Launcher size optimization for a crewed Mars mission

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Abstract

Several proposals exist for human Mars mission architectures. An important question to resolve is to determine the most appropriate size of the launcher to minimize the costs, without compromising with risks, efficiency and future developments. Strategic choices are proposed. A fundamental choice is the direct to surface option, one that greatly simplifies the architecture of the mission and avoids a complex and costly LEO assembly of a giant vehicle. The second is aerocapture for Mars orbit insertion. The third is the choice of the EDL systems with highest TRL in order to minimize the risks of the mission and at the same time to avoid possible cost overruns due to qualification issues. Minimization is achieved for a crew of three. It is shown that an LEO capacity of the order of 100 to 110 tonnes is sufficient to carry out a Mars mission using 5 heavy launchers. This result is of particular interest for the countries currently developing super heavy launchers with such capacity, like the Starship and SLS in the USA, Long March 9 in China and similar developments in Russia. If Europe were also interested in the design of a super heavy launcher, it is shown here that it could be based on Vulcain or Prometheus engines. As the mission is rather simple and optimized with high TRL, the mission could be affordable. A roadmap is also suggested with appropriate preparatory missions for a human Mars exploration program.

Keywords: Human mission to Mars, super heavy launcher, Mars direct

Acronyms:

EDL: Earth, Descent and Landing
ERV: Earth Return Vehicle
IMLEO: Initial Mass in Low Earth Orbit
LEO: Low Earth Orbit
MAV: Mars Ascent Vehicle
NASA: National Aeronautics and Space Administration
TMI: Trans-Mars Injection

1. Introduction

Elon Musk recently proposed a new architecture for human missions to Mars [14]. It is based on a super heavy launcher and a single giant spaceship that could be sent directly to the surface of Mars, refuel using local resources and come back to Earth. The simplicity of the concept contrasts with the complex architectures proposed by NASA in 2009 and 2014 [7,8,9], in which it is suggested to assemble several giant spaceships in Earth orbit by means of numerous heavy launches. An important task is to determine the most appropriate launcher capacity for a mission to Mars, taking into consideration the development and production costs, the risks, the usefulness of the rocket for other missions, possible international collaboration and future trends. This question is especially important for all countries in which a super heavy launcher is under development for human missions beyond low Earth orbit, such

as the SLS and Starship in the US, the successor of the Yenisei program in Russia and Long March 9 in China [6,12,13,14]. In Europe, although there is often a call for a human spaceflight program, there is currently no official plan to develop a super heavy launcher [2]. However, in 2004, the European Space Agency released a preliminary study on a possible architecture for a human mission to Mars [1]. At the end of the report, it was stated that it was a first attempt and that further investigations would be needed to improve the proposal but, to our knowledge, no other studies have been published. While the choice of a super heavy launcher does not make much difference for a conceptual study, it is proposed here to take as a case study a possible European design for the main rocket that could send humans to Mars as a contribution to the earlier study. The main issue addressed in this paper is to determine the most appropriate LEO payload of that super heavy rocket based on available launchers, trying to find the best tradeoff between the various decision making criteria such as feasibility, simplicity, risks and costs.

In order to address the question, several important assumptions are made in the paper about the architecture of the mission:

- The objective is to send at least 3 astronauts to the surface of Mars and to bring them back to Earth. This number has been suggested in the literature as a minimum required to reduce human factor risks to acceptable levels [19,20]. A greater number would nevertheless be preferable.
- The strategy to assemble a giant rocket in LEO and to send astronauts and cargo to Mars should be based, if possible, on available launchers.
- The architecture of the mission should be effective and as simple and efficient as possible in order to minimize development and qualification costs.
- It should be smart enough to pave the way for a long term strategy for space exploration, reducing both recurrent and long term costs [9].

An important issue is to determine the best propulsion system for interplanetary transportation. In several NASA studies, it is argued that scenarios based on chemical propulsion require much more mass in LEO and a longer and more complex assembly process than other scenarios based on nuclear thermal propulsion or on a combination of chemical propulsion with solar or nuclear electric propulsion [7,8]. However, in several papers, it has been highlighted that these comparisons were biased by inappropriate options, such as the elimination of aerocapture for Mars orbit insertion, without looking for possible tradeoffs [9,21,25]. Considering chemical propulsion options, aerocapture for Mars orbit insertion is indeed a key choice that should be set on top of the tradeoff tree and should drive the design of the mission and the choice of the other options (such as the interplanetary propulsion system), and not the other way round [9,20,21]. As NASA architectures do not provide clear advantages over chemical based architectures [7,8,17,21,27], and as an important criterion of the study is the cost, it is proposed here to focus on mission architectures based on chemical propulsion only.

In order to address the problem, the paper is organized as follows. In the next section, strategic choices are presented and discussed. The central question of the following section is the determination of the minimum launcher capacity for a direct shipment from Earth to Mars. This question is addressed first for the interplanetary crewed vehicle, secondly for the interplanetary cargo that transports the Mars ascent vehicle to the surface, and thirdly for the return vehicle that has to wait in Mars orbit while the crew is on the surface. Once the minimum payload capacity is defined for the launcher, it is proposed to look at possible options to develop a “Super Ariane” using existing engines and boosters. Finally, a possible roadmap and an estimation of the development costs are provided.

2. Strategic choices

2.1 Direct to surface

Numerous original architectures have been proposed to send humans to Mars [7,8,9,14,17,21,25,27,28]. A first group of architectures is rather simple in essence, like Mars Direct, Mars Semi-Direct, or Musk's proposition [14,17,21,27,28]. In another group, there are NASA architectures and futuristic approaches based on nuclear thermal engines, or electric propulsion technologies supplied by nuclear power plants or giant solar panels [7,8,9]. The most relevant difference between the two groups is the direct or indirect option to reach the surface for the space vehicle transporting humans from Earth. Indeed, in the first group, the interplanetary vehicle has a dual role and lands on Mars, while in the second group of architectures, there is a transfer of the crew onto a specific landing vehicle on arrival in Mars orbit. Considering the long term vision of human missions to Mars, as no clear benefit has been proven yet in terms of transfer time or initial mass in LEO (see [9,14,21]), the simplest solution is preferable. This was the driving principle of the Zubrin or Musk architectures (it was called "Mars Direct" or "Mars Semi-Direct" by Zubrin [27,28]). It is proposed here to try to follow that principle and to set as a key preliminary assumption that the interplanetary crew transportation vehicle will also be the vehicle landing on the Martian surface. We will explain why it is also a very efficient option in the following sections.

