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Design optimisation and analysis of very high power transportation system to Mars

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This paper presents a preliminary design for a crewed transportation system to Mars, looking at both the vehicle and mission design trade-offs. The mission requirements are for a return mission to Mars, with a variable stay time on/around the planet. The vehicle is designed for a minimum crew of 3 plus 50 t cargo, and assumes in-orbit assembly and re-fueling options. A nominal vehicle design was developed using a bimodel nuclear propulsion system. A multi-objective optimisation was run examining the trade-offs between flight times, and total duration, for different Mars stay times, planetary conjunction angles, dry masses and engine sizing, and the mission trajectories.
Keywords: Nuclear propulsion, Mars mission, Design optimisation, Mission analysis, ,

1. Introduction

As early as the 1940s, nuclear derived technology was recognised as a fundamental enabler for future space travel. Landmark studies, such as by Shepherd and Cleaver [18] concerning nuclear thermal rocketry (NTR) and nuclear electric propulsion for interplanetary travel, helped prompt initial practical investigations into nuclear propulsion systems. Subsequent investigations included the Soviet nuclear thermal rocket engine development program for the RD-041x, Atomic Energy Commission Project Rover [10] and the NASA Nuclear Engine for Rocket Vehicle Application (NERVA) program [9], the latter incorporating a promising series of successful terrestrial test firings, prompting inclusion into several proposals for crewed missions to Mars [2, 8, 22]. Since then, advances in materials, magnetic fields and plasma dynamics have supported the development of numerous appealing concepts [1]. Currently, many of the major space agencies and governments have programmes examining nuclear propulsion for space exploration, in particular targeting Mars, such as NASA and DARPA, ESA, UKSA, Roscomos, China, and India.

There a few main types of nuclear propulsion systems proposed for space travel: nuclear thermal propulsion using fission reactions, and nuclear electric propulsion (NEP), which uses a nuclear reactor to power an electric propulsion system (and generator). A particular advantage of NEP is the independence from solar power, which decreases proportionally to r^{-2}, thus making it attractive for deep space applications though at a cost in efficiency due to the conversion between thermal and electric power. Multi-modal options exist, which use NTR with some electrical power generation capabilities to power the spacecraft. In 2001, ESA started development on a bimodal design, Nuclear Thermal Electric Rocket (NTER), which could increase the specific impulse over solid fuel NTR by adding a thermal induction heater downstream of the reactor core [5, 6].

Given the directly proportional relationship between total dose and radiation flux/time of exposure, logical mitigation strategies are to either reduce the radiation flux via some form of protective shielding, or to decrease the overall transfer time. Research into active/passive shielding is ongoing, but it is currently accepted that with the significant volume of shielding required for a Hohmann-type transfer to Mars, the costs to deliver this additional payload to LEO would be largely prohibitive to the overall mission. As such, it is prudent to examine the propulsion options available to reduce transfer time. Of note here are additional operational constraints determined by transfer time, such as the total mass/volume of consumables and habitation structure, and the physical/psychological wellbeing of the crew, all of which add further incentive to reduce time in space. While traditional LO2/LH2 chemical propulsion systems may still remain practical for Earth orbit and Lunar missions, their characteristic high thrust/low specific impulse results in largely infeasible mass fractions when applied to interplanetary trajectories. Given the resultant need for higher power density and specific impulse I_{sp}, a viable alternative is the use of nuclear reactions.

A study was undertaken to examine the trade-offs, given near-term future technologies, for the preliminary design of a high powered transportation system to Mars. The study examined the mission performance and a vehicle configuration study with numerical models for structural mass, radiation, propulsion, habitat and consumables, and a structural analysis of the separation truss between the spacecraft, including crew habitat module, and the nuclear engine. The system analysis focuses on a nominal crewed mission to Mars, as it is the more limiting of the options of crewed versus cargo-only. The requirements and assumptions for the system are: Earth and Mars one-way journey travel time of less than 90 days, spacecraft carrying a minimum of 50 tons of cargo with a minimum of 3 crew, and in-orbit manufacturing and refuelling facilities are assumed as operational around both Earth and Mars. The launch and landing segments of the mission are not considered.

A multi-objective optimisation solver is used to examine trade-offs in the mission and trajectory designs, and the driving vehicle design parameters including engine sizing, and gross and dry vehicle masses. For a semi-cycler based mission architecture [11], single and return legs were analysed independently and together, using continuous and on-off thrust models. The stay time on Mars was a variable parameter, set to values up to 100 days, to understand the impact on optimal set of timings of a such Earth-Mars-Earth trip.

2. System models

A study of different spacecraft configurations and designs was undertaken examining studies on proposed configurations. Key drivers were identified, such as the type and design of the nuclear propulsion system, radiation shielding and mitigation measures necessary for a crewed mission, radiator sizing, and mass modelling of the crew habitat module. Numerical models for the vehicle were developed as part of the mission design trade-off, in particular looking at the mass scaling laws.

