. If for the TCD we use stepper drives with 1000 steps per total control range, then the discreteness of variation of a control torque by one step of the thrust regulator rotation angle will be about 15–20 Nm/step, and to create the maximum control torque, only 10–20 drive rotation steps are required. Discreteness per one step of rotation of the drive in terms of the control angular acceleration will be about $1°/s^2$/step. At the same time, when using a 4-chamber engine of the same drives for controlling the total thrust of the engine, the discreteness of the control torque will be about 1.5 Nm/step, and the control angular acceleration of the order of $0.1°/s^2$/step, thereby providing greater "softness" of the angular motion control.

Thus, the use of a 4-chamber engine compared with the use of four independent engines significantly reduces mutual restrictions and interconnection of control channels, and also provides sufficient softness of the spacecraft angular motion control.

2.2.2. The main characteristics of a braking engine. As an example, for the range of lander masses (at the time of parachute separation) of 1200–1400 kg, requirements were developed for a 4-chamber braking engine operating on hydrazine decomposition products. Its use requires an additional section for warming up the catalyst in the engine chambers be included in the list of control operations. For this purpose, the engine is turned on four seconds before the start of its thrust control. In this time interval, at a fixed value of the TCD angle, the engine thrust increases from zero to ~1500 N. Switching on the engine to the "warming-up" mode is allowed even during the descent by parachute.

In the general case, the throttle characteristic of the braking engine is a nonlinear function of the dependence of the thrust on the TCD rotation angle, but in the operating range of the TCD rotation angles, it can be approximated by a linear function with saturation in the region of the maximum thrust with the form $P = Pr + Sd(Del(1) - Delr) \leq P_{\max}$, where Pr is the thrust value at the tuning point; Sd is the thrust derivative with respect to the TCD rotation angle; D is the angle of the TCD rotation; $Del(1)$ is the value of the TCD angle at the tuning point of the throttle characteristic; $P_{\max}$ is the maximum thrust. Figure 3 shows the braking engine thrust depending on the TCD rotation angle.

The working range of rotation angles of the TCD was from –72.5° to +90°, so that the braking thrust value of the engine can be adjusted from a minimum thrust of 2943 N to a maximum thrust of 12737 N. In

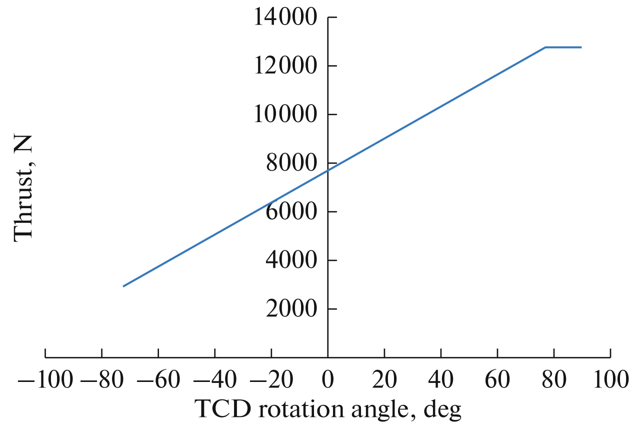

Fig. 3. Braking engine throttle characteristic.

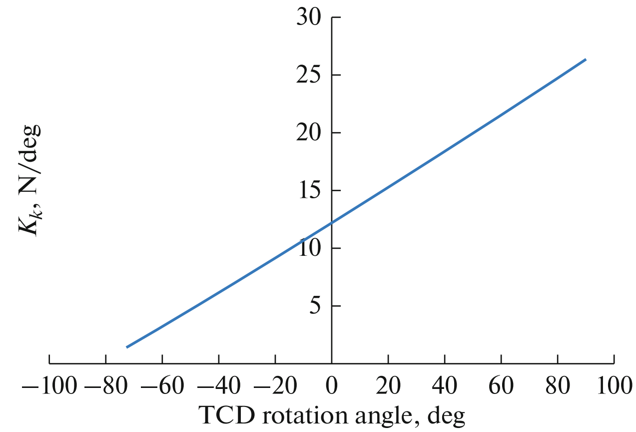

Fig. 4. Proportionality coefficient, K_k.