2.2 Aerocapture

As pointed out by several authors, architectures based on chemical propulsion are efficient only if aerocapture is made possible for Mars orbit insertion [9,11,17,21]. There are indeed 2 advantages with aerocapture:

- First, a space vehicle following an interplanetary trajectory must decelerate for insertion into Mars orbit. The ΔV depends on the desired orbit and the velocity on arrival. It is in general in the range 1 to 1.9 km/s at the end of a Hohmann transfer [3]. Without aerocapture, the consequence of such a ΔV would be the necessity to add a heavy propulsion stage, which would double the mass of the vehicle at Mars entry [8]. With aerocapture, a heat shield and other systems would be required, but the mass would only increase by 30% to 40% [4,5]. In addition, if it is a lander, which is assumed here with the "direct to surface" option for the habitable module, a heat shield would be needed anyway and the additional mass would therefore be even less.
- Secondly, if it is desired to reduce the travel time of the following flights, the velocity would be increased at departure from Earth and the velocity on Mars arrival would be greater, resulting in a greater ΔV for Mars orbit insertion. Without aerocapture, the impact could be high with new propellant requirements and additional mass for the propulsion stage, which in turn might necessitate an LEO assembly of a giant spaceship. With aerocapture, provided that it is still possible to follow the required corridor, the impact would be quite low. In fact, aerocapture allows minimizing the impacts of planetary configuration and Earth departure velocity on the size and mass of the interplanetary space vehicle, which is important for reaching a standardization of mission architectures and to prepare technological evolutions.

The feasibility of aerocapture remains to be demonstrated. However, it is generally acknowledged, even by NASA, that aerocapture is feasible and efficient, provided that the shape of the vehicle is compact (no module or escape vehicle hanging on the side as illustrated Fig. 1) [7,8]. According to Braun and Manning, aerocapture is possible with a lift to drag ratio on the order of 0.3 (biconic shape) [3]. An important issue is to achieve a reasonable ballistic coefficient (e.g., less than 200kg/m^2) in order to start deceleration in the highest layers of the Martian atmosphere and to keep deceleration rates

below 4 g, which is the maximum that can be tolerated by astronauts [3]. The entry corridor is narrow, about 1° large, but it is within navigation capabilities of current guidance, navigation and control (GNC) technologies, which is confirmed by a recent study [11]. Aerobraking by means of several passages in the atmosphere is another option, but as it takes in general several months to reach the low orbit, it is a waste of time for the astronauts. Direct entry is possible, but the arrival planetary configuration and the location of the landing site may not be appropriate for a direct entry. The weather might also be a problem. Going to orbit first is therefore preferable in order to adapt the time and location of the entry point. Aerocapture is therefore assumed in this architecture. Importantly, the implementation of the aerocapture option is easier if the vehicle arriving from Mars is compact. This is indeed the case for the direct to surface strategy (no LEO assembly of a complex vehicle) and one of the key reasons for its efficiency.

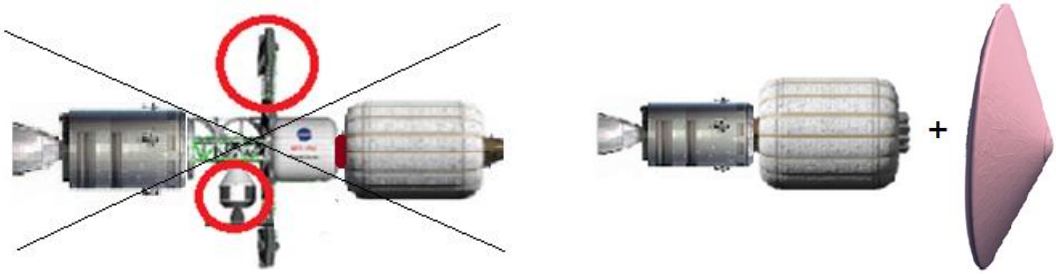


Fig. 1. Left: A complex train of several modules with docking ports on the side is not appropriate for aerocapture. Right: A compact shape makes aerocapture easier.

2.3 Entry, descent and landing

A critical issue in a Mars mission architecture is the test and qualification of EDL (Entry, Descent and Landing) systems during the preparatory phase [3,18]. Many difficulties have to be overcome to be able to land 20 to 40 tonnes, or even more if the Starship is considered. In order to minimize propellant needs and therefore the overall mass of the lander, the thin atmosphere of Mars can help braking during the descent phase. However, because of the generally high ballistic coefficient, large heat shields have to be used. Ideally, as the TRL is high, a possible option is to use a rigid heat shield with a conic shape [3,17]. The problem is that atmospheric braking would be sufficient only with very large diameter heat shields (between 12 and 30 meters), which is impractical due to fairing constraints [7,8]. Moreover, as the use of parachutes is not possible for heavy landers, optimization suggests the use of two heat shields, one for the hypersonic regime at entry into the atmosphere and another one for the supersonic regime [7,8]. This option is very interesting because a hypersonic heatshield is already needed for the aerocapture phase. If the same heatshield can be used, this is another mass saving for the interplanetary vehicle and another reason explaining the efficiency of the Direct to surface strategy. An assessment of different EDL strategies has been carried out by NASA [16], which suggests the use of inflatable heat shields [16]. Some tests have already been carried out in the Earth's atmosphere and the results are promising [23]. According to the NASA assessment, the best strategy for Mars is to use two inflatable decelerators that would be deployed during different regimes [23,26]. For the study presented in this paper, as it is aimed at calculating the minimum mass of the launcher, it is necessary to calculate the mass of the interplanetary vehicle (payload of the launcher), and therefore to estimate the mass of EDL systems. It is proposed here to consider EDL systems proposed in the NASA reference mission and to assume that the EDL systems to landing vehicle mass ratio is constant. An estimate of the EDL systems mass provided in the NASA study is summarized Table 2 [8]. **As a first approximation, the arrival mass is exactly twice the mass of the payload.** This estimate will

help us determine the requirements for the initial mass in the LEO and the size of the launcher (see next sections).

Table 2. Mass of EDL systems as a percentage of arrival mass (NASA data [8]).

