Maximizing Upper Stage Performance As a Method of Enabling Near-Term Manned Mars Free-Return Mission

EXTENDED AND AMENDED EXCERPT FROM *SPACE IS MORE* TEAM REPORT.

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ABSTRACT

The paper describes the launch system architecture employed by the *Space is More* team in its Inspiration Mars Contest mission proposal, designed to overcome the severe payload mass limitations of current and proposed launch systems. Existing know-how and experience in in-orbit payload assembly is noted and focus is placed on maximizing upper stage capability. The feasibility of near-term serious improvements in that regard is demonstrated. A Boeing's Large Upper Stage is modeled and scaled-up through a simple tank stretch, numerical simulations being then conducted to investigate the outcome. The project uses both literature data analysis and independent deep-space simulations to determine stage performance on a number of free-return Mars flyby trajectories, thereby defining the options for the *Space is More* manned mission proposal. An allowed payload mass growth of approximately 50% or more is reported for high-energy trajectories, greatly improving the trade-offs that can be made between the spacecraft's capability, design margins and overall feasibility, though it should be noted that the utilized stage model still shows room for improvement. The study relies heavily on the Python programming language and its scientific libraries, like SciPy, PyKEP and PyGMO.

KEYWORDS: free-return trajectories, Mars flyby, upper stage, numerical burn simulations, Python scientific computing

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1. Introduction

During the 16th Annual International Mars Society Convention, the Mars Society announced the launch of an international engineering competition for student teams to propose design concepts for the architecture of the Inspiration Mars mission [1]. The task was to design a two-person Mars flyby mission for 2018. The following work is an extended and amended version of Chapter II from the finalist *Space is More* team design report (which can be downloaded from [2]), with improved text formatting. Other report chapters contain additional information that may help put this highly focused work in perspective.

Very early in the project we made the decision to put the mission on a relatively short, free-return trajectory with no big deep space maneuvers, for a number of technological, political, medical and psychological reasons. When, after this decision, we concentrated on formulating an early estimate of the mass value for our craft and turned to current systems for comparisons, we immediately realized they had certain relevant limitations. Currently used launch vehicles have a stated LEO payload mass no higher than approx. 29 metric tons (Delta IV Heavy, [3]). This corresponds to a Mars free-return payload no higher than 5 metric tons [4] – at least an order of magnitude too little for a manned mission. Systems to be introduced in the future improve this figure to about 10 metric tons for Falcon Heavy [5] and about 20 metric tons for SLS with an advanced upper stage, this last configuration having the bare minimum capacity required for a manned Mars mission.

However, in January 2018 over five decades will have passed since the first on-orbit rendezvous, made by Gemini 6 and 7 on December 15th 1965. Over fifty years will also have elapsed since the first successful docking, made on March 16th 1966, when Gemini 8, under the command of Neil Armstrong, rendezvoused and docked with an unmanned Agena Target Vehicle. By the end of the decade the rendezvous & docking technology was known well enough to be used in critical phases of Apollo flights. In the 90s, the collapse of the Soviet Union and opening of the Russian space program allowed the Space Shuttle to fulfill one of its original roles – building and servicing a space station. Both MIR and ISS missions led to a great increase in skill and knowledge. As of now, NASA and its international partners have mastered extremely complex, long term on-orbit assembly of structures weighing hundreds of tons out of smaller parts and pieces that can be sent to orbit individually, within the current launch space and weight limitations. We can build literally every element of a manned interplanetary spacecraft using current launch and on-orbit assembly technology, except for one: *propulsion*. There is, currently, no technology for building rocket stages from parts in orbit. Furthermore, we will not have access to a sufficiently developed in-orbit cryogenic propellant storage by 2018. In consequence, the biggest constraint for deep space missions is upper stage capability – we may be able to assemble functional payloads weighting hundreds of tons, but, with such a disproportion between assembly and propulsion technology, we cannot send them anywhere.

Thus, the following sections explore the possibility of maximizing upper stage capability for a manned flyby mission (and, therefore, S/C capability), within limitations imposed by current technology and the constraints related to running the mission in 2018. Firstly, an already proposed high performance stage is modified and modeled, then its performance is benchmarked at the number of real-world trajectory options. The question of upper stage payload and the way of transferring that payload to the rendezvous point – in this case into LEO – is not covered here. That information can be found in the other chapters of the *Space is More* report.

2. SLS – carrier of the extended Large Upper Stage

NASA's Space Launch System with a DUUS/LUS class upper stage is the most capable launch vehicle that could be available for a 2017/2018 launch opportunity [6]. However, as stated in that source, creating an upper stage fit for an Inspiration Mars mission would require a significant increase in development speed. Given this requirement, we consider the development of a generic multi-purpose rocket/upper stage combo for a manned mission program an architectural mistake, for reasons explained above.

The best concept for short and mid-term deep space exploration without in-space depots, including a Mars flyby of this sort, calls for using an SLS rocket as, essentially, a carrier for a big, cryogenic, maximum-performance LH2/LOX upper stage, which, if the mission is planned for the near future, can just be a modified DUUS/LUS with stretched tanks (fig. 1), as presented in this paper. The following section contains a summary of the most important parameters and models relevant to our modified or extended LUS scenario. However, it must be borne in mind that we are limited practically to using only publicly available data. Furthermore, it seems probable that much of the information needed for a professional analysis of such a mission concept is ITAR restricted. In consequence, what follows is as good an analysis of an extended LUS scenario as we could create in our information and tool restricted position.

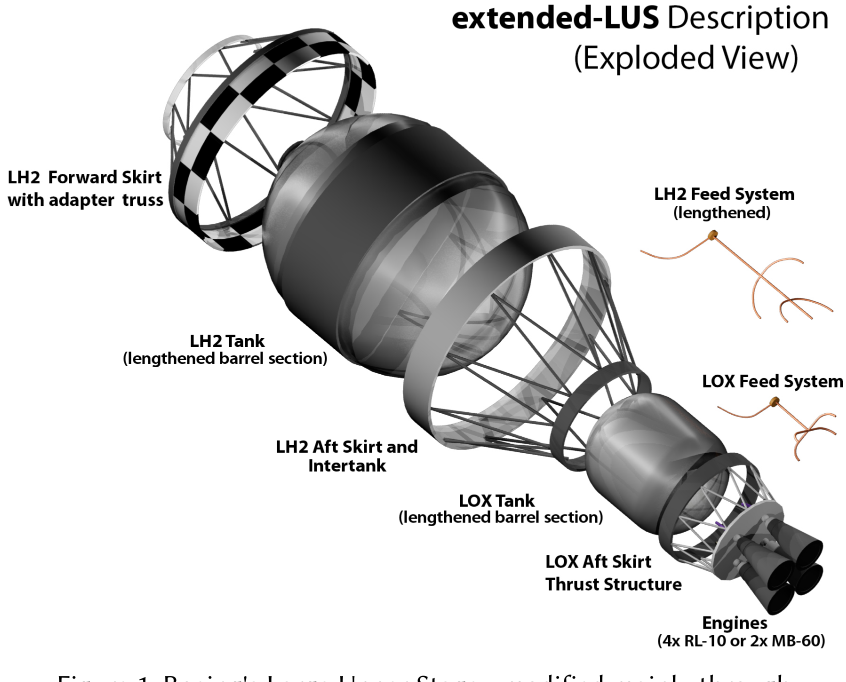


Figure 1: Boeing's Large Upper Stage - modified mainly through a tank stretch

2.1 LUS Parameters

We chose the Boeing Large Upper Stage as the basis for a "big upper stage" concept, one we then named extended LUS or ext-LUS, mainly because of its light design, focused on minimizing residues, and a high performance combination of engines and fuel, as well as its large, but not excessive overall size and the fact that its key (if preliminary) parameters were made publicly known in the publication *The Space Launch System Capabilities with a New Large Upper Stage* [4]. A summary of known parameters of a RL10 and a MB60 LUS is given in table 1.

	LUS 4xRL-10	LUS 2xMB-60
LH2 (TLI mission)	15573.64 kg	-
LO2 (TLI)	91105.39 kg	-
RCS prop	649.54 kg	-
Dry mass	11853.73 kg	-
Stage total mass (TLI)	119182.30 kg	-
Usable propellant (TLI)	105000 kg	105000 kg
Usable propellant (LEO mission)	63000 kg	63000 kg
Residual propellant	1679.03 kg	-
LEO parking orbit	240.76 km	240.76 km
LEO payload	93100 kg	97000 kg
Tanks propellant load max	125000 kg	125000 kg
Isp	462.5 s	465 s

Table 1: Boeing LUS parameters

The information about the 4x RL-10 configuration is fairly detailed. The higher performing 2x MB-60 is something more of a mystery, while 1x J2-X is not presented here at all, due to its poor high-C3 performance. Given the data, we chose RL-10 as “safe”, base option, but also present calculations for MB-60, considering that MB-60 LH2/LOX liquid rocket engine development is to be undertaken by JAXA in support of the Space Launch System (SLS) Program [7]. Its design is heavily based on finished work, with over 90 % of the components being complete [8]. Please note that while MB-60 is referred to as MARC-60 in recent documents (the acronym that stands for *Mitsubishi Aerojet Rocketdyne Collaboration – 60 klbf engine*), this paper still uses the older, more widespread name.

2.2 Scaling From LUS to ext-LUS

It must be noted that the scale up model is limited in its finesse. That is due, firstly, to the fact that we do not know the exact parameters and behavior of an SLS rocket and, secondly, because even if the data were available, writing a reliable numerical rocket launch simulator would probably be impossible within the project completion deadline. The change in lower stack performance due to change in mass of upper composite is unknown, and would probably be significant, given the SLS's hydrolox core. In consequence of all those complicating variables, we simplified the model by making the ext-LUS wet mass fixed at one specific mission point based on the Boeing's LEO mission scenario.

The equation is as follows:

$$_{ext}LUS\ gross\ mass\ =\ Ascent\ prop.\left(LEO\right)+\ Dry\ mass\ +\ RCS\ prop.\ +\ Residual\ fuel\ +\ LEO\ payload$$

In other words, the stretched stage alone weighs as much as original LUS with its maximum LEO payload. The result is 170 282.3 kg for fully fueled 4x RL-10 ext-LUS. We assume that, given no change in gross mass, the amount of propellant consumed during ascent to LEO is the same as in Boeing's reference, i.e. 63 metric tons. This gives 107 282.3 kg mass in LEO.

For a 2x MB-60 ext-LUS, RCS propellant and residual propellant values were assumed to be the same. although, in reality, the second value might be a little lower, due to the reduced number of fuel lines in the thrust structure area. Precise change of dry mass is unknown and educated guess estimates had to be made.

Two MB60 engines weight about as much as four RL-10 engines (2604 lb vs. 2656 lb), while a single J2-X engine is about two times heavier. However, subtracting 4x RL-10 mass from LUS dry mass and adding 1xJ2-X mass gives a result 1334 lb higher than the stated J2-X

LUS dry mass. From there, an educated guess was made. Assuming the savings come from eliminating three out of four per-engine stage structure components, one per-engine stage structure component weights about 200 kg.[1] Thus, 2x MB-60 LUS dry mass would be about 400 kg lower than in a 4x RL-10 LUS. That means 173 782.3 kg for fully fueled 2x MB-60 ext-LUS and 110 782.3 kg after ascent to LEO.

The next step in the calculations involved scaling the stage up, to accommodate all this mass. We assumed that dry mass will scale proportionally to the propellant tank capacity stretch, as the ratio of propellant mass to tank mass depends on propellant density, pressure and material properties [9], all three remaining constant. However, excluded from scaling are some known elements with constant mass (like engines, thrust structure or intertank). Since, in either configuration, the engines weighs about 1200 kg and an individual engine structure component weighs about 200 kg, this constant mass was cautiously assumed to be 3000 kg for 4x RL-10 and 2600 kg for 2x M-B60, a rather undervalued figures. The original LUS has to carry the same gross mass (with payload), just distributed differently – it may be assumed that there would be no significant weight penalty from the different distribution that our model does not account for. Residual fuel mass was assumed to scale proportionally to the extension in tank capacity. Values obtained under such assumptions agree with historical data [10]. The RCS fuel value was assumed to be constant, i.e. the same as in a normal LUS, and was treated as an inert mass, unconsumed till the end of the TMI burn, like a ballast.

Furthermore, later in the design process of our ext-LUS, we added additional hardware to the stage – its description and the rationale for its inclusion is presented in the other chapters of the *Space is More* report. Here, we provide just a sketch of its mass breakdown for calculation clarity, since we present the final figures for our ext-LUS configuration below. The stage modifications mass breakdown is given in table 2.

The rest of the mass, up to the limit based on the data point, is fuel. The model in its mathematical form is presented in Equations 1 below and embedded with all LUS data in an EXT_LUS Python module, where a Fixed Point Iteration method is used for solving. The results are presented in table 3 and used as the basis for an ext-LUS payload analysis.

It is worth noting that the model described above assumes presence of a big, 8.4 m diameter payload fairing, weighing around 10-12 tons. Such a fairing would be unnecessary for an ext-LUS flight – a much smaller, jettisonable aerodynamic "cap" could be used instead, resulting in increased performance. A detailed analysis of such an increase is not available but we estimate that for a fast, free-return Mars mission it should be several hundred kilograms, since a 1 mt increase in LUS LEO payload mass in our model results in a payload increase of approximately 400 kg for C3's around 39 km^2s^{-2}. Also short fairing and unmanned, "no-payload" character of a flight allow for stretching the upper stage beyond 60 ft in length, the current Not To Exceed design requirement for an SLS upper stage [11].

