Interplanetary Mission Design Using Multi-Disciplinary Design Optimization

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ABSTRACT

A detailed discussion is presented for a software tool developed to conduct interplanetary mission design trade studies and to identify optimum mission parameters. The Interplanetary Design Framework Software (IDFS) code supports these trade studies by determining optimum interplanetary missions using a diverse set of user inputs. A Graphical User Interface accepts trade parameters selected from the IDFS database or from direct user input. Both graphical and tabular output is provided. IDFS core components include: (i) Launch vehicle performance calculation, (ii) Planetary ephemeris propagation, (iii) Planetary position and velocity vector computation, (iv) Interplanetary trajectory determination, (v) Delta velocity calculation, (vi) Usable payload mass and time of flight computation, and (vii) Optimization of delta velocity, usable payload mass, and time of flight using Multidisciplinary Design Optimization (MDO) techniques. Benchmarking results are also presented. Finally, IDFS is utilized to identify an optimum interplanetary mission and the trade results are presented, discussed, and evaluated.

1. INTRODUCTION

Not since the Apollo Program of the late 1960s and early 1970s has there been such an earnest effort to place men on the surface of a distant body. This is due in large part to President George W. Bush's 2004 announcement of an initiative to explore Mars via a permanent Moon base [1]. The President's proposal, in conjunction with the recent backing of Congress, places both manned and unmanned interplanetary travel at the forefront of future space endeavors [2].

In parallel with the renewed interest in interplanetary exploration, the MDO technique has continued to come to the forefront as a formal method used in the design of complex systems. MDO allows the effects of coupling between various interacting design variables to be numerically explored and the sensitivity of a design to each factor to be investigated.

Several numerical MDO methods currently exist and are well documented in the public domain. Further, there exists a serious worldwide effort to expand MDO technology and theory. For example, the AIAA has formed a technical committee to act as a forum for those active in development, application, and teaching of MDO. Similarly, NASA-Langley has an MDO branch whose mission is to lead in the identification, development, and demonstration of MDO methods.

While the Moon and Mars are the current focus of NASA planning, several other possible interplanetary missions are also of extreme interest. Saturn's moons Enceladus and Titan and Jupiter's moon Europa are of particular significance due to the likely presence of liquid water on the surface of those bodies [3]. As Jupiter is the closest of the outer planets, robotic "Mars like" missions to explore that planet and her many intriguing satellites must also be seriously considered.

Motivated by the heightened interest in interplanetary missions, the myriad of other possible high value interplanetary missions, and the rise of MDO, a software tool has been developed. The tool, known as IDFS, enables first order analysis of interplanetary mission design trade studies to be conducted efficiently and accurately. In addition, IDFS uses MDO methods to allow the complex trade space of interplanetary missions to be readily evaluated numerically.

2. BACKGROUND

The IDFS tool, whose high level flow is illustrated by Figure 1, is a MATLAB® based software program. IDFS is principally comprised of a mathematical model that determines a series of transfer orbit trajectories from Earth to any planet in the solar system across a user specified launch and arrival day windows. The corresponding departure and arrival ΔVs, departure and arrival propellant requirements, launch and arrival dates, and the resulting Payload System Mass (PSM) are computed for each unique trajectory. The results produced by this model are then analyzed, using the Simplex and Matrix Experiment MDO techniques, to identify the most advantageous case. The optimum case identified is a function of the user inputs and which of the six optimization options identified below is selected by the user.

- Find maximum Payload Systems Mass
- Find minimum Time of Flight
- Goalseek Payload System Mass

- Goalseek Time of Flight
- Find minimum Time of Flight for a given Payload System Mass
- Optimize Low Earth Orbit Altitude for Maximum Payload System Mass

2.1. METHODOLGY

At a high level, IDFS uses a MATLAB® mathematical model to calculate launch vehicle performance, determine planetary positions, identify interplanetary trajectories, compute ΔVs, derive propellant and payload system masses, and determine of the resulting optimum mission as outlined below:

- Launch vehicle performance – The Energy equation or a curve fit to manufacturer performance data
- Planetary orbital position – Keplerian orbit elements
- Interplanetary trajectories – Lambert's Problem
- ΔV computations – Patched-Conic Approximation
- PSM derivation – Rocket engine equation
- Optimum case determination – MDO Simplex and Matrix Experiment techniques