Element of spacecraft	Mass (tonnes)	% of total arrival mass
Arrival mass before aerocapture	80.6	100%
Avionics and separation structure	1.9	2%
Entry RCS (wet)	7.4	9%
Hypersonic IAD	10.6	13%
Supersonic IAD	2.1	3%
Descent stage (wet propulsion system + landing mechanisms)	18.6	23%
Landed mass	50.4	63%
Payload	40	50%

Though NASA suggested the use of inflatable heat shields, another option is to use deployable flexible heat shields (se Fig. 2), which has also been studied by NASA with the ADEPT concept and a European team [16,24]. According to a previous study, a 34-tonne vehicle can be efficiently slowed down in the Martian atmosphere if the diameter of the heat shield is of the order of 12 meters [17]. Within an 8 meter-large fairing, which would be more or less the constraint of a launcher for a Mars exploration program, the diameter of a deployable heat shield could expand to 12-14 meters. The advantage of a deployable heat shield is the possibility of adapting the diameter to the phase of the descent. In a hypersonic regime, the diameter could for example be limited to 12 meters and, in a supersonic regime, it would be expanded to 14 meters. The TRL of such systems is relatively low but their complexity is not high. Guidance, navigation and control systems, as well as thermal protection systems, have already been studied by several European teams with demonstrators (ARD, IXV, and soon the Space Rider) and the use of deployable heatshields has also been explored with promising results [16]. This last option is feasible only if the landing vehicle is not too heavy but it would indeed be the case for the vehicle that is considered in this study with the constraint of a direct to surface strategy and the minimum size for the launcher.

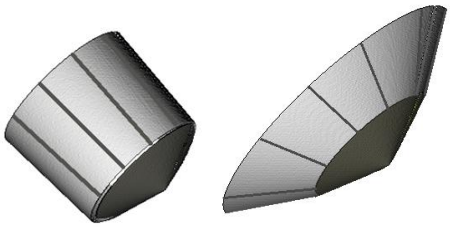


Fig. 2. Deployable heat shields could be a key EDL technology (left, the heat shield is stored and right it is wide open).

3. Designing the launcher

3.1 Method

Huge launchers are not needed for satellites or robotic missions. For a human Mars exploration program, however, a huge launcher is required for two reasons. First, as the interplanetary flight

lasts at least 6 months, a large volume is needed for the crew and a significant amount of consumables is needed for life support. Secondly, as heat shields are usually very large, the diameter of the fairing also has to be very large. Conversely, the heavier the launcher, the more difficult and risky the launch. It is indeed necessary to increase the thrust, the number of engines, the number of pipes and pumps, the size of the tanks and the resistance of materials. The assembly of the vehicle is also more difficult and costly with the use of dedicated cranes and surface transportation systems. Launchers of the Saturn V or SLS class, with 100 to 130-tonne LEO capacity, are probably close to a maximum in terms of height and mass. The most suitable LEO capacity should therefore be the result of a tradeoff between two opposing principles. In order to determine the best LEO option, it is proposed here to calculate the minimum size and mass of the interplanetary vehicles that have to be sent to Mars: the crewed vehicle, the Mars ascent vehicle, and the Earth return vehicle. The most suitable LEO capacity must be sufficient to send all these vehicles to Mars without LEO assembly and at the same time the launcher must be as small as possible so that it can be used for other missions beyond Earth orbit (e.g., the Moon or asteroids).

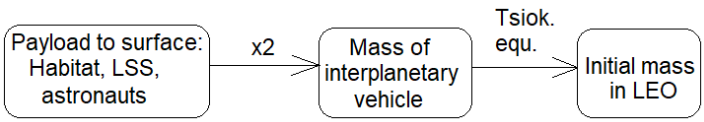


Fig. 3. *Method for the calculation of the required LEO capacity*

The minimum LEO capacity of the launcher is determined in several steps, see Fig.3. First, the mass of the minimum payload sent to the Mars surface is estimated. Then, aerocapture and EDL systems are considered in order to estimate the mass of the interplanetary vehicle on Mars arrival (see section 2.3). From the LEO, it is assumed that **the required ΔV for the trans-Mars injection maneuver is 3.7 km/s**. To infer the mass of the propulsion system for that maneuver, the Tsiolkovsky equation is used. The propulsion system is based on chemical propulsion. **A specific impulse of 450 seconds (e.g., Ariane-Vulcain engine) and a structural to propellant mass ratio of 10% (Ariane 5) are assumed**. The total mass of the interplanetary vehicle thus corresponds, in a first approximation, to the LEO capacity required for the launcher.

3.2 Minimum size of launcher for crew transportation

Let us examine the impact of the size of the crew on each vehicle [19]. For that purpose, we propose to recall the values proposed by NASA to estimate the mass of the habitable module as a function of the crew size and the trip duration (Table 3) [8]. Remarkably, there is a significant mass increase between a 600-day and a 1000-day trip. NASA did not provide intermediate values for other durations but the points are aligned and it is easy to determine the regression lines to obtain the mass of the module for any given duration. The following formulas can therefore be used, with d the number of days:

- 3 astronauts: $mass_3 = 16.585\ d + 13545$ (1)
- 4 astronauts: $mass_4 = 20.178\ d + 15015$ (2)
- 6 astronauts: $mass_6 = 28.255\ d + 17101$ (3)

Table 3. Mass of the habitable module (kg) as a function of the crew size and number of days, according to NASA data (NASA Design Reference Architecture, annex 2, 2014, page 370 [8]).

Crew size \ duration (d)	**600**	**800**	**1000**
3	23505	26794	30139
4	27128	31144	35199

6	34039	39694	45333

An important parameter in determining the mass is the number of days. In the proposed architecture, the crewed interplanetary vehicle is also the crewed landing vehicle sent to the surface. The duration of the trip is between 6 and 8 months, depending on the trajectory and perhaps the choice of a faster transit. Two important issues have to be considered:

- If everything goes well, the vehicle lands safely on the surface close to the MAV and perhaps to other assets. In this case, other consumables (or even another habitat) can be provided by another module and the minimum duration in the crewed vehicle would be equal to the duration of the trip. However, some contingencies have to be considered if, for example, the landing is a few kilometers away from the expected position. It is assumed here that the autonomy of the habitable module must be at least one month on the surface. Ideally, of course, the autonomy should be as long as the stay on the surface, but the minimum is considered here for the demonstration of feasibility to send the crewed vehicle to Mars and to determine the minimum LEO capacity of the launcher.
- It might happen that something goes wrong and the mission to the surface must be aborted. In this case, several abort strategies are possible. The first strategy is to impose a specific trajectory to enable a "free return" to Earth using the same vehicle. In general, this option leads to a total trip duration of about 2 years. The second option is to transfer the crew onto another vehicle, typically the Earth return vehicle, which must be waiting in Mars orbit. A third option is also possible: A free return trajectory is chosen but the crew is nevertheless transferred onto the ERV, which is sent on the same trajectory. If the second and third options are chosen, one month of complementary consumables is probably acceptable as a minimum to take these abort strategies into account.