2.1 Vehicle models

Crewed missions are driven by the requirements to minimised crew radiation exposure, thus minimising flight times [7]. As the reduced flights time translate, at present, into nuclear propulsion, this also strongly drives the design of the spacecraft. In particular the need for shielding, both for the crew and spacecraft systems, and maximising the distance between the crew and the reactor. The volume of the habitat is directly driven by the number of crew and duration of flight, and on the mission architecture (e.g., if the transportation habitat doubles as part of the surface habitation module on Mars).

The nominal spacecraft configuration is shown in Figure 1, with a truss connecting the nuclear thermal propulsion (NTP) system (engine, reactor and shield), with the rest of the spacecraft. With respect to the shielding, the habitat should sit within the shadow.

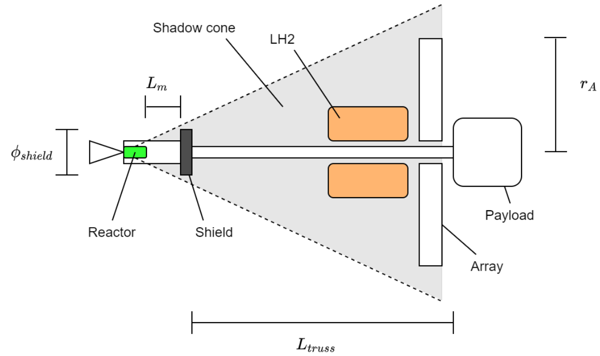

Fig. 1: Nominal spacecraft configuration

2.1.1 Propulsion model

The ESA-developed bi-modal Nuclear Thermal Electric Rocket (NTER) engine is a promising hybrid thermal-electric solution for future missions to Mars. The NTER engine concept incorporates high thrust nuclear thermal propulsion and, by virtue of bleeding off a small amount of hydrogen propellant through a generator within a closed thermodynamic cycle, can both generate electrical power and improve the overall engine I_{sp} [6]. An engine sizing model was defined for the NTER performance specifications (see Table 1), and modified via a linear scaling parameter $n = [0.1,\ 10]$ for the thrust and subsystem mass.

Propulsion system parameter	Baseline ($n = 1$)
Power	507 MW
Thrust	111.2 kN
Specific impulse I_{sp}	921 s
Propulsion system mass	23.50 t
Propellant mass flow rate $\dot{m}_{prop}$	12 kg/s

Table 1: Performance specifications of the bimodal NTER concept [6]

2.1.2 Habitat

Salotti et al. [17] present a method to estimate habitat mass breakdown based on the NASA TransHab specification. This study applies subsystem specific scaling coefficients to estimate the reduced component mass/volume for 3 people relative to the established NASA 6-person habitat concept, as detailed in Table 2.

Consumables were divided in the following groups: 1) food 2) fecal canisters, urine pre-filters, and trash bags 3) personal hygiene kit, hygiene consumables, wipes, towels, and health care consumables 4) clothing and 5) per-

	Crew = 6		
Component	Mass (kg)	Volume (m^3)	% reduction
Power system	5840	–	22.5
Avionics	290	0.1	0
ECLSS	3950	19.1	15
TMS	1260	5.3	15
Crew accommodations	4210	29.7	22.5
EVA systems	870	2.9	30
Structure	2020	–	22.5
(inc. 30% margin)	4920		
Spares	4180	1.4	15
Crew	6×80	–	–
Radiation shelter	6×739.36		
Total (no consumables)	32536.16		

Table 2: Mass breakdown, adapted from Drake [4], Salotti et al. [17] with radiation shelter estimates from Simon et al. [19]

sonal stowage and operational supplies. Consumable requirements are largely taken from Lopez et al. [12]. Consumption rates for all logistics goods are defined using historical usage/resupply data from the ISS in combination with data from the Advanced Life Support Baseline Values and Assumptions Document, Human Integration Design Handbook, and Orion Commercial Crew Development design values. Mass scaling models were developed for each category, depending on the number of crew members, flight durations, and stay time on/around Mars.

2.1.3 Spacecraft structure

To account for the mass of components such as propellant tanks, external subsystems (e.g., communications arrays) and external structure (e.g., truss), the overall structural mass is assumed 10% of the initial spacecraft mass in LEO.