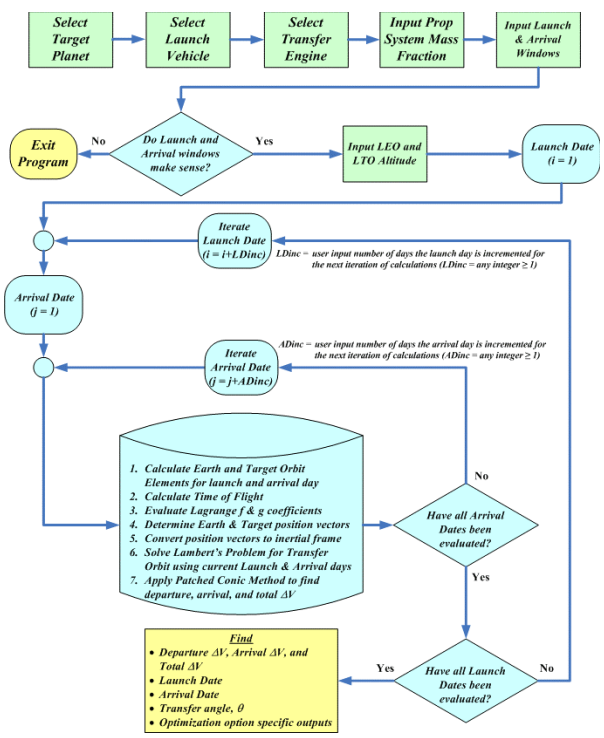

Fig. 1: IDFS High-Level Flow Diagram

2.1.1. Launch Vehicle Performance

IDFS allows the user to either select one of 12 launch vehicles from a database or to input launch vehicle performance directly. For the provided launch vehicles, performance is modeled using one of two approaches; performance curves or the energy equation.

For those launch vehicles that have published performance curves, the Least Squares Fit utility provided by MATLAB® is used to derive a second order mass to LEO altitude. These equations provide the launch vehicle throw weight, in kg to LEO where LEO altitude, h in kilometers, is input by the user. The performance as a function of altitude, known as Total System Mass (TSM) or $m_{TSM,}$ for 3 of the launch vehicles is provided by the Eqns. 1-3 below [4]-[6].

$$m_{TSM\text{-}AtlasV} = 21436.632 - 4.638h + 0.00064h^2 \quad (1)$$

$$m_{TSM\text{-}Angara5} = 25543.911 - 5.359h + 0.0007h^2 \quad (2)$$

$$m_{TSM\text{-}DeltaIV} = 25712.256 - 3.546h + 0.00024h^2 \quad (3)$$

Performance estimates for the remaining launch vehicles use the circular orbit velocity equation, (Eq. 4), the energy equation (Eq. 5), and manufacturer performance data. Once the energy to the reference orbit is known, the performance (m_{TSM}) to any low Earth orbit can be approximated by assuming E is constant, using the user provided altitude h, and Eq. 5 to derive a result for the total system mass, m_{TSM}. Eq. 6 shows the result of this derivation.

$$v = \sqrt{\mu_{earth} \big/ (r_{earth} + h)} \quad (4)$$

$$E = \frac{1}{2} m_{TSM} v^2 + m_{TSM} gh \quad (5)$$

$$m_{TSM} = \frac{2E}{v^2 + 2gh} \quad (6)$$

2.1.2. Planetary Position Calculation

Six Keplerian orbit elements are used to calculate the heliocentric radius vector of the Earth and the target planet for each Launch Day/Arrival Day pair. The six elements include: 1) Semi-major axis, a; 2) Inclination, i; 3) Eccentricity, e; 4) Longitude of the Ascending Node, Ω; 5) Longitude of Perihelion, $\tilde{\omega}$; and 6) Mean Longitude, L. The J2000 values and change rates per day for each of the six elements for all nine planets are obtained from NASA-Goddard's website [7].