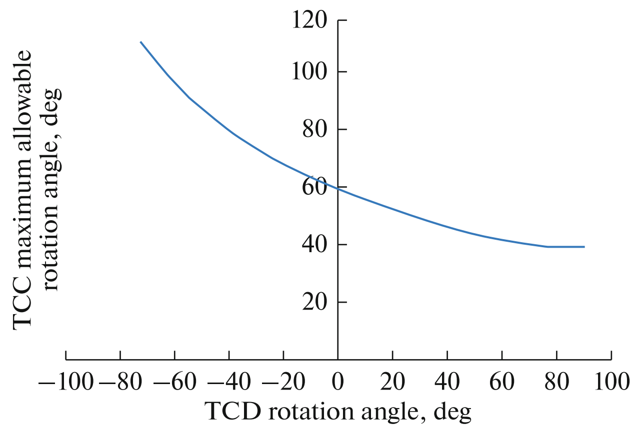

Fig. 5. TCC maximum angle.

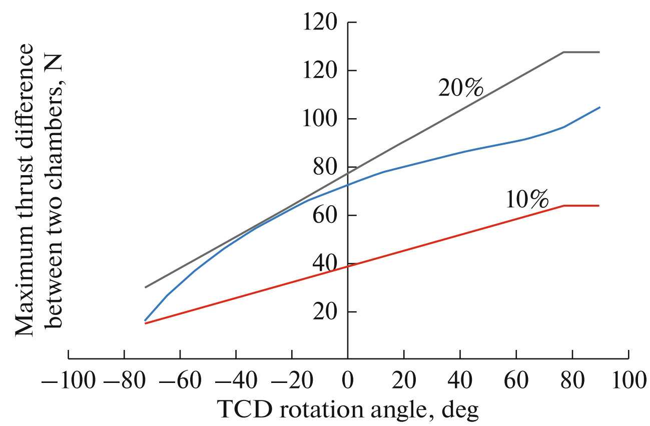

Fig. 6. Achievable thrust difference of chambers.

the warming-up mode, the TCD rotation angle should be equal to −95°, while the steady-state value of the braking engine thrust at the end of this mode is about 1500 N.

At the zero position of the TCC driver, the nominal thrust of each chamber is equal to one fourth of the total thrust of the engine, $Prk(i) = Ptr/4$. To control the angular movement, the thrust difference of two diametrically opposite chambers is regulated by the TCC drives. The thrust difference for a pair of chambers is determined by the ratio $\Delta Prk = K_k Del(2,3)$ where the proportionality coefficient K_k nearly linearly depends on the total thrust of the engine (the angle of the TCC rotation) (see Fig. 4), increasing the efficiency of angular control while increasing the total engine thrust.

Thus, depending on the total thrust of the engine, the thrust difference of a pair of chambers within 10–20% of the total thrust of these chambers can be used to control the angular motion (Fig. 6).[1]

[1] The end of the article follows.

REFERENCES

Artem'ev, A.V., Bykov, A.I., Rykov, E.V., and Markachev, N.A., Modeling the processes of soft lunar landing of a spacecraft. Development of means and methods of ground testing using the dynamics of a similar spacecraft model, *Vestn. NPO im. S.A. Lavochkina*, 2019, no. 1, pp. 31–38.

Likhachev, V.N. and Fedotov, V.P., A method for estimating the direction of gravitational acceleration in the active phase of a spacecraft landing on the Martian surface, *Sol. Syst. Res.*, 2017, vol. 51, no. 7, pp. 563–565.

Likhachev, V.N., Sikharulidze, Yu.G., and Fedotov, V.P., Main braking phase for a soft moon landing as a form of trajectory correction, *Sol. Syst. Res.*, 2013, vol. 47, no. 7, pp. 535–541.

Likhachev, V.N., Sikharulidze, Yu.G., and Fedotov, V.P., Final touchdown phases and a guidance and control methodology for a soft Moon landing, *Sol. Syst. Res.*, 2014, vol. 48, no. 7, pp. 497–506.

Zhukov, B.I., Zaiko, Yu.K., Likhachev, V.N., Sikharulidze, Yu.G., et al., Adaptive terminal algorithm for Moon landing, *Vestn. NPO im. S.A. Lavochkina,* 2012, no. 4, pp. 92–102.

Translated by G. Dedkov