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Abstract

Since the early history of Space Exploration, Mars conquest has been the most important target. After Apollo mission's Moon landing, several concepts and projects, concerning a mission on the Red Planet, were developed. One of the most important contributes was given by Buzz Aldrin, who theorized the use of particular kind of orbits, called cycler orbits, as baseline for an enduring Mars colonization. A cycler orbit is a kind of orbit which repeats every integer multiple of synodic period and which encounters two bodies with a precise schedule. In the case, the bodies considered are Earth and Mars. It is possible to inject a space station in the cycler orbit which allows a continuous transfer of a crew from Low Earth Orbit to Mars Low Orbit and vice-versa. Small taxi vehicles are used to rendezvous the cycler station from the two bodies, significantly reducing the amount of propellant. In this paper a mission architecture based on this new concept was analysed, in order to develop an alternative mission profile compared to the actual architectures proposed for human missions. The work starts with an analysis of several classes of cycler. Through a trade-off analysis an unique class of cycler was identified as baseline for a further mission analysis. The mission analysis consists of an evaluation of orbital perturbation, the computation of ΔV required for injection and rendezvous manoeuvres and an identification of close approach windows of the cycler with the two planets, allowing an evaluation of mission duration. Eventually, the presented mission concept was compared with more classical concepts, focusing on a key figure to enable Mars colonization, sustainability.

Keywords	Cycler Orbit, Mars, Interplanetary, Human Exploration
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Earth-Mars Cyclers for a sustainable Human Exploration of Mars

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Abstract

Since the early history of Space Exploration, Mars conquest has been the most important target. After Apollo mission's Moon landing, several concepts and projects, concerning a mission on the Red Planet, were developed. One of the most important contributes was given by Buzz Aldrin, who theorized the use of particular kind of orbits, called cycler orbits, as baseline for an enduring Mars colonization.

A cycler orbit is a kind of orbit which repeats every integer multiple of synodic period and which encounters two bodies with a precise schedule. In the case, the bodies considered are Earth and Mars. It is possible to inject a space station in the cycler orbit which allows a continuous transfer of a crew from Low Earth Orbit to Mars Low Orbit and vice-versa. Small taxi vehicles are used to rendezvous the cycler station from the two bodies, significantly reducing the amount of propellant.

In this paper a mission architecture based on this new concept was analysed, in order to develop an alternative mission profile compared to the actual architectures proposed for human missions. The work starts with an analysis of several classes of cycler. Through a trade-off analysis an unique class of cycler was identified as baseline for a further mission analysis. The mission analysis consists of an evaluation of orbital perturbation, the computation of ΔV required for injection and rendezvous manoeuvres and an identification of close approach windows of the cycler with the two planets, allowing an evaluation of mission duration.

Eventually, the presented mission concept was compared with more classical concepts, focusing on a key figure to enable Mars colonization, sustainability.

Finally the proposed architecture should be seen as a preliminary assessment of an alternative solution to the currently proposed enabling architectures for the Martian exploration.

Keywords: *Cycler, Orbit, Mars, Interplanetary, Human Exploration.*

Nomenclature

a: Semi-Major Axis
D: Atmospheric Drag
e: Eccentricity
F: Thrust
I_{sp}: Specific Impulse
m: Mass
r: Radius
S: Earth-Mars Synodic Period
t: Time
T_0: Standard Temperature
T_c: Normalized Temperature Coefficient
v: Velocity
v_{Earth}: Earth Circular Velocity
v_∞: Hyperbolic Excess of Speed
γ: Flight Path Angle
δ: Turn Angle
η: Efficiency
η_0: Standard Efficiency
ν: True Anomaly

Ψ: Leftover Angle
ω: Angular Velocity

Acronyms/Abbreviations
AOP: Argument Of Periapsis
AU: Astronomical Unit
DSM: Deep Space Manœuvre
DRA : Design Reference Architecture
ECLSS: Environment Control Life Support System
EML: Earth-Moon Lagrangian
LEO: Low Earth Orbit
LMO: Low Mars Orbit
RAAN: Right Ascension of the Ascending Node
RdV: Rendezvous
SLS: Space Launch System
SMA: Semi-Major Axis
SOI: Sphere Of Influence
TOF: Time Of Flight
WDV: Water Delivery Vehicle

1. Introduction

Actual vision for Human Mars Exploration are based on huge transportation architectures that ensure the provision of comfortable conditions and safety to the crew. Indeed, sustaining the human life in an harsh environment in during the Earth-Mars transfer and return is one of the most noticeable problem during this kind of missions. The consequence is the need of large system and consequently the need of enormous amount of propellant, leading to several heavy lift launchers (5 to 8) [7]. This can be accepted for a single mission but it is impracticable and not sustainable for an extensive and continuous Mars Exploration campaign.
A possible way out is the implementation of a complex architecture that allows utilization of several smaller systems, each one in charge of small portion of journey. The most important segment of this architecture will be a space station orbiting on a particular heliocentric orbit, called cycler orbit. This station allows the implementation of a recurrent channel to link Earth and Mars orbits that need to be injected in orbit just once at the beginning of its life. Once the vehicle is on the correct orbit, it can be used over and over again with very limited propellant mass for correction manoeuver. This is enabled by the intrinsic nature of the cycler orbits.
A cycler orbit is a special kind of trajectory which encounters two bodies on a regular schedule. In the case, Mars and Earth are the bodies involved. The cyclers repeat every whole number multiple of synodic period between the two planets. Nevertheless, external perturbation forces as solar pressure and gravitational perturbations cause the orbit degradation with consequential station keeping manoeuvres. Moreover, for some classes of cycler, a rotation of apses line is necessary in order to guarantee cycler repetition. If this manoeuvre is not required, the cycler is called ballistic.
Taking advantage from this theory, one or more vehicles will be equipped with all needs for a long transfer leg while smaller taxi vehicles could link Earth and Mars orbit to the cycler vehicle. The architecture will be completed with a staging post in Mars orbit to support and refuel the Mars Taxi.
Given these considerations, the Earth-Mars cycler problem seems to be worthy of study and consideration.