2.2 Flight dynamics

The vehicle dynamics are defined via the following expressions for the rates of change of state vector $\mathbf{x} = [r, \theta, v_r, v_\theta, m_{prop}, m_{cons}]$ in an inertia reference frame, where r is the radial distance from the centre of the current sphere of influence (SOI), θ is the angular position, v_r is the radial velocity, v_θ is the angular velocity, m_{prop} is the mass of propellant and m_{cons} is the mass of consumables.

$$\dot{r} = v_r \tag{1a}$$

$$\dot{\theta} = \frac{v_\theta}{r} \tag{1b}$$

$$\dot{v}_r = \frac{v_\theta^2}{r} - \frac{\mu}{r^2} + \frac{\tau T_{max}}{m} \sin \alpha_{eng} \tag{1c}$$

$$\dot{v}_\theta = -\frac{v_r v_\theta}{r} + \frac{\tau T_{max}}{m} \cos \alpha_{eng} \tag{1d}$$

$$\dot{m}_{prop} = -\tau \dot{m}_{prop} \tag{1e}$$

The scalar variable $\tau = [0,\ 1]$ is the throttle controlling the fraction of thrust applied, α_{eng} is the angle of the thrust vector and T_{max} is the maximum available engine thrust.

The SOI for both Earth and Mars is given by,

$$SOI = a_{\mathrm{E/M}} \left(\frac{m_{\mathrm{E/M}}}{m_{sun}} \right)^{2/5} \tag{2}$$

where a is the semi-major axis and m is the planetary mass. A further assumption is that of perfectly circular orbits of Earth and Mars. This is to disentangle the initial mission analysis from dependence on considering specific launch dates/windows, instead allowing a more general approach that may be applicable across many future scenarios. The reference frames are centered on the primary gravitational body, either Earth, the Sun, or Mars depending on the mission phase.

3. Approach

The full mission profile is divided into three distinct segments: the outbound (Earth to Mars) transfer, the stay time at Mars, and the inbound (Mars to Earth) transfer. Within the two transfers, three further phases were defined based on the SOI: low-altitude planetary departure, Earth-Mars (or Mars-Earth) interplanetary transfer, and planetary capture to low altitude orbit.

The relationship between these segments, with respect to spacecraft design parameters (including engine and habitat sizing, consumable provisions, structural mass and propellant requirements), is examined through a series of multi-objective problem formulations. The numerical system models were used within a multi-objective optimisation framework MODHOC, capable of solving non-linear constrained optimal control problems and a multi-disciplinary design optimisation problems.

Continuous control profiles for thrust were included to determine the optimal engine switching structure for each case. In this study, the throttle is defined as a continuous parameter evaluated at the control nodes. Using this information, switching control problem variations were formulated (i.e., on/off per phase, with optimisable phase flight times) to further refine solutions.

Problem objectives were the minimisation of total trajectory transfer time (i.e., not including Mars stay time), and gross vehicle mass (including any in-orbit refuelling whilst in LMO or LEO).

$$\min_{\mathbf{u}\in\mathbf{U},\mathbf{d}\in\mathbf{D}} \sum_{i=1}^{N} (t_{f,i} - t_{0,i}) \tag{3a}$$

$$\min_{\mathbf{u}\in\mathbf{U},\mathbf{d}\in\mathbf{D}} m(t_0) \tag{3b}$$

where N is the number of flight trajectory phases defined, t_0 and t_f are the initial and final times respectively, and

$$\mathbf{u} = \begin{bmatrix}\boldsymbol{\alpha}_i & \boldsymbol{\tau}_i & (\Delta t)_i\end{bmatrix} \tag{4}$$

is the trajectory control vector where $(\Delta t)_i = t_{f,i} - t_{0,i}$ is the time of flight for the i^{th} phase, $\boldsymbol{\alpha}$ is the set of thrust direction angles and $\boldsymbol{\tau}$ is the set of throttle values at every control node in the the i^{th} phase. The optimisation vector of static design parameters is given by,

$$\mathbf{d} = \begin{bmatrix}\phi_p(t_0) & P_{eng} & m_{str}\end{bmatrix} \tag{5}$$

where ϕ_p is the angle between Earth and Mars at the initial time, P_{eng} is the reactor power, m_{str} is the structural mass of the spacecraft. The gross mass $m_0 = m(t_0)$ is a sum of the calculated masses for the structure, engine system, propellant and tanks, crew, habitat and consumables.

A number of assumptions were used for the mission analysis:

- Earth and Mars are assumed to follow circular Sun-centered orbits, such that the optimal phasing between Earth and Mars and the overall mission architecture may be examined without reference to specific launch dates/transfer windows.
- Prior to each transfer, the spacecraft is assumed positioned in a circular low-altitude planetary parking orbit (LEO, or LMO).
- A sphere of influence (SOI) trajectory approach is adopted, where the spacecraft is assumed bound by only the dominant gravitational force within a particular mission phase. For example, Earth-centered from LEO to the SOI of Earth, Sun-centered for interplanetary space and Mars-centered between the SOI of Mars and LMO.
- Specific launch systems are not considered in detail, though the availability of in-orbit construction whilst in LEO is assumed.
- Similarly, the availability of in-orbit refuelling in LMO is assumed.
- No advanced mid-flight events, e.g., mass jettisoning, refuelling, aero-braking or gravity assist fly-bys are considered, though the optimiser may include deep space manoeuvres whilst in interplanetary space.