To compute planetary position IDFS acquires the launch and arrival window dates from the user in the Gregorian calendar format. IDFS converts the launch

and arrival dates to Julian format and propagates the six orbit element values from epoch forward in time. Eqns. 7 and 8 calculate the elapsed time since the epoch with T_1 = launch day at Earth (elements with a subscript of 1) and T_2 = arrival day at the target (elements with a subscript of 2). Eqns. 9-10 illustrate orbit element propagation forward from the epoch.

$$T_1 = LaunchDay - epoch \tag{7}$$

$$T_2 = ArrivalDay - epoch \tag{8}$$

$$a_1 = \left(a_{epoch1} + a_{rate1}\right)T_1 \tag{9}$$

$$a_2 = \left(a_{epoch2} + a_{rate2}\right)T_2 \tag{10}$$

After the Earth and target orbit elements are updated to the launch and arrival days, Mean Anomaly, M, Eccentric Anomaly, E, and True Anomaly, ν are calculated as shown by Eqns. 11-13. Using True Anomaly from Eq. 13, the heliocentric radius vectors for earth and the target are computed using Eqn. 14. The heliocentric Cartesian coordinates for the planetary radius vectors **r** (r_x, r_y, and r_z) are given by Eqns, 15 – 17 [8]. Note Eqn. 12 is solved by Newton's Method.

$$M = L - \tilde{\omega} \tag{11}$$

$$E = E - e\sin(E) \tag{12}$$

$$\nu = 2\left(a\tan\left[\tan\left(\frac{E}{2}\right)\sqrt{\frac{(1+e)}{(1-e)}}\right]\right) \tag{13}$$

$$r = a\left[\frac{1-e^2}{1+e\cos(\nu)}\right] \tag{14}$$

$$r_x = r\left[\cos(\Omega)\cos(\nu+\tilde{\omega}+\Omega) - \sin(\Omega)\sin(\nu+\tilde{\omega}-\Omega)\cos(i)\right] \tag{15}$$

$$r_y = r\left[\sin(\Omega)\cos(\nu+\tilde{\omega}+\Omega) + \cos(\Omega)\sin(\nu+\tilde{\omega}-\Omega)\cos(i)\right] \tag{16}$$

$$r_z = r\left[\sin(\nu+\tilde{\omega}-\Omega)\sin(i)\right] \tag{17}$$

Following Earth and target heliocentric radius vector calculation, the heliocentric velocity vectors are found for both planets. The magnitude of the velocity vector is determined using Eq. 18. The result of Eq 18 is used in Eqns. 19 – 21 to find the heliocentric x, y, and z components of the two velocity vectors [8].

$$V = \sqrt{\mu_{sun}\left(\frac{2}{r} - \frac{1}{a}\right)} \tag{18}$$

$$V_x = V\left[-\cos(\Omega)\sin(\nu+\tilde{\omega}+\Omega) - \sin(\Omega)\cos(\nu+\tilde{\omega}-\Omega)\cos(i)\right] \tag{19}$$

$$V_y = V\left[-\sin(\Omega)\sin(\nu+\tilde{\omega}+\Omega) + \cos(\Omega)\cos(\nu+\tilde{\omega}-\Omega)\cos(i)\right] \tag{20}$$

$$V_z = V\left[\cos(\nu+\tilde{\omega}-\Omega)\sin(i)\right] \tag{21}$$

2.1.3. Interplanetary Trajectory Determination

IDFS utilizes a MATLAB® math model that solves Lambert's Problem to calculate the interplanetary trajectories. This approach allows the minimum energy trajectory to be found, using time of flight and the two planetary positions, even though an infinite number of trajectories exist between departure and arrival points.

IDFS begins at the first day in the launch window and calculates the Earth's position for that day as described above. Similarly, the target's position is calculated for the first day in the arrival window. Time of Flight (ToF) is determined from the elapsed time between those two days. Once the time of flight, t_f and the departure arrival positions are known, the solution to Lambert's Problem is determined using Lagrange's formulation (Eq. 22), where N is the number of revolutions about the sun for the transfer orbit, and α and β are defined by Eqns. 23 and 24 [9].