1.1 Solar system model

For an initial estimation is necessary to give some hypotheses to proceed with an analytical calculation of orbital parameters, exploiting the resolution of Lambert's problem. Indeed, a perpetual cycler exists only in a theoretical simplified solar system, which respects the following hypotheses:

- The Earth-Mars synodic period is 2.143 years.
- Earth, Mars and cycler orbits lie in ecliptic plane.
- The cycler trajectory is conic and prograde.
- Only the Earth has sufficient mass to provide gravity assist manoeuvres.
- Gravity assist manoeuvres occur instantaneously.

At the initial time, a clockwise angle is chosen so that the spacecraft will encounter Mars after leaving Earth. After Mars encounter, the spacecraft may encounter the Earth again. If the Earth encounter happens with the same Earth-Mars

angle, the spacecraft may return to Mars using the same shape heliocentric orbit, hence it is a cycler trajectory. The orbit could be found as a solution of Lambert's problem.
Let now T, an integer number of synodic period, be the time to repeat a cycler trajectory. It is also known the position of the spacecraft at the Earth encounters. Thus, the conditions for the trajectory are:

- $T = nS$
- $r(t = 0) = [a_E, 0]$
- $r(t = nS) = [a_E\cos(2\pi nS), a_E\sin(2\pi nS)]$

Where $a_E = 1\ AU$ is the SMA of Earth orbit, r the spacecraft radius from the Sun and n is an integer number. With these conditions the trajectories could be found as a solution of Lambert's problem.

1.2 Cycler characterization

Each cycler class, with a different number of synodic period may have multiple solutions. Every solution can be uniquely identified by specifying:

- *n*, time to repeat in integer synodic periods.
- If the solution is long-period, short-period or unique-period.
- *r*, the number of revolutions, rounded down to the nearer integer.

Then, we can denote a cycler with an acronym composed by three elements with the form 'nPr', where P is either 'L', 'S', 'U', depending on the period of solution. For example, the Aldrin's cycler has the following acronym: 1L1.
A first consideration can be developed about the turn angle required to keep the shape and repetitiveness of cycler orbit. Since the Earth-Mars synodic period is not an integer multiple of Earth revolution period, when the spacecraft returns to Earth after *n* synodic periods, the Earth will accomplish a fraction of revolution ahead of where it was when the spacecraft left. For example, it is possible to consider the case *n = 1*, which means that, the trajectory taken, repeats every synodic period, equal to 2.143 years. In that time, the Earth orbits around the Sun in a not-integer number of years. This angle, called leftover angle, could be computed as follows:

$$\Delta T = mod(T, 2n) \qquad (1)$$

Where *T* is the TOF and *n* the number of synodic periods. Thus, knowing the Earth angular velocity and the time fraction, it is clear to compute the required leftover angle. Therefore, in order to keep the shape and repetitiveness of cycler orbit we have to rotate the line of apses. The rotation is achievable either with a fly-by with Earth or with a powered manoeuvre. The second case is necessary when the hyperbolic fly-by orbit perigee has and altitude smaller than *200 km* from Earth surface. The cyclers which rotates the line of apses through a fly-by are called ballistic cycler.
When *n* is a multiple of seven, the Lambert's problem becomes degenerate and the cycler becomes a resonant transfer. In this case the line of apses does not need to be rotated, this is the case of VISIT 1 and VISIT 2 cyclers.
A second consideration could be introduced discriminating inbound from outbound trajectories. Indeed, cycler orbit usually crosses Mars orbit in two points. Depending on launch date, Mars is encountered at the first or second crossing. An encounter at first crossing minimizes the TOF from Earth to Mars. A cycler used in this way is called 'outbound cycler' because it travels from Earth to Mars. Conversely, if the cycler is launched in a different date such that it will encounter Mars at second crossing, it will minimizes the TOF of travel from Mars to Earth. This trajectory is called 'inbound cycler'. The difference between an inbound and outbound cycler is the launch date, not the shape of cycler trajectory.

2. Mission architecture overview

As introduced previously, the utilization of cycler orbits to reach the LMO requires the establishment of a complex architecture, which is made up by simpler systems in charge for a small part of the journey.
Of course, the most important system involved is the cycler station, a heavy architecture which has to support and guarantee the health and safety of the crew for a long travel in deep space with a large number of mission.
Other critical elements of the mission are the taxi vehicles which are going to link the LEO or LMO with the cycler station thanks to an hyperbolic orbit RdV. Designing the RdV manoeuvres is both a critical and challenging task. Since it is accomplished through a hyperbolic orbit, the main purpose is the design safe manoeuvres in order not to jeopardize the crew life in the case of failures.
The last critical segment of transport system is the Mars ascent vehicle which can guarantee frequent links between Mars surface and LMO exploiting in situ resources.