3.1 *Optimisation algorithm*

A multi-objective optimisation algorithm MODHOC (Multi-objective Direct Hybrid Optimal Control solver) [14, 16] was used for the multidisciplinary design optimisation. MODHOC is based on DFET, a transcription method for nonlinear multi-phase optimal control problems [20], with MACS, a population based memetic multi-objective optimisation algorithm [15] and mathematical programming solvers*. MODHOC was initially developed under an ESA NPI on Multi-Objective Hybrid Optimal Control Problems. The software has been successfully used for the trajectory and design optimisation of vertical and horizontal launch systems [21], deployment of constellations [3], interplanetary exploration missions [23] and the design of multi-debris removal missions. It has also been extended to incorporate the treatment of uncertainties, applied to the robust trajectory optimisation of a spaceplane [13].

*DFET and MACS are available open source under the SMART Optimisation and Optimal Control toolbox, `github.com/strath-ace/smart-o2c`

4. Results

4.1 Full transfer with continuous thrust

Results are presented for a full mission: outbound Earth to Mars transfer, stay time around Mars, return Mars to Earth transfer. The propulsion system is assumed to have continuously variable throttle (applying between 0-100% of available thrust). The problem was formulated to maximise the time spent in the vicinity of Mars with respect to the total mission time. In line with other mission analyses for Mars cycler and semi-cycler architectures, the optimised solutions utilise the opposition-class trajectory as depicted in Figure 2. The transfer specifications have a 176.5 day outbound trajectory, 43.8 day stay on Mars and an inbound trajectory of 313.2 days. The proportion of time spent in flight (489.7 days) to the total mission duration (533.5 days) is unfavourably large considering the additional health risks posed by prolonged exposure to radiation. Additionally, the close pass to the Sun on the inbound leg (just within the orbit of Venus) will pose additional health risks given the increased dosage.

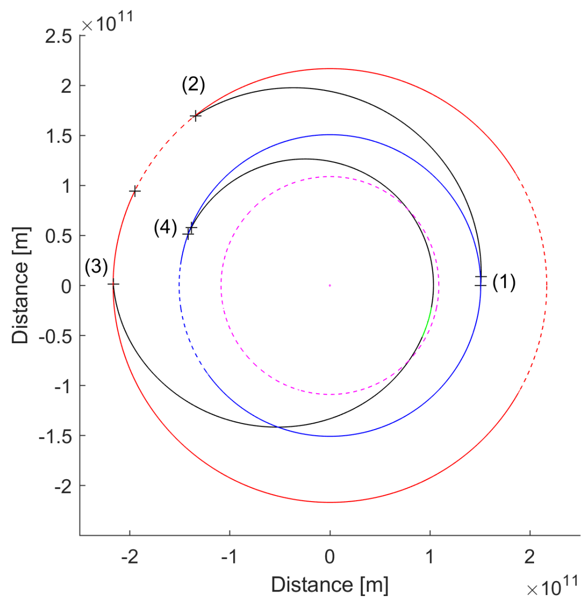

Fig. 2: Two-way transfer with continuous thrust, including (1) Earth departure, (2) Mars arrival, (3) Mars departure and (4) Earth arrival

4.2 Transfer legs with bang-bang thrust

In order to better represent the switching (or bang-bang) control) typically seen for impulsive high-thrust manoeuvres, the single outbound transfer trajectory is instead represented by an increased number of time-phases, with the engine throttle control fixed as either fully on or fully off. The problem is formulated as a multi-objective problem to examine the trade off between transfer duration and the initial gross mass of the spacecraft in LEO or LMO.

Figure 3 shows the Pareto-optimal set of design solutions considering the minimisation of both initial mass in LEO and transfer duration. A heavy dependence on propellant mass is observed, which along with a large relative variation in engine sizing corresponds to increased Δv requirements for the faster, more aggressive transfer trajectories. The corresponding transfer trajectories are presented in Figures 4 and 5. The inclusion of an assumed switching structure in engine thrust profile is seen to have a markedly positive effect on the reduction of initial mass relative to transfer duration. While the minimum time solution remains roughly unchanged, the slope of the Pareto front at lower transfer durations is much more drastic, resulting in significant mass reductions, with an assumed minimum around 100 tonnes.

