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Trajectory Analysis and Possible Architecture for Manned Venus and Mars Flyby Mission in 2021-2023

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Abstract

A possibility of carrying out a double flyby mission with Venus and Mars encounters for the crew of 2 with Earth departure in 2021 is investigated. Free-return Earth-Venus-Mars-Earth trajectories with a single chemical propulsive maneuver at Earth are found for different launch dates from 11/17/2021 to 01/20/2021 and Mars flyby altitudes of 200…1500 km. Three key trajectory parameters including delta-V requirements for the trans-Venus injection maneuver, time of flight and Earth's atmosphere reentry speed are analyzed. A possible mission architecture requiring 6 launches and on-orbit assembly in low Earth orbit is developed with an emphasis on making maximum use of already flown and well-proven technologies. The transportation system including a flyby vehicle, two upper stages, two fuel tankers and six launch vehicles is outlined. The launch vehicles include one Ariane-5, three Proton-M, one Angara-A5, and one Soyuz-2 rockets. The mass of the departure vehicle stack in parking LEO is estimated to be 96300 kg with the mass of the flyby vehicle of 21500 kg. A nominal trajectory for the mission is selected with Earth departure on 11/17/2021, Earth return on 06/27/2023, a total flight time of 588 days, Mars and Venus flyby altitudes of 300 km and 12000 km, respectively, and an Earth's atmosphere reentry speed of ~ 13 km/s. A launch window with a 7-day width is defined. A concept of operations is developed. The results of this work were presented as part of Team Russia's project at the International Gemini Mars Design Competition organized by the Mars Society in 2015-2016.

Keywords: interplanetary trajectory design, mission design

Nomenclature

h	altitude, km
V	velocity, km/s
V_∞	hyperbolic excess velocity, km/s
ΔV	change in velocity, km/s

Subscripts:

Arr	Earth arrival
Dep	Earth departure
Entry	Earth's atmosphere reentry
M	Mars
TVI	trans-Venus injection
V	Venus
α	apogee
π	perigee

Acronyms/Abbreviations

AEB	auxiliary equipment bay
ATV	Automated Transfer Vehicle
DN-a	active docking node
DN-p	passive docking node
DVS	departure vehicle stack
ECLSS	Environmental Control and Life Support System
EVME	Earth-Venus-Mars-Earth
FT	fuel tanker
FV	flyby vehicle
HLUS	heavy-lift upper stage
ISS	International Space Station
KVTK	"oxygen/hydrogen heavy class" upper stage
LEO	low Earth orbit
LM	living module
LW	launch window
PS	propulsion system
SM	service module
TOF	time of flight
TS	transportation system
TVI	trans-Venus injection

1. Introduction

Sending humans to Mars has been a long-time dream for space enthusiasts on Earth and a challenge to the governments of the countries developing space technologies. Today, manned missions to Mars are official part of the national space programs of the US, Europe, Russia, and China. Apart from that, a number of private organizations have expressed their interest in developing technologies that will contribute to human exploration, settling on and even terraforming the red planet [1]. Though sophisticated manned missions to

Mars with landing on its surface are still considered to be quite a long-term goal, in the recent years there has been an interest in designing a manned Mars flyby mission that could be flown as early as in the nearest decade.

A mission of that kind would be a precursor to more complex Mars endeavors demonstrating some crucial developments necessary for future Mars missions which will include orbiting the red planet and landing on its surface [2]. Some examples of these new developments are a long duration ECLSS capability, human radiation protection during interplanetary space journeys, long term human flight operations, and the ability to perform a safe landing after reentering the Earth's atmosphere at higher velocities than previous missions required. At the same time, because of the overall simplicity and much lower energetic (delta-V) requirements, there is hope that a flyby mission, relying heavily on existing and near-term technologies, could become a viable and relatively cheap option for an early human mission to Mars. In fact, Tito et al. [2] suggest that, in the times of emerging commercial space sector, such a mission may prove to be cheap enough to be financed privately.

In 2013, Tito et al. [2] studied a possibility to launch a human Mars flyby mission, dubbed "Inspiration Mars", as early as in 2018. The mission was based on a "fast" free-return trajectory first investigated in [3] which required a single trans-Mars injection maneuver and returned the crew back to Earth after performing a close Mars flyby in a relatively short time of 501 days. The crew of two (a man and a woman) were to fly in a SpaceX Dragon-like capsule weighing ~ 10,000 kg. The capsule was to be launched into the trans-Mars interplanetary trajectory by the perspective Falcon Heavy launch vehicle.

In 2014, Hughes et al. [4] explored another class of fast Mars free-return trajectories which used a Venus gravity assist. As a result, an alternative launch opportunity for an early Mars flyby mission with favorable trajectory characteristics was found in 2021.