Therefore, up to six independent systems are involved in the mission architecture:

- **Cycler Vehicle:** one medium sized orbital habitable infrastructure linking LEO to LMO. The vehicle, supporting crew life during the legs, will be equipped with radiation shielding and high regenerative ECLSS. Once its final orbit is achieved, the cycler maintains it through electric propulsion to minimise propellant consumption. Depending on the selected cycler, more than one vehicle might be required. In order to satisfy its functionalities a first estimate of its mass is of the order of 100 t wet, 80 t dry.
- **Earth taxi:** an Orion-like vehicle which can host the crew from Earth surface to LEO and then perform RDV with the CTV. Clearly the Taxi must be equipped with a strong thermal protection system (e.g. a heat shield to sustain both the aero-capture and the entry manoeuvres will be necessary).
- **Earth staging post (optional):** a small infrastructure used as support to Earth taxis.
- **Mars Taxi:** equivalent to Earth taxi but different requirements and design features due to the different environment. The identification of a common design between the two would improve the sustainability.
- **Mars staging post:** an orbiting infrastructure with capabilities of on Mars orbit refuelling is required to limit the ascent mass of Mars taxi. About 40 t of propellant are needed for the RdV manoeuvre with the CTV starting from a parking LMO. The staging post would be of the same kind of the one proposed by Lockheed Martin Corporation in [15], which has already the same performance and capabilities.
- **Mars ascent vehicle:** the most critical element of architecture, used as ascent module from Mars surface. The definition of this element is not threated in this paper and for reference similar capabilities can be found in the Lockheed Martin Corporation study cited in [15].

Both outbound and inbound scenarios are represented in the next figure.

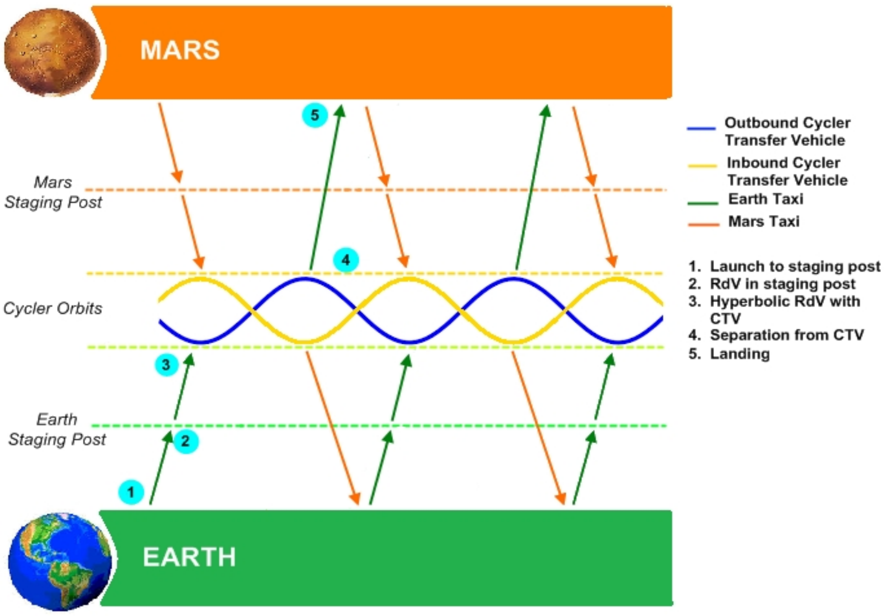

Figure 1: Cycler based transportation scenario

3. Mission analysis

According to [1], there are nineteen classes of cyclers representing the most promising solutions, indeed, not all of them are suitable for dissertation purposes. For example, some classes of cycler, such as 1L2, 2L4, 3L6 and 4S8, are all equivalent to Earth orbit. Other cycler trajectories do not intersect the Mars orbit because they have aphelion radius shorter than mean Mars radius. However, if the aphelion radius is just slightly below the Mars one, taking into account Mars orbit eccentricity and some small correcting manoeuvres the two trajectories may intersect with each other; this is the case of cycler 6S9.
The most famous and studied class is 1L1 (i.e. Aldrin's cycler) and [3] presents a deep study on it. Therefore, the purpose of the section is to evaluate briefly each class of cycler in order to gather data for a following trade-off analysis: a single cycler will be chosen as baseline for the mission architecture.

3.1 Cycler identification

In general, an orbit is univocally determined by six parameters, usually by two vectors, position and velocity, referred to an inertial frame in three dimensions. Since the orbits are circular and they lie in a single plane and the orientation of the orbits in this plane is not an interesting problem, we neglect the AOP, the inclination and the RAAN .
Thanks to the resolution of Lambert's problem, which is not a critical task, the orbital parameters are known. The algorithm were implemented by [1], [2], [3], [4], thus, only the main data are shown.
All the other orbital features, could be computed and found thanks to the algorithm provided by [6], they are neglected in the dissertation because they are not worth of interest.