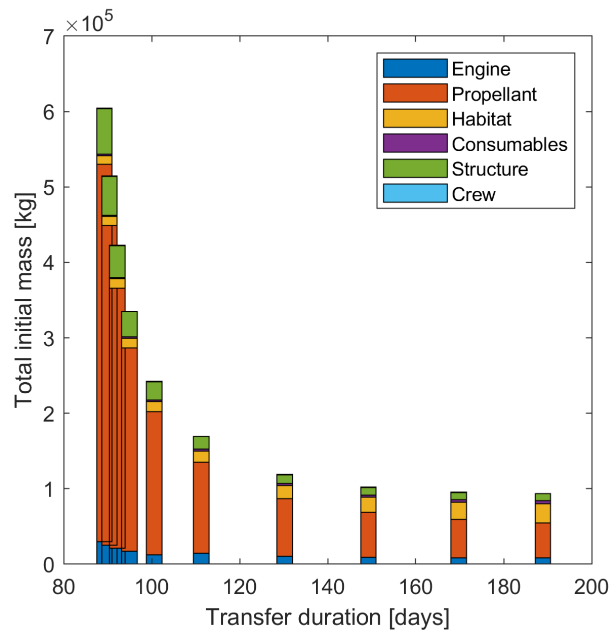

Fig. 3: Pareto set for initial mass against transfer duration

Figure 6 shows the Mars-escape and Earth-capture phases for the minimum mass and minimum time solutions, including the corresponding interplanetary trajectories.

5. Conclusion

A range of numerical models were developed, covering the major systems of a crewed interplanetary transport vehicle and mission, in order to further analysis the performance of both the vehicle and the mission options.

The models were all scalable to determine their mass, around a nominal value or set of values, in order to examine the design trade-offs within the multi-disciplinary design optimisation.

The problem formulation and test case settings and results for the mission analysis were described. Mission critical subsystems identified during the initial project literature review were translated into continuous numerical relationships.

The test cases focused on optimising the vehicle design, specifically the engine sizing, propellant masses, dry masses and mass of consumables required, and the trajectory. Transfer trajectories between Earth and Mars were evaluated separately and as part of a complete mission architecture with respect to single and multiple performance objectives.

The objective functions were set as the total transfer duration, and the gross vehicle mass. The analyses used a multi-objective optimal control solver, for non-linear constrained problems with multiple phases and within a multidisciplinary design optimisation framework.

Of particular interest was the effects of varying the stay time at Mars with respect to the minimisation of total time spent in transfer. It was found that the relative durations of mission segments (Earth to Mars, Mars stay and Mars to Earth) seemed to conform to distinct groups of solutions within the objective space. This represents a quantifiable set of transfer opportunities/mission architectures differentiated by total propellant requirement and desired duration of surface activities at Mars.

An additional finding was the universal agreement between test cases to reduce the size of the propulsion system to the lower end of the performance spectrum represented by the developed sizing relationships. Though these performance requirements still represent a fundamental shift from traditional chemical propulsion systems (particularly in terms of specific impulse), the implication is that current/future development within NTR technology should focus on improving system reliability, including investigations into extended operational cycles and restartability, rather than base performance statistics.

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References

[1] (2021). *Space Nuclear Propulsion for Human Mars Exploration*. National Academies of Sciences, Engineering, and Medicine, Washington, DC.

[2] Braun, W. V. (1969). Manned Mars Landing: Presentation to the space task group. Technical report.

[3] Di Carlo, M., Ricciardi, L., and Vasile, M. (2016). Multi-objective optimisation of constellation deployment using low-thrust propulsion. In *AIAA/AAS Astrodynamics Specialist Conference*.

[4] Drake, B. G. (2009). Human exploration of mars design reference architecture 5.0. Technical report, NASA.

[5] Dujarric, C. (2001). A new propulsion concept for interplanetary missions. *ESA Bulletin*, 108:66–71.

[6] Dujarric, C., Santovincenzo, A., and Summerer, L. (2013). The nuclear thermal electric rocket: A proposed innovative propulsion concept for manned interplanetary missions. *Progress in Propulsion Physics*, pages 293–312.

[7] Durante, M. (2014). Space radiation protection: destination mars. *Life sciences in space research*, 1:2–9.

[8] Edward Jr, A. (1971). *Comparison of Trajectory Profiles and Nuclear-propulsion-module Arrangements for Manned Mars and Mars-Venus Missions*. National Aeronautics and Space Administration.

[9] Finseth, J. (1991). Rover nuclear rocket engine program: Overview of rover engine tests, final report. Technical report, Sverdrup Technology.

[10] Koenig, D. R. (1986). *Experience Gained from the Space Nuclear Rocket Program (Rover)*. Los Alamos National Laboratory.

[11] Landau, D. F. and Longuski, J. M. (2004). A reassessment of trajectory options for human missions to Mars. In *Collection of Technical Papers - AIAA/AAS Astrodynamics Specialist Conference*, volume 2, pages 836–871.

