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DEVELOPMENT OF THE PROPULSION CONSTRUCTION AND THE TRAJECTORY OF THE SPACECRAFTS FOR THE STUDY OF MARTIAN PLANETARY SYSTEM

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The article provides a brief description of the flight scheme of a prospective automatic spacecraft intended for the study of Mars and its satellites by remote and contact methods. At the near-Martian expedition site, it is planned to first bring the vehicle into orbit of the artificial satellite Deimos, and then landing on Phobos with the subsequent delivery of its soil to Earth. The main ballistic characteristics of the spacecraft flight conditions at all stages of the flight at launch after 2025 are given. The time frames for the five starting periods are considered - in 2026, 2028, 2030, 2033 and 2035. The launch of the spacecraft on the flight path to Mars is performed by a heavy class launcher. The article describes the design of the vehicle, propulsion systems of its modules and flight scheme at all stages – from launch from the Earth to landing on Phobos, and returning back to Earth.

The article describes the propulsion systems of the main spacecraft units proposed for the mission implementation – the propulsion module, the flight landing platform and the return vehicle. The designs of these units are provided in the work. Flight schemes have been developed in accordance with their characteristics, which allows conducting remote study of Deimos, making a soft landing on the surface of Phobos, and then delivering samples of its soil to Earth.

The project should be developed on the basis of the spacecraft launch from the Vostochny launch site by the Angara-A5 launch vehicle and the KVTK upper stage. An alternative variant of the construction of a spacecraft involves the use of a vehicle of a lighter class - perspective Soyuz-5 launch vehicle and Fregat-SBU upper stage. In this case, the engine module is excluded, and the flight and landing module is replaced by a heavier version with larger tanks.

Both proposed options for constructing a spacecraft make it possible to implement the developed trajectory, while ensuring full-time operation of the target equipment and conducting a set of experiments during a given period of active spacecraft existence.

Keywords: spacecraft, propulsion system, trajectory, flight scheme, interplanetary transfer, Mars, Phobos, Deimos.

РАЗРАБОТКА КОНСТРУКЦИИ ДВИГАТЕЛЬНЫХ УСТАНОВОК И ТРАЕКТОРИЙ КОСМИЧЕСКИХ АППАРАТОВ ПРОЕКТА ДЛЯ ИССЛЕДОВАНИЙ ПЛАНЕТНОЙ СИСТЕМЫ МАРСА

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В статье приведено краткое описание схемы полета перспективного автоматического космического аппарата (КА), предназначенного для исследования Марса и его спутников дистанционными и контактными методами. На околомарсианском участке экспедиции предполагается сначала вывести аппарат на орбиту искусственного спутника Деймоса, а затем совершить посадку на Фобос с последующей доставкой его вещества на Землю. Даны основные баллистические характеристики условий полета КА на всех этапах при запуске после 2025 г. Рассмотрены временные рамки для пяти пусковых периодов – в 2026, 2028, 2030, 2033 и 2035 гг. Выведение КА на траекторию перелёта к Марсу выполняется с помощью ракеты-носителя (РН) и разгонного блока (РБ) тяжёлого класса. В статье приведено описание проектного облика аппарата, двигательных установок (ДУ) его модулей и схемы полёта на всех этапах – от старта с Земли до посадки на Фобос и возврата обратно к Земле.

В статье рассмотрены предлагаемые для реализации миссии двигательные установки основных блоков космического аппарата – двигательного модуля, перелётно-посадочной платформы и возвращаемого аппарата. Приведены описания их конструкций. В соответствии с их характеристиками разработаны схемы полёта, позволяющие провести дистанционные исследования Деймоса, осуществить мягкую посадку на поверхность Фобоса, а затем доставить образцы его грунта на Землю.

Проект должен быть разработан, исходя из запуска космического аппарата с космодрома «Восточный» при помощи РН «Ангара-А5» и РБ «КВТК». Альтернативный вариант построения КА предполагает использование средств выведения более лёгкого класса – перспективную РН «Союз-5» и РБ «Фрегат-СБУ». В этом случае двигательный модуль отсутствует, а перелётно-посадочный модуль заменяется на более тяжёлую версию с баками большего объёма.

Оба предлагаемых варианта построения КА позволяют реализовать разработанную траекторию, при этом обеспечить штатную работу целевой аппаратуры и провести комплекс экспериментов в течение заданного срока активного существования КА.

Ключевые слова: космический аппарат, двигательная установка, траектория, схема полёта, межпланетный перелет, гравитационный манёвр, Марс, Фобос, Деймос.

Introduction. In spite of significant successes in fundamental space research conducted by interplanetary stations and orbiting astrophysical observatories, a number of questions remain unsettled: questions about the origin of the solar system and its evolution; about the origin of life on Earth and, possibly, on other planets. In this regard, the study of small celestial bodies becomes interesting, which allows us to obtain valuable information about the early stages of the genesis of the bodies of the solar system.

At the same time, Phobos is of particular interest for study as it is a small body and one of the two (along with Deimos) satellites of Mars. The small sizes of these bodies exclude internal heating, any tectonic activity, therefore their substance is the initial material of the protoplanetary cloud. Based on this, many believe that the substance of Phobos is the primary material of the solar system and is of exceptional interest.