Cycler	Aphelion [AU]	V∞ Earth [km/s]	V∞ Mars [km/s]	Shortest transfer time [days]	Turn Angle Required [°]	Eccentricity
1L1	2,23	6,54	9,75	**146,00**	84,00	0,39
2L2	2,33	10,06	11,27	**158,00**	134,00	0,45
2L3	1,51	5,65	3,05	**280,00**	135,00	0,24
3L4	1,89	11,78	9,68	**189,00**	167,00	0,42
3L5	1,45	7,61	2,97	**274,00**	167,00	0,27
3S5	1,52	12,27	5,45	**134,00**	167,00	0,30
4S5	1,82	11,23	8,89	**88,00**	167,00	0,40
4S6	1,53	8,51	4,07	**157,00**	167,00	0,30
5S4	2,49	10,62	12,05	**75,00**	134,00	0,48
5S5	2,09	9,08	9,87	**89,00**	134,00	0,40
5S6	1,79	7,51	7,32	**111,00**	135,00	0,33
5S7	1,54	5,86	3,67	**170,00**	135,00	0,25
5S8	1,34	4,11	0,71	**167,00**	136,00	0,17
6S4	2,81	7,93	12,05	**87,00**	83,00	0,49
6S5	2,37	6,94	10,44	**97,00**	84,00	0,42
6S6	2,04	5,96	8,69	**111,00**	84,00	0,35
6S7	1,78	4,99	6,66	**133,00**	85,00	0,29
6S8	1,57	4,02	3,90	**179,00**	85,00	0,23
6S9	1,40	3,04	1,21	**203,00**	86,00	0,17

Table 1 Most promising cyclers features

3.2 Earth fly-by analysis

From the starting hypotheses, the Earth is the only planet with a sufficient mass to provide a gravity assist, modifying velocity vector of the cycler spacecraft. This operation is needed to rotate the line of apses of the required leftover angle, helping us to identify ballistic cyclers.
The approximation used is the patched conics one, in which the Solar System is treated with spheres of influence. Furthermore, Earth SOI is supposed negligible than its mean radius from Sun, thus, the spacecraft will encounter the Earth exactly at a distance equal to $1\ AU$. This analysis does not take into account the gravitational effect on the Moon.
The resolution of the hyperbolic trajectory during a planet fly-by is well described by [5]. Taking into account the work developed by [3], in which only Aldrin's cycler is deeply analysed, we carried on a similar study in order to gather data for a further RdV analysis with the taxi vehicles.
Considering a bodies system lying on the same orbital plane, rotational matrixes are not required for a coordinate system change. Indeed it will be sufficient using Carnot's theorem to solve velocity triangles. Indeed, since the excess of speed is computed as follows:

$$v_\infty = \sqrt{v^2 + v_{Earth}^2 - v v_{Earth} cos\gamma} \qquad (2)$$

The turn angle required is computed solving the fly-by ΔV equation:

$$\Delta V = 2v sin\gamma = 2v_\infty sin\delta \qquad (3)$$

This analysis also allowed the identification of ballistic cyclers: only 6S8 and 6S9 are included in this group. The ΔVs provided by gravity assist are respectively $5.45\ km/s$ and $4.20\ km/s$.

3.3 Trade-off

At this point, the main features of several cycler orbits are determined by the earlier analysis, thus it is possible to choose the best cycler orbit for our objectives. After the orbit selection, this will be deeply analysed and will be used in all the dissertation.

The trade-off process begins by selecting some features, between those previously found, which must describe each class of orbit in the most general way, for the purpose of trying to describe the cycler in an objective way. These features are called figures of merit.

In this case five figures of merit were identified, each with a relative weight ranging from 1 to 5 :

- Total ΔV RdV (relative weight of 5): this figure of merit represents the amount of propellant required to the taxis for rendezvous and docking operations: the lower the ΔV value the lower the fuel consumption will be. An high value of this feature leads to an over-sizing of taxi dimensions; considering that one of the study objective is the reduction of the total amount of propellant in this trade-off analysis the ΔV RdV is the heaviest figure;
- ΔV fly-by additional (relative weight of 4): this figure represents the powered manoeuvre necessary to rotate the line of apses of the cycler orbit. Only three cyclers are ballistic cycler, in fact the rotation resulting from fly-by manoeuvre is enough to guarantee the cycler repetition. The other ones require an additional ΔV. The weight of this figure is significant, because it will determine the amount of propellant required to the Cycler Vehicle;
- Repeatability (relative weight of 3): the last figure substantially represents the ideal number of Cycler Vehicle required to have a mission every synodic period. This number is equal to 2n, so an high value of n require a large number of vehicle; therefore cycler with n=6 requires an ideal number of 12 Cycler Vehicle to guarantee a travel every synodic period. Clearly this is a scenario extremely expensive and complex, hence would be less sustainable. One way of reducing the costs is decreasing the number of Cycler Vehicles. For this reason this figure is quite meaningful and it has a medium weight in this analysis;
- Mechanical Energy (relative weight of 2): this parameter is considered in order to provide a univocal parameter which could describe the cycler orbit injection. Nevertheless, the orbital strategy for injection could be different from case to case, losing the objectivity required for this analysis. Therefore, a dissertation based on mechanical energy was preferred, because the energy is a univocal and common figure for each cycler. The weight assigned is not the lowest because it is directly linked to the amount of propellant needed for orbit injection;
- Shortest transfer time (relative weight of 1): this figure of merit represents the shortest transfer time between Earth and Mars, provided in
- Table 1. Naturally due to the manned mission considered, the lower the value the better the result, because a greater number of travel day leads to less day on Mars. As can be seen, this figure of merit has the lowest weight; that is because the range between the various cases is not particularly significant and this features is less binding than the others.