[12] Lopez, P., Schultz, E., Mattfeld, B., Stromgren, C., and Goodliff, K. (2015). Logistics needs for potential deep space mission scenarios post Asteroid Redirect crewed Mission. In *IEEE Aerospace Conference Proceedings*.

[13] Ricciardi, L., Maddock, C., and Vasile, M. (2020). Robust trajectory optimisation of a TSTO spaceplane using uncertainty-based atmospheric models. In *AIAA International Space Planes and Hypersonic Systems and Technologies Conference*.

[14] Ricciardi, L. A., Maddock, C. A., and Vasile, M. (2018). Direct solution of multi-objective optimal control problems applied to spaceplane mission design. *Journal of Guidance, Control, and Dynamics*, 42(1):30–46.

[15] Ricciardi, L. A. and Vasile, M. (2015). Improved archiving and search strategies for multi agent collaborative search. In *International Conference on Evolutionary and Deterministic Methods for Design, Optimization and Control with Applications to Industrial and Societal Problems (EUROGEN)*, Glasgow, UK.

[16] Ricciardi, L. A., Vasile, M., and Maddock, C. (2016). Global solution of multi-objective optimal control problems with multi agent collaborative search and direct finite elements transcription. In *2016 IEEE Congress on Evolutionary Computation (CEC)*, pages 869–876. IEEE.

[17] Salotti, J. M., Heidmann, R., and Suhir, E. (2014). Crew size impact on the design, risks and cost of a human mission to Mars. In *IEEE Aerospace Conference Proceedings*.

[18] Shepherd, L. R. and Cleaver, A. V. (1948). The atomic rocket. I. *J. Brit. Interplanetary Soc.*, 7.

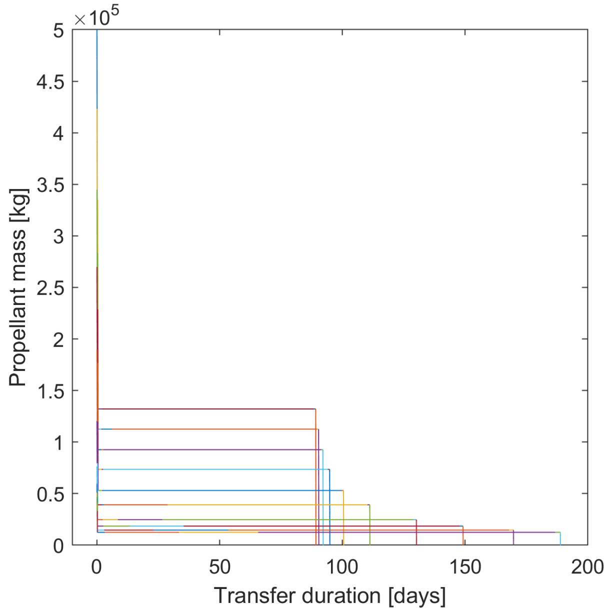
[19] Simon, M., Clowdsley, M., and Walker, S. (2013). Habitat design considerations for implementing solar particle event radiation protection. *International Conference on Environmental Systems*.

[20] Vasile, M. (2010). Finite elements in time: A direct transcription method for optimal control problems. In *AIAA/AAS Astrodynamics Specialist Conference, Guidance, Navigation, and Control and Co-located Conferences*, Toronto, Canada.

[21] Vasile, M. L., Maddock, C., and Ricciardi, L. (2018). Multi-objective optimal control of re-entry and abort scenarios. In *Space Flight Mechanics Meeting*.

[22] Willis, E. A. (1971). *Comparison of Trajectory Profiles and Nuclear-propulsion-module Arrangements for Manned Mars and Mars-Venus Missions*. NASA.

[23] Zuiani, F., Kawakatsu, Y., and Vasile, M. (2013). Multi-objective optimisation of many-revolution, low-thrust orbit raising for destiny mission. In *23rd AAS/AIAA Space Flight Mechanics Conference*, Kauai, Hawaii.



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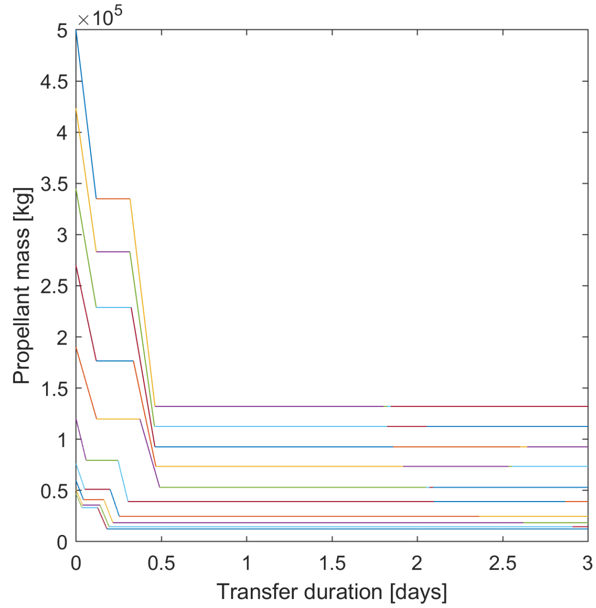