All in all, the number of days has to correspond to a minimum of 10 months to demonstrate the feasibility. As it is of primary importance to minimize the duration of the trip, we propose to examine 2 scenarios. First, no effort is made and the duration is 9 months plus one month for contingency operations. Secondly, with a mass penalty of 3 tonnes for the payload of the launcher in order to take more propellant, the duration is reduced to 6 months plus one month for contingency. In each case, the mass of the interplanetary vehicle is calculated as follows: The mass of the habitat at departure is set according to the number of crew members and the duration injected in formulas (1) to (3). During the trip, there is a consumption of food, water and other fluids and some wastes are produced. These wastes can be ejected just prior to arrival on Mars. The mass of wastes is estimated in a first approximation at 0.5 tonnes per astronaut (mass of food per astronaut for the outbound trip, NASA data). This mass is deducted from the mass at departure to obtain the mass of the habitable module (payload) at Mars arrival. Then, the mass of the EDL systems is assumed to be equivalent to the mass of the payload (assumed in previous section). The total mass of the interplanetary vehicle at departure is therefore twice the mass of the habitable module at Mars arrival plus the mass of wastes.

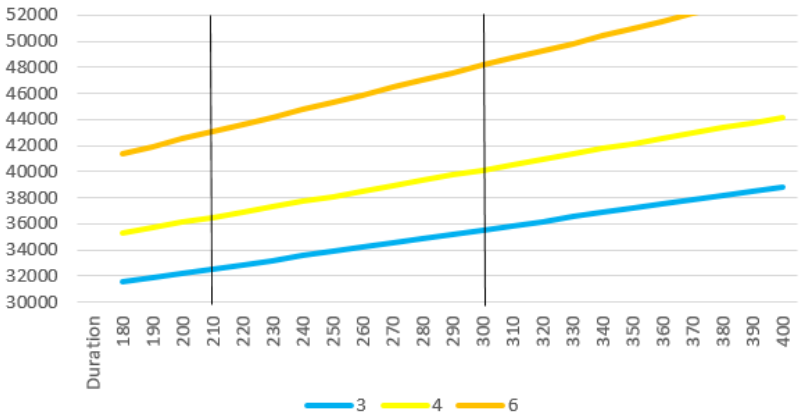


Fig. 4*. Mass of crewed vehicle as a function of crew size (3, 4 or 6 astronauts) and required duration. Horizontal axis: Duration in days. Vertical axis: Mass of vehicle in kg. Vertical lines: Minimum duration for fast trip +30 days and minimum duration for normal trip +30 days.*

The results are presented in Fig. 4. Considering the amount of consumables for a crew of 3 astronauts, the energy required for a relatively fast Earth-Mars transit, the mass of the systems for an aerocapture maneuver and the mass of entry, descent and landing systems, our study leads to three important conclusions:

- On Mars arrival, the minimum mass for an interplanetary 3-crew vehicle and a direct to surface option is of the order of 34 tonnes (36 tonnes in LEO).
- Each supplementary astronaut increases the required LEO mass by approximately 12 tonnes.
- From LEO, assuming chemical propulsion for the trans-Mars injection maneuver, the minimum mass for that interplanetary vehicle is about 100 tonnes.
- The recommended minimum LEO capacity for a launcher is therefore of the order of 100 tonnes.

Importantly, this minimum is achieved for a very short stay on the surface with a small amount of consumables and very few tools and exploration equipment. This is not practical, but it is possible to ship consumables, surface vehicles and scientific tools in another vehicle prior to the human flight. This module would typically be sent 2 years prior to the crewed spaceship. This is a good trade-off. It is indeed much simpler to organize a rendezvous on the Mars surface with 2 small modules (one not critical) rather than assembling a giant spaceship in low Earth orbit and trying later on to land on Mars with a huge module. Another benefit of the proposed strategy is that the second module could serve as a backup habitable module and could therefore increase the safety of the mission. A possible issue is to make sure that both modules will land at the same location with good accuracy (less than 1 kilometer distance). This constraint has already been taken into account in NASA studies with an accurate control of the descent trajectory and possible lateral navigation in the last phase of the descent for pin-point landing [8]. If a problem occurs despite the precautions, a Mars ascent vehicle should also be present on the surface to allow a possible come back to LEO at any time.

Interestingly, the propellant mass penalty for the reduction of trip time to 6 months (180+30=210 days' duration requirement) is more or less equivalent to the benefit in terms of consumables for the habitable module. An in-depth study has to be carried out for a more accurate estimation. A complementary result, which deserves to be mentioned, is the strong impact of the crew size on the mass of the vehicle. For the crew vehicle, the supplementary mass per astronaut is approximately 12 tonnes. However, as the stay on the surface is much longer than the trip to Mars (about 500 days), it must be noted that the impact on the second vehicle is greater and, therefore, the feasibility of sending

the remaining consumables, scientific tools and rovers in the same vehicle might become more difficult. For example, with a maximum LEO capacity of 130 tonnes, 5 astronauts could be sent in the crew vehicle for a direct trip to Mars but the requirements for the second vehicle might exceed the 130-tonne LEO capacity, especially if a heavy pressurized rover is in the payload.

To summarize, the requirements for a crew of 3 are proposed here. The mass of the interplanetary spaceship is presented in Table 4. The second interplanetary spaceship is not provided but its mass budget is not critical if the mass of rovers and scientific tools is reasonable.

Table 4. LEO mass of crewed spaceship.