This paper investigates the possibility of carrying out a double flyby mission (with Venus and Mars encounters) in 2021 more closely. Free-return Earth-Venus-Mars-Earth trajectories using a single chemical propulsive maneuver at Earth are found for different launch dates and Mars flyby altitudes. The corresponding requirements for the transportation system in terms of delta-V, the Earth's atmosphere reentry speed and flight duration capabilities are defined. A possible transportation system to conduct the mission for the crew of 2 is outlined with an emphasis on making maximum use of already flown and well-proven technologies. At the same time, an option with more comfortable living conditions for the crew is considered (compared to the original Inspiration Mars proposal) – the flyby vehicle includes not only the landing capsule (with the mass of 7,000 kg) but also a living module with the mass of 14,500 kg. The launch window and the nominal interplanetary trajectory for the mission are defined. A concept of operations for the mission requiring 6 launches and on-orbit assembly in low Earth orbit (LEO) is developed.

The results of this work were presented as part of Team Russia's project [5] at the International Gemini Mars Design Competition organized by the Mars Society in 2015-2016 [6, 7].

2. Methods

2.1 Trajectory Computation

A free-return trajectory selected for the mission uses a Venus flyby before the Mars encounter. The trajectory requires a single propulsive maneuver at Earth (trans-Venus injection – TVI), includes two gravity assists and ends with the descent module entering the Earth's atmosphere. Chemical propulsion is used for the TVI maneuver.

The flight path consists of the following segments: (1) *Earth departure* – the TVI maneuver performed in LEO sets the flyby vehicle (FV) on the Earth-Venus transfer trajectory; (2) *Earth-Venus heliocentric leg*; (3) *Venus flyby* – the gravity assist from Venus sets the vehicle on a Venus-Mars transfer trajectory; (4) *Venus-Mars heliocentric leg*; (5) *Mars flyby* – the gravity assist from Mars sets the vehicle on a free-return Mars-Earth trajectory; (6) *Mars-Earth heliocentric leg*; (7) *Earth return* – the FV approaches the Earth on a geocentric hyperbolic trajectory; the descent module separates from the FV, re-enters the Earth's atmosphere and performs landing.

In order to compute Earth-Venus-Mars-Earth (EVME) trajectories an approximate computational model based on the following assumptions was used:

- Sun gravitational force was assumed to be the only force affecting the FV on the heliocentric legs (2), (4), (6) of the trajectory (i. e. heliocentric legs were approximated by conic solutions);
- planets' spheres of influence were considered to be points [8, 9], i. e.: each heliocentric leg was assumed to begin in the center of the departure planet and to end in the center of the arrival planet; the gravity assists (3), (5) were approximated by the instantaneous turns of the corresponding hyperbolic excess velocity vectors; the time of flight was considered to be the sum of the three heliocentric legs' durations;
- the TVI delta-V for the Earth departure (1) and the reentry speed for the Earth return (7) were derived from the corresponding hyperbolic excess velocities at the Earth's sphere of

influence for a 300 km initial circular LEO altitude and a 100 km reentry altitude;

- NASA JPL DE430 ephemerides were used for computing positions and velocities of the planets [10, 11];

The PTC Mathcad software was used to compute flyby trajectories with the problem of finding a trajectory for fixed launch date and Mars flyby altitude having been transformed into a problem of solving a system of 6 nonlinear equations with 6 variables. (For these two parameters fixed the other trajectory parameters such as hyperbolic excess velocities, time of flight, Venus flyby altitude, etc. are explicitly derived from the solution of the nonlinear system.)

In order to estimate gravity losses for the TVI maneuver performed by the elements of the transportation system designed the equations of motion of the departure vehicle stack (DVS) were numerically integrated for the Earth departure segment of the trajectory. A simplified force model was used which included only the Earth's central gravitational force and the DSV thrust; the thrust vector was considered to be directed along the DVS velocity vector.

2.2 Transportation System Design

The main strategy while designing the transportation system (TS) was to use existing technologies or their near-term modifications wherever it was possible. For some of the elements (such as launch vehicles), existing vehicles were included in the TS in their entirety; for other elements of the TS, parts of existing vehicles or their modifications were used as their "building blocks" (for example, "building blocks" for the living module of the FV included two modified sections of the ATV as well as a modified compartment of an ISS module).

Modifications expected to be made to the existing prototypes included changing their size (to fit in the payload fairings of the selected launch vehicles or to enlarge living space for the crew), adding additional equipment and features (such as docking nodes, refueling systems, or measures to increase vehicle life duration) or, vice versa, excluding some elements from the design (such as excessive fuel tanks).

Mass estimations for the TS elements were obtained by scaling from the corresponding prototypes and took into account the modifications mentioned above. The Tsiolkovsky formula was used to estimate the amount of fuel needed for on-orbit assembly operations, the theoretical delta-V capability for performing the TVI maneuver, and the delta-V reserve for possible post-TVI trajectory corrections and attitude control*.

3. Calculation

3.1 Interplanetary Trajectory Analysis

Three main characteristics of an Earth-Venus-Mars-Earth (EVME) trajectory determine the possibility of its practical implementation: (1) *delta-V* required for performing the TVI which determines the launch scheme and launch vehicles selection and is limited by the capabilities of existing launching technologies; (2) *time of flight* (TOF) which is limited by existing capabilities of human health preservation under the conditions of long-duration space flights; (3) *atmosphere reentry speed* limited by existing landing technologies.