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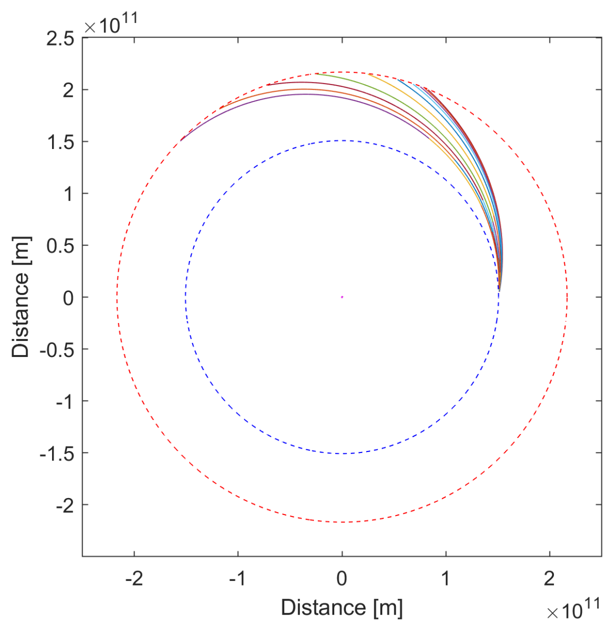

Fig. 4: Pareto set for propellant mass across transfer duration and corresponding transfer trajectories