Crewed spaceship	**Mass (kg)**
Habitable module for a crew of 3 astronauts, 210 days life support	17,000
EDL systems (equal to payload mass at Mars entry)	15,500
Complementary propellant for reduction of trip to 6 months	3,000
Total mass for TMI	**35,500**
Total mass in LEO	**99,000**

3.3 Minimum size of launcher for MAV transportation

Another important issue is to bring the astronauts back to Earth. Space X proposes to reuse the giant vehicle that lands on the surface [14]. This is a risky approach for several reasons:

- First, the plume of the giant vehicle could make holes in the ground and make the terrain unstable and dangerous for landing and, at the end of the mission, dangerous for takeoff [7].
- Secondly, the refueling of such a vehicle would require huge amounts of propellant and therefore significant infrastructure in terms of surface power and ISRU systems.
- Thirdly, the maintenance of a giant vehicle on the surface of Mars would be complex and the qualification for the next flight could be difficult.
- Finally, there is no launch backup system in case of problems during the descent and landing on Mars.

Another option, which was considered by Zubrin and NASA for a "semi-direct" architecture, is to send an Earth return vehicle (ERV) to Mars orbit and to land only a small Mars ascent vehicle (MAV) on the surface, the latter being used at the end of the stay to join the former [7,8,21,27]. In order to reduce the risks, the MAV must be as small as possible, shielded during the descent and landing, and its propulsion system would not be used at all before the launch from Mars, making it easier to qualify the vehicle for launch [8]. The mass of the MAV depends on various options. Assuming a propulsion system based on LOX/LCH4, it is possible to send a wet MAV to the surface of Mars, or to produce oxygen from local resources, and maybe methane. As NASA carried out a detailed analysis of the requirements and design of the MAV, it is proposed here to consider it as the baseline of our study [8]. For the requirements, it is assumed first that an ERV is parked in a 33800x250 km orbit, as proposed in the NASA scenario. The MAV has to be launched from Mars in order to dock with the ERV in that orbit. The ΔV from the Mars surface to that Mars orbit is 5625 m/s (NASA estimate). Taking time margins into account, the life support systems of the MAV should be able to sustain the lives of the astronauts for a minimum of 43 hours. As a fueled MAV would be too heavy for a direct launch from Earth, it is proposed to follow the NASA strategy: The propulsion system of the MAV is chemical and uses methane and oxygen. Methane is brought from Earth but oxygen is produced on Mars using *in situ* resources. This process is well known and NASA detailed the technical needs as well as the impact in terms of mass [7,8].

The mass of the landing vehicle is determined in several steps. First, based on NASA data, an estimate of the mass of the crew's cabin is given. As NASA provided an estimate for a 4-crew and a 6-crew cabin, a regression line is determined and an extrapolation is proposed to estimate the mass for a crew of 3, see Table 5. Then, following the NASA approach, the mass of the propulsion system required for Mars ascent is examined. The ascent vehicle proposed by NASA has two stages with tanks placed around the crew cabin. According to NASA, the total ascent mass is 38, 076 kg and the total descent mass (no LOX, no astronaut and no cargo) is 13,536 kg. Based on the NASA study for a 6-crew cabin and NASA estimate of the propulsion systems' mass, a proportional extrapolation is proposed for the mass of the propulsion systems for a 4-crew and a 3-crew MAV. Once the mass of the MAV is calculated, the mass of the landing vehicle can be estimated. Several options are possible, especially for the engines of the MAV that can have a dual use, descent and ascent, or not. For the sake of simplicity, and also in order to maximize the probability of non-failure, it is assumed here that the MAV is simply a payload of the lander (it was also the choice in the Apollo program for the design of the lunar ascent vehicle). Importantly, as for NASA, it is also assumed that the oxygen tanks of the MAV are empty for the descent and have to be filled using an ISRU system deployed on the surface after landing. The mass of that system has to be taken into account in the payload for landing. Our results are presented in Table 6.

Table 5: Mass estimates (kg) for the cabin of the MAV (extrapolations from NASA study [8, page 263]). NASA data is colored grey.

	6 astronauts (NASA data)	**4 astronauts (NASA data)**	**3 astronauts extrapolation**
Command and data handling, GNC, comm. and tracking, power, thermal, ECLS, EVA, human factors	1, 188	1, 031	953
Structures	1, 308	778	513
Crew and worn equipment	0 (742 for ascent)	0 (506 for ascent)	0 (388 for ascent)
Cargo	250	250	250
Non propellant fluids	61	41	31
Total crew cabin	**2, 807**	**2, 100**	**1, 747**
Total + 30% growth	**3, 649**	**2, 730**	**2, 271**

Table 6: Total mass (kg) of the interplanetary vehicle carrying the MAV to the surface. NASA data is colored grey.

			6 astronauts (NASA data)	4 astronauts extrapolation	3 astronauts extrapolation
Payload for descent	MAV	Cabin mass	3, 649	2, 730	2, 271
		Propulsion systems for ascent (w/o LOX)	9, 887	7, 397	6, 153
		Total MAV	13, 536	10, 127	8, 424
	ISRU system		936	936	936
	Fission power system		7, 000	6, 000	5 ,000
	Total payload mass		**21, 472**	**17, 063**	**14 ,360**
Mass of EDL systems equal to payload (see section on EDL)			**21, 472**	**17, 063**	**14, 360**
Total at Mars entry (kg)			**42, 944**	**34, 126**	**28, 720**

All in all, for a crew of 3 astronauts, the mass of the MAV is of the order of 8.4 metric tons. Then, the specifications of the interplanetary vehicle that transports the MAV and ISRU systems to the surface can be inferred and, in turn, the specifications of the launcher can be determined. According to our calculations, the LEO capacity requirement for the launcher is less than 100 metric tons. This is an important result, because the minimum LEO capacity of 100 tonnes for the crew vehicle is also sufficient to send the MAV to the surface. Another important finding is that the mass rapidly grows with the number of astronauts. With a 100-tonne LEO capacity, a MAV for 4 astronauts could probably be sent to Mars, but not a MAV for 5 or more astronauts. In fact, as a first approximation, the mass of the crew landing vehicle is identical to the mass of the MAV for crews between 3 and 5 astronauts (provided that heavy rovers and complementary consumables are sent in another lander). A heavy launcher with a 100-tonne LEO capacity would therefore have a double justification.