In order to explore how these parameters change depending on the launch date and Mars flyby altitude, various EVME trajectories were found for the opportunity of 2021. The computations were conducted for Mars flyby altitude range of 200…1500 km and launch dates from 11/17/2021 to 01/20/2021. The altitude of the initial circular LEO was assumed to be 300 km; the Earth's atmosphere reentry altitude 100 km.

The results are presented in Fig. 1-5. Fig. 1, 2 show the hyperbolic excess velocities for Earth departure ($V_{\infty,Dep}$) and Earth arrival ($V_{\infty,Arr}$) as functions of launch date. The corresponding TVI delta-V (ΔV_{TVI}) and atmosphere reentry speed (V_{Entry}) are shown in Fig. 3, 4. Fig. 5 illustrates the time of flight as a function of launch date. All the graphs are shown for several Mars flyby altitudes ($h_{\pi M}$).

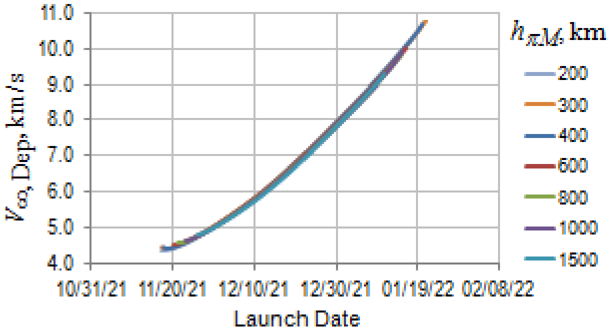

Fig. 1. Hyperbolic excess velocity (Earth departure)

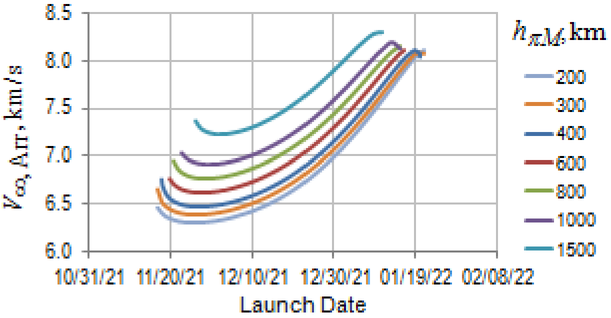

Fig. 2. Hyperbolic excess velocity (Earth arrival)

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* Estimation of the delta-V needed for the trajectory corrections and attitude control was not part of this work. The corresponding numbers reflect specifics of the TS design and are subject to further investigation.

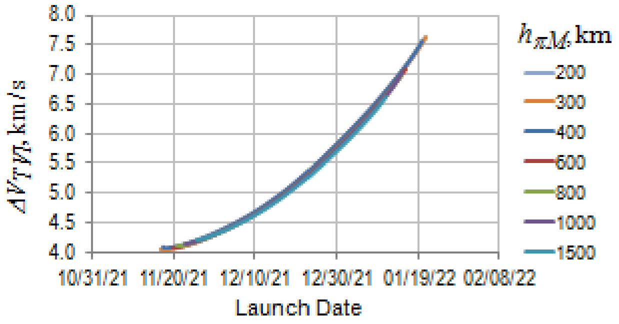

Fig. 3. TVI delta-V (LEO altitude 300 km)

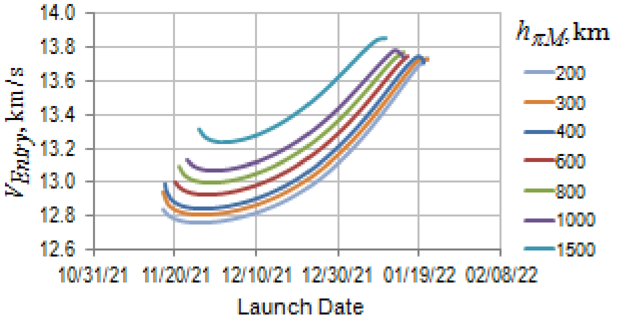

Fig. 4. Atmosphere reentry speed (at 100 km altitude)

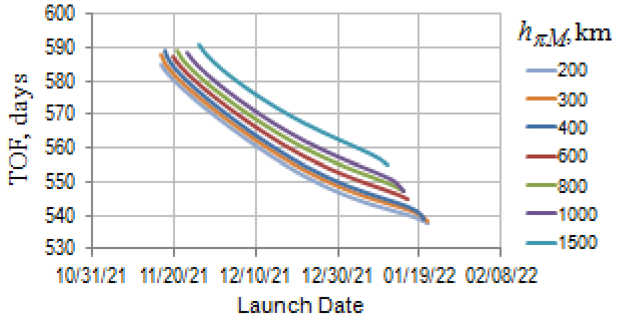

Fig. 5. Time of flight

Fig. 1, 3 show that $V_{\infty,Dep}$ and related ΔV_{TVI} have a minimum at the beginning of the date range and then increase (later launch dates correspond to greater values of $V_{\infty,Dep}$ and ΔV_{TVI}). However, these characteristics are very little dependent on the Mars flyby altitude (for the given range of $h_{\pi M}$).