$$t_f = \sqrt{\frac{a^3}{\mu_{sun}}}\left[2N\pi + \alpha - \beta - \left(\sin(\alpha) - \sin(\beta)\right)\right] \tag{22}$$

$$\sin(\alpha) = \sqrt{\frac{s}{2a}} \tag{23}$$

$$\sin(\beta) = \sqrt{\frac{s-c}{2a}} \tag{24}$$

IDFS implements a modified Battin algorithm to solve Lambert's Problem. The approach was developed by the European Space Agency's (ESA) Advanced Concepts Team (ACT) and is superior to other methods in that it avoids the singularity that commonly occurs at a transfer angle of 180º [10]. To solve Lambert's Problem by the ACT method, the transfer angle θ between the Earth's and target planet's position vectors is determined as shown by Eq. 25. Once θ is known, the chord length of the transfer orbit, c, the semi-perimeter length of the transfer orbit, s, and the minimum energy semi-major axis length, a_m, of the transfer orbit are all computed using Eqns. 26-28. From the results of Eqns. 26-28, the dimensionless parameter λ is found by Eq. 29.

$$\theta = a\cos\left[\frac{r_{1x}r_{2x} + r_{1y}r_{2y} + r_{1z}r_{2z}}{r_1 r_2}\right] \tag{25}$$

$$c = \sqrt{1 + r_2^{\,2} - 2r_2\cos(\theta)} \tag{26}$$

$$s = \frac{1 + r_2 + c}{2} \tag{27}$$

$$a_m = \frac{s}{2} \tag{28}$$

$$\lambda = \frac{\cos\left(\frac{\theta}{2}\right)\sqrt{r_2}}{s} \tag{29}$$

Following the determination of θ, c, s, a_m, and λ the transfer orbit solution conic is found using these values to solve Eqns. 22-24. Once a transfer orbit solution is found for the unique launch day/arrival day pair the Patched Conic Approximation method (see Figure 2) is used to calculate the departure and arrival ΔVs.

In this method, spacecraft motion is simplified by assuming only two bodies exist in the problem at any point in the transfer orbit. The fundamental premise of the two-body system is that only the spacecraft and the celestial body it orbits need be considered since the dominating influence on the spacecraft's motion is the celestial body whose "sphere of influence" it is within.

Using this two-body concept, the velocity of the transfer orbit with respect to both the Earth and the target, V_{Depart} and V_{Arrive}, are determined by finding the Lagrange *F* and *G* coefficients from Eqns. 32-35 and using those results in Eqns. 30 and 31 [8]. Note, P is the semi-latus rectum of the transfer ellipse.

$$V_{Depart} = \frac{1}{G}\left(r_2 - F r_1\right) \tag{30}$$

$$V_{Arrive} = \dot{F} r_2 + \dot{G} \; V_{Depart} \tag{31}$$

$$F = 1 - \frac{r_2}{P}\left[1 - \cos(\theta)\right] \tag{32}$$

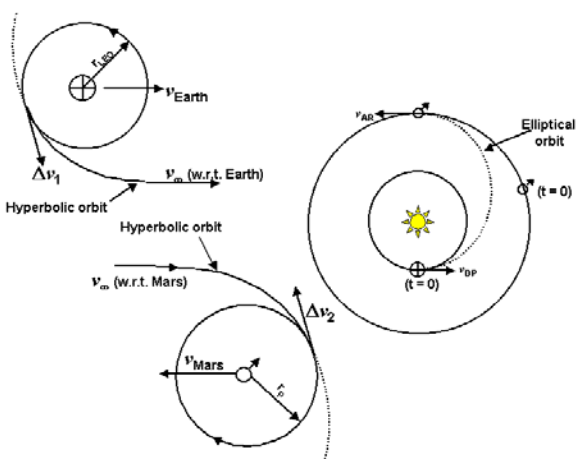

Fig. 2: The Patched-Conic Method [11]

$$G = \frac{r_1 r_2 \sin(\theta)}{\sqrt{\mu_{sun} P}} \tag{33}$$

$$\dot{F} = \sqrt{\frac{\mu_{sun}}{P}}\left[\tan\left(\frac{\theta}{2}\right)\left(\frac{1-\cos(\theta)}{P} - \frac{1}{r_2} - \frac{1}{r_1}\right)\right] \tag{34}$$

$$\dot{G} = 1 - \frac{r_1}{P}\left[1 - \cos(\theta)\right] \tag{35}$$

The methodology developed by ACT and implemented in IDFS, overcomes the known singularity problem at θ = 180°. The approach provides a generic orbit solution to the boundary problem as shown by Eq. 36. Using Eq. 23 to find α and substituting the result into Eq. 36 yields accurate values of ΔV versus ToF for the entire real axis. This occurs as the method avoids the use of Langrage coefficients as required for the solution to Eqns. 32-35. However, the algorithm in Eq. 36 will not converge at θ = 0° as log(0) is indeterminate [10].