1 Note: the authors are aware that the RL10 and J2X thrust structures are, in reality, quite different – this is a simplified calculation assumption.

Docking compartment:		
	Hatch door	80 kg
	Airlock + thermal insulation	200 kg
	NDS	370 kg
Tank interior accomodations:		
	ECLSS pipes	50 kg
	wall attachments	1.5 kg
	g-bike attachment	0.5 kg
Total stage modifications mass:		702 kg

Table 2: Stage modifications mass breakdown

	4xRL10 ext-LUS	2xMB60 ext-LUS
Stage burnout mass:	17408.4 kg	17315.3 kg
Residual fuel	2081.4 kg	2130.3 kg
Usable prop:	89873.9 kg	93467.0 kg
Tank stretch (by volume):	23.964 %	26.878 %
Total in-LEO mass:	107282.3 kg	110782.3 kg

Table 3: ext-LUS stage data

$$
\begin{cases}
LUS\ burnout\ mass = LUS\ dry\,mass + RCS\ fuel + LUS\ Residual\ fuel \\
{}_{ext}LUS\ usable\ LEO\ prop. = LUS\ LEO\ payload + LUS\ burnout\ mass - {}_{ext}LUS\ burnout\ mass \\
Tank\ stretch = \dfrac{{}_{ext}LUS\ usable\ LEO\ prop. + ascent\ prop. + {}_{ext}LUS\ Residual\ fuel - LUS\ max\ prop.\,load}{LUS\ max\ prop.\,load} \\
{}_{ext}LUS\ Residual\ fuel = LUS\ residual\ fulel \times (1 + Tank\ stretch) \\
{}_{ext}LUS\ burnout\ mass = LUS\ dry\ mass + (LUS\ dry\ mass - mass\ const.) \times Tank\ stretch + Stage\ mods \\
\qquad + {}_{ext}LUS\ Residual\ fuel + RCS\ fuel
\end{cases}
$$

Equations 1: Scaling-model equations

3. Trajectories and ext-LUS performance

An ideal trajectory for a Mars flyby should have a number of properties, which can, in reality, be mutually exclusive, due to celestial mechanics. The most important ones include:

- No big DSMs. They not only add risk and complexity to the mission, but also increase deep-space-stack mass in an ineffective manner, requiring it to carry a relatively low Isp propellant (currently, it has to be hipergolic) and expend it without the help of the Oberth effect (excluding a close Mars flyby itself).
- The shortest flight time possible. Flight time has a significant impact on ECLSS design requirements and parameters (i.e. mass), the crew's psychological and medical safety, as well as the chances of completing the project (political and public perception and support).
- Minimal required C3. Characteristic energy (C3) is twice the specific orbital energy of an object escaping the Earth gravitational field, therefore it has a direct impact on potential payload mass.
- Escape trajectory asymptote declination equal to or lower than the launch site latitude, and permitting flight within range constrains. Declinations higher than launch site latitude require ascent to orbit with an inclination higher than minimal, thereby causing payload reductions. A launch azimuth exceeding range constrains could make the mission impossible or cause severe payload reductions due to the dogleg maneuver.
- To comply with IM competition rules, the launch window should lie in 2018.

After weighting each point and discussing the relevant pros and cons, a fast, free-return trajectory, very similar to the IM reference trajectory, was chosen as the most compatible with the above goals. Therefore, we present, first, a short description of a tool used for payload assessment, then, second, an analysis based on the data provided in the Inspiration Mars feasibility study and in its sources, and then, finally, an analysis based on independent deep-space simulations.

3.1 Authorial Payload-Assessment Tool

An authorial Python code was written for payload mass assessment. A numerical simulation of a burn starting from 130nm parking orbit was programmed, using NumPy and SciPy libraries and a spherical Earth gravity model, to account for gravity losses, since an extended, RL-10 or MB-60 engined stage has a relatively low thrust-to-weight ratio. To optimize performance, a two-burn sequence with single coast period in between, on an elliptical, high apogee orbit, was selected[2] and hardcoded. The code consists of four closely connected functions that are placed in an burn_sym module. The main function of the four, named c3, returns by itself only C3 value for a given payload mass, however. Nelder-Mead simplex algorithm is incorporated to determine payload mass for a given C3 when needed, which may seem inefficient as it stacks two single-variable minimizations, one inside the objective function of the another. The benefit of this approach is reliable convergence to the global minimum. Appendix A on page 17 presents the obtained ex-LUS C3-payload chart.

The simulations take in the account propellant loss due to boil-off, but it is estimated, rather than calculated precisely, due to data availability limitations. Specifically, no exact figures concerning LUS/DUUS predicted in-space boiloff rate were found. Current LH2/LOX upper stages, like the Centaur, are prepared only for missions lasting no longer than a few hours and involving carrying payloads targeted at GEO. Consequently, they have either minimal or no in-space insulation, which results in boil-off of approximately 1.5-2.0

2 The radiation risk associated with this choice did not noticeably increase the mission's overall radiation budget.

percent of full fuel load per day [12]. If one of those existing stages were modified accordingly, with near-term passive systems it could go down to as little as 0.1 percent per day. A purpose built stage could further improve this figure by about an order of magnitude [13]. There is no publicly available information about planned LUS insulation. To accommodate wet workshop requirements, we assume an ext-LUS would have a modified, habitat-rated insulation similar to the one used in a DUUS. Namely, a *Spray on Foam Insulation* (SOFI) plus *Multi Layer Insulation* (MLI) combo, under a MLI aerodynamic ascent shield, as presented in DUUS paper [14]. A DUUS has a total stage life up to seven days, but this figure also includes other constrains. Since a DUUS's performance parameters usually match or exceed LUS's parameters and no other information is publicly available on this issue, no special ext-LUS estimate was made.

The time between an ext-LUS's orbital insertion and TMI is assumed to be four days, considered by us as the minimum for in orbit operations in terms of architecture design, and equal to planned LEO loiter time of Constellation-era Ares's V EDS [15], designed to perform a very similar mission. Over the course of four loiter days the slope of an instantaneous C3 function suggests that the boil-off rate should be no greater than 0.3 %/d, and rates lower that 0.2 %/d are not necessary. Thus, a 0.2 %/d boil-off rate was taken as baseline. Plugging boil-off rates of existing stages, however, has a severe impact on the achievable payload.

3.2 ext-LUS Payload on IM Reference Trajectory

For purposes of this analysis, the instantaneous C3 achievable in subsequent days of LEO loitering was plotted, using Matplotlib, against an IM trajectory C3 curve taken form [6]. Maximum payload mass was automatically optimized. Note that the C3 line is purely hypothetical, as the stage would have a tight *Trans Mars Injection* (TMI) window due to orbit precession and (to a lesser degree) required departure hyperbole evolution.

Figures 2 and 3 presents the two most important charts, attained for a minimal-energy departure scenario (on 2018-01-04), complemented with essential information. The payload mass is rounded to hundreds for display.

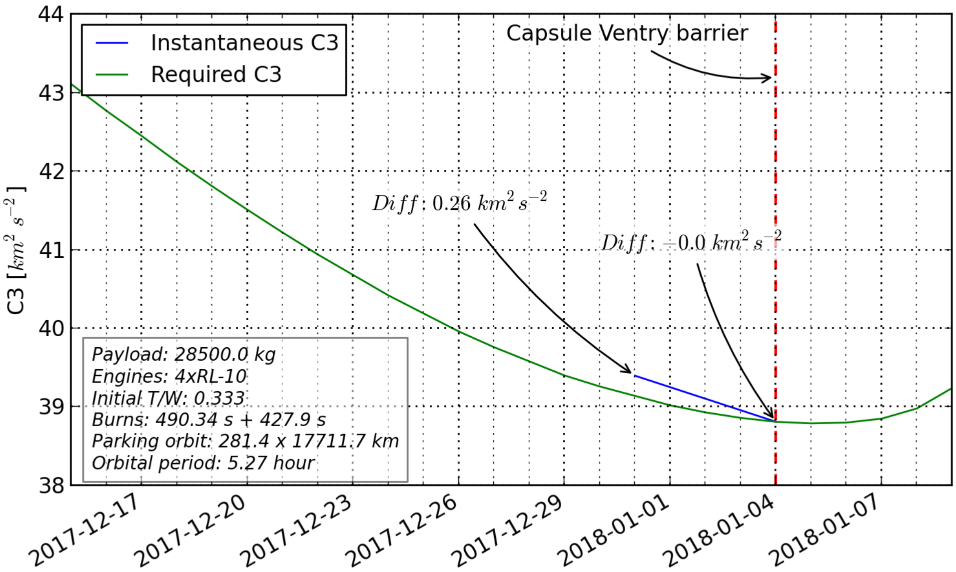


Figure 2: Stretching the 4x RL-10 based stage gives considerable payload increase. Note the slope of inst. C3 function (at 0.2 %/d boil-off).

In payload figures, 702 kg of stage modifications are treated as the stage dry mass and not as payload. With stage modifications mass set to zero, the payload for the 04.01.2018 TMI is 29 431 kg for a RL10 and 31 937 kg for a MB60 based ext-LUS, respectively.

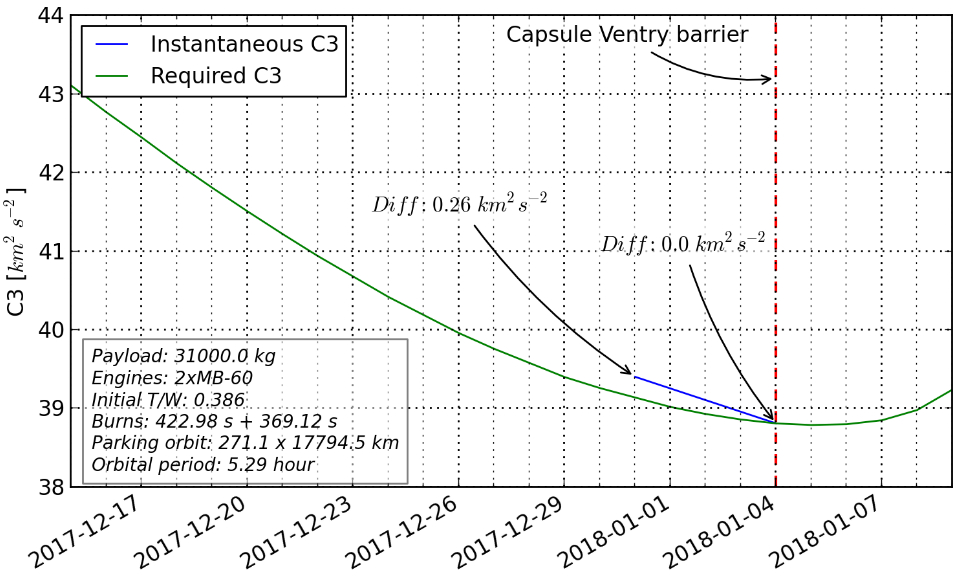


Figure 3: The 2x MB-60 based stage performance.

3.3 Independent Trajectory Modeling

For the purposes of this analysis another tool was developed, on top of the PyKEP and PyGMO libraries. A problem, based on modified PyKEP's *Multiple Gravity Assist with one Deep Space Maneuver* (MGA-1DSM) built-in module code was constructed and optimized using PyGMO's *Non-dominated Sorting Genetic Algorithm* (NSGA II) and *Self-adaptive Differential Evolution* (jDE). Due to MGA-1DSM heritage, a practical total mission ΔV limit of 1 cm·s^{-1} was used to classify a trajectory as essentially free-return. This is two to three orders of magnitude less than TCM expenditures of a current one-way unmanned Mars missions (see Appendix C: *Historical TCM ΔV Magnitudes*). Discussed earlier payload-assessment tool was used for payload mass figures. Where appropriate, Earth reentry speed was limited to 14200 m/s (inertial, at the height of 56.5 km above surface) in simulation constrains.

What's presented further in the following subsections is, firstly, a description of the modules used to generate and handle results, then, secondly, the results of a trajectory options investigation.

3.3.1 MGA-1DSM Class Modifications

The core of a problem's objective function was not tweaked and represents the original GNU GPL 3 licensed code written and maintained by Dario Izzo et al. [16] from ESA *Advanced Concepts Team*. The part responsible for fitness vector return was modified to include penalties for exceeding total ΔV limit and reentry speed limit, as well as allow for three-objective optimization. During single-objective optimization run, the code basically search for any trajectories that comply with limits, that include maximum C3, time of flight, minimal planet distances as well as mentioned ΔV and reentry speed. During multi-objective run, the algorithm “sees” directly also reentry speed and C3 energy as a fitness indicators. A compare-fitness function was reimplemented to ensure a more convenient behavior.

Larger changes were applied to PRETTY (responsible for description printing) and PLOT methods of the MGA-1DSM class, which were consolidated together and modified to:

- Return (when called for) a vector of trajectory parameters like dates of encounters, flyby distances, reentry velocity, etc. for a given solution vector;
- Allow for spacecraft-planet distance plot over time;
- Provide a more descriptive code of the simulation than in the objective function, by keeping comments and print statement near the relevant code parts.