Once we have identified and assigned a relative weight to each figures of merit, it is possible to execute the trade-off analysis. Traditional approach for the trade-off has been used; for each figure the assigned weight is multiplied by a score ranging from 1 to 10, which is proportional to the specific figure. Thus the total score for a certain kind of orbit may be expressed as follows:

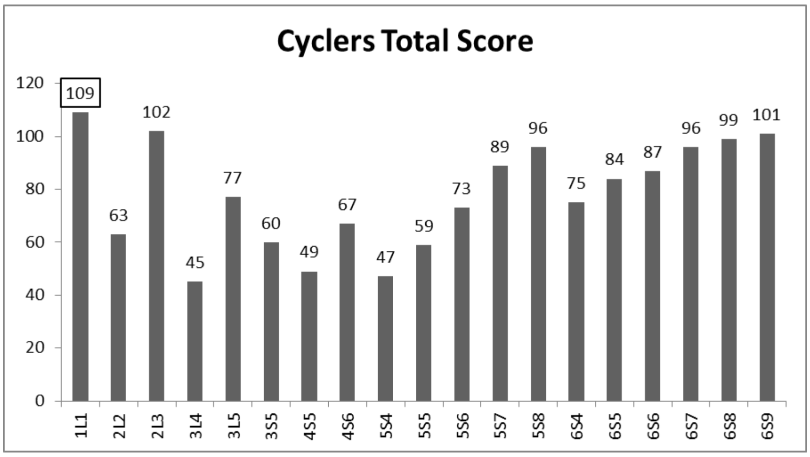

Figure 2 Trade off results

As it is possible to see, the highest score belongs to the "1L1" cycler, called Aldrin cycler. Also cyclers with n=5 and n=6 present high scores: the "5S8" is very affordable in terms of total ΔV RdV, the "6S9 " is in addition a ballistic cycler; on the other hand, unfortunately, these cyclers are inconvenient in terms of repeatability (in view of the large n) and the high energy necessary to the cycler orbit injection. The second highest score belongs to the "2L3" cycler, characterized by excellent values repeatability and ΔV RdV, but likewise with an high energy.
The Aldrin cycler represents a good compromise between all the figures of merit, so it is chosen as baseline for further mission analysis. In the next chapter the selected solution will be described with more details by removing some simplifications and using optimizations in order to reduce the cost of the rendezvous manoeuvres which is the week point of 1L1.

4. RDV strategies

The cycler scenario requires that the taxi vehicle performs a rendezvous with the Cycler Vehicle while the cycler is in a hyperbolic orbit in the proximity of the planet.
While spacecraft rendezvous has been studied for some time, there are unique challenges involved in rendezvous with spacecraft in hyperbolic orbits. One risk is a delay in the scheduled burn time resulting in a lost rendezvous opportunity. Another risk is being marooned in deep space if the taxi spacecraft enters a hyperbolic transfer orbit and misses the rendezvous. There are proposed methods to manage these risks, however. If the taxi carries enough propellant, it can still perform the rendezvous in the off-nominal scenario that it must depart one or two orbits late. A large amount of fuel would also be able to re-circularize the taxi orbit in the event that it is already on a hyperbolic approach but cannot achieve the final docking.
Two different RdV approach are considered in this study, one more secure for the crew in the case of docking failure and one more advantageous in terms of fuel consumption:

1. Elliptic rendezvous
2. Hyperbolic rendezvous

4.1 Elliptic rendezvous

This formulation assumes a two-impulse rendezvous for achieving a rendezvous between elliptic and hyperbolic orbits, in which the first impulse raising the transfer orbit apoapsis and the second nulling out the remaining relative velocity compared to the Cycler Vehicle at the hyperbola periapsis. This method look like an Hohmann.
The total ΔV required for the taxi RdV is composed by the two impulsive manoeuvres: in order to calculate the value of these ΔV, some assumption are used:

- Taxi vehicle starts in a 400 km circular orbit both Earth and Mars side;
- Taxi is optimally phased for the RdV manoeuvres at start;
- Taxi and Cycler Vehicle stay in a coplanar plane;
- Manoeuvres are impulsive and instantaneous.

The total ΔV required for the taxi RdV is the sum of this two manoeuvres. This method can be implemented by the use, for example, of three impulsive manoeuvres in a bi-elliptical transfer orbit, in order to reduce the total ΔV costs with, however, an increase in terms of time.

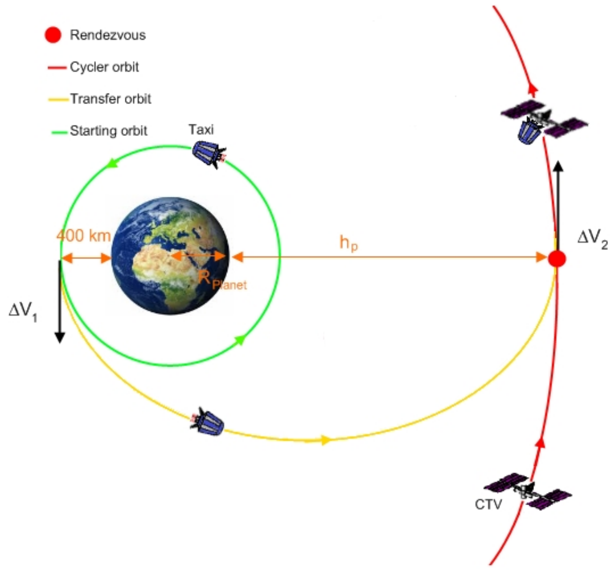

Figure 3: Elliptic rendezvous manoeuvre

The main advantages of this method relies in the relative simple "backup" manoeuvres in case of RdV failure: since the transfer orbit for the rendezvous is an ellipse, it could be retraced using the same instantaneous burns, in order to return in the circular orbit around the planet.