This fact stimulated the beginning of Phobos research in the 80s of the last century, and the study was continued in the current century. At present the research is represented by a program studying the system of Mars and its satellites. Its next stage is planned to be carried out within the framework of the Federal Space Program of Russia for 2025–2035 [1; 2].

The main scientific objectives of the project are aimed at solving a wide range of problems related, first of all, to the genesis of the solar system. The solution of the main problem should be provided by studying the physical and chemical properties of the relict substance from Phobos. Other scientific tasks include:

– studies of the physical and chemical characteristics of Phobos as a celestial body not only by remote methods, but also by contact methods, as well as studies of samples of Phobos matter delivered to Earth, which will allow us to come closer to understanding the origin of Martian satellites and, possibly, the origin of satellite systems from other planets;

– determination of detailed parameters of the orbital and proper rotation of Phobos, which is important for studying the internal structure of this small body and the evolution of its orbit;

– conducting remote studies of Mars, Phobos and Deimos by scanning the surface in various spectral ranges;

– studies of the physical conditions of the environment near Mars – electric and magnetic fields, the characteristics of the interaction of the solar wind with the plasma environment of Mars, including the registration of oxygen ions "escaping" from the Martian atmosphere, which will expand the understanding of water history on Mars;

– research of variations of Mars atmospheric characteristics.

The implementation of these tasks will significantly expand knowledge of the composition and structure of small bodies of the solar system. In addition, the reserve of this project can be used for further research of other celestial bodies, as well as for the subsequent stage of development of asteroids and planet satellites.

1. The construction of the spacecraft propulsion system. Bringing the spacecraft on the flight path for the study of the Mars system is carried out by the upper stage (US) combining the units of the launch vehicles. The spacecraft is designed to solve the following problems:

– flight to Mars and arrival in the orbit of the artificial satellite of Mars;

– entering the orbit of Deimos and researching it by remote methods;

– the transition to the orbit of Phobos and the study of it by remote methods;

– landing on Phobos, taking soil samples, conducting research;

– start from Phobos and flight to Earth;

– landing on Earth.

The main modules of the spacecraft are: flight and landing platform (FLP), return vehicle (RV) and descent vehicle (DV). FLP and RV include propulsion systems (PS), the engines which should operate on asymmetric dimethylhydrazine and amylin fuel components. At this stage of the work, two versions of the PS of FLP are considered, one of which provides the presence of a detachable propulsion module (PM) on the platform. There are trusses between the PM – FLP and the FLP – RV modules that are load-bearing power elements, each of them is an all-welded structure in the form of a truncated prism. The modules are mounted with trusses with pyro-bolts of the separation system.

PM as a part of the spacecraft is designed to perform the following operations:

– correction of the trajectory at the flight stage, from the moment of separation from the upper stage to braking before going into orbit as an artificial satellite of Deimos;

– stabilization along the pitch and yaw channels during the operation of the main engine.

Low-thrust engines (LTE) of the PM of the FLP ensure the launch of the main engine, as well as the orientation and stabilization of the spacecraft. The PM is controlled by the FLP of the SC. The PM is developed on the basis of US "Frigate".

The structural basis of the block consists of six spherical shells with a diameter of 1360 mm spaced evenly and welded into each other. Four shells are used as fuel tanks: two fuel tanks and two oxidizer tanks. Two spheres present airtight instrument containers. Since the volume of the oxidizing agent is slightly larger than the volume of the fuel, the oxidizing tanks have been fully implemented and are being introduced into the fuel tanks. So it has become possible to ensure full filling of the tanks with fuel, achieving the most efficient performance of the available volumes [3]. The spherical shape is optimal in terms of weight neutralizing the action of internal pressure. The power circuit of the tank block is a torispherical structure, consisting of six mutually intersecting spherical shells, hermetically connected to each other through expansion frames by argon-arc welding.

To transfer longitudinal loads from the launched spacecraft, the sealed containers are "pierced" by 8 power rods – one for each fuel tank and two for the instrument compartment, thus forming a truss structure. Each bar above and below ends with a complex milled support. The payload adapter is attached to the upper supports with the help of threaded rods, and the pyromechanical pusher locks of the separation system mounted on the transition compartment are joined to the lower supports [4].

It is supposed to use the engine C5.92 developed by the Design Bureau of Chemical Engineering named after A. M. Isaev as an engine for the PM of the FLP.

The development of the PM of the FLP assumes the maximum use of the reserve of the PM of the flight module of the "Phobos-Grunt" and "Luna-Glob" projects [5; 6]. The supporting structure for the PM is a block of tanks, which is a welded structure of four spherical fuel tanks with cylindrical spacers between them. Fuel tanks have an internal cavity volume of (0.150 ± 0.002) m^3 [(150 ± 2) l] each: two tanks for storing and supplying fuel and two for an oxidizing agent.

The project considers two design options for the fuel tank:

– with a metal separation diaphragm;

– with an elastic displacing device (EDD).