3.3.2 Evolution Handler

An Evolution_Handler class was written to streamline problem's parameters setting, optimization runs and results storage. The code allow for separated single objective and multi objective runs and storage in a separated Pandas DataFrames (containing both solutions vectors and human-readable trajectory parameters) as well as injecting single objective optimization results to multi-objective problem's populations. HDF5 format is employed for on-disk permanent DataFrames storage.

Alongside Evolution_Handler the trajectories module also contains a hdf_plots class that allow for ad-hoc plotting of HDF5-stored data.

3.3.3 Improving Planet Ephemeris

All of the work was done on PyKEP's compilation that predated a V1.1.3 PyKEP release [17], which introduced non keplerian solar system planet ephemerides based on JPL low-precision polynomial expansion [18]. This drawback of the installed software resources was realized and a fix had to be applied, as the previous implementation give position errors in order of millions of kilometers, some three orders of magnitude larger than the current.

Given that the scope of this study is highly focused on 2018 free-return Mars flyby trajectories the fix developed was a special rather than general solution. A jplephem 1.2 Python package coupled with Jet Propulsion Laboratory DE421 ephemeris was utilized to obtain high precision position data, to which a set of classical orbital elements was fit over a period of 30 days. This 30-day period had to be manually adjusted to cover a time window for a given spacecraft-planet encounter event. Using orbital elements adjusted in a such way a standard PyKEP's planet object is constructed. The written planets module alongside standard fitting routines has a “shortcut” - namely fast methods returning sequences of planets build from hard-coded earlier-adjusted orbital elements. The position errors during fitting period fall within several hundreds of kilometers from DE421 ephemeris. This is more than enough accuracy for a trajectory model that does not include any perturbations. Example errors plot is provided in figure 4.

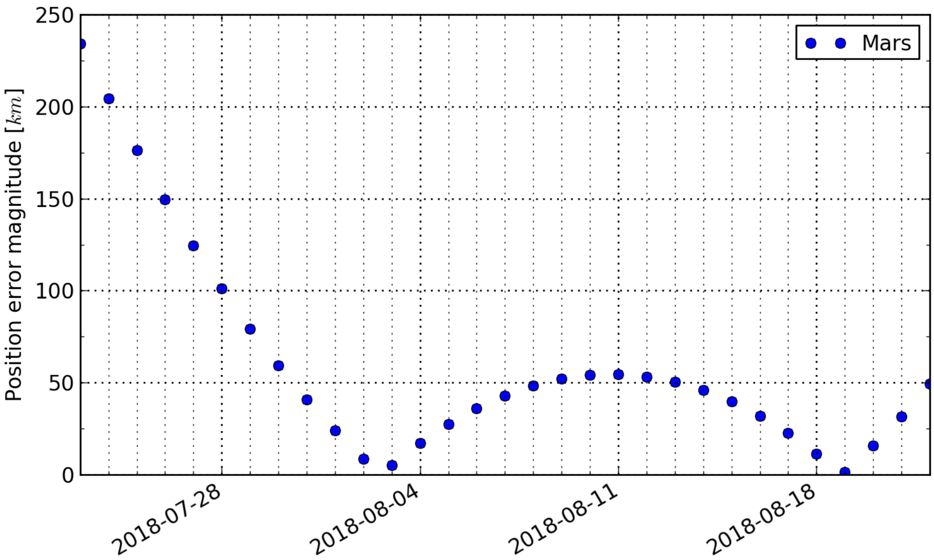


Figure 4: Mars position errors during 2018-flyby window.

3.3.4 Recreating a 500-Day Class Trajectory

Evolutionary algorithm optimization yielded a slightly lower minimal C3 energy requirement and slightly moved minimal-energy departure date than for the reference IM trajectory. Table 4 presents the maximum-payload trajectory parameters found, while figure 5 shows its visualization. Figure 6 on page 12 presents spacecraft-planet distances during the

Leg 1 228.4447 d	Earth depart:		2018-Jan-05 03:19:39
		C3	38.20 km²s⁻²
		Dec	-6.789 deg
		Rt Asc	271.091 deg
Leg 2 271.7509 d	Mars flyby:		2018-Aug-21 14:00:04
		Flyby dist	100.044 km
	Earth arrival:		2019-May-20 08:01:23
		Vinf	8815.840 m/s
		Entry Vel	14198.086 m/s
ext-LUS	4x RL-10	Payload	28800 kg
		Parking orbit	281.2 x 17555.7 km
	2x MB-60	Payload	31400 kg
		Parking orbit	271.0 x 17636.9 km

Table 4: Results of an optimization focused on maximizing payload mass

flight for Earth, Mars and Venus. Declination of the outgoing asymptote allows for a launch from KSC to LEO with the lowest inclination possible. Entry velocity is on upper end of Orion ERP capability. The simulation is somewhat sensitive to the minimal Mars distance allowed during swing-by. There is number of inconsistent figures about the planned IM flyby distance, ranging from 100 km, through 100 miles, to "closer than ISS is to Earth". For purpose of the minimal C3 evaluation, this distance was set to 100 km as in [5]. The payload figures, as previously, exclude stage modification mass and are rounded to hundreds of kilograms. For the 2xMB60 based ext-LUS with stage modifications mass set to zero and following the presented trajectory, the maximum achievable payload mass peaks at 32 334 kg.

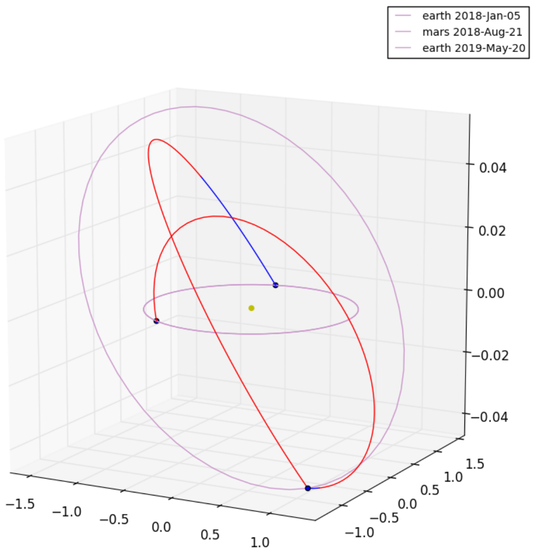


Figure 5: Visualization of a maximum-payload trajectory for an early 2018 window.

An example of a free-return trajectory providing larger margins, with flyby distance set to 200 km and C3 set to 39.60 km²s⁻² is presented in table 5. Note that in this variant the launch date changes to the December 2017.

If a payload trade-off were considered acceptable, with a higher C3 and/or minimal distance Mars approach, Earth reentry speed could be further reduced. A reduction is also possible if some small deep space maneuvers are added. Table 6 presents a trajectory optimized for Earth reentry speed, that reduced the speed by about 485 m/s compared to C3-optimized solution, with C3 boundary set to 45.1 km²s⁻², a shortest distance to Mars of 100 km and a cumulative DSM magnitude limit set to 10 m/s of ΔV, a value comparable to the nominal total TCM expenditure of a current robotic Mars missions.

Other trajectory options besides the 500-day variant were also studied. They were discarded for the present mission, but two variants are presented in a following subsections as alternatives. In addition, Appendix B on page 18 contains the *Trajectory Pareto Fronts* for all depicted options, made from HDF-stored results.

Leg 1 234.8961 d	Earth depart:	2017-Dec-28 02:25:45	
		C3	39.60 km^2s^{-2}
		Dec	-9.492 deg
		Rt Asc	264.882 deg
Leg 2 267.2184 d	Mars flyby	2018-Aug-19 23:56:06	
		Flyby dist	200.323 km
	Earth arrival	2019-May-14 05:10:40	
		Vinf	8569.385 m/s
		Entry Vel	14046.387 m/s
ext-LUS	4x RL-10	Payload	28000 kg
		Parking orbit	281.5 x 17914.9 km
	2x MB-60	Payload	30500 kg
		Parking orbit	271.2 x 17999.9 km

Table 5: Larger-margin free-return trajectory data

Leg 1 249.7882 d	Earth depart:	2017-Dec-08 17:04:43	
		C3	45.10 km^2s^{-2}
		Dec	-11.973 deg
		Rt Asc	249.165 deg
		DSM after	29.912 days
		DSM mag.	0.45139 m/s
Leg 2 254.4066 d	Mars flyby:	2018-Aug-15 11:59:46	
		Flyby dist	102.333 km
		DSM after	171.236 days
		DSM mag.	9.55739 m/s
	Earth arrival:	2019-Apr-26 21:45:20	
		Vinf	8010.594 m/s
		Entry Vel	13712.631 m/s
ext-LUS	4x RL-10	Payload	24800 kg
		Parking orbit	282.6 x 19367.3 km
	2x MB-60	Payload	27200 kg
		Parking orbit	272.0 x 19468.6 km

Table 6: Reentry speed optimized trajectory data

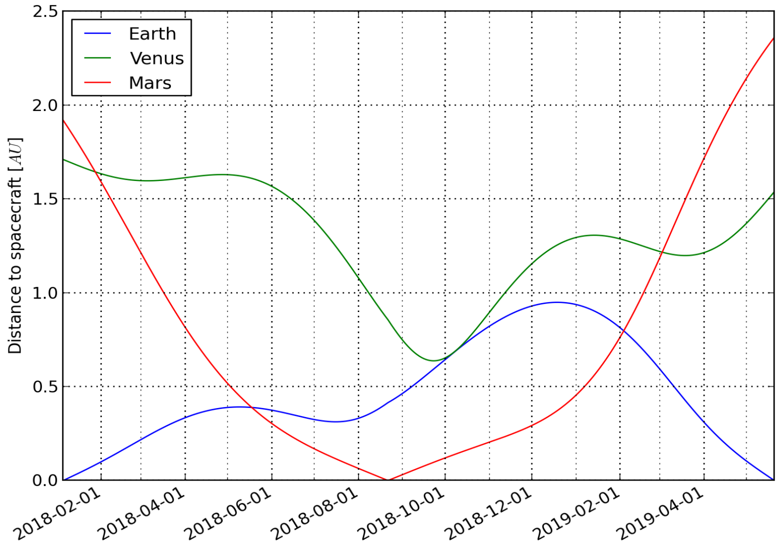


Figure 6: Spacecraft-planet distances (in AU) over time for a C3-optimized trajectory

3.3.5 Alternative 730-Day Class Trajectory

The parameters of a more classical, two-year free-return flyby trajectory were also investigated, despite the fact that flight time alone would make this an undesirable choice. A launch window was found in 2018 (around May). Example trajectory parameters are shown in table 7, figure 7 shows its visualization.

It is clear from the data, that this trajectory would also have a few other drawbacks. First, the flight time is not only significantly longer, but also divided unevenly between legs, the second one being six and a half times longer than the first. The ship would arrive at Mars about three months after launch, and then the crew would have to wait 21 months to come home. This is undesirable for both psychological and political reasons. Secondly, the Mars flyby distance is large, usually tens of thousands of kilometers, and potentially almost 100 000 km. At that distance, Mars would be only around 7.5 times larger than the Moon as seen from Earth. Thirdly, a large portion of the second leg is spent further away from the Sun, behind Mars orbit.

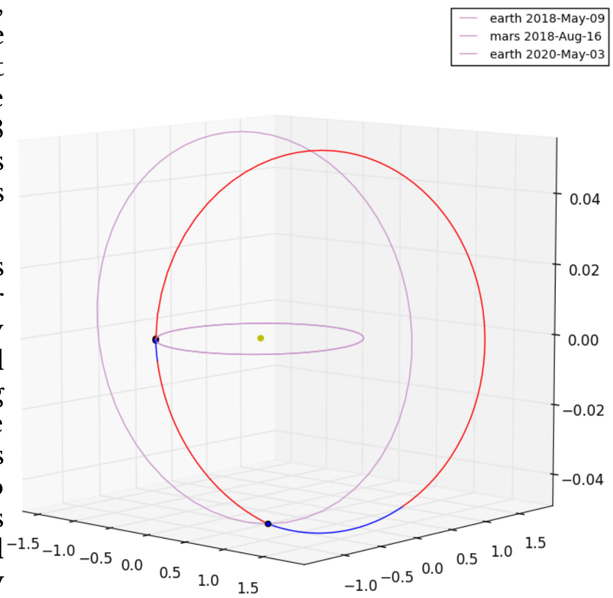


Figure 7: Visualization of a 730-day class trajectory. Note the trajectory part outside the orbit of Mars

The trajectory has two main advantages, however. Firstly, a lower required C3 launch energy – less than 27.5 km^2s^{-2}, which allows for a bigger payload. The second advantage is a much lower reentry velocity – about 12300 m/s – bringing reentry conditions back into known territory.

Leg 1 98.8874 d	Earth depart:	2018-May-09 18:15:29	
		C3	27.34 km²s⁻²
		Dec	-27.408 deg
		Rt Asc	323.759 deg
Leg 2 625.6642 d	Mars flyby:	2018-Aug-16 15:33:17	
		Flyby dist	97998.290 km
	Earth arrival:	2020-May-03 07:29:41	
		Vinf	5269.066 m/s
		Entry Vel	12313.800 m/s
ext-LUS	4x RL-10	Payload	36500 kg
		Parking orbit	278.9 x 14906.8 km
	2x MB-60	Payload	39300 kg
		Parking orbit	269.2 x 14963.4 km

Table 7: Parameters of a two-year free-return trajectory.