4.1 Hyperbolic *rendezvous*

This manoeuver can be broken down into four phases [11], or simply in four burns, which are considered impulsive here too.
Starting from the same low circular orbit as the previous case, the first manoeuvre achieves an high-energy orbit, but does not escape; this is useful in case of failure of the second burn, because this transit orbit is an ellipse and the taxi will return at the starting point after one complete revolution. The second burn is the "escape burn", in which the taxi achieves the same V_∞ as the Cycler Vehicle. The third manoeuvre cancels most of the relative velocity between the taxi and the cycler spacecraft and the final burn, that is not treated in this study, is the "docking burn".

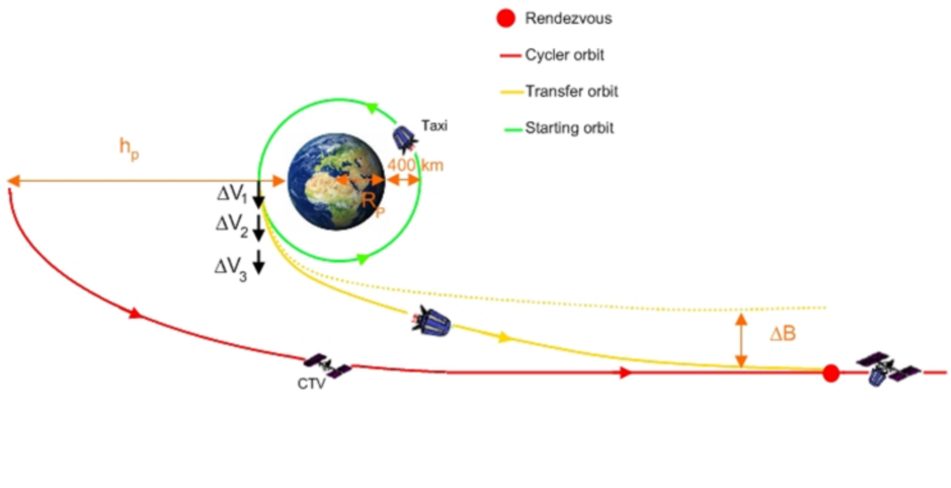

Figure 4: Hyperbolic rendezvous manoeuvre

This method is less expensive than the previous one (for the outbound worst case, the required ΔV for a hyperbolic RdV is 4,77 km/s, compared to 6,22 km/s required by an elliptic RdV), but is more dangerous for the crew: in fact, if the docking phase fails, the taxi gets lost in space. In order to prevent this deadly possibility, redundant propulsion system and extra propellant could be considered.

5. Optimization

Various studies about the attempt to improve the main Aldrin cycler criticality, which is the taxi RdV costs, are present in the Earth-Mars cycler literature. These studies propose trajectory optimizations by using some particular software: the reasoning behind these optimizations is the use of low-thrust in order to reduce the spacecraft approach velocity at planets sphere of influence, therefore decreasing the hyperbolic excess velocity ($V\infty$) and changing the fly-by hyperbola characteristics in order to lower the ΔV necessary for the taxi RdV. All this is possible by carrying out more expensive deep space manoeuvres; so these manoeuvres made by the Cycler Vehicle will not be necessary only for guarantee the orbit repeatability, but also for optimizing the trajectory.
The literature optimizations taken into account for this study are: Rauwolf, et al. [8], that use the SAIC's version of the Chebytop computer code to calculate the optimal trajectory solutions with normalized parametric mass performance data; Friedlander, et al. [9]; two different optimizations by Chen, et al. [10], that use Jet Propulsion Laboratory's Mission Analysis Low-Thrust Optimization (MALTO) tool to construct the Aldrin cycler trajectories.

5.1 Optimization choice

Among the various optimizations considered, the most advantageous one in terms of taxi fuel consumption for the elliptic RDV has been selected, both for the Earth and Mars side.
After that, by comparing the relative worst case for the outbound and the inbound trajectories, it has been analysed in detail the mission schedule, the cost of Cycler Vehicle deep space manoeuvres and evaluating an initial estimate of the transportable mass.
In the following table the worst case of every optimization both outbound (for the Earth RdV) and inbound (Mars RdV) are depicted. Moreover, to evaluate possible future development, the values of this kind of manoeuvre in case of departure from Lagrangian point (L1) are shown.

	Earth		**Mars**
Cycler optimization	*LEO [km/s]*	*L1 [km/s]*	*LMO [km/s]*
Rauwolf et al.	***6,217***	*2,159*	*12,922*
Friedlander et al.	*7,031*	*2,522*	*12,304*
Chen et al. (1)	*10,320*	*6,404*	*11,346*
Chen et al. (2)	*9,394*	*4,685*	***5,520***

Table 2. delta-V for Taxi RDV with different optimization

As can be seen in the table, the best outbound cycler trajectory is the Rauwolf et al. [8] optimization, and the best inbound is Chen et al.[10], so these optimizations will be considered in the rest of this work.