To solve the tasks, a tank with a metal diaphragm is a more preferable structural solution for supplying fuel components (FC) as relating to the EDD. Providing the safety of the diaphragm operability during two years under the conditions of the spacecraft operation in the Mars planetary system in the presence of thermal cycling and thermal expansion of the FC is a rather laborious and technically difficult process with a large number of assumptions. Fuel is taken from a spherical tank through a perforated tube fixed on both sides with an EDD installed in it. The boost gas presses on the surface of the EDD and squeezes the fuel into the line. The EDD material consists of polyphene fabric and fluoroplastic films that provide chemical resistance and high elasticity. Polyphene fabric reduces the tendency of the material to crack and improves resistance to complex bending deformations. To give the shape and dimensions of the internal contour of the fuel tank, the EDD is made by heat welding from molded segments (lobes), which are welded into hemispheres, and then into the sphere.

The choice of one of the presented options for the construction of the fuel tank to a certain and not significant extent affects the mass and the composition of the pneumohydraulic circuit of the PS of the FLP to a certain and not a significant extent.

A pressure sensor, temperature sensors, pyro valves, a filter, a pneumatic main valve, check valves, an electric heater with three control temperature sensors are attached on each tank. There are two mounting brackets in each tank which allow it to be attached to the frame; elements for installing refueling panels and assemblies, lodges for fastening two composite high-pressure cylinders (SMKB 25–340) with a volume of 25 l (internal diameter 356 mm) [7]. Installation of fuel and gas lines was performed on the outer surface of the tanks and structural elements of the PS of the FLP.

Brackets are welded to the cylindrical spacers of the tank block for mounting the valve panel, the rods of the stabilization engine blocks. Blocks of the low-thrust engines are mounted on four identical rods made of AMg6. Five low-thrust engines are installed on each bracket: one engine with a thrust of ~ 1.27 kgf (~ 12.45 N) and four engines with a thrust of ~ 5.5 kgf (~ 53.9 N).

The second version of the design of the FLP propulsion system suggests that it includes a propulsion system that performs the functions of the propulsion module. The supporting structure for the FLP propulsion system is a block of main tanks, which are a welded structure of four spherical fuel tanks with cylindrical spacers between them. Orientation and stabilization system (OSS) tanks are placed In the spacers. Brackets are welded to the cylindrical spacers of the tank block for mounting the panel of the valve block, the rods of the engine blocks of the OSS and the truss for installing the main engine with a thrust of ~ 2030 kgf (~ 19900 N). The number of OSS engines and their propulsive power values are similar to the first option.

The supporting structure for the propulsion system of the RV is a block of tanks, which is a prefabricated structure of four spherical fuel tanks, connected through flanges by spacers to each other. Fuel tanks have a volume of fuel cavity (0.125 ± 0.002) m^3 [(125 ± 2) l] each: two tanks for storing and supplying fuel and two for an oxidizing agent.

The project considers two design options for the fuel tank:

– with a metal separation diaphragm;

– with an internal tank capillary type device (ITCTD).

The use of a metal diaphragm requires confirmation of its operability in the flight conditions of the spacecraft during the launch escape system. The design of the tank with ITCTD does not have moving mechanical parts and provides the supply of fuel components to the fuel lines without gas inclusions. The composition of the ITCTD includes a gas-separation device and capillary type localization elements near the intake surface. To increase the efficiency of the ITCTD, most of its elements and units are made of capillary-porous mesh materials.

A pressure sensor, temperature sensors, a pyrovalve, a filter, a pneumatic main valve, check valves, an electric heater with three control temperature sensors are attached on each tank. Each tank has elements for installing refueling panels and components, lodges for fastening two SMKB 25–340 cylinders.

Brackets for attaching the valve block panel, stabilization engine blocks and the truss of correction engine blocks are welded to the structural elements and tanks., Two stabilization engines with a thrust of ~ 1.27 kgf (~ 12.45 N) each are installed on every bracket.

The correction engine block is an assembly of two engines with a total thrust of ~ 79 kgf (~ 784 N) on a circular plate, which is fixed to the rest of the structure with a carbon fiber truss. Heat is removed from working engines by a heat accumulator with subsequent discharge through a heat pipe to a radiator, mounted with a bracket on a plate of a motor correction unit.

The installation of fuel and gas lines was made on the outer surface of the tanks and structural elements of the RV propulsion system.

2. The flight pattern. When developing the flight pattern, the following restrictions were taken into account.

Due to the spacecraft's battery capacity limitation, the duration of shadow intervals when flying in orbits of the artificial Mars satellite should not exceed 3 hours.

There should be no shadow intervals in orbits for observations:

– observations of Deimos;

– orbit of observation and quasi-synchronous orbits of Phobos.

Important dynamic operations (maneuvers, landing on Phobos, etc.) should not take place when Mars sets behind the Sun when observed from Earth.

The article analyses several options for flight patterns for launching from the Earth in the range from 2025 to 2035, in accordance with the promising Federal Space Program covering this decade. In the period under review, for the optimal flight of the spacecraft to Mars by the criterion of the maximum of its mass delivered to the satellite's orbit on this planet, five launch periods can be distinguished – in October 2026, November 2028, December 2030, April 2033 and July 2035. Further, these options for flight patterns are numbered according to the year of launch from Earth.