3.4 Sizing the Earth return vehicle

3.4.1 Main principles

For the Earth return vehicle, it has been clearly shown in a previous study that the best option is to split that vehicle into two parts, to send them directly to Mars and to link them in Mars orbit [21]. Indeed, in the context of a NASA mission, the total mass of the habitable module, the wet propulsion system for the outbound trip, the atmospheric Earth re-entry capsule and the systems for Mars orbit insertion was so high that 2 SLSs would be required to send all modules to LEO, and then to Mars. In addition, if a giant spaceship is assembled in LEO with an Orion-like capsule hanging on its side (NASA concept), aerocapture at Mars would be very difficult and risky. If the capsule is set atop the main propulsion system of the inbound trip, aerocapture is made possible and efficient. There are therefore good reasons to split that ERV into two smaller vehicles and to link them in Mars orbit rather than any other option (see Fig. 5). The difficulty would be to perform the rendezvous automatically. However, there is considerable experience in automatic docking in LEO and the maneuver can be performed a long time before the crew is sent to Mars.

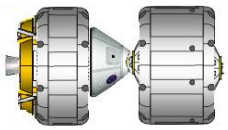


Fig. 5. *After Mars orbit insertion, the heatshield is jettisoned and a rendezvous is programmed between ERV part 1 (propulsion system for return and capsule on the left) and ERV part 2 (habitable module on the right) to assemble the full ERV. Another docking system is available on the right for the Mars ascent vehicle.*

3.4.2 ERV, part 1

With the limitation of 36 tonnes to Mars, 2 parts might be considered too little. As the mass of consumables is driven by the number of days in space, it is important to look at the requirements for the habitable module. In the consumables budget mass, it is necessary to take backup options into account. If it is required to abort landing, it should be possible for the crew to join the ERV and to wait there for the start of the launch window for the return. As a consequence, there should be enough consumables for approximately 700 days (450 days in orbit and 250 days for the inbound trip). The habitable module of the return is therefore heavier than the habitable module of the outbound trip. Sending a heavier payload is made possible because there are mass savings on EDL systems (no landing). Another optimization is made possible by jettisoning the excess consumables or the numerous wastes before the departure of the return trip (also proposed in NASA architectures). By doing so, some propellant can be saved or the trip time can be reduced. Some calculations have been made. The mass of aerocapture and orbit insertion systems can be roughly inferred from Table 2. The supersonic IAD and the descent stage are not needed anymore. Thus, avionics represents 2%, hypersonic IAD 13%, and wet RCS of the order of 5% of the entry mass, which all together corresponds to 40% of the payload mass. Supplementary tonnes of propellant might also have to be provisioned for orbit adjustment and orbit rendezvous. However, it is assumed here that the rendezvous will be performed by the second ERV module, which carries the capsule and the propulsion system for the return (see next section).

Table 7. Mass of ERV, part 1.

Mass of ERV, part 1 (kg)	**Crew of 6**	**Crew of 4**	**Crew of 3**
Habitable module for 700 days (from NASA data, see equations (1) to (4))	36,900	29,100	25,150
Aerocapture systems (assumed equal to 40% of payload mass)	19,200	11,600	10,000
TOTAL	**56,100**	**40,700**	**35,150**

Using the same mass ratios, the mass of the ERV, part 1, has been calculated for different crew sizes, see Table 7. According to our results, it seems possible to use a heavy launcher with a 100-tonne LEO capacity for sending to Mars, provided that the crew is limited to 3 astronauts. However, as the estimate is close to the limit (36 tonnes), an in-depth study is required to check the feasibility of the 3-crew option. For a crew of four, 40 tonnes are required for this module, which corresponds to 113 tonnes in LEO.

3.4.3 ERV, part 2

The second vehicle carries the capsule and the propulsion system for the TEI maneuver. Let us make an estimate of its mass according to the number of astronauts. For the capsule, a good estimate can be provided by modules that have already gone into space. Because of the need for a dedicated propulsion system for the TEI maneuver, there is no need to attach a service module to the capsule. As already pointed out in a previous study, for a mission to Mars, the heavy Orion spacecraft (8.9 tonnes without service module) is not justified [21]. A capsule is needed only for the last day of the mission for the re-entry into the Earth's atmosphere. Even the Command Module of the Apollo program, which had a mass of 5.8 tonnes, would be oversized. As the specifications are not exactly the same and we also need a docking system, we propose to average at 6 tonnes. For a crew of 4, a linear extrapolation is proposed between the crew of 3 and crew of 6 cases. For the inbound trip, the propulsion system must provide a ΔV of 1.5 km/s (NASA data). The payload of the return is the habitable module (its mass is given in Table 7) and the capsule. As consumables of the habitable module were provisioned for 700 days, there are excess consumables (or wastes if astronauts had to live there), which can be jettisoned before the TEI maneuver. This excess is estimated at 1.2 tonnes per astronaut (same as NASA). The mass of the propulsion system is calculated using Tsiolkovsky's equation. For the sake of simplicity, it is assumed an ISP of 360 seconds (methane + oxygen) and a structural mass to propellant mass ratio of 12%. The results are presented in Table 8.

Table 8. Mass of ERV, part 2.

Mass of ERV, part 2 (kg)			**Crew of 6**	**Crew of 4**	**Crew of 3**
Payload at Mars arrival	Capsule for Earth re-entry.		8,900	7,000	6,000
	Propulsion system for TEI maneuver, payload = mass of habitat + capsule, ΔV=1.5 km/s (NASA data), ISP=360s	Propellant	21,812	17,687	15,568
		Structure (12%)	2,617	2,122	1,868
	Complementary propellant + systems for orbit rendezvous and docking (5 %)		1,666	1,340	1,172
	Total		**34,996**	**28,150**	**24,608**
Aerocapture systems (assumed 40% of payload)			13,998	11,260	9,843
TOTAL (kg, rounded)			**48,994**	**39,409**	**34,450**

Once again, the total mass is below the 36 tonnes limit only if the crew size is limited to 3 astronauts. This is not surprising, as the mass of consumables per astronaut is high, consumables are a major contributor of the mass of the payload and the mass of many systems is closely related to the mass of the payload (e.g., heat shield, propellant for RCS, propellant for TEI). The heavy launcher of minimum 100-tonne LEO capacity that was assumed for crew transportation and MAV transportation could therefore also be used to transport each of the two parts of the ERV.