$V_{\infty,Arr}$ and related V_{Entry} (see Fig. 2, 4) also have a minimum at the beginning of the date range though this minimum is shifted to the right from that one of $V_{\infty,Dep}$ and ΔV_{TVI}. At the same time $V_{\infty,Arr}$ and V_{Entry} (unlike $V_{\infty,Dep}$ and ΔV_{TVI}) are also sensitive to the Mars flyby altitude – lesser $h_{\pi M}$ correspond to lesser $V_{\infty,Arr}$ and V_{Entry}.

Flight time is maximal at the beginning of the date range and decreases for later dates. At the same time, lesser Mars flyby altitudes also correspond to lesser flight times.

Therefore, the trajectories with the lowest $h_{\pi M}$ possible are preferable for the date range considered. The minimum $h_{\pi M}$ is limited by the altitude of the Martian atmosphere and safety of the crew in case of failure to correct possible deviations from the nominal trajectory. The safety issues require further investigation. For this paper the flyby altitude of 300 km was assumed to be safe.

The choice of optimal launch date is not so obvious as in this case decreasing $V_{\infty,Dep}$ (ΔV_{TVI}) means increasing the time of flight. However, decreasing $V_{\infty,Dep}$ seems to be preferable to decreasing TOF since essential increase of $V_{\infty,Dep}$ yields in not so essential decrease of TOF. Thus, it is safe to assume that optimal launch dates will be located closer to the beginning of the date range.

The transportation system for the mission should ensure the following capability: $V_{\infty,Dep} > 4.4$ km/s ($\Delta V_{TVI} > 4.04$[†] km/s), V_{Entry} ~ 12.8…13 km/s, the TOF up to 600 days.

3.2 Transportation System Design

A mission architecture which requires 6 launches and on-orbit assembly was devised to ensure the capability defined by the interplanetary trajectory analysis. The transportation system for this architecture includes the following components (see Fig. 12): *flyby vehicle* (FV) which consists of a living module (LM) docked with a modified Soyuz spacecraft; *2 upper stages* which perform the TVI maneuver (the upper stages docked with the FV form the departure vehicle stack (DVS)) including the KVTK[‡] oxygen/hydrogen upper stage and the heavy-lift upper stage (HLUS); *2 fuel tankers* (FT) for the HLUS and *6 launch vehicles* (1 Ariane-5, 3 Proton-M, 1 Angara-A5, and 1 Soyuz-2).

3.2.1 Flyby Vehicle

As it was mentioned earlier, the flyby vehicle consists of a living module docked with a modified Soyuz spacecraft. The living module serves as living quarters for the crew during the flight. Main functions of the Soyuz spacecraft are crew delivery from Earth to the departure vehicle stack waiting in initial parking LEO and crew return to Earth after the flyby as well as in case of emergency prior or during the TVI maneuver.

[†] Ideal delta-V capability, without gravity losses taken into account.

[‡] KVTK is Russian abbreviation for "oxygen/hydrogen heavy class".

Both Soyuz and LM are used for trajectory corrections and attitude control.

The living module consists of three compartments – the spherical transfer compartment, the crew compartment and the service module (see Fig. 6). The transfer compartment is similar to those of Zarya and Zvezda modules of the ISS. The compartment has an observation dome which is analogous to the ISS Cupola but is smaller and lighter. There is a passive docking node for Soyuz and a high gain antenna mounted on the compartment. The Crew Compartment is based on the ATV's Integrated Cargo Carrier pressurized compartment but it is 1 meter longer to give the crew more space inside. The Service Module of the LM is similar to that one of the ATV but has less fuel tanks and is shorter than the ATV's equivalent. It also has a passive docking node between its engines for the heavy-lift upper stage to dock and larger solar arrays. The total mass of the LM in the parking orbit is 14500 kg.

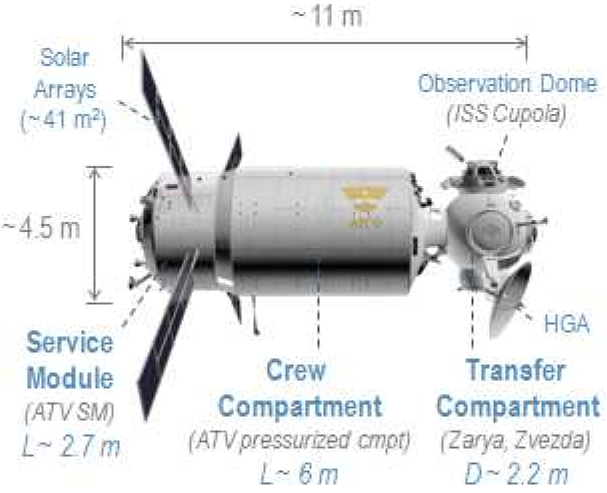

Fig. 6. Living module

Modified Soyuz spacecraft is based on Soyuz MS. The major and most challenging modification is to be made to its descent capsule which is to ensure safe entry, descent and landing on Earth for the atmospheric entry speeds of at least ~ 12.8…13 km/s[§]. Another major modification is extending the spacecraft's lifetime up to 600 days in docked condition. It also should be suited for the crew of two (instead of three). The mass of the modified Soyuz in the parking orbit is 7000 kg.