$$X = \log\left[1 + \cos\left(\frac{\alpha}{2}\right)\right] \tag{36}$$

2.1.4. Delta Velocity Computation

Once the departure and arrival velocities are found, the required ΔV for each trajectory can be computed. IDFS finds these velocities using vector subtraction, as shown by Eqns. 37 and 38, to account for both the hyperbolic velocity on the transfer ellipse and the velocity of the Earth and target planet (see also Fig. 2).

$$\vec{V}_{E\infty} = \vec{V}_{Depart} - \vec{V}_{Earth} \tag{37}$$

$$\vec{V}_{T\infty} = \vec{V}_{Arrive} - \vec{V}_{Target} \tag{38}$$

Once the transfer orbit velocity of the spacecraft with respect to Earth and the target planet are computed, ΔV can be found. In each case ΔV is the difference between the spacecraft's velocity around the planet and the velocity of the spacecraft with respect to the planet. The circular orbit velocities are found by using the user provided orbital altitudes, h, in Eq. 39. Once the orbital velocities are known ΔV_{Depart}, ΔV_{Arrive}, and ΔV_{Total} are found from Eqns. 40-42 respectively.

$$V_{Orbit} = \sqrt{\frac{\mu_{Planet}}{h}} \tag{39}$$

$$\Delta V_{Depart} = V_{E\infty} - V_{LEO} \tag{40}$$

$$\Delta V_{Arrive} = V_{T\infty} - V_{LTO} \tag{41}$$

$$\Delta V_{Total} = \Delta V_{Depart} + \Delta V_{Arrive} \quad (42)$$

The ΔVs are then saved for future evaluation and the entire process is repeated for the next launch /arrival day pair. The next iteration is performed by holding the launch day constant and incrementing the arrival day until each case in the arrival window is considered. Once each arrival day has been used, the launch day is incremented and the process is repeated until the ΔVs for all possible pairs have been calculated.

2.1.5. Payload System Mass Calculation

Payload System Mass (PSM), m_{PSM}, is the mass of the "usable" Payload delivered to the target. Propellant and Propulsion Subsystem mass are not included in the computed PSM. The propellant needed to complete the mission is calculated by IDFS while the engine and propellant tank mass are found from user inputs. Before calculating PSM, the required propellant for both the departure and arrival ΔVs must be determined using Eq.43 (derived from the rocket engine equation).

$$m_1 = m_0\left(1 - e^{-\Delta V / gI_{sp}}\right) \quad (43)$$

To find the required departure propellant, m_{TSM} is obtained from Eqns. 1-3, 6 or by user input. TSM is then substituted into Eq. 43 with specific impulse, I_{sp}, obtained as a user input or from the database. The result of Eq. 44, $m_{prop\text{-}depart}$ yields the propellant required to depart Earth. The wet departure mass, $m_{wet\text{-}depart}$, is then calculated as shown in Eq. 45 with the result used in Eq. 46 to compute the propellant required to generate the arrival ΔV.

$$m_{prop-depart} = m_{TSM}\left(1 - e^{-\Delta V_{Depart} / gI_{sp}}\right) \quad (44)$$

$$m_{wet-depart} = m_{TSM} - m_{prop-depart} \quad (45)$$

$$m_{prop-arrive} = m_{wet-depart}\left(1 - e^{-\Delta V_{Arrive} / gI_{sp}}\right) \quad (46)$$