5.2 Mission schedule

Taking into consideration the chosen trajectories, it is possible to define exactly the launch date, the time of flight between the planets, the arrival date and the subsequent return date. Assuming that after seven synodic period (≈15 years and seven Earth/Mars transit) Earth and Mars return exactly in the same initial conditions, a possible mission schedule is:

Earth departure date	***Travel duration [d]***	***Mars arrival date***	***Mars operation [d]***	***Mars departure date***	***Travel duration [d]***	***Earth arrival date***	***Mission duration [days]***
6-Dec-2028	*151*	*6-May-2029*	*650*	*15-Feb-2031*	*166*	*31-Jul-2031*	***967***
14-Jan-2031	*142*	*5-Jun-2031*	*680*	*15-Apr-2033*	*206*	*7-Nov-2033*	***1028***
5-Mar-2033	*145*	*28-Jul-2033*	*723*	*21-Jul-2035*	*143*	*11-Dec-2035*	***1011***
23-May-2035	*160*	*30-Oct-2035*	*663*	*23-Aug-2037*	*153*	*23-Jan-2038*	***976***
10-Aug-2037	*155*	*12-Jan-2038*	*616*	*20-Sep-2039*	*176*	*14-Mar-2040*	***947***
15-Sep-2039	*169*	*2-Mar-2040*	*594*	*17-Oct-2041*	*186*	*21-Apr-2042*	***949***
20-Oct-2041	*163*	*1-Apr-2042*	*611*	*3-Dec-2043*	*213*	*3-Jul-2044*	***987***

25-Nov-2043	*151*	*24-Apr-2044*	*642*	*26-Jan-2046*	*166*	*11-Jul-2046*	***959***
2-Jan-2046	*142*	*24-May-2046*	*672*	*26-Mar-2048*	*206*	*18-Oct-2048*	***1020***
22-Feb-2048	*145*	*16-Jul-2048*	*715*	*1-Jul-2050*	*143*	*21-Nov-2050*	***1003***
10-May-2050	*160*	*17-Oct-2050*	*687*	*3-Aug-2052*	*153*	*3-Jan-2053*	***1000***
28.Jul.2052	*155*	*30-Dec-2052*	*609*	*31-Aug-2054*	*176*	*23-Feb-2055*	***940***
2-Sep-2054	*169*	*18-Feb-2055*	*587*	*27-Sep-2056*	*186*	*1-Apr-2057*	***942***
7-Oct-2056	*163*	*19-Mar-2057*	*604*	*13-Nov-2058*	*213*	*14-Jun-2059*	***980***

Table 3. Mission opportunities from June 2028 to February 2059

5.3 Cycler deep space manoeuvres

As previously mentioned, the Cycler Vehicle must perform deep space manoeuvres in order to maintain the cycler repeatability (by rotating the line of apses) and to have the ideal hyperbolic excess velocity at planets encounter. The values of the DSM for outbound and inbound cycler are shown in the following table; the outbound DSM are reported in [8], the inbound DSM are calculated from [10].

OUTBOUND			**INBOUND**		
Cycler optimization	**DSM [km/s]**	**Date (Earth passage)**	**Cycler optimization**	**DSM [km/s]**	**Date (Mars passage)**
Rauwolf et al.	0	6-Dec-2028	*Chen et al.*	1,464	15-Feb-2031
Rauwolf et al.	0	14-Jan-2031	*Chen et al.*	1,597	15-Apr-2033
Rauwolf et al.	0,211	5-Mar-2033	*Chen et al.*	0,357	21-Jul-2035
Rauwolf et al.	0,679	23-May-2035	*Chen et al.*	3,766	23-Aug-2037
Rauwolf et al.	0,671	10-Aug-2037	*Chen et al.*	3,334	20-Sep-2039
Rauwolf et al.	0	15-Sep-2039	*Chen et al.*	1,711	17-Oct-2041
Rauwolf et al.	0	20-Oct-2041	*Chen et al.*	0,792	3-Dec-2043
Rauwolf et al.	0	25-Nov-2043	*Chen et al.*	1,464	26-Jan-2046
Rauwolf et al.	0	2-Jan-2046	*Chen et al.*	1,597	26-Mar-2048
Rauwolf et al.	0,211	22-Feb-2048	*Chen et al.*	0,357	1-Jul-2050
Rauwolf et al.	0,679	10-May-2050	*Chen et al.*	3,766	3-Aug-2052
Rauwolf et al.	0,671	28.Jul.2052	*Chen et al.*	3,334	31-Aug-2054
Rauwolf et al.	0	2-Sep-2054	*Chen et al.*	1,711	27-Sep-2056
Rauwolf et al.	0	7-Oct-2056	*Chen et al.*	0,792	13-Nov-2058

Table 4. Outbound and Inbound cyclers deep space manoeuvres [8] [10]

As can be seen in Table 4, the inbound DSM are more expensive. in order to cope with these manoeuvres budget Solar Electric Propulsion is selected. Assuming, for example, an ISP = 6000 s (Thrust = 4 N) and an initial Cycler Vehicle mass of 100 t, the inbound fuel consumption is of 40 t while the outbound fuel consumption is of 5 t without considering refuel operation during the approximately 30 years of operations. This leads to a refuelling scenario made of 3 Falcon Heavy for the inbound and 1 for the outbound. It is important to specify that the assumption of this high specific impulse is due to the fact that the study carried out is futuristic, where it is reasonable to think of electric propulsions much more advanced than the current ones.

5.4 Taxy RdV

Using the chosen optimizations, we consider the hyperbolic rendezvous to transfer the taxi from the circular low orbit to the Cycler Vehicle. The outbound worst case (1st November 2026 case) results in a total ΔV cost of 4,77 km/s for the Earth RdV, while the inbound worst case on Mars results in a ΔV of 4,1 km/s.

If we assume that the taxi is equipped with a cryogenic propulsion system with Isp = 450 s and the propulsion module has a structural coefficient ε of 0.1, we can use the Tsiolkowski rocket equation to determine the final mass that will dock with the Cycler Vehicle and thus that will arrive close to the target planet.