3.3.6 2021 Venus-Mars Flyby Trajectory

A double flyby trajectory for 2021 was investigated as a backup, in case of schedule slips, providing a nearly four year time margin with a late-November 2021 launch window. Example trajectory parameters are given in table 8, while figure 8 show its visualization and figure 9 presents the plot of a spacecraft-planet distances. The minimal Mars flyby distance was again set to 100 km. This family of trajectories is characterized by high departure asymptote declinations, approaching 60° (over two times higher than the KSC latitude), which significantly decrease in-LEO payload. The SLS+LUS payload delivery capability to such inclined LEO orbits is unknown, thus the methodology used in earlier sections could not be directly reapplied here.

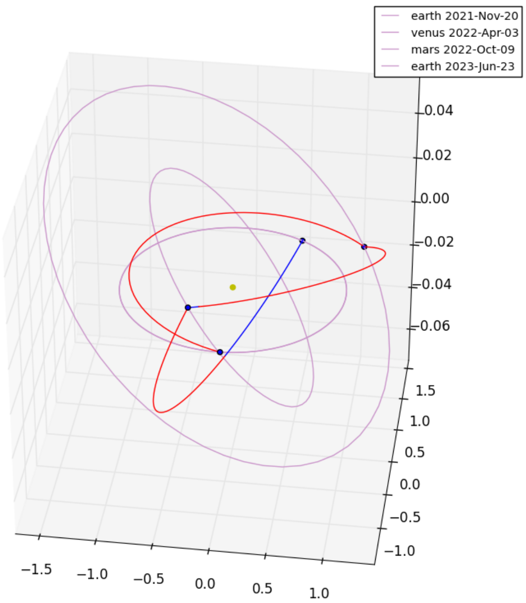


Figure 8: Visualization of a Venus-Mars flyby trajectory

To allow for payload mass assessment, an Atlas V in its most powerful single-core configuration was adopted as a benchmark vehicle. Its Mission Planner's Guide [19] in revision 10a contains the payload-altitude charts for orbits with inclinations equal to 28.6°, 51.6° and 63.4° for CCAFS launch site but also 63.4° and 90° for VAFB launch site. A huge penalty can be seen in 63.4° CCAFS figure, caused by a dogleg maneuver imposed by range constrains that limit the orbital inclination to 57° for a nominal, payload-optimized launch vehicle ascent. Thus, 63.4° CCAFS figure was from start dismissed as irrelevant, as with cautious trajectory picking the required ext-LUS parking LEO orbit inclination permits an optimized ascent. On the other hand, the 90° figure for VAFB was totally out of the range of our interest.

From the remaining curves the payload values were read at 200km altitude, to which 2138 kg [19] of inert Centaur stage mass was added to form an Atlas V 552 IMLEO estimates. These values were normalized and plotted, forming figure 10 on page 15. To estimate the values between the datapoints a 2nd order polynomial was used (as the curves for higher-

Leg 1 133.9883 d	Earth depart:		2021-Nov-20 22:28:49
		C3	19.79 km^2s^{-2}
		Dec	-56.917 deg
		Rt Asc	297.759 deg
Leg 2 188.4018 d	Venus flyby:		2022-Apr-03 22:11:56
		Flyby dist	10798.904 km
Leg 3 257.3046 d	Mars flyby:		2022-Oct-09 07:50:30
		Flyby dist	100.304 km
	Earth arrival:		2023-Jun-23 15:09:05
		Vinf	6311.037 m/s
		Entry Vel	12794.367 m/s

Table 8: Parameters of a double-flyby trajectory

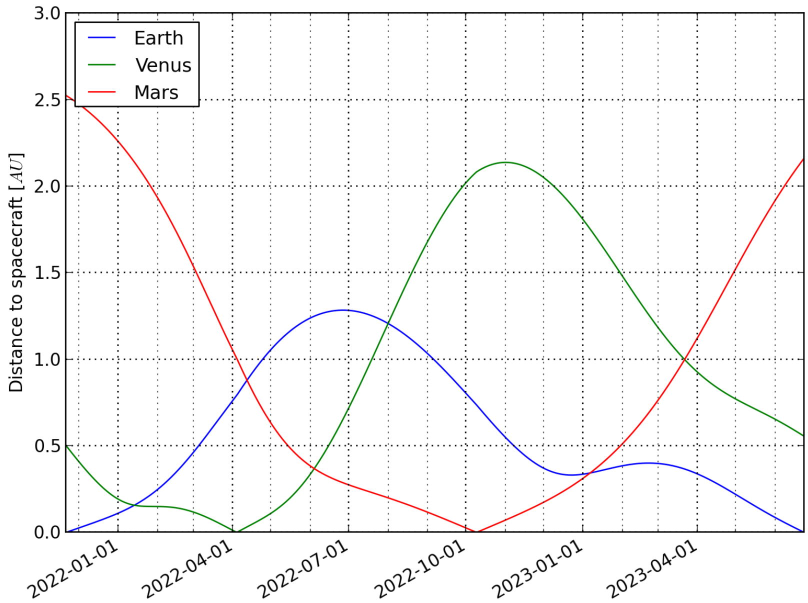


Figure 9: Spacecraft-planet distances (in AU)

orders were virtually the same). Normalization allowed for easy remark of IMLEO mass hit magnitude. The 6.24° difference in launch site latitude between CCAFS and VAFB can be safely not taken in the account, as the relative error it contributes to the analysis is negligible.

Therefore, mentioned figure 10 effectively contains the approximate IMLEO scaling factors for a range-clear ascent. For an orbit with inclination equal to 56.9° the scaling factor is equal to 0.90. To get the ext-LUS TMI-payload estimates, a virtual fuel "offload" that decreased the total mass by 10% was applied to the loitering stage before running the burn simulation. The rough results are 39 metric tons of payload mass for MB-60 version and 36 metric tons for RL-10 version, at C3 equal to 19.79 km^2s^2, values comparable to a two-year, C3=27.34 km^2s^2 trajectory.

In addition, since the Stardust Sample Return Capsule survived Earth reentry at 12.8 km/s [20], the entry velocity of the Earth Reentry Pod as calculated for this variant should also be within the currently known manageable range.

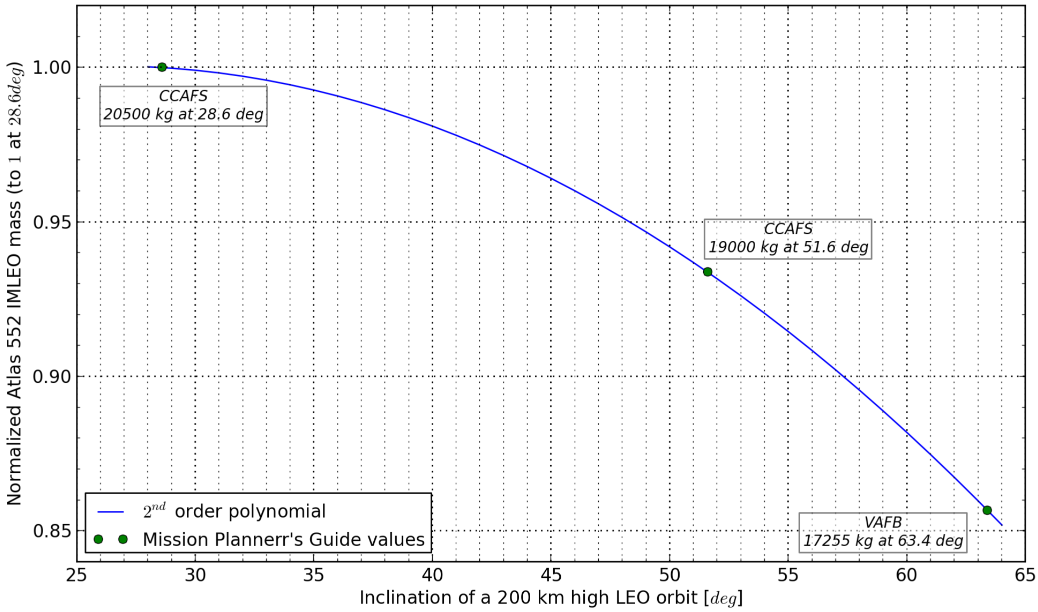


Figure 10: Approximate Atlas V 552 IMLEO mass hit vs. orbit inclination

4. Summary

The modification of the upper stage, described at the beginning of this analysis, is pretty simple in principle – stretching the propellant tanks to accommodate more propellant. Due to our limited tools and access to data, a non-optimized scaling model with maximum reasonable simplicity was used to scale the stage up to the fixed data point which, as mentioned for example in [4], may not be optimal itself – beyond a certain point, more fuel does not result in more payload.

A fast, free-return flyby trajectory was chosen, allowing us to reuse Inspiration Mars findings in the first part of the ext-LUS payload-trajectory study, giving apples-to-apples C3/reentry speed comparisons in a second part, as well as useful payload mass comparisons. Using proposed architecture with an ext-LUS instead of a classical, LUS/DUUS scenario results in a considerable increase in payload mass for the same C3, up to 45% for the conservative, RL-10 based stage. Independent trajectory simulations further lowered the requirements and, together with MB-60 engines, enable a payload mass increase of about 60%, up to a whopping 65% (when using both the ext-LUS maximum achievable payload and a SLS+DUUS figure from [6]). Thus, a RL-10 based stage with an IM reference C3 was chosen as our “safe-bet” option (giving a final payload of 28 500 kg), and a MB-60 based stage with the independently found minimum-C3 trajectory was chosen as “best SIM-achieved” (with a final payload of 31 400 kg). Both variants were used equally in the payload architecture design process, as well as plugged directly into payload mass margin calculations.

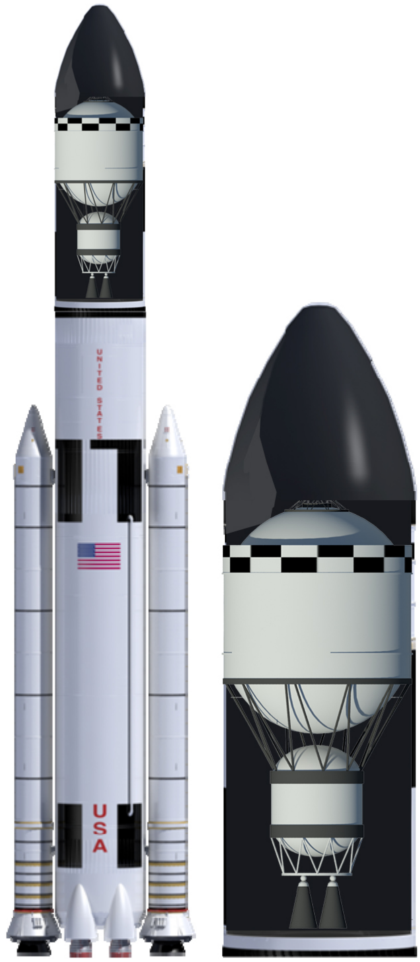
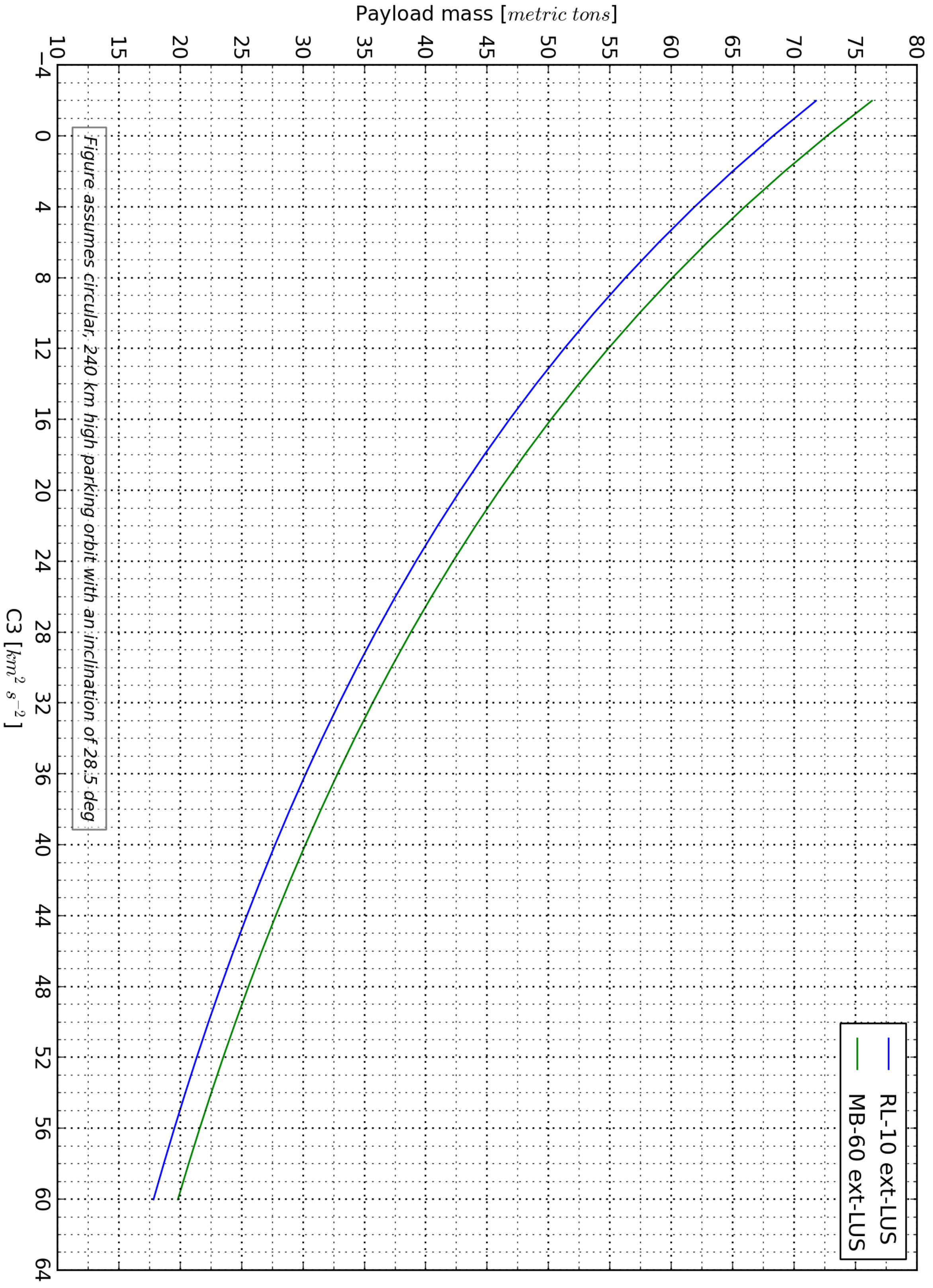


Figure 11: ext-LUS on top of the SLS rocket, with a short aerodynamic shroud used

Presented payload figures do not take into account the performance increase caused by shortened fairing. Other possibilities for increasing payload are also available, besides using the aerodynamic “cap” (see figure 11) and optimizing the scaling model. For example, incorporating the *Integrated Vehicle Fluids* [21] system into the ext-LUS design could result in a performance increase of about 10% - 20% of a stage's dry mass.