Once the mass of the propellant needed to perform the arrival ΔV is known, the total arrival mass, $m_{total\text{-}arrive}$, is obtained utilizing Eq. 47. When the wet arrival mass is determined, Eq. 48 is used to calculate the PSM. In Eq. 48, the engine mass is again from the database or a user input, the propellant tank mass is computed using the total propellant mass (Eq. 49), and the propellant tank mass fraction, m_{pmf}, is provided by the user. The IDFS default m_{pmf} is 4% based on the propellant mass tank fraction of the Space Shuttle's External Tank and the Apollo Command Service Module.

$$m_{total-arrive} = m_{wet-depart} - m_{prop-arrive} \quad (47)$$

$$m_{PSM} = m_{wet-arrive} - m_{engine} - m_{\tan k} \quad (48)$$

$$m_{\tan k} = m_{pmf}\left(m_{prop-depart} + m_{prop-arrive}\right) \quad (49)$$

2.1.6. Optimum Case Identification

IDFS allows the user to select from one of six possible optimization options. These options include: 1) Find maximum PSM; 2) Find minimum ToF; 3) Goalseek PSM; 4) Goalseek ToF; 5) Find minimum ToF for a specified PSM; and 6) Optimize LEO altitude for maximum PSM. In each case, IDFS optimizes only within the constraints of the user input parameters.

2.1.6.1. Find Maximum Payload System Mass

To find the maximum PSM, IDFS performs a two-step optimization process. IDFS first calculates the arrival, departure, and total ΔVs. It then employs the MDO Simplex approach to identify the minimum total ΔV, (which corresponds to the maximum PSM case). When the minimum total ΔV is identified, PSM is calculated as discussed above.

2.1.6.2. Find Minimum Time of Flight

IDFS identifies the minimum ToF by using the input parameters only. The tool merely determines minimum ToF by finding the elapsed time between the launch window end and the departure window start. The associated departure and arrival ΔVs are computed to determine the total ΔV for the ToF. Once the total ΔV is found, the PSM is calculated.

2.1.6.3. Goalseek Payload System Mass

The IDFS Goalseek PSM routine allows the user to specify a range of Payload System Masses to be delivered to a remote planet. The algorithm evaluates the user input parameters to determine if such a goal can be met. The goal is found by using the MDO Matrix Evaluation technique. IDFS first calculates the total ΔV.

Once the total ΔV is found PSM is determined and compared to the PSM goal. If the current PSM is within range of the PSM goal, all information associated with that launch day/arrival day pair, including the ToF, is saved. On subsequent iterations, the last saved PSM is compared to the PSM for the current launch day/arrival day pair. If the PSM from the current iteration is in range and closer to the PSM

goal than the last saved PSM, the current PSM and its related information are saved. However, if the two PSMs are equal, the mission with the shortest ToF, and it associated design characteristics, is saved.

2.1.6.4. Goalseek Time of Flight

Like the PSM Goalseek, IDFS uses the MDO Matrix Evaluation method to find the ToF goal. Similar to the previous optimization techniques, the total ΔV and PSM for each launch/arrival day pair is determined. If the ToF calculated for the current iteration is within the goal range, all launch/arrival day pair data is saved.

On succeeding iterations, the last saved ToF is compared to the current ToF for the launch/arrival day pair. If the latest ToF is closer to the ToF goal then the saved ToF, it and the related information are saved. However, if the two ToFs are equal, then the PSMs are compared and the case with the greatest PSM is saved.

2.1.6.5. Find Minimum ToF for a Given PSM

Perhaps the most useful IDFS tool for the planning of manned interplanetary missions is the Find Minimum ToF for a Given PSM optimization routine. IDFS, using both the standard set of input parameters and a range of acceptable PSMs provided by the user, identifies the minimum ToF for the given PSM.

In a fashion identical to the routines described earlier, the total ΔV, PSM, and ToF for the current launch /arrival day pair are found. As before, if the PSM calculated for the current iteration is within the goal range, the data associated with that pair is saved.

On subsequent iterations, the last saved PSM is contrasted against the current PSM. If the current PSM is within the specified range and its associated ToF is shorter than the saved ToF, then the current iteration's information is saved. If however, the two ToFs are equal, the mission with the greatest PSM is retained.