The article analyzes several versions for flight patterns for launching from the Earth in the range from 2025 to 2035, in accordance with the promising Federal Space Program covering this decade. In the period under review, five launch periods for the optimal flight of the spacecraft to Mars by the criterion of its maximum mass delivered to the satellite's orbit of this planet can be distinguished – in October 2026, November 2028, December 2030, April 2033 and July 2035. Further, these versions of flight patterns are numbered according to the year of their launch from Earth.

The spacecraft flight pattern is close to that previously developed for the "Phobos-Grunt" project [8–11]. In accordance with the objectives of the expedition and the construction of the spacecraft and the restrictions on the operation of its on-board systems, a flight pattern including the following stages was developed:

– launch from the Earth, placing in an Earth-escape trajectory;

– Earth – Mars flight;

– braking and ascent to the initial orbit of the artificial Mars satellite (AMS);

– flight in the orbits of the AMS with remote sensing of Mars;

– transition to the Deimos observation orbit (OO);

– Deimos observation orbit flight for conducting its remote sensing;

– Deimos – Phobos flight;

– transition to the Phobos observation orbit;

– phased approach of the spacecraft orbit to the Phobos orbit to distances allowing landing;

– landing on the surface of Phobos;

– conducting contact research, taking soil samples and loading it into the descent vehicle;

– the start of the returned vehicle along with the descent vehicle from the landing module platform, the ascent of the descent vehicle to the base orbit, while the landing module remains on the surface of Phobos and continues to carry out the scientific program;

– phased ascent of the returned vehicle to the escape orbit;

– the returned vehicle flight along the Mars – Earth trajectory;

– separation of the descent vehicle from the returned vehicle on approach to Earth;

– the entry of the descent vehicle into the Earth's atmosphere, landing, search and evacuation of the descent vehicle with Phobos soil samples.

The following is a detailed description of the expedition stages.

2.1. Flight to Phobos

2.1.1 Earth-Mars Flight. Placing the spacecraft into the flight trajectory to Mars is carried out with the help of a heavy-class launch vehicle. The "Angara-A5" LV with the "KVTK" upper stage is considered as the main variant of the launch vehicle. Also, as an alternative, the following options for launching the spacecraft were analyzed:

"Angara-A5" LV with the "DM-03" upper stage,

"Soyuz-5" LV with the "DM-03" upper stage,

"Soyuz-5" LV with the "Frigate-SBU" upper stage.

The Earth-Mars flight starts from the moment the spacecraft enters the Earth-escape trajectory and ends with the minimum distance approaching Mars [12]. The duration of the flight is from 6 to 11 months, depending on the expected launch period.

Three corrections are planned on the interplanetary section of the flight. The first correction is carried out on the 5–10th day of the flight, it can reach 25 m/s. The second correction is carried out approximately on the 65th day of the flight, its value will not exceed 10 m/s. The third correction is carried out from 4 to 2 weeks before the approach to Mars and can reach 15 m/s.

The characteristics of the flight trajectories selected for the expedition are presented in tab. 1. The values of the payload mass displayed on the flight trajectory to Mars are shown in tab. 2.

Table 1

The main characteristics of the Earth-Mars flight trajectories

	Version				
Characteristics	2026	2028	2030	2033	2035
Earth launch date	31.10.2026	23.11.2028	24.12.2030	20.04.2033	10.07.2035
Duration of the flight to Mars, days	311	300	283	199	198
Arrival to Mars	07.09.2027	19.09.2029	03.10.2031	05.11.2033	24.01.2036
Asymptotic departure speed from the Earth, km/s	3.050	3.020	3.279	3.038	3.469
Characteristic departure speed from the Earth, km/s	3.639	3.631	3.702	3.636	3.688
Asymptotic approach speed to Mars, km/s	2.570	2.969	3.479	3.310	2.630
The characteristic injection speed to the orbit of the AMS, km/s	0.769	0.972	1.260	1.161	0.798

Table 2

The payload mass placed on the Mars flight trajectory

Launch vehicle	Version				
	2026	2028	2030	2033	2035
“Angara-A5” + “KVTK”	7500	7500	7350	7500	7200
“Angara-A5” + “DM-03”	5900	5900	5800	5900	5650
“Soyuz-5” + “DM-03”	3300	3300	3200	3300	3100
“Soyuz-5” + “Frigate-SBU”	3950	3950	3850	3950	3750

2.1.2 Flight in the sphere of Mars action, Phobos rendezvous. The near-Martian part of the expedition consists of the stages listed in tab. 3. Time frames and duration data are also given there. It should be noted that when planning the duration of the stages, both the requirements for conducting scientific experiments and the limitations of the spacecraft onboard systems were taken into account:

– when flying along elliptical orbits of the AMS with a large apocenter radius (initial, first, third transitional, as well as prelaunch), the duration of the shadow sections does not exceed 3 hours;

– when flying in Deimos and Phobos observing orbits, as well as in the area of rendezvous with Phobos, shadows are absent;

– landing on Phobos is carried out when the angle of the Sun – Earth – Mars is more than 5 degrees.