Considering two different types of launchers, Falcon Heavy and SLS Block 2, that are capable of carrying in LEO respectively 55 t and 130 t and analysing separately the three manoeuvres ΔV1, ΔV2, ΔV3, the rocket equation brings to the following results:

- Falcon Heavy, payload on cycler: 15,79 t
- SLS Block 2, payload on cycler: 37,32 t

5.5 Descent Phase

When the Cycler Vehicle, with the taxi attached, arrives near to the target planet, the taxi leaves the cycler with the crew on board and carries out the descent phase. On Mars the final speed is too high to perform a direct entry, thus a preliminary aerocapture manoeuver must be implemented to slow down the spacecraft before performing the descent phase and allowing the landing on the surface. The aerocapture represents a critical phase for the crew survival, because the resultant accelerations must be maintained below a certain limit, sets in this case to 5 g.
Considering an hyperbolic approach, the trajectory with the greatest atmospheric entry speed at Mars (125 km) turns out to be of 12,5 km/s and is considerably less for some dates.
Assuming an entry FPA of about -10° in Mars atmosphere (125 from the surface) and a gradually decreasing the downward lift that is used to maintain constant altitude (50 km) during the aero-cruise phase [12], in order to exit with a the desired speed of 4.2 km/s, the limit of 5 g is exceeded in two cases (6th December 2028 and 14th January 2031 Earth departures). For this reason, it is necessary to adopt a solution to reduce the acceleration peaks. These strategies include for example:

1. Use drag to reduce velocity before reaching aero-cruise altitude;
2. Entry in atmosphere with a steeper angle in order to bleed off more speed prior to aero-cruise;
3. Reduce entry speed with the propulsion system;
4. Use propulsive thrusting during the initial aero-cruise phase (0,358 km/s for the worst case [12]).

Setting the aerocapture phase in a way that the g-limit is respected and the exit speed from the Mars atmosphere is approximately 4,2 km/s, the taxi travels through an elliptic trajectory that goes up to 6000 km from Mars surface (crossing Phobos orbit) before re-entering into the atmosphere and finally landing to Mars surface. Considering this second atmospheric entry speed of 4,2 km/s, the propulsive ΔV necessary to perform the landing is 0,58 km/s [13].
On the Earth side the atmospheric entry speeds are close to Orion performances. Therefore a direct descent to Earth surface using the atmospheric drag is considered feasible without additional manoeuvers [14].

5. Sustainability

A useful figure of merit to underline the benefits coming from an architecture similar to the cycler one comes from a sustainability comparison between the Cycler and the NASA DRA 5.0 [7]. With the term “Sustainability” it is intended the costs related to the required launches and orbiting elements deployed to build up the missions for an extended exploration campaign covering 14 synodic periods. DRA 5.0. plan relies on 5 SLS Block II launches per mission for the in-space segment (thus not comprising the Mars Ascent Vehicle). The Cycler option would need 3 SLS Block II launches for each Cycler Vehicle injection on orbit in order to start the cycle (thus only the first time, due to the fact the Cycler Vehicle will be re-utilized for the subsequent journeys). Having a Cycler Vehicle in the outbound and another one in the inbound leg, a total amount of 6 SLS Block II launches would be required. Subsequently 1 SLS Block II launch the Earth Taxi that would be needed to transfer the crew to the Cycler Vehicle for each mission plus an Ariane64 for the cargo delivery. The need of logistic is based on preliminary estimation and shall be refined in future studies. Additionally the propellant for the Mars Taxi, that has to be refuelled in Mars orbit, can be provided by the same concept (called WDV) described [15], which has a performance of 40 t of LOX/LH2, compatible with the Mars Taxi needs. This can be launched with one SLS Block II once per synodic period. Finally there may be various strategy for the Cycler Vehicles refuelling: it could be used “light” launchers (with small amount of propellant) at each Earth-passage of the Cycler Vehicles, or less frequently with bigger launchers such as Falcon Heavy. With this last strategy the 3 launches are required for the inbound CTV (1 every 5 cycles) and 1 launch every 14 cycles for the outbound CTV. This approach is compatible with a 100 ton wet CTV.
Therefore, after 14 synodic periods, NASA DRA 5.0 approach would require 70 SLS launches while the Cycler architecture would require only 34 SLS, 14 Ariane64 (cargo) and 4 Falcon Heavy (refuel) cutting launch costs of about 50%. Additionally the Cycler vehicle would be reused for 14 missions avoiding the construction of 14 transfer vehicles.
Moreover the travel performed in the firsts synodic periods could be used as an intermediate step between Cis-Lunar missions and the Martian surface exploration. The crew, after the RdV with the Cycler, can remain on board for a 2 years mission in Deep Space, testing the harsh conditions before the Martian surface infrastructure will provide the

capability to land and to ascent safely. This approach would spread the financial efforts deploying the infrastructure in a stepwise manner increasing the sustainability of the Martian exploration.

6. Conclusion

The use of Earth-Mars cycler as baseline for enduring mission towards Mars is an alternative concept which offers several advantages. In the long term the cycler based architecture offers a great saving of propellant and launches. It is clear that the mission concept allows frequent transfer between Earth and Mars with a deep re-utilization of the vehicle. The exploitation of Earth-Mars cycler is surely useful in the case of an enduring colonization of Mars but it can be compared with NASA DRA5.0 also in case of a single mission, deploying the transportation system gradually spreading the cost of the infrastructure across several years.
In the near term this study will focus on the functional analysis and a better definition of the involved system in order to validate the concept here proposed.
Eventually, this feasibility study is important because gives the bases to several further studies. We hope that the efforts will contribute to a future enduring Martian colonization, in order to fulfil another giant leap for mankind.

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