2.1.6.6. Optimize LEO for Maximum PSM

The optimum LEO case is found by first computing the maximum PSM, based on the user input mission design parameters, at a LEO of 150 km. LEO is then stepped at 50 km increments to an altitude of 1000 km. At each altitude, the maximum PSM is again calculated and compared to the previous case. The optimum case is identified and data saved and output for that case.

3. RESULTS

PSM is of course fundamental to the design of all interplanetary missions. That is, the PSM that can be delivered to the target planet is essential to the design and success of any mission. All aspects of the mission design are affected by this very limited resource. Consequently, every mission necessarily seeks to optimize PSM so to maximize the benefit.

Conversely, for crewed space missions, reducing the ToF, and thus the overall exposure of the crew to space environments, is also a mission design driver. ToF can also drive cost for unmanned missions as longer missions require additional mission ground support.

Unfortunately, the optimum method of shortening ToF is to traverse high-energy interplanetary trajectories. These high-energy interplanetary trajectories require massive propellant loads that consume a large fraction of the mass and thus diminish the utility of the mission. Further, delivering large propellant loads, even to LEO, equates to high launch costs. Accordingly, minimizing the ToF and maximizing the PSM are two goals that are diametrically opposed. This dichotomy of ToF vs. PSM for space mission design illustrates the need for tools like IDFS to find optimum mission designs.

The conclusion that PSM and ToF are the two most important aspects of mission design shaped the design of IDFS. The tool could merely have calculated trajectories and ΔVs, but this information is only moderately relevant. Each of the optimization tools was developed with the goal of helping the user find the best combination of PSM and ToF.

3.1. IDFS Outputs

IDFS produces outputs in two forms. The input trade parameters and the results of the trade are provided as text in the MATLAB® command window. In addition, each IDFS optimization routine generates six plots that illustrate the results of the input trade parameter analysis. Two example plots are shown in Fig. 3 and 4.

Each optimization routine echoes the user input parameters to the command window so that the user can view the values input in conjunction with the results. This facilitates iteration of trade parameters. Input parameters displayed by IDFS include the launch vehicle, target planet, number of launches, aerobrake ΔV, launch and arrival windows and window step size, LEO and LTO altitudes, rocket engine specifics, and propellant tank mass fraction.

The IDFS MATLAB® command window outputs include launch vehicle performance, total, departure, and arrival ΔV, optimum mission launch and arrival days, PSM and ToF for the optimum case, optimum mission transfer angle θ, departure and arrival propellant mass, and propulsion system mass.

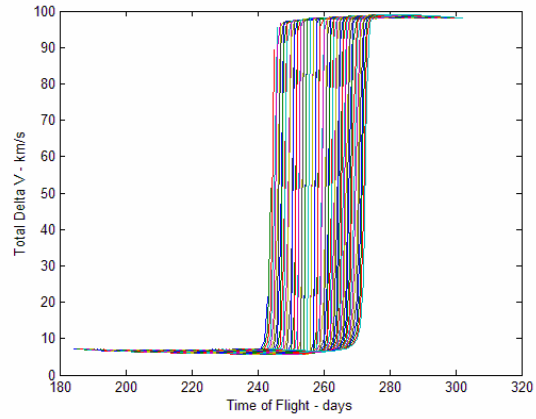

Fig. 3: Total ΔV vs. ToF

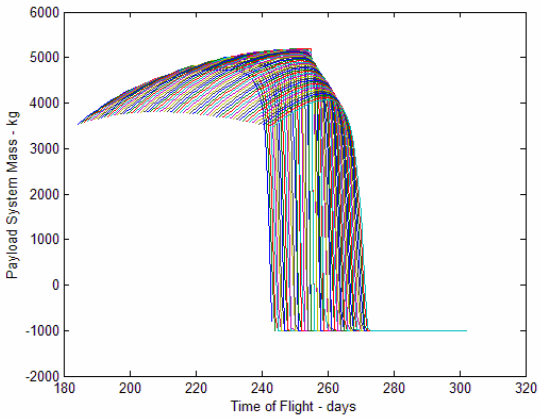

Fig. 4: PSM vs. ToF

Several indicators of the trade parameter validity can be assessed from the tool outputs. Specifically, a small or negative PSM occurs when there are large arrival or departure ΔVs or both. Small PSMs ultimately result from a θ not near 180° (a Hohmann transfer). Small θs (less than ~90°) imply launch and arrival windows too close together or that, due to the relative motion of the planets, the window locations are not optimal.