The FV mass budget is presented in Table 1. The cumulative mass of the transfer compartment, the crew compartment and the service module of the LM is assumed to include the mass of the following flight systems: power, thermal control, communication, computer, motion and navigation. Fuel reserve provides delta-V capability of ~ 90 m/s and 100 m/s for the LM and Soyuz respectively, for trajectory corrections and attitude control. The propulsion system parameters for the LM and Soyuz are shown in Table 2.

Table 1. Mass budget of the FV in the parking LEO

Item	Mass, kg
Living module, incl.	**14500**
transfer compartment	1000
crew compartment	4400
service module	2800
mass reserve (ECLSS, crew supplies)	5700
fuel reserve	600
Soyuz, incl.	**7000**
fuel reserve	660
Total FV Mass	**21500**

Table 2. LM and Soyuz propulsion systems

Item	Thrust, N	Exhaust Velocity, m/s
LM main engines	4x490	3200
Soyuz main engine	2940	3200

3.2.2 Upper Stages and Fuel Tankers

Two upper stages are used to perform the TVI maneuver – the KVTK oxygen/hydrogen upper stage and the heavy-lift upper stage (HLUS). The HLUS is launched into initial parking LEO partially fueled; two fuel tankers are used to deliver additional fuel to the HLUS needed to perform the departure maneuver.

The KVTK (see Fig. 7, Tables 3, 4) is the only piece of a new technology in the transportation system designed. The upper stage is currently under development at the Khrunichev State Research and Production Center in Russia with its first launch planned for 2021. For the flyby mission proposed it should be modified to have an active docking node to dock with the heavy-lift upper stage.

Fig. 7. KVTK upper stage

Table 3. Mass budget of the KVTK in the parking LEO

Item	Mass, kg
Dry mass	3500
Fuel mass	19000
Total mass	**22500**

[§] There is hope that this modification can be made in the near future as under the Soviet Zond program, which explored possibilities of manned circumlunar loop flights in 1969-1970, two descents of Soyuz-type entry modules were successfully performed for the atmosphere entry speeds about 11 km/s.

Table 4. KVTK propulsion system

Item	Thrust, N	Exhaust Velocity, m/s
Main engine	73575	4690

The heavy-lift upper stage (see Fig. 8, Tables 5, 6) is based on the third stage of the Proton-M rocket. The modifications are to be made to its service module to increase its lifetime and add the refueling capability. An active docking node should be added to the construction to ensure docking with the living module. The main engine of the third stage of Proton-M can be ignited only once, so an auxiliary equipment bay (AEB) with its own propulsion system is to be used to transfer the upper stage from its initial launch orbit to the parking orbit where the stage is docked to the LM. This additional propulsion system is assumed to be based on the ATV propulsion system with the same four main engines and 28 attitude control thrusters. The auxiliary bay is also to have a passive docking node for the fuel tankers and the KVTK. The bay is jettisoned before the ignition of the HLUS.

Fig. 8. Heavy-lift upper stage (top – HLUS with AEB; bottom – HLUS without AEB)

Table 5. Mass budget of the HLUS in the parking LEO

Item	Mass, kg
Dry mass, incl.	5700
AEB (to be ejected)	1000
Fuel mass	46600
Total mass	**52300**

Table 6. HLUS propulsion system

Item	Thrust, N	Exhaust Velocity, m/s
Main engine	582000	3180
Propulsion system on AEB	4x490	3200

The fuel tankers (see Fig. 9, Tables 7, 8) are mainly based on the ATV design, but they have smaller diameters to fit under the Proton's fairing. The Integrated Cargo Carrier is to be modified to contain tanks with fuel for the heavy-lift upper stage. At the same time, the tanker's own propulsion system will contain less fuel – just to lift the tanker's initial launch orbit and provide the docking with the HLUS.

Fig. 9. Fuel tanker

Table 7. FT mass budget

Item	Mass, kg
Dry mass	7500
Fuel for HLUS	15000
Fuel mass	500
Total mass	**23000**

Table 8. FT propulsion system

Item	Thrust, N	Exhaust Velocity, m/s
Main engines	4x490	3200

3.2.3 On-orbit Assembly

On-orbit assembly for the mission requires 6 launches. Each launch vehicle delivers its payload to initial circular orbit with the altitude of 175 km and the inclination of 51.6°. The payload then uses its own engines to transfer itself to the parking orbit where the assembly takes place. The parking orbit is a circular orbit of 300 km altitude and 51.6° inclination.

The assembly is implemented in the following order: (1) the LM is the first to be launched atop the Ariane-5 rocket; (2) after the LM is in the parking orbit, the partially fueled HLUS is launched atop the Proton-M rocket; the HLUS docks with the LM and waits for the first FT to arrive; (3) another Proton-M launches the tanker which docks to the HLUS, has the fuel transferred to the upper stage, undocks, performs a braking maneuver, de-orbits and burns up in the Earth's atmosphere; (4) the second tanker delivers the fuel in the same way using the third Proton-M rocket; the heavy-lift upper stage is now fully fueled; (5) the modified Soyuz with the crew is launched atop the Soyuz-2 rocket; it docks to the LM; (6) the last launch is the KVTK upper stage[**] atop the Angara-A5 launch vehicle; the KVTK docks to the HLUS, and the

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[**] The KVTK has to be launched last due to the short storage time of its fuel.

assembly of the DVS is completed. The stack is ready to perform the TVI maneuver.