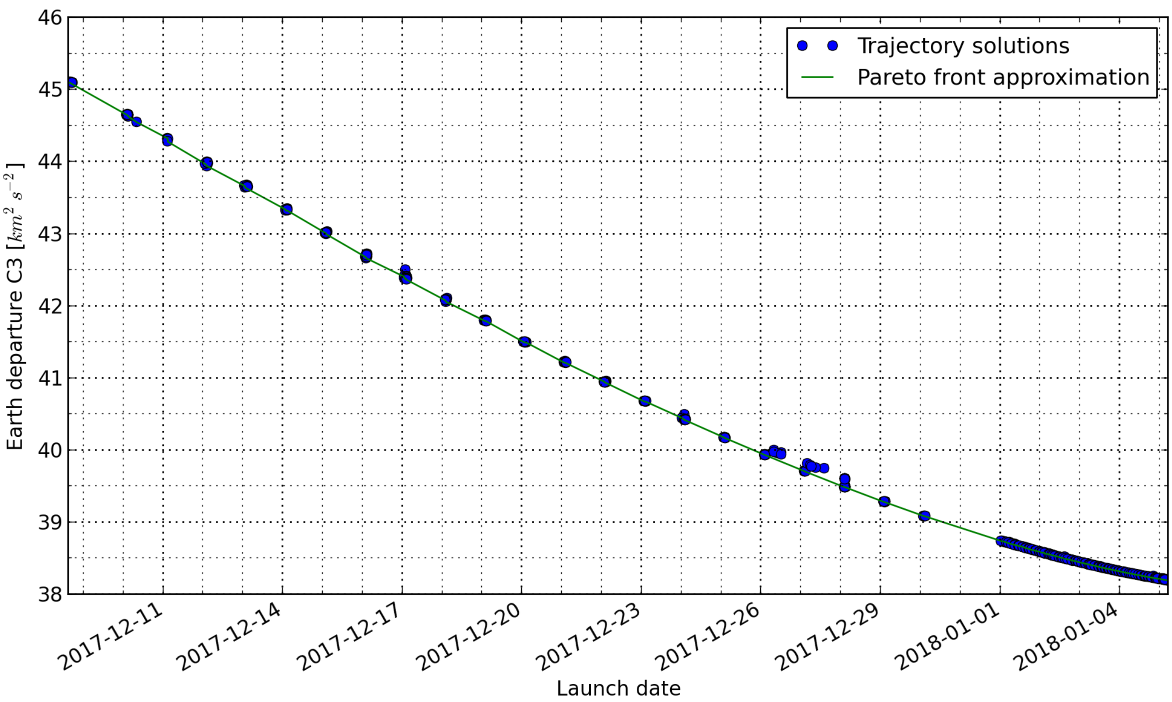
It can easily be concluded that the payload, although much bigger than in a scenario proposed by Inspiration Mars, could be launched in one go by a Falcon Heavy that has a LEO capacity of 53 metric tons [22], and then rendezvous and dock with the loitering stage. With over half a century of experience in conducting rendezvous and docking operations, this should not be considered a disadvantage of the presented concept. If such a proven operation were a disadvantage, how many centuries would we have to wait for a manned Mars flyby mission?