The formation of the Deimos observation orbit. In the area of the pericenter of the approach trajectory, the propulsion system (PS) is switched on, and the spacecraft transfers to the three-day orbit of the AMS, hereinafter referred to as the “initial” one. The pericenter height of this orbit will be ~ 800 ± 400 km, and the apocenter height – 79 thousand km. The value of the braking impulse will be from 0.8 to 1.3 km/s.

A scheme of the approach area and the initial stage of stay in the AMS orbits and the entry into the Deimos observation orbit is shown in fig. 1.

After the decision is made to transfer the spacecraft to the first transitional orbit in the apocenter of the initial orbit, a second maneuver is performed, which combines the spacecraft's orbit plane with the Deimos orbit plane, and the pericenter radius rises to the observation orbit radius (~ 20.2 thousand km). The characteristic speed of the maneuver is 285 m/s. The orbital period of the resulting orbit will be 3.8 days.

The third maneuver at the pericenter of the transitional orbit of the spacecraft is transferred to the Deimos observation orbit – a circular synchronous orbit with a period equal to the period of Deimos.

Phobos observation orbit flight. The scheme of injection into Phobos observation orbit is shown in fig. 2. An almost circular orbit with an average radius of 9.9 thousand km, which is approximately 535 km above the Phobos orbit, was chosen as its orbit. The orbital period of the observation orbit is 8.3 hours. The spacecraft is transferred to it in two maneuvers – the fourth powered phase forms an elliptical orbit (a second transitional one) with the desired radius of the pericenter, and the fifth transfers the device into a circular orbit.

In the observation orbit (OO), autonomous navigational observations of Phobos begin, since the apriori accuracy of knowledge of its ephemeris (3σ is almost 20 km) is insufficient neither for further approach of the SC orbit to the Phobos orbit, nor for landing on its surface. Navigation measurements are carried out using an on-board TV camera in those areas of OO where it is possible to make the lighting conditions and where the distance from the spacecraft to Phobos does not exceed 1500 km. To clarify the Phobos ephemeris and its gravitational constant, it is necessary to conduct navigation measurements in at least three rendezvous, which may take almost a month. Therefore, the minimum necessary spacecraft flight time along OO is 1.5 months. Carrying out navigation observations from the spacecraft is supplemented by high-precision trajectory measurements of the spacecraft's orbit from the Earth. As a result of the joint processing of all navigation information, the mutual motion of the spacecraft and Phobos before the transition to a closer to Phobos quasi-synchronous orbit will be predicted with limiting errors of ± 3 km.

Formation of a pre-landing quasi-synchronous orbit. For the direct rendezvous of the spacecraft with Phobos and providing an autonomous landing system with high-precision navigation and ballistic data, a quasi-synchronous orbit (QSO) was chosen. A feature of this orbit is its short range (no more than 70 km) of the spacecraft from Phobos during the entire flight interval, which allows to increase the resolution of television images and, therefore, to increase the accuracy of measuring the spacecraft – Phobos line of sight.

To select the QSO, a study of the trajectory of the spacecraft in the restricted problem of three bodies was made: Mars, Phobos and the spacecraft. As a result, many QSO were built, providing landing at any longitude and, if necessary, the possibility of a repeat landing [13–15]. The lighting conditions, the provision of radio communications with tracking stations, and other conditions determine the landing time at a given point. The landing point and time designate the point and moment of descent from the QSO, which are the initial data for calculating this orbit. The transition to it is carried out by two pulses: 36 and 30 m/s.

Important scientific experiments will be conducted during the flight on the QSO. The planned flight time on the QSO is 30 days.

Table 3

The main near-Martian flight stages

Flight stage/ Version	2026	2028	2030	2033	2035
Arrival to Mars, ascent to the initial orbit, flight in the initial and intermediate orbit	07.09.2027	19.09.2029	03.10.2031	05.11.2033	15.01.2036
flight on Deimos observation orbit	25.11.2027	15.10.2029	15.12.2031	01.06.2034	10.02.2036
flight on the observation orbit and Phobos QSO	20.01.2028	15.12.2029	20.09.2032	15.08.2034	10.07.2036
Landing on Phobos, take-off of the spacecraft, flight on the base orbit	20.04.2028	01.03.2030	01.12.2032	01.12.2034	01.11.2036
Prelaunch orbit flight	25.08.2028	25.03.2030	15.12.2032	20.12.2034	15.04.2037
Transition to the Earth-escape trajectory of return to Earth	12.09.2028	01.11.2030	28.01.2033	08.05.2035	12.07.2037
Arrival to Earth, entry into the atmosphere, landing	11.08.2029	07.07.2031	02.09.2033	22.11.2035	08.04.2038