3.3 Trans-Venus Injection Maneuver

The TVI maneuver is divided into two propulsive burns (see Fig. 10): (1) the KVTK performs the first burn which lifts the departure vehicle stack from the parking LEO to an elliptic transfer orbit; the KVTK is then jettisoned; (2) in the perigee of the transfer orbit the HLUS performs the second burn injecting the HLUS-FV stack into a hyperbolic departure trajectory; after the burn the HLUS is jettisoned.

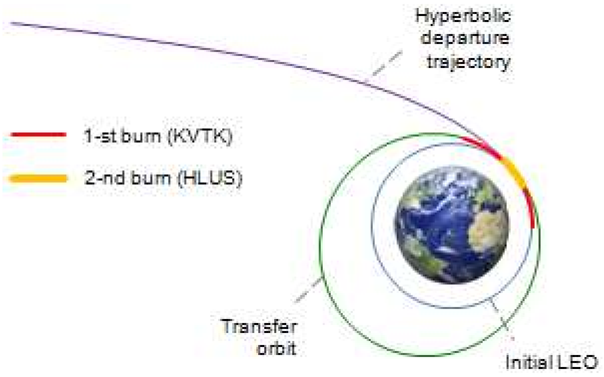


Fig. 10. Earth departure trajectory

The total delta-V capability of the DVS is 4194 m/s which theoretically enables launching the FV to a departure trajectory with a hyperbolic excess velocity of 4770 m/s. However, due to gravity losses the excess will be lesser. To estimate the influence of gravity losses on the value of $V_{\infty,Dep}$ provided by the DVS, the equations of motion of the vehicle stack were numerically integrated for the Earth departure segment of the trajectory. The results are shown in Table 9.

Table 9. Earth departure trajectory

TVI phases	ΔV, m/s	Orbit	Orbit parameters
before TVI	-	circular LEO	$h = 300$ km
KVTK burn	1013	elliptic transfer orbit	$h_\pi = 424$ km $h_\alpha = 5230$ km
HLUS burn	3181	hyperbolic departure trajectory	$h_\pi = 437$ km $V_{\infty,Dep} = 4625$ m/s*

* this value refers to the ideal delta-V (i.e. delta-V which does not include gravity losses) of 4140 m/s.

Therefore, with gravity losses having been taken into account the maximum hyperbolic excess velocity provided by the DVS for initial 300 km LEO is 4625 m/s (the gravity losses are ~ 55 m/s).

3.4 Earth Return

The final segment of the free-return FV trajectory is a geocentric hyperbolic orbit that intersects Earth's atmosphere. On approach to Earth the Soyuz vehicle separates from the living module, performs trajectory corrections and maneuvers to the attitude necessary for the further descent module separation and its meeting the entry requirements. After the separation the descent module with the crew enters the atmosphere and performs landing on the surface. The entry speed for the descent module is ~ 12.8…13 km/s.

Exploring possible landing trajectories was not the aim of this paper. It was assumed that a trajectory with two atmospheric entries [12, 13] can be used for the descent – in this case after the first entry the descent module exits the atmosphere on a suborbital trajectory and then reenters and performs its final descent to the ground. Such type of an atmospheric descent for Soyuz-type entry modules was first successfully performed under the Soviet Zond program for circumlunar missions (atmospheric reentry speed ~ 11 km/s).

3.5 Launch Window and Nominal Trajectory

For the trajectories with 300 km Mars flyby altitude, launch windows (LW) were determined in relation to constraints on $V_{\infty,Dep}$ (ΔV_{TVI}) and V_{Entry} (see Table 10).

Table 10. Launch windows for different constraints

Constraints	LW (2021)	LW width, days
$V_{Entry} \leq 12.9$ km/s	11/18 – 12/14	27
$V_{Entry} \leq 13$ km/s	11/17 – 12/21	35
$V_{\infty,Dep} \leq 4625$ m/s	11/17 – 11/23	7

The third row of the table refers to the delta-V limitations of the departure vehicle stack. The table shows that the delta-V capability of the DVS is the main constraint defining the width of the launch window. Combination of the constraints on $V_{\infty,Dep}$ and V_{Entry} (rows 2 and 3 in Table 10) yields in the following characteristics of the launch window: launch dates 11/17/2021…11/23/2021; window width of 7 days; $V_{\infty,Dep}$ = 4.400-4625 m/s; V_{Entry} = 12.8…13 km/s; TOF = 579…588 days.

Table 11 contains the characteristics of the nominal trajectory corresponding to the launch date of 11/17/2021 (the beginning of the launch window). Fig. 11 shows the trajectory projection on the plane of the ecliptic. Nominal flight time for the mission is 588 days; the Mars flyby altitude 300 km; the Venus flyby altitude ~ 12000 km.