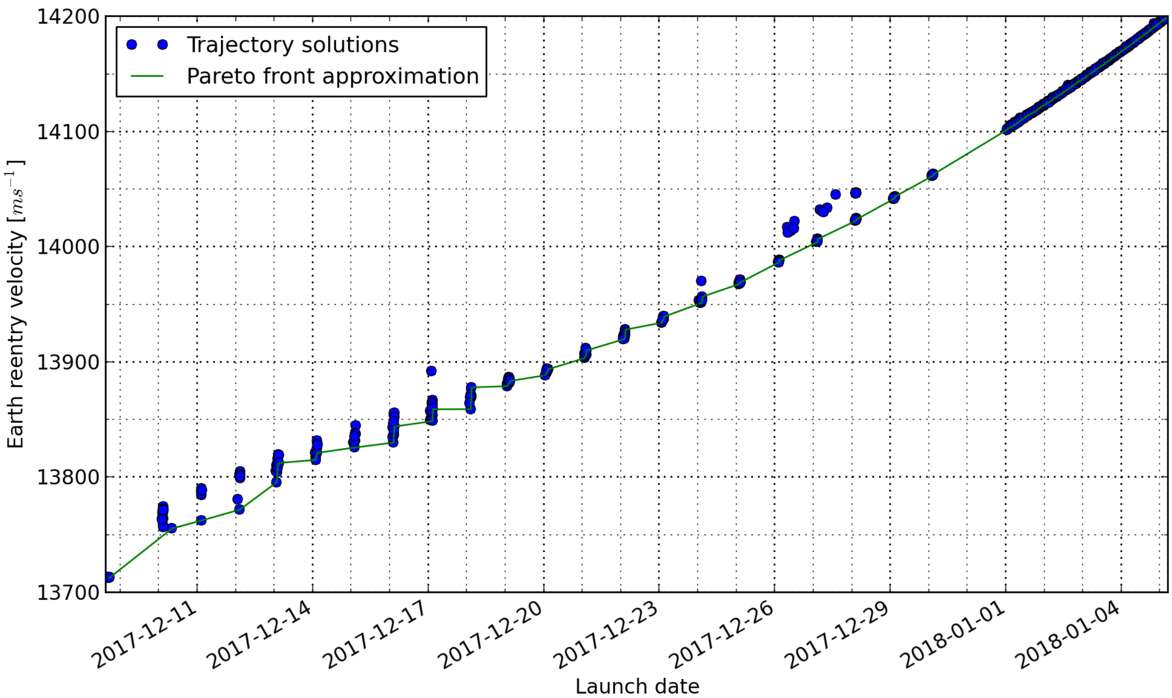
Appendix A: ext-LUS C3-Payload Chart

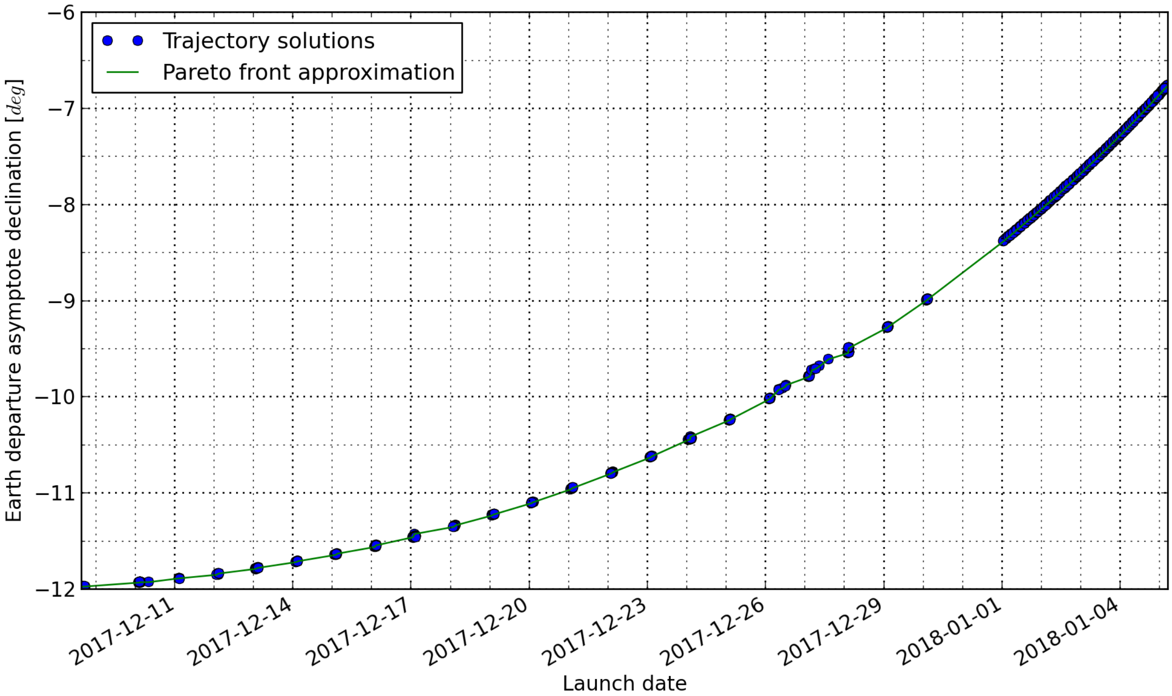


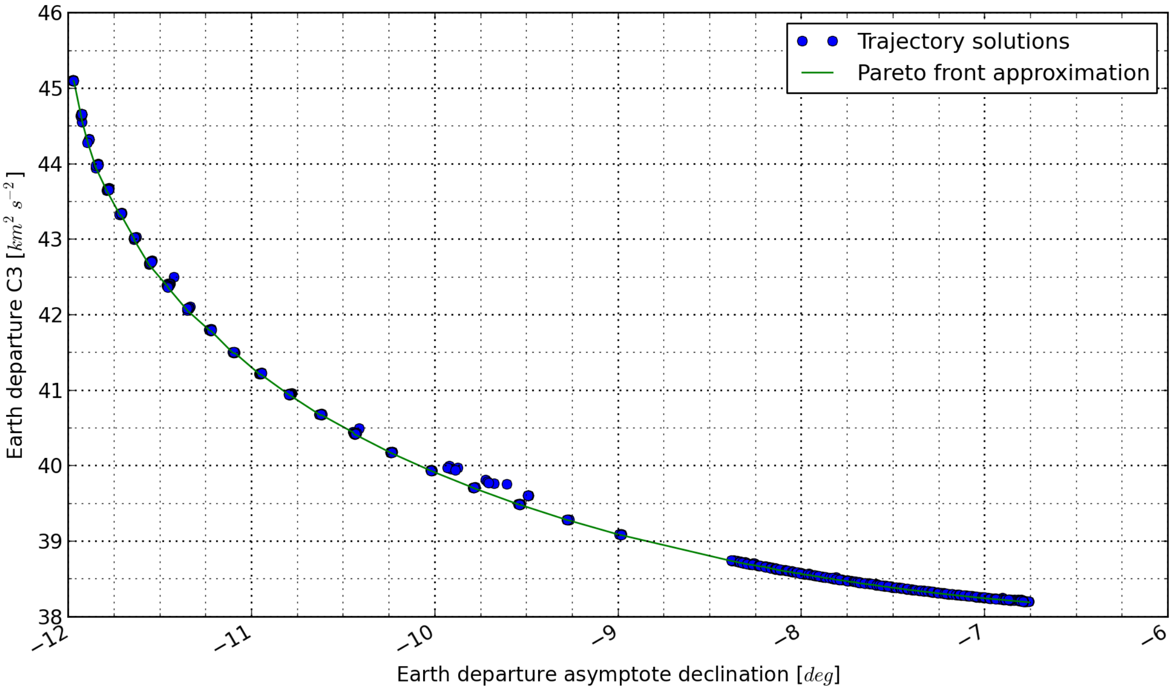
Appendix B: Trajectory Pareto Fronts

Appendix B.I: The 500-day class trajectory

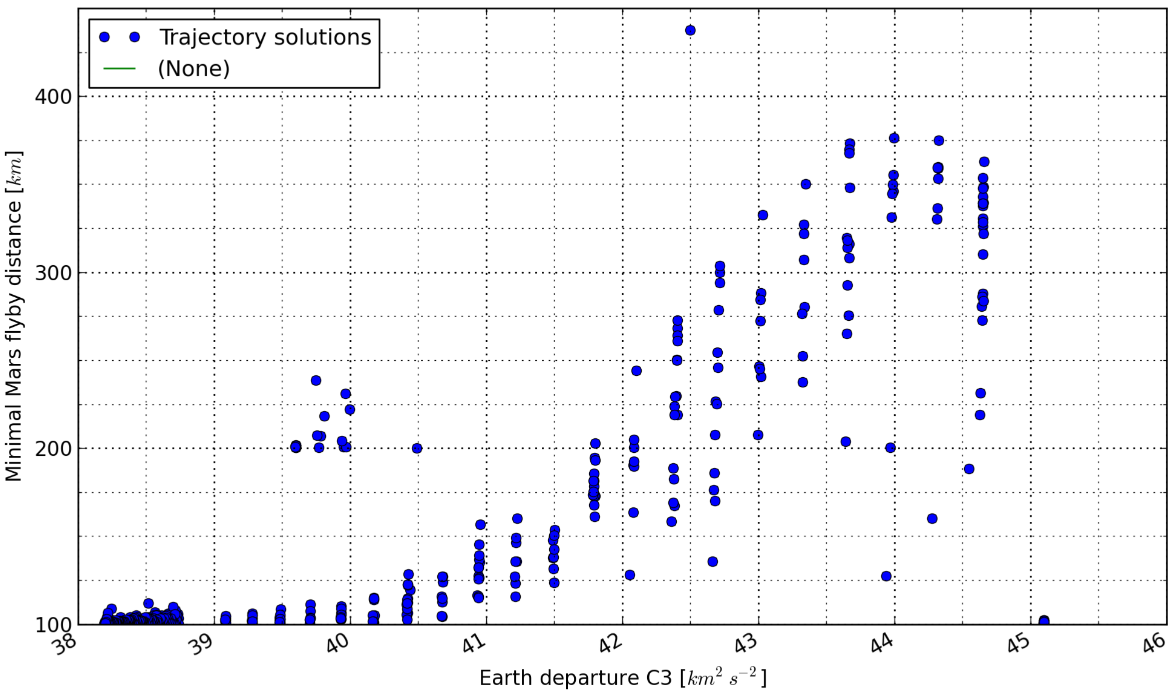


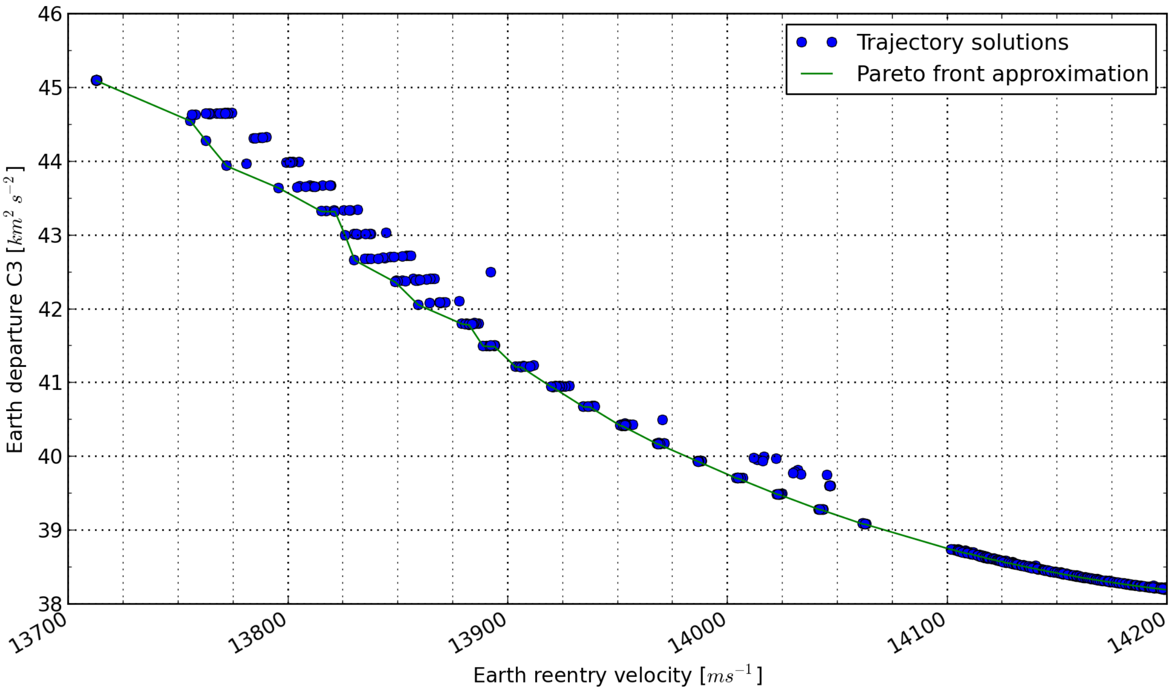



Trajectory solutions
Pareto front approximation
Earth departure asymptote declination [deg]
Launch date
−6
−7
−8
−9
−10
−11
−12
2017-12-11
2017-12-14
2017-12-17
2017-12-20
2017-12-23
2017-12-26
2017-12-29
2018-01-01
2018-01-04

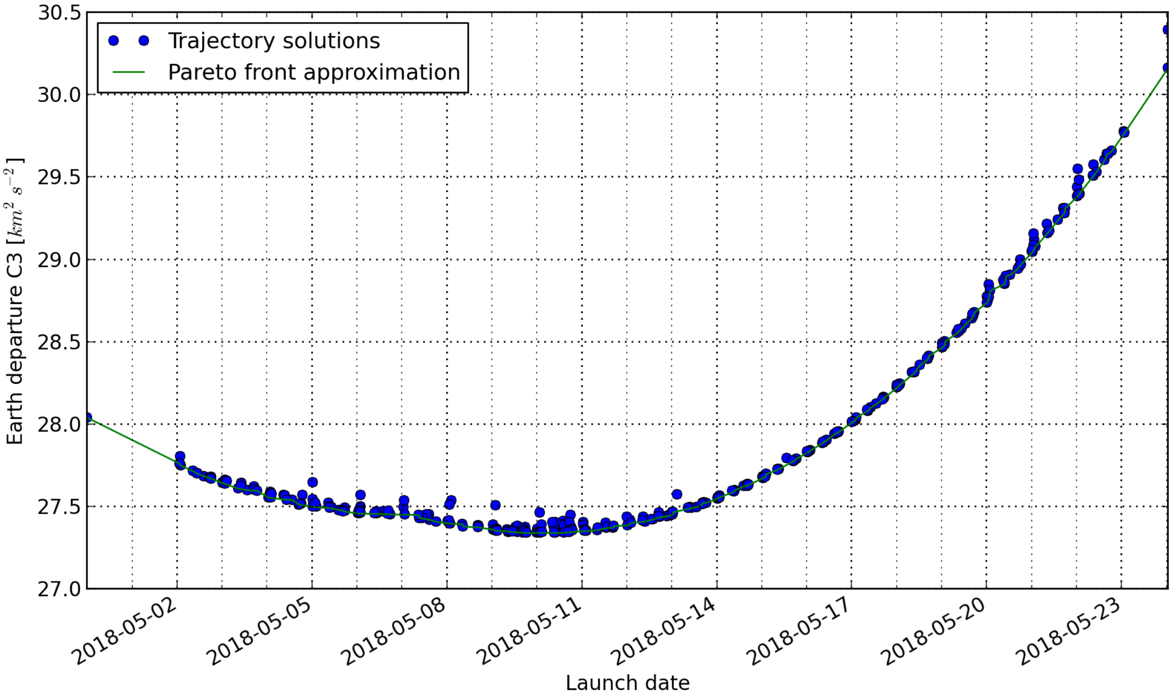

Trajectory solutions
Pareto front approximation
Earth departure C3 [$km^2\ s^{-2}$]
Earth departure asymptote declination [deg]
46
45
44
43
42
41
40
39
38
−12
−11
−10
−9
−8
−7
−6

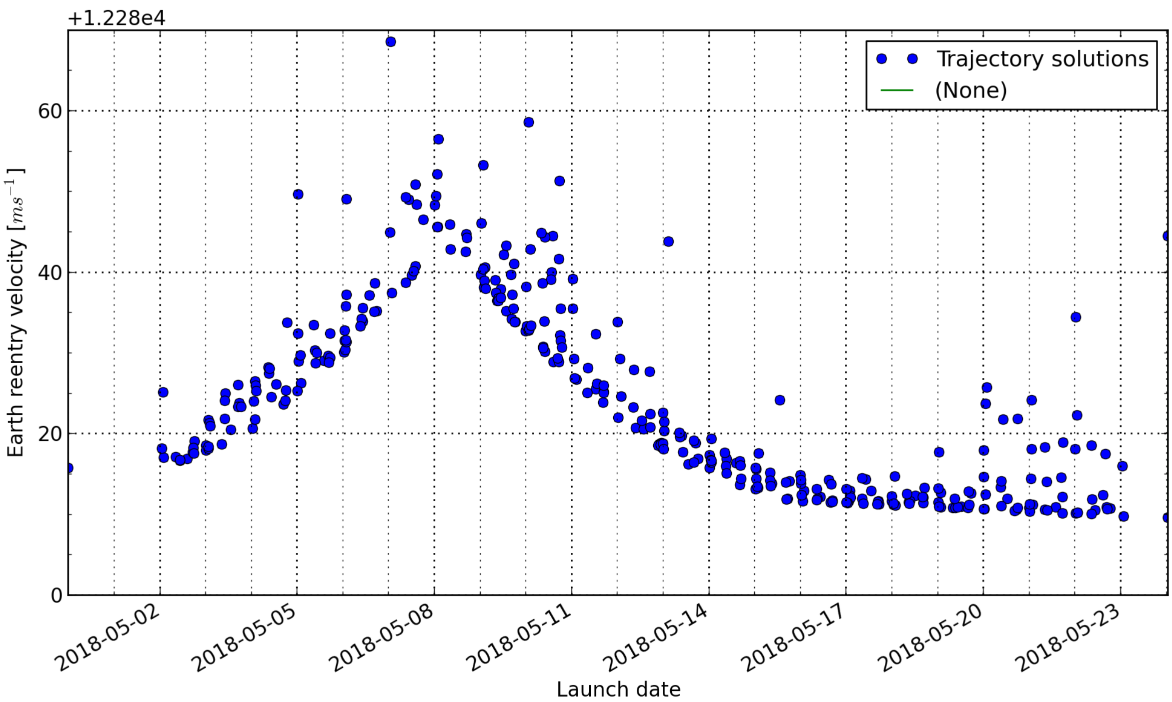
Appendix B.I: <u>The 500-day class trajectory</u>