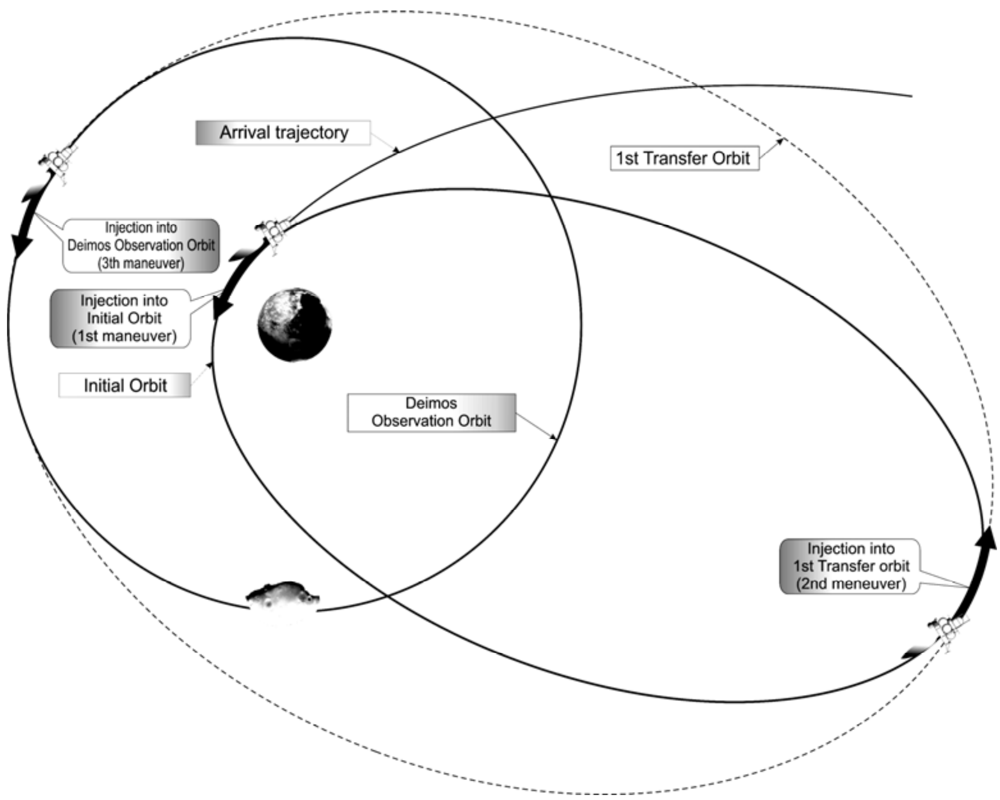

Fig. 1. The scheme of the initial stage of the flight on the AMS orbit
and injection into Deimos observation orbit

Рис. 1. Схема начального этапа полета по орбите ИСМ
и выхода на орбиту наблюдения Деймоса

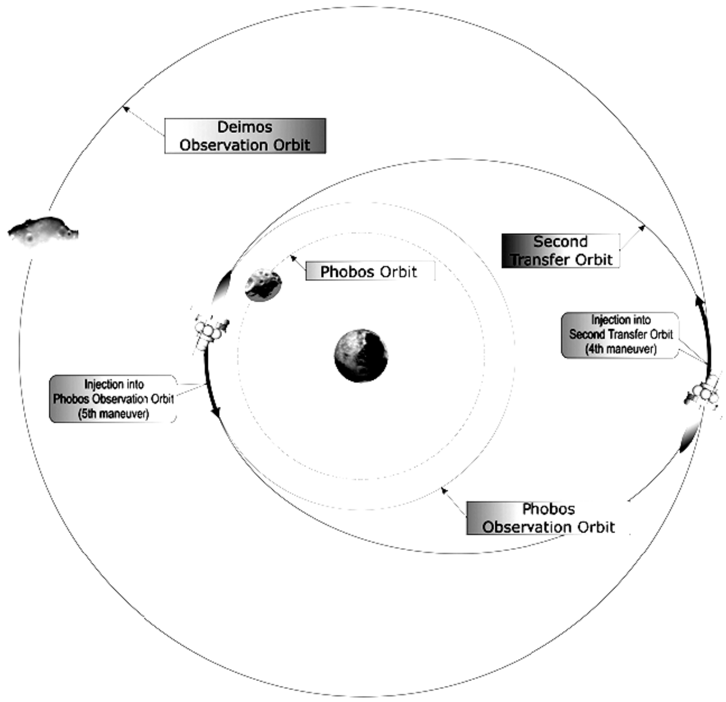

Fig. 2. Maneuvering scheme for injection into Phobos observation orbit

Рис. 2. Схема проведения маневров для выхода на орбиту наблюдения Фобоса

2.1.3 Rendezvous and landing on Phobos. Landing of the spacecraft on Phobos is carried out autonomously. To ensure the landing conditions, the spacecraft must be brought to a point that is located at an altitude of no more than 60 km above the proposed landing area. When preparing the spacecraft for landing and during landing, the following requirements must be met:

– a few days before the start of the landing session, a television image of the landing area should be obtained;

– the angle of the Sun – Phobos – SC should be in the range from 20° to 70° during the landing session;

– joint radio visibility of the spacecraft from the tracking stations in Ussuriysk and Medvezhikh Lakes;

– radio visibility of the Earth in the permissible range of drive angles of a high gain antenna (HGA);

– a forecast of the spacecraft's motion relative to Phobos at the time of departure from the QSO with errors not exceeding 3 km in position and 1 m/s in speed;

– the operability of the main on-board systems providing landing should be checked before the start of the landing session;

– the possibility of repeating the landing session is provided if the maneuver of the descent from the QSO has not been started.