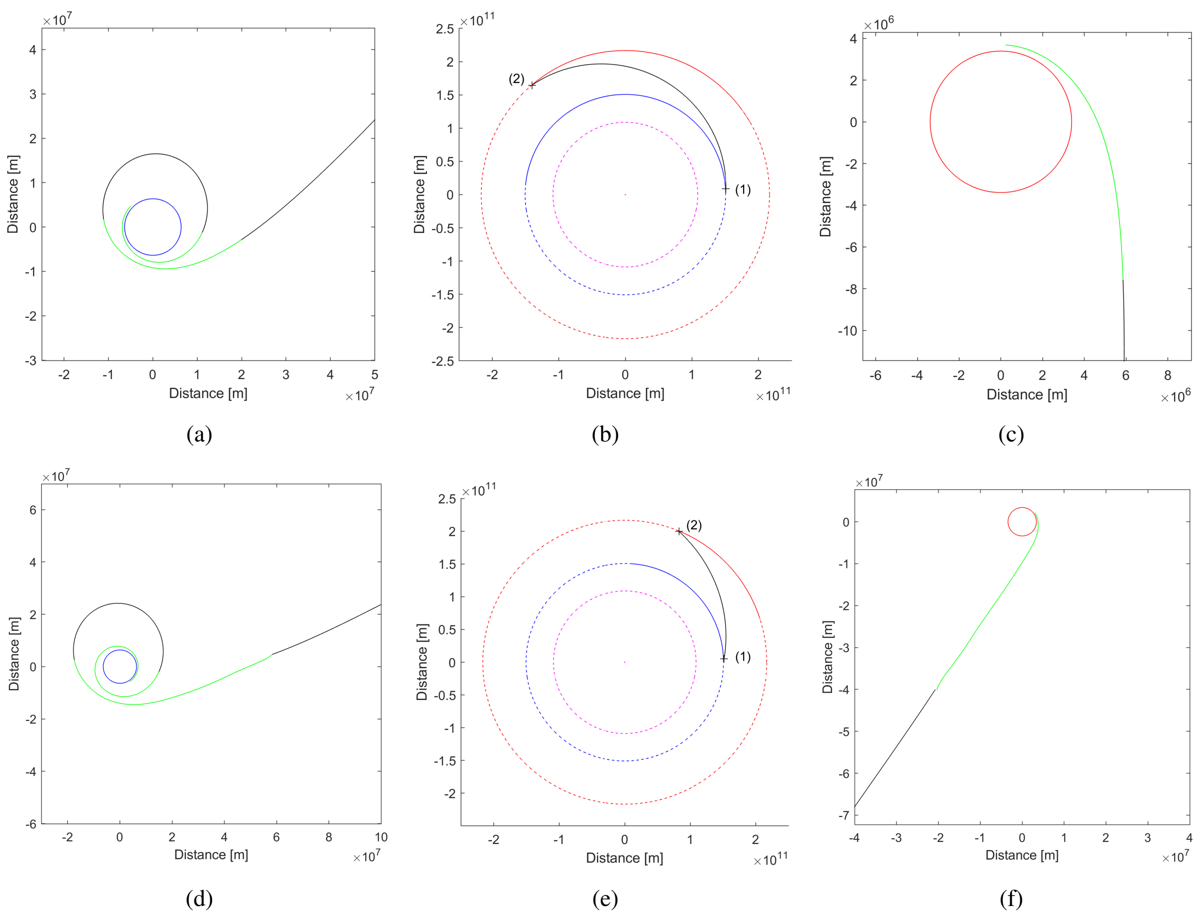

Fig. 5: Escape/Capture manoeuvres and transfer trajectories for minimum mass (a,b,c) and minimum time (d,e,f) solutions, including (1) Earth departure and (2) Mars arrival. Green lines represent ’engine-on’ segments

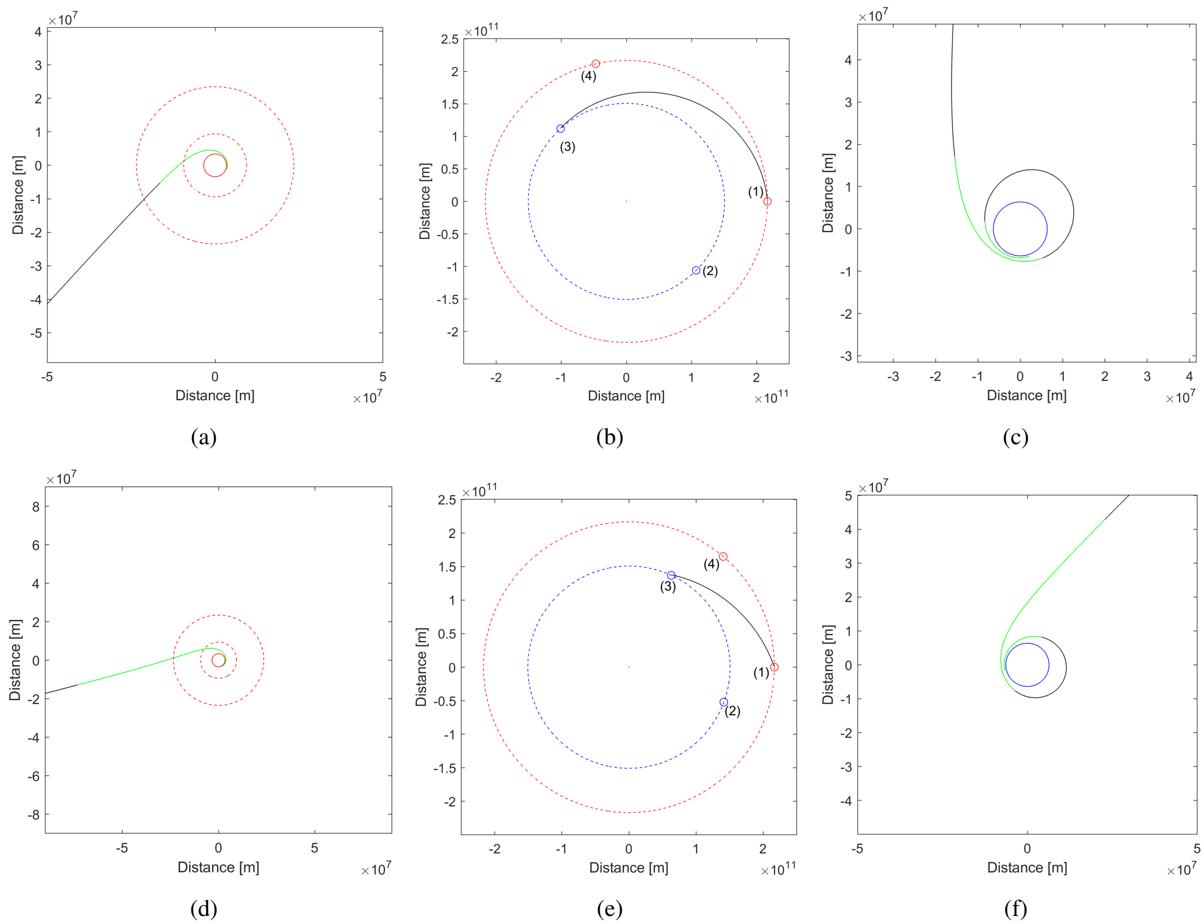

Fig. 6: Escape/Capture manoeuvres and transfer trajectories for the minimum mass (a,b,c) and minimum time (d,e,f) solutions, including (1) Mars departure, (2) Earth position at Mars departure, (3) Earth arrival and (4) Mars position at Earth arrival. Green lines represent 'engine-on' segments.