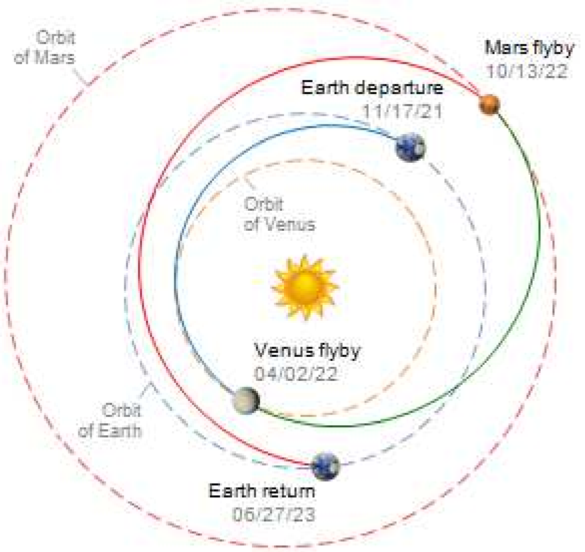


Fig. 11. Nominal interplanetary trajectory (projected on the plane of the ecliptic)

Table 11. Nominal interplanetary trajectory

Leg	TOF, day	Inclination*, deg
Earth-Venus	136.7	5.8
Venus-Mars	193.8	2.3
Mars-Earth	257.5	0.1

* Inclination of the trajectory plane to the ecliptic

4. Results and Discussion

This paper investigated a possible architecture for a double flyby mission with Venus and Mars encounters in 2021 which would use existing technologies or their near-term modifications. The mission summary is presented in Table 12. The transportation system and concept of operations are shown in Fig. 12, 13.

Table 12. Double flyby mission with Venus and Mars encounters: summary

Parameter	Value
Launch window (year 2021)	11/17…11/23
Launch window width	7 days
Earth departure (nominal)	11/17/2021
Venus encounter (nominal)	04/02/2022
Mars encounter (nominal)	10/13/2022
Earth return (nominal)	06/27/2023
Venus flyby altitude	1200 km
Mars flyby altitude	300 km
Time of flight	588 days
TVI delta-V	4100-4200 m/s
Atmospheric reentry speed	12.8…13 km/s
Number of launches	6
Mass of the departure vehicle stack (assembly LEO 300 km)	96300 kg
Mass of the flyby vehicle	21500 kg
Number of crew	2 persons

An attempt to make maximum use of yet existing and well-proven technologies was made. However, it proved to be impossible to completely avoid taking advanced developments into consideration. Two of such developments are worth paying special attention to.

One of the main problems of manned interplanetary space flights, which is still waiting for its solution, is the development of a descent module enabling safe crew return to Earth for atmospheric entry speeds exceeding the escape velocity. The mission architecture presented in this paper assumes that for early Mars missions this problem can be solved by modifying the Soyuz descent module. At the time of conducting calculations for the paper (late 2015 – early 2016), the idea behind choosing Soyuz was that it might be possible to modify the well-proven Soyuz technology in time for the double flyby mission in 2021 (considering the fact that in 1969-1970 Soyuz-type descent modules already performed landing on Earth with atmospheric entry speeds of about 11 km/s). Today, however, as the launch date for the 2021 opportunity is much closer, it seems unlikely that Soyuz can be modified in time.

One more piece of new technology which is crucial for the proposed mission architecture is the KVTK upper stage. This upper stage was chosen for the mission because of its high specific impulse which enables to accomplish the mission in 6 launches. Today, however, the KVTK is at the development stage. Its first launch is planned for 2021, i.e. for the same year as the mission launch date, which means that unforeseen delays regarding the KVTK development may result in missing the launch window which itself is quite short.

A possible way to deal with the problem of missing the launch window is to search for backup trajectories for the mission which would have similar or better trajectory parameters but correspond to later launch dates. According to [4], the next possibility of implementing a double flyby mission with similar parameters will occur approximately 32 years after the 2021 opportunity – which is likely to be too late for an early human mission to Mars. Therefore, a backup free-return trajectory would include only a single Mars gravity assist. A single Venus flyby trajectory can also serve as an interesting alternative to the Mars flyby option for an early interplanetary manned mission. Notably, the Venus flyby option has the advantage of considerably shorter flight times (~ 350-370 days [4]). At the same time, the Venus mission would still allow testing some crucial technologies necessary for future human interplanetary missions including those to Mars.

Finally, it would also be interesting to explore a possible mission architecture which would use a second upper stage similar to the HLUS instead of the KVTK. In this case, however, the mission would most probably require at least 7 or even 8 launches.