The landing scheme on Phobos involves the use of a trajectory consisting of four sections:

– exit from the QSO;

– flight from the QSO to a point located above the landing area;

– vertical descent;

– precision braking.

The section of the flight from the QSO to the point located above the landing area begins with the maneuver of descent from the QSO and ends when it reaches the given point. Trajectory corrections are provided during the flight.

When the spacecraft moves in the vertical descent section, the simplest method of compensating the horizontal velocity component is used. If it exceeds the threshold value, the engine is turned on, which compensates for it. At the same time, a position error is accumulated in the horizontal plane. Therefore, the descent velocity must be large enough so that a large error is not accumulated during the descent.

In the area of precision braking, the vertical velocity component is gradually extinguished to a value which allows the contact with the surface: 1.5–2 m/s. In this case, the lateral component of the velocity vector should not exceed 1 m/s.

2.2 Return to Earth

2.2.1 Start from Phobos. The takeoff and exit of the RV to the AMS basic orbit is carried out according to the following scheme:

1. During the period of RV being on the surface of Phobos, the orientation of its longitudinal axis and, if possible, the coordinates of the landing point are specified. Relevant data is transmitted to Earth.

2. At a given time, the RV is separated from the landing module in the direction of its longitudinal axis with a speed of ~ 1 m/s and begins a passive flight maintaining the initial orientation of the longitudinal axis, stabilizing with the help of stabilization engines (SE).

3. After 50–60 seconds of a passive stabilized flight, the propulsion system of the RV is switched on, with the help of which additional delta-velocity maneuver is performed in the same direction to a speed of ~ 10 m/s.

4. Passive flight (stabilized in the same direction) during 1000 s to a distance of ~ 10 km from the Phobos surface.

5. The search for the Sun is carried out, and the longitudinal axis is brought to it.

6. A speed impulse of 20 m/s is emitted to the propulsion system of the RV in the direction of the Sun, after which the RV is in the intermediate orbit, hereinafter referred to as take-off, which touches the Phobos orbit at the take-off point and is 300–350 km away from it at the opposite point. Starting from entering this orbit and beyond, with the exception of short periods when active maneuvers and corrections are carried out, the RV makes a flight, being guided by the longitudinal axis +X on the Sun. The RV can take up to 10 days in the take-off orbit, after which there is a high probability of collision with Phobos, so before that time it should be transferred to the base orbit.

7. Passive flight for about three days, during which the Earth's connection with the RV is established, trajectory measurements are made, the parameters of the received AMO orbit are specified, the KPI is calculated and planned for the next maneuver to complete the launch to the basic orbit.

8. At the estimated time, the RV rotates in the required direction, and an impulse of ~ 20 m/s is generated, after which the RV is in the base orbit and again switches to the solar orientation.

Taking into account the spacecraft landing site (on the back side of Phobos), a circular orbit is chosen as the base orbit, the height of which is 300–350 km lower than the height of the Phobos orbit, and the orbital period is 7.23 hours, i.e. approximately 26 minutes less than the Phobos orbit period.

2.2.2. Transition to return trajectory and Mars-Earth flight. Tab. 4 shows the main characteristics of the return trajectories. The Mars-Earth flight time is from 6 to 11 months, depending on the version.

The transition of the RV from the base orbit to the flight trajectory to Earth, as well as the ascent to the observation orbit, is carried out according to a three-pulse scheme, but only in the reverse sequence. It includes the following elements:

– acceleration with the help of the propulsion system of the RV into a transitional three-day orbit with a pericenter radius equal to the radius of the base orbit, by the apses line lying in the plane of the Phobos orbit;

– passive flight in the third transitional orbit (at least ~ 5 turns or ~ 15 days) with trajectory measurements from the Earth and refinement of its parameters;

– apogee maneuver with the help of the propulsion system of the RV to lower the height of the pericenter to 1000-500 km and change the inclination to the value required for subsequent acceleration to the Earth (this orbit is called the pre-launch);

– passive flight in a prelaunch orbit (also at least ~ 5 turns or ~ 15 days) with trajectory measurements from the Earth and refinement of its parameters;

– acceleration on the selected date to the trajectory of departure to Earth, the beginning of the flight of the returned vehicle to Earth.

The total characteristic velocity of the transition from the base orbit to the Earth return trajectory for the considered options is from 1900 to 2450 m/s. The scheme of the take-off section is shown in fig. 3.

Navigation on the return trajectory should provide direct entry into the Earth's atmosphere and landing in a given area with the subsequent search and evacuation of DM. To solve this problem, five spatial corrections will be required on the flight path. The total correction impulse will not exceed 90 m/s.