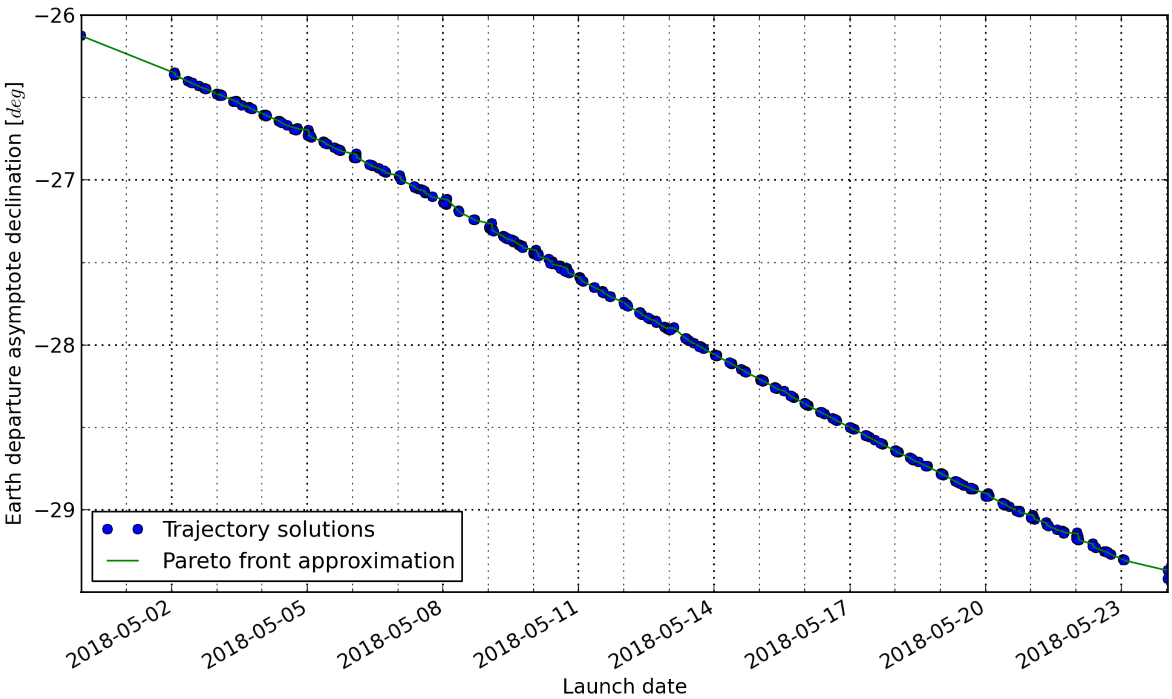


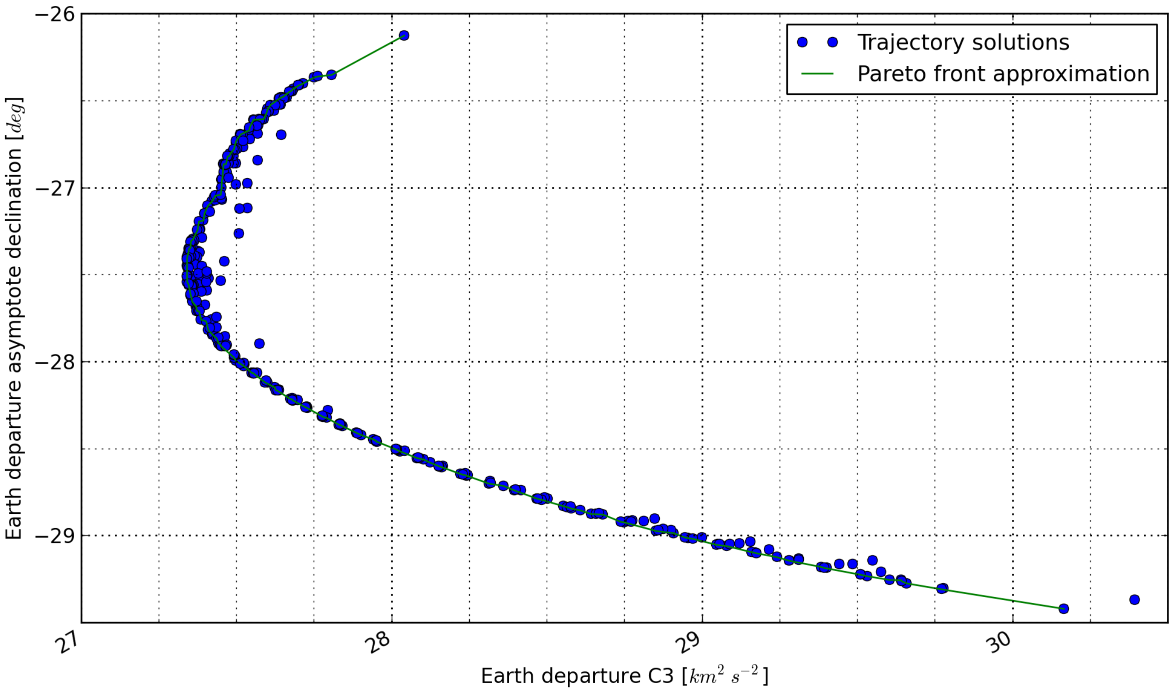
APPENDIX B.II: THE 700-DAY CLASS TRAJECTORY

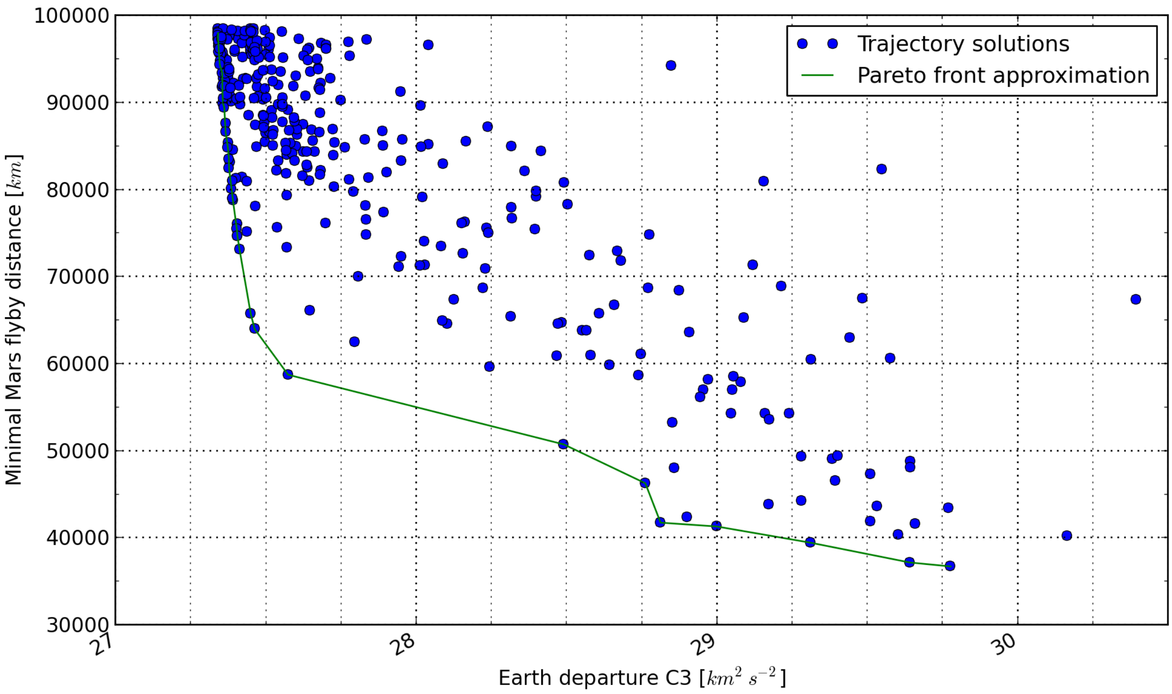




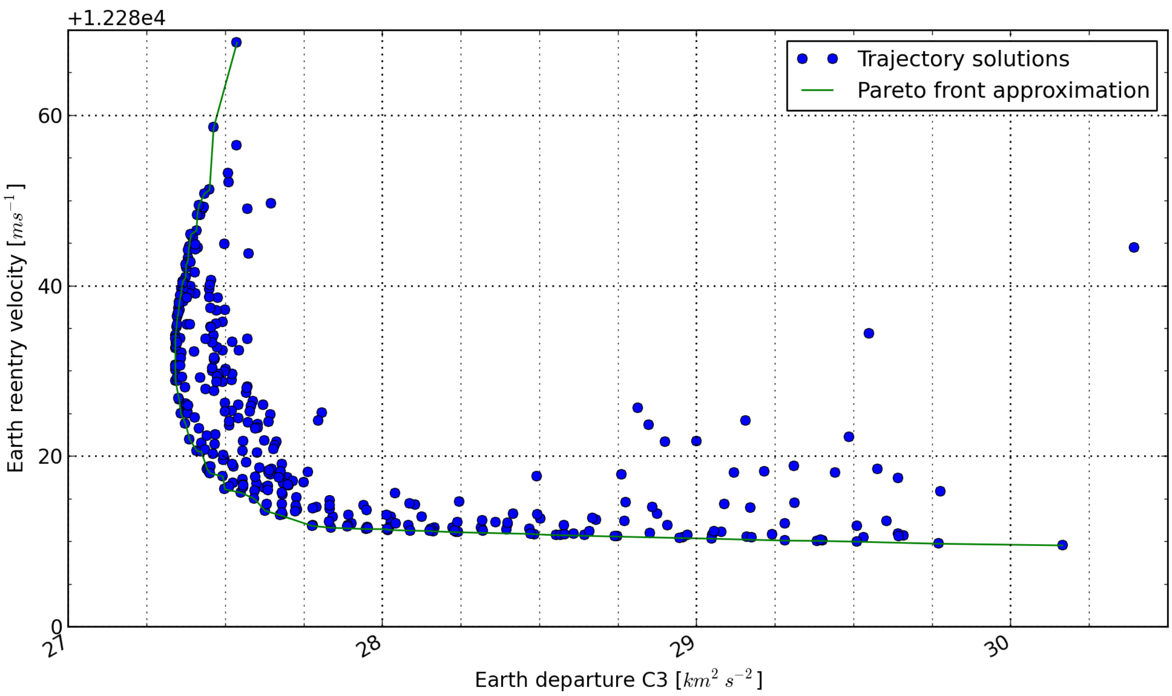
Appendix B.II: The 700-day class trajectory



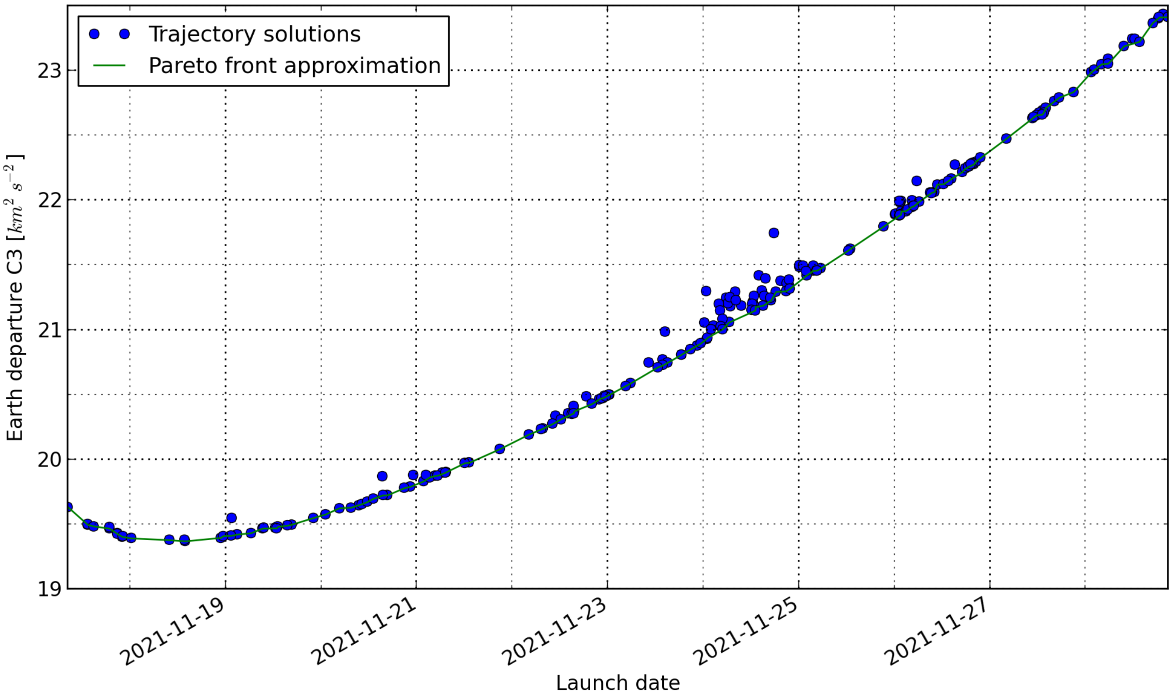




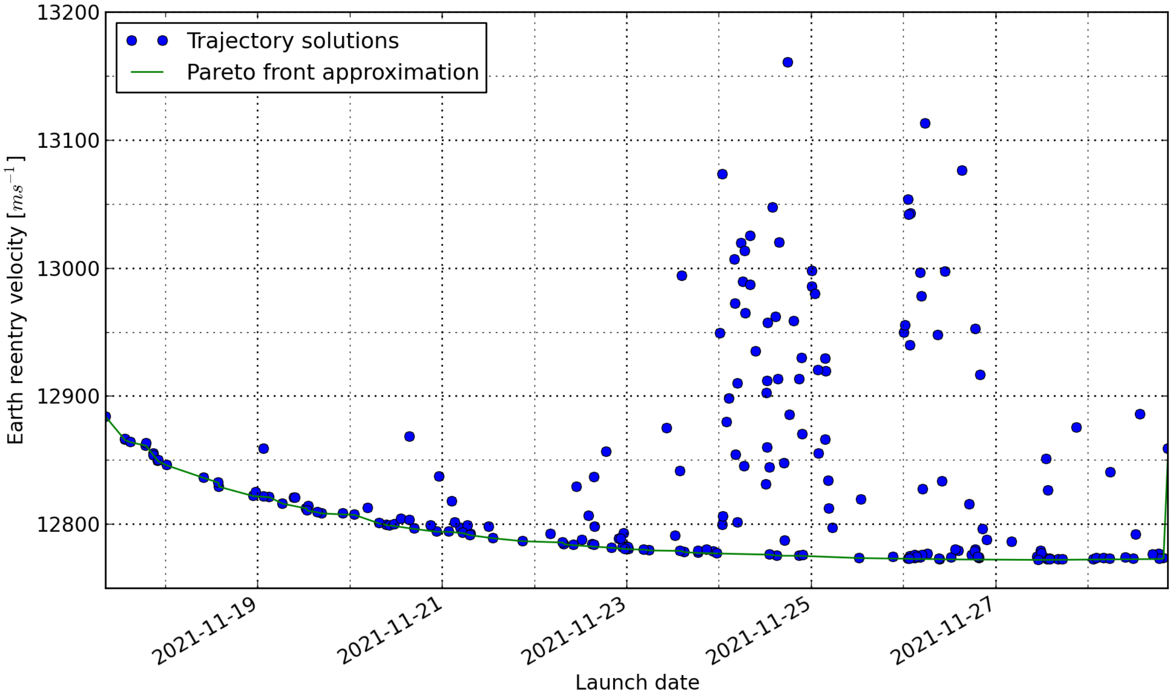
Trajectory solutions
Pareto front approximation
Minimal Mars flyby distance [km]
100000
90000
80000
70000
60000
50000
40000
30000
27
28
29
30
Earth departure C3 [$km^2\ s^{-2}$]