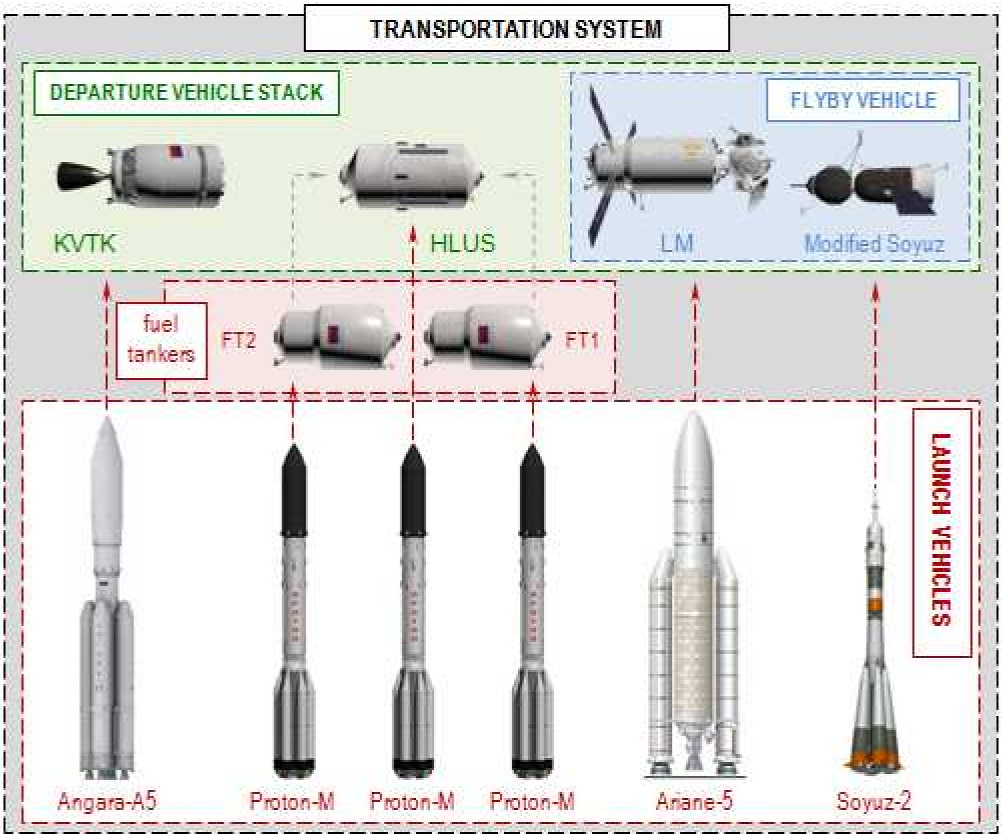
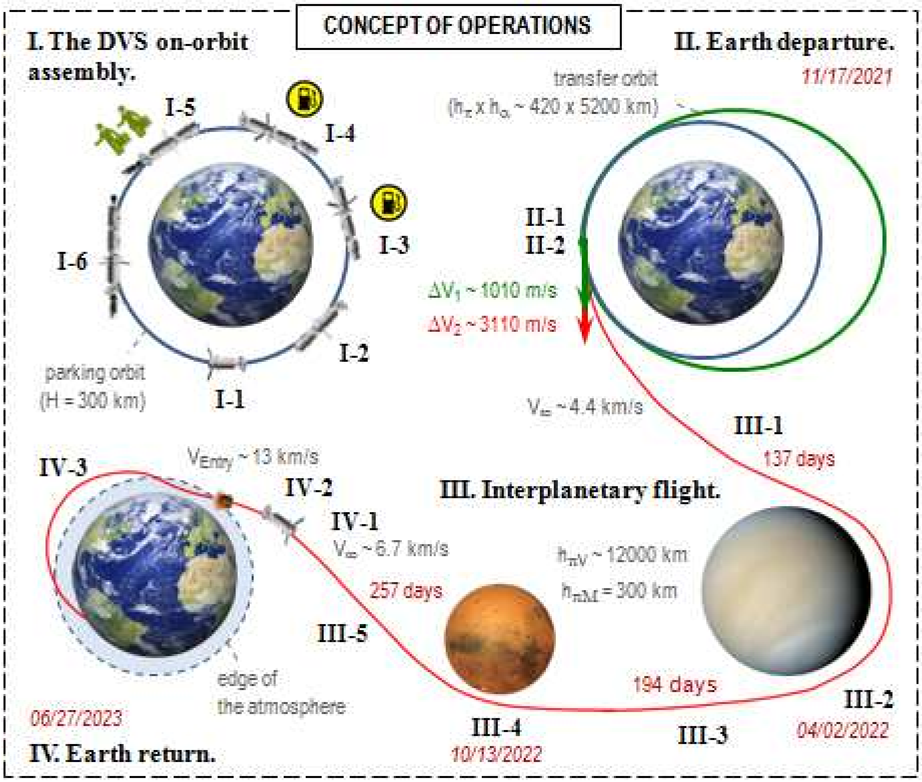


Fig. 12. Transportation system



I-1 LM launch into parking LEO
I-2 HLUS launch, docking to LM
I-3 First FT launch, fuel delivery to HLUS
I-4 Second FT launch, fuel delivery to HLUS
I-5 Soyuz launch, docking to LM
I-6 KVTK launch, docking to HLUS-FV stack

II-1 KVKT burn
II-2 HLUS burn

III-1 Earth-Venus trajectory leg
III-2 Venus flyby
III-3 Venus-Mars trajectory leg
III-4 Mars flyby
III-5 Mars-Earth trajectory leg

IV-1 Approaching Earth
IV-2 Soyuz separation from FV, descent module separation from Soyuz
IV-3 Atmospheric descent and landing

Fig. 13. Concept of operations

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