Conclusion. The main modules of the spacecraft are: flight and landing platform, return vehicle and descent vehicle. The proposed design options for the propulsion systems of the FLP and RV modules make it possible to implement the developed trajectories, while ensuring the regular operation of the target equipment and conducting a series of experiments for a given period of active existence.

1. The developed flight scheme satisfies all the requirements and restrictions imposed on the functioning of the spacecraft, and provides the possibility of a successful expedition for the range of launch dates in 2025–2035.

2. When designing the trajectories of the spacecraft, the restrictions on the operation of on-board systems and NKU are taken into account. The analysis of the ballistic conditions of the spacecraft functioning at all stages of the flight is carried out.

3. The length of the parts varies significantly depending on the launch date. The total duration of the expedition will be about 3 years.

4. The propulsion system of the FLP should provide a total characteristic speed of maneuvers of 3.2 km/s, the propulsion system of the RV – 2.45 km/s.

An alternative version of the spacecraft with reduced mass provides the solution to the expedition problem by replacing the Deimos observation orbit from circular to synchronous elliptical and saving about 500 m/s.

Table 4

The main characteristics of the Mars-Earth flight trajectories

Characteristics	Version				
	2026	2028	2030	2033	2035
Start date from the orbit of the AMS	06.09.2028	01.11.2030	28.01.2033	08.05.2035	12.07.2037
Flight duration, days	339	247	217	198	269
Arrival to Earth	11.08.2029	07.07.2031	02.09.2033	22.11.2035	08.04.2038
Asymptotic departure speed from Mars, km/s	2.481	2.406	2.391	2.952	3.541
Characteristic acceleration speed with OISM	0.726	0.692	0.685	0.963	1.298
Asymptotic approach speed to the Earth, km/s	4.441	5.620	4.050	3.031	3.522
Absolute velocity of entry into the Earth's atmosphere, km/s	11.932	12.420	11.793	11.483	11.625

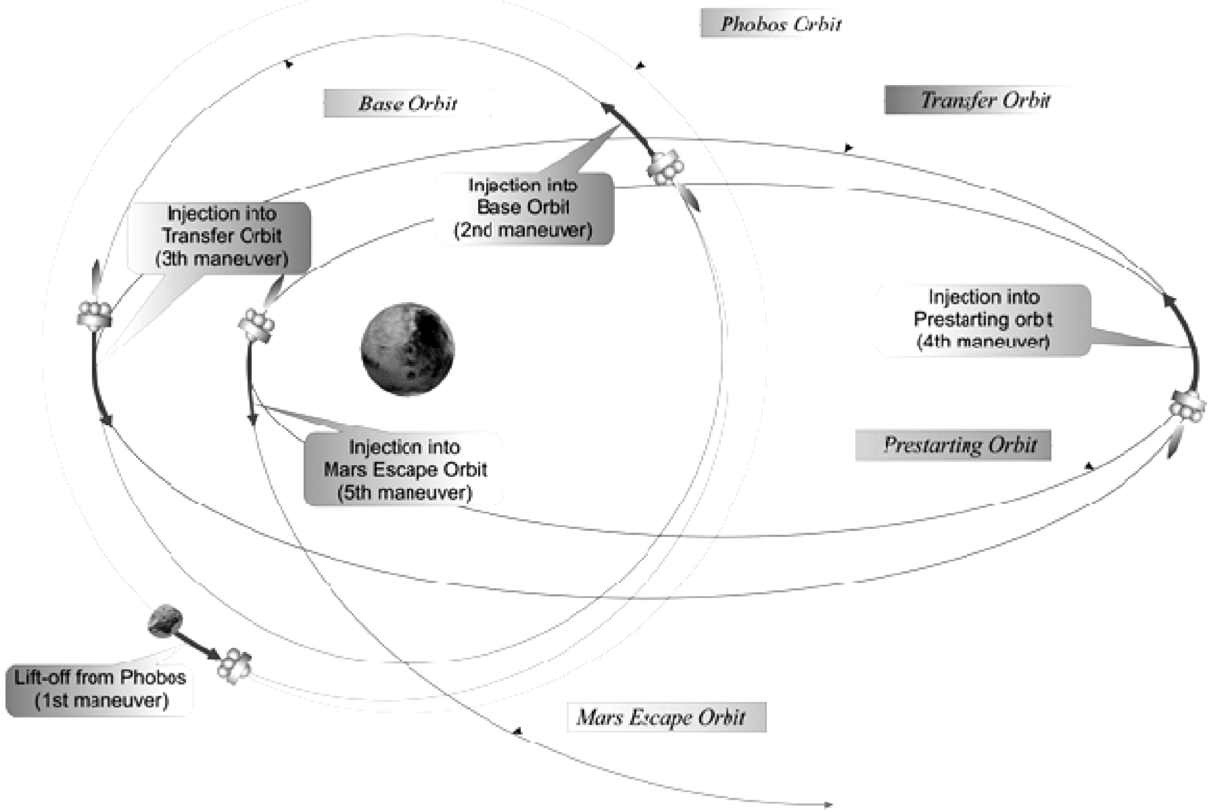

Fig. 3. Injection scheme to back-to-Earth trajectory

Рис. 3. Схема выхода на траекторию полета к Земле

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