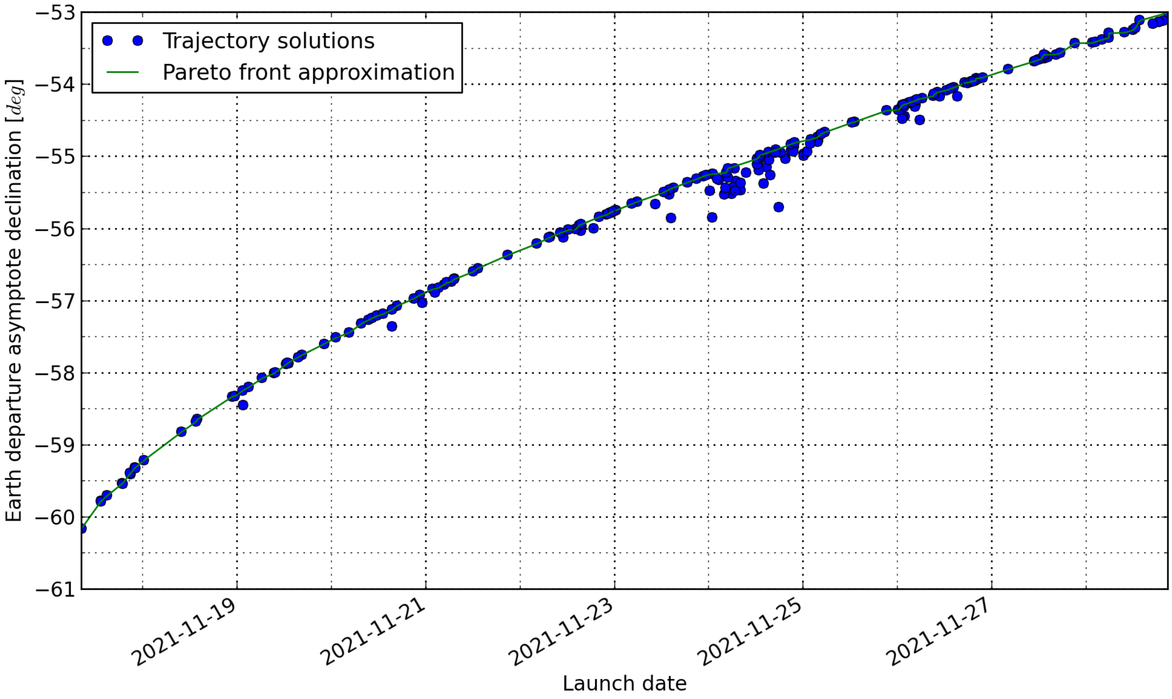
+1.228e4
Trajectory solutions
Pareto front approximation
Earth reentry velocity [ms^{-1}]
60
40
20
0
27
28
29
30
Earth departure C3 [$km^2\ s^{-2}$]

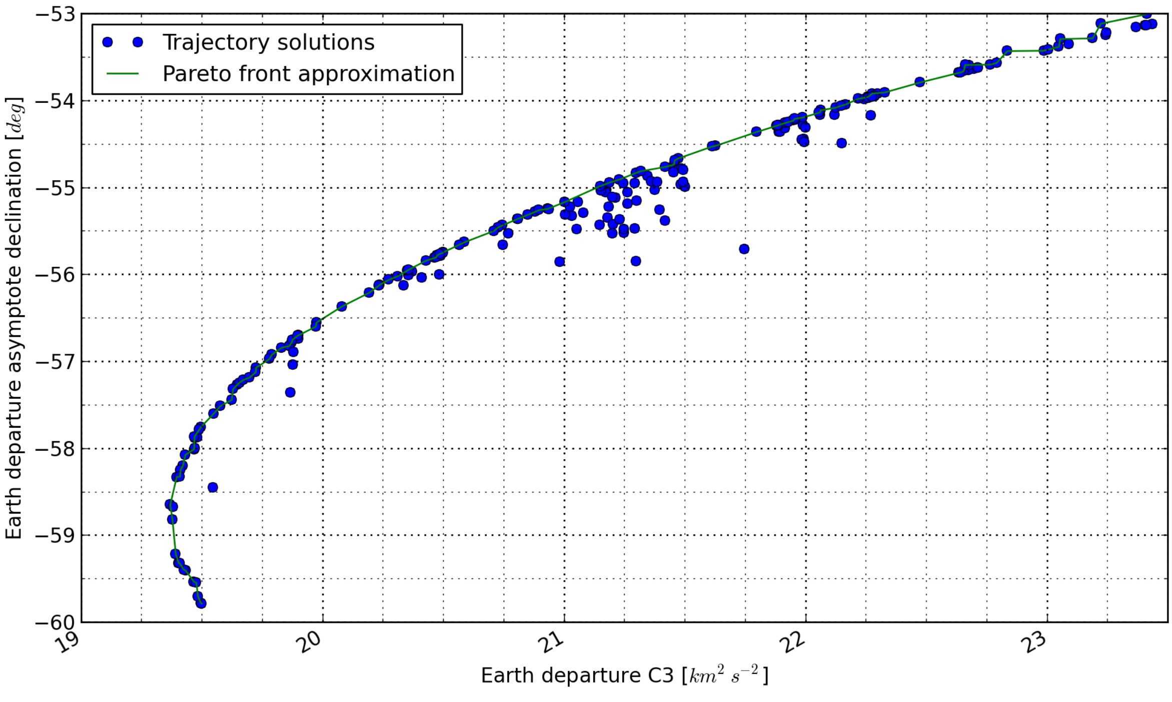


Trajectory solutions
Pareto front approximation
Earth departure C3 [$km^2\ s^{-2}$]
19
20
21
22
23
2021-11-19
2021-11-21
2021-11-23
2021-11-25
2021-11-27
Launch date

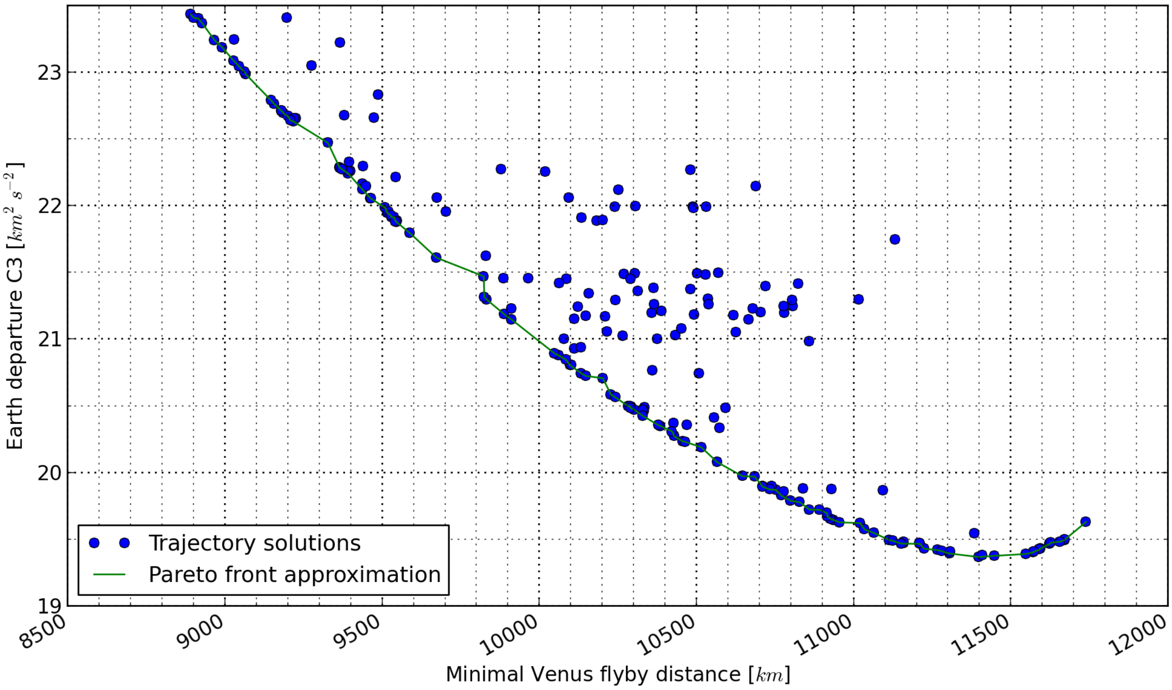


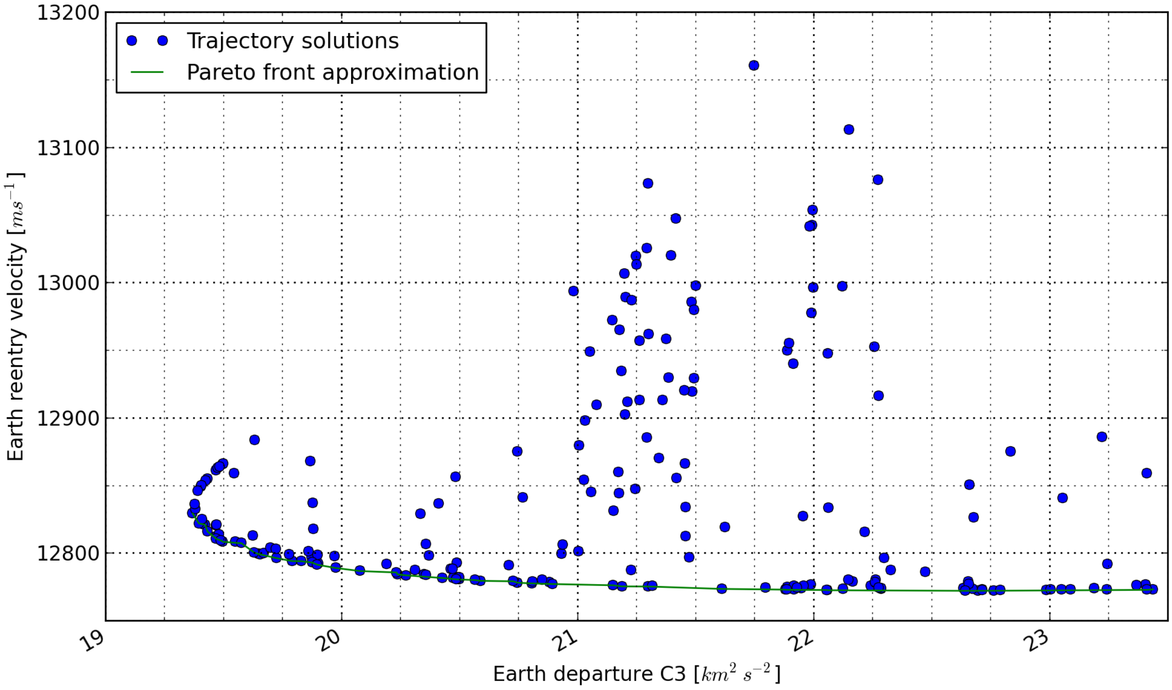
Trajectory solutions
Pareto front approximation
Earth reentry velocity [ms^{-1}]
12800
12900
13000
13100
13200
2021-11-19
2021-11-21
2021-11-23
2021-11-25
2021-11-27
Launch date


Trajectory solutions
Pareto front approximation
Earth departure asymptote declination [deg]
−53
−54
−55
−56
−57
−58
−59
−60
−61
2021-11-19
2021-11-21
2021-11-23
2021-11-25
2021-11-27
Launch date


Trajectory solutions
Pareto front approximation
Earth departure asymptote declination [deg]
−53
−54
−55
−56
−57
−58
−59
−60
19
20
21
22
23
Earth departure C3 [$km^2\ s^{-2}$]

Appendix B.III: The 2021 Venus-Mars flyby trajectory





Appendix C: Historical TCM ΔV Magnitudes

MRO [23]	
TCM1	7.80 m/s
TCM2	0.754 m/s
Total	8.554 m/s

MER Opportunity [25]	
TCM1	16.172 m/s
TCM2	0.534 m/s
TCM4	0.107 m/s
Total	16.813 m/s

MER Spirit [25]	
TCM1	16.460 m/s
TCM2	6.008 m/s
TCM3	0.577 m/s
TCM4	0.025 m/s
Total	23.07 m/s

Mars Pathfinder [24]	
TCM1	30.077 m/s
TCM2	1.5959 m/s
TCM3	0.1120 m/s
TCM4	0.0186 m/s
Total	31.8035 m/s

MSL [26]	
TCM1	5.6350 m/s
TCM2	0.7119 m/s
TCM3	0.0420 m/s
TCM4	0.0104 m/s
Total	6.3993 m/s

Table 9: TCM magnitudes and total TCM expenditures for chosen Mars-Targeted missions. In all cases TCM1 is the most costly as it removes injection errors and planetary protection bias.

Phoenix [27]	
Mean values (from Monte-Carlo simulations)	
TCM1	14.6 m/s
TCM2	5.7 m/s
TCM3	1.4 m/s
TCM4	0.25 m/s
TCM5	0.16 m/s
TCM6	0.41 m/s
Total	22.52 m/s
Total TCM ΔV at 99% probability	
window opening:	52.4 m/s
window day 16:	48.5 m/s
window close:	46.5 m/s
TCM budget limit:	56.0 m/s

Table 10: ΔV budgets have to be considerably larger than the Table 9 values to ensure that probability of a mission success exceeds an acceptable threshold. Provided above are results of a preliminary preflight simulations for a Phoenix spacecraft. Note that TCM budget limit is almost three times larger than total simulated mean TCM expenditure.

Appendix D: License, Changes and GitHub Repository

GitHub repository:

The repository is accessible at https://github.com/arkhebuz/spaceismore and contains most up-to-date version of this document. Code developed during the research will be also published there (probably in upcoming weeks/months).

List of changes:

- Revision 1.1, September 15, 2014: added keywords and copyright information, revised "consultation" section, other minor changes.
- Revision 1.0a, September 09, 2014: added comment to chart in Appendix A, rewritten Appendix D, changed position of figure 11.
- Revision 1.0, August 09, 2014: original document commited to GitHub repository.

License:

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