3.5 Use of existing or under development launchers

As previously explained, the minimum LEO capacity of a super heavy launcher for a direct trip to Mars without LEO assembly is of the order of 100 tonnes. This is an important result because the specifications for current super heavy launchers under development (SLS, Starship, Long March 9, Yenisei) are around, or a little greater than, 100 tonnes, which is sufficient to carry out the direct to surface strategy as proposed in this paper [6,12,13,14]. For a supplementary astronaut, the LEO capacity must be of the order of 112 to 115 tonnes. For the Starship vehicle, according to Space X,

there is no LEO assembly but a long refueling is required using other Starships before starting the Mars transfer maneuver [14]. If Space X succeeds in qualifying the vehicle for landing on Mars and coming back to orbit at the end of the mission, there is obviously no need to change that strategy. However, if the robustness or the safety of the approach is a concern, as the Starship payload (habitable module and fairing included) exceeds 100 tonnes to LEO (close to 150 tonnes according to Musk), Space X could abandon the refueling strategy and choose instead to include a third stage in order to keep going to the Mars surface directly but with a much smaller vehicle and a smaller crew. For the return, a MAV must also be sent to the surface by means of a direct trajectory using another Starship launch, as suggested in this paper, but the ERV could perhaps be sent to Mars orbit in a single payload. This is precisely the "Semi-Direct" architecture that was proposed by Zubrin [27].

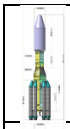
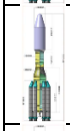
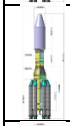
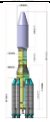
4. Case study: going to Mars with a Super Heavy Ariane

4.1 Mission architecture

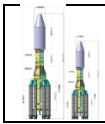
In previous sections, it has been shown that a launcher with a 100-tonne LEO capacity would be sufficient to implement a human mission to Mars by means of direct trajectories, without LEO assembly. Each payload would be an interplanetary vehicle with a dedicated propulsion system and a 36 tonne maximum payload to be sent into Mars orbit or to the planet's surface. For Mars orbit insertion, aerocapture is made possible for all vehicles and explains the good tradeoff in terms of initial mass in LEO. Assuming the existence of that launcher, 5 heavy launches would be required to send 5 interplanetary vehicles to Mars, see Table 9.

Table 9. Mars Mission architecture.

First phase, at least 2 years before the human flight:

	A cargo is sent to the surface of Mars. The payload is the MAV. It includes methane but does not include oxygen, which will be produced on Mars using ISRU.
	A cargo is sent to the surface of Mars. In the module, there are consumables, rovers and scientific tools. It is sent before the human flight.
	A cargo is sent to Mars orbit. The payload is the habitable module of the ERV.
	A cargo is sent to Mars orbit. The payload is the wet propulsion system of the ERV and the Earth's atmosphere re-entry capsule. The two parts of the ERV are joined in Mars orbit by means of an automatic docking maneuver.

Second phase: Human flight.

	The main vehicle is launched and placed in LEO. For safety reasons, the crew is sent to LEO in a man-rated launcher. The crew is transferred onto the interplanetary vehicle in LEO. Then a TMI maneuver is carried out and the crewed vehicle is sent to Mars.

Third phase: Return to Earth.

	After approximately 450 days on the surface, the crew uses the MAV to come back to LEO and join the ERV. TEI is performed to send the crew back to Earth. The last day of the trip, the capsule is used for the Earth's atmosphere re-entry.

4.2 Super Heavy Ariane specifications

In 2006, considering high demanding missions, a study from Astrium/CNES suggested the use of a super heavy Ariane 5 launcher to send heavy payloads to LEO [10,22]. It was based on 5 Vulcain II engines for the first stage, plus six solid boosters. The LEO capacity was of the order of 100 tonnes and the Mars transfer capacity around 36 tonnes. In this study, it is proposed to deepen the analysis and to determine the specifications of a Super Heavy Ariane launcher with a 100-tonne LEO capacity using Ariane 6 elements. Small differences are expected:

- The Vulcain 2.0 engine is replaced by the Vulcain 2.1 version. The thrust remains the same: 1350 kN.
- The solid boosters are different. In the Ariane 6 version, the P120C boosters are smaller but twice as numerous with a thrust of 3,500 kN each on average (more than 4,000 kN at lift-off). The duration of the thrust is 130 seconds as for Ariane 5.
- For the second stage, it is proposed to choose an adaptation of the core stage of Ariane 6. A Vulcain 2.1 engine would provide the remaining thrust to reach LEO. A similar choice was proposed in the Astrium study based on Ariane 5 engines [10].

It may be noted that in the new Ariane Next program, in order to recover the first stage of the launcher, the reusable Prometheus engine, which is based on LCH4-O2, is proposed [15]. In this case, 7 Prometheus engines would be required instead of 5 Vulcain to obtain approximately the same thrust.

Fig. 6. *Core stage and boosters for the Super Ariane 6 launcher.*

Table 10. Mass estimations for the Super Heavy Ariane

System		Extrapolated total mass in tonnes
Core stage with 5 Vulcain 2.1	Empty mass	80,000
	Propellant	700,000
8 P120C boosters	Empty mass	88,000
	Propellant	1,136,000
Upper stage with Vulcain 2.1	Empty mass	17,000
	Propellant	85,000
Payload to LEO	Vehicle for TMI	64,000
	Payload to Mars	36,000
Total (tonnes)		2,206,000

At lift-off, a total thrust of about 36,000 kN would be provided by 5 Vulcain 2.1 engines and 8 P120C. See illustration Fig 6. With an expected total mass of the order of 2,200,000 tonnes (see Table 10), the acceleration would be 1.8g at lift-off, which is acceptable for a human spaceflight. Once in LEO, as for

the Apollo program, a check-out of the systems would be carried out before the trans-Mars injection maneuver (TMI). The 100-tonne LEO payload would be an interplanetary spaceship with a chemical propulsion system and one of the modules needed for the Mars mission (see previous sections). For the TMI maneuver, if a Hohmann trajectory is chosen, the propulsion system must provide 3.696 km/s in the worst Earth Mars planetary configuration (Earth close to perihelion and Mars close to aphelion). For such a ΔV, the wet mass of that propulsion system would be of the order of 64 tonnes, while the payload would be limited to 36 tonnes [20]. Importantly, thanks to the Oberth effect, a small ΔV increase would significantly decrease the duration of the trip. In previous sections, it was highlighted that shortening the outbound trip to 180 days (35% reduction) necessitates reducing the payload (interplanetary vehicle) mass by only 3 tons.

4.3 Roadmap

Key elements of the mission have to be developed and qualified [18]:

a) Ariane Super Heavy with upper stage for trans Mars injection.
b) Dual use habitable module for 3 astronauts.
c) *In situ* resource utilization system to produce oxygen.
d) Mars ascent vehicle.
e) Rendezvous in Mars orbit and return vehicle.
f) Atmospheric Earth re-entry capsule for 3 astronauts.

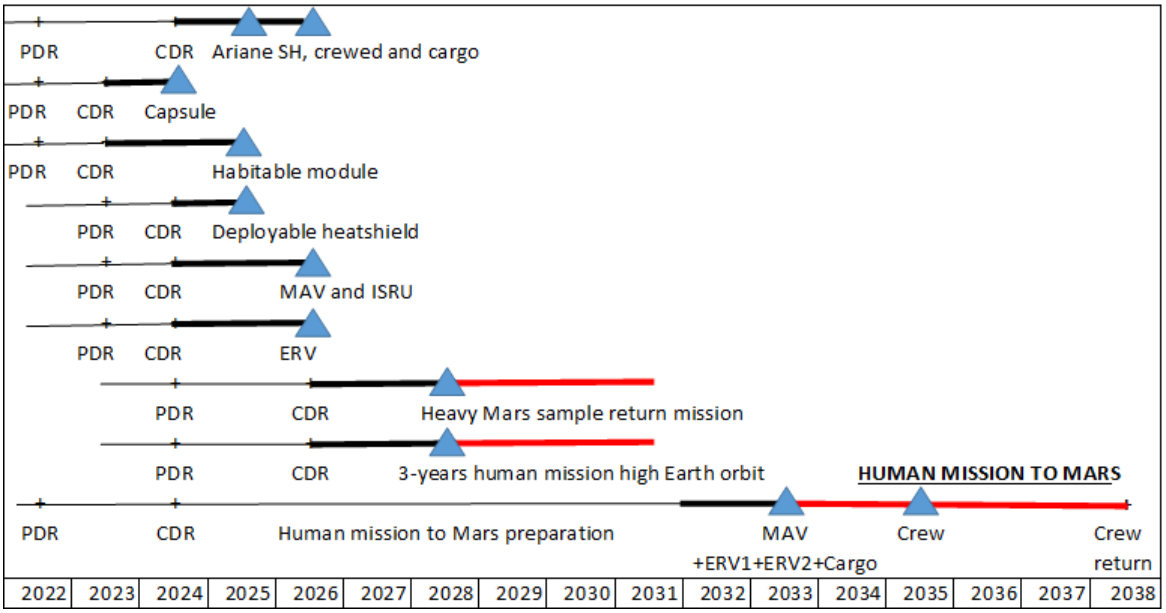


***Fig. 7**. Proposed roadmap.*

Two preliminary space missions would be appropriate and sufficient to qualify the key elements of the Mars mission, see Fig. 7 [18]:

- A 3-year manned mission in high Earth orbit with several rotating crews. The objective would be to qualify b) and at the same time maturing a) and f). This mission is important to make sure that life support systems are effective and safe with appropriate lifetimes. It is also an opportunity to study psychological issues and to gain experience on monitoring a distant crew with communication delays.

- A heavy Mars sample return mission. The objective would be to qualify c), d) and e) and at the same time maturing a). Collecting Mars samples and bringing them back to Earth would be an added advantage.

4.4 Technology Readiness Levels

Many systems needed for a Mars mission are already or will be soon mastered in Europe. The most critical issue is probably the qualification of the entry, descent and landing systems for Mars. ESA has a good expertise on heat shields and thermal control systems (e.g., IXV mission) but specific tests would have to be carried out in the Martian environment. A TRL estimation of the main systems is provided Table 11.

Table 11. TRL estimation.

Element	System	TRL
Ariane Super Heavy based on existing elements	Vulcain 2.1	9
	P120C booster	9
	Tanks	6
	Integration and control of multiple engines	6
	Other (structural, thermal, etc.)	6
Ground segment	Launchpad for Ariane Super Heavy	6
	Integration building	6
Capsule	Heat shield	7
	Life support systems	6
	Spacesuit	5
	Other systems	6
Interplanetary vehicle	Engines: Prometheus (projection)	6
	Heat shield for aerocapture	6
	Other	6
Payload of first interplanetary vehicle: Mars ascent vehicle (MAV)	Entry, descent and landing systems	3
	MAV, propulsion system for ascent	3
	In situ resource utilization systems (O2 prod. from atm. CO2)	3
Payload of second interplanetary vehicle: Earth return vehicle, ERV1	Wet prop. system: Based on Prometheus (projection)	6
Payload of third interplanetary vehicle: Earth return vehicle, ERV2	Habitable module	6
Payload of fourth interplanetary vehicle: Lander with habitat	Entry, descent and landing systems	3
	Habitable module	6
Payload of fifth interplanetary vehicle: Lander with cargo	Entry, descent and landing systems	3
	Cargo (e.g., surface vehicles, tools and backup consumables)	6

5. Conclusion

A minimum payload of around 100 tonnes to LEO has been determined to avoid a complex LEO assembly for a direct trip from Earth to the surface of Mars. This minimum may not be an optimum, especially if more than 3 astronauts are desired on the surface. The mass of the crewed vehicle rapidly grows with the mass of consumables and, if the capacity of the launcher is not sufficient for the

proposed direct to surface strategy, key options might not be feasible anymore (e.g., aerocapture) and the IMLEO would increase more in proportion than the number of astronauts. For example, duplicating the mission to send 6 astronauts to the surface would lead to an IMLEO of 1000 tonnes, while a NASA study suggested that for a crew of 6, using a giant spaceship and eliminating the aerocapture option, the IMLEO would be greater than 1200 tonnes [7]. If 4 astronauts are absolutely needed, the recommendation is therefore to design a launcher with 115-tonne LEO capacity.

A case study has been proposed with 5 launches of a theoretical Super Heavy Ariane, which exploits existing Vulcain engines (or 7 Prometheus) and solid P120 boosters. Thanks to aerocapture, the IMLEO would be around 500 tonnes, which is much less than current NASA estimations. Another advantage is the possible integration of rigid, deployable and dual use heat shields, which would reduce the complexity of EDL qualification and EDL risks. All in all, as TRL are high for many systems, this concept seems feasible at relatively low cost and can be implemented in 15 years. Thanks to a clear decomposition of the mission into 5 independent interplanetary flights, it is possible to assign to one or several partners one or several such flights. Some elements of the mission, for instance the habitable module or the Mars ascent vehicle, could be designed and built by international partners. The total costs would thus be shared among the partners. If the context is not favorable, Europe, or perhaps a small group of European countries, would nevertheless have all the potential to carry out everything alone.

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