Finally, IDFS produces negative PSMs when that result is driven by the inputs. Though negative PSMs do not make sense physically, they are provided so that an increase in even a negative PSM, suggests to the user in which direction to adjust the launch and/or arrival windows such that the ToF should move in a direction that produces a larger PSM. Note that a negative PSM is the result of subtracting the larger Propulsion system mass from the smaller resulting PSM (see Eq. 48).

3.2. Benchmarking

Benchmarking of IDFS was performed by two separate code-to-code comparisons. The first validated IDFS planetary position calculations by contrasting the results produced by IDFS and JPL's Horizons website [11]. The results in Table 1 show IDFS accurately calculates, to 0.1 m, all planetary position vectors.

A second code-to-code comparison was performed on the IDFS ΔV calculations. This evaluation, whose results are provide in Table 2, is based on the output of a software model utilized by industry analysts to perform first order interplanetary mission trajectory design. As can be seen from the results below, the ΔV determined by IDFS is extremely accurate [12].

Table 1: Planetary Position Benchmark

Radius (km)	Mercury			Venus		
	IDFS	JPL	% Err	IDFS	JPL	% Err
Rx	52638550.8	52638550.8	0.0%	-54092361.3	-54092361.3	0.0%
Ry	-22688214.2	-22688214.2	0.0%	92784072.7	92784072.7	0.0%
Rz	-6683556.1	-6683556.1	0.0%	4391857.2	4391857.2	0.0%
Radius (km)	Earth			Mars		
	IDFS	JPL	% Err	IDFS	JPL	% Err
Rx	95541839.9	95541839.9	0.0%	103303558.3	103303558.3	0.0%
Ry	-118007885.5	-118007885.5	0.0%	-182950046.9	-182950046.9	0.0%
Rz	2227.6	2227.6	0.0%	-6371754.3	-6371754.3	0.0%
Radius (km)	Jupiter			Saturn		
	IDFS	JPL	% Err	IDFS	JPL	% Err
Rx	-772138493.7	-772138493.7	0.0%	635076135.5	635076135.5	0.0%
Ry	243284053.6	243284053.6	0.0%	-1355728654.5	-1355728654.5	0.0%
Rz	16269944.9	16269944.9	0.0%	-1671976.7	-1671976.7	0.0%
Radius (km)	Uranus			Neptune		
	IDFS	JPL	% Err	IDFS	JPL	% Err
Rx	2966641523.4	2966641523.4	0.0%	3704547277.0	3704547277.0	0.0%
Ry	-485416521.3	-485416521.3	0.0%	-2538175432.8	-2538175432.8	0.0%
Rz	-40136794.8	-40136794.8	0.0%	-33237926.6	-33237926.6	0.0%
Radius (km)	Pluto					
	IDFS	JPL	% Err			
Rx	671955176.4	671955176.4	0.0%			
Ry	-4767573386.5	-4767573386.5	0.0%			
Rz	315848193.6	315848193.6	0.0%			

Table 2: ΔV Benchmark

Item	Case 1			Case 2		
	IDFS	LM	% Diff	IDFS	LM	% Diff
Departure ΔV	5.989	5.989	0.00014%	3.839	3.840	0.00055%
Arrival ΔV	7.839	7.839	0.00000%	2.634	2.634	0.00000%
Total ΔV	13.828	13.828	0.00006%	6.473	6.473	0.00033%
TOF (days)	149	149	0.00000%	200	200	0.00000%
Launch Day	2/8/2018	2/8/2018		3/12/2016	3/12/2016	
Arrival Day	7/7/2018	7/7/2018		9/28/2016	9/28/2016	
Target	Mars	Mars		Mars	Mars	
Low Orbit Alt	400	400		400	400	
Item	Case 3			Case 4		
	IDFS	LM	% Diff	IDFS	LM	% Diff
Departure ΔV	10.369	10.369	-0.00009%	6.459	6.459	0.00017%
Arrival ΔV	10.699	10.699	-0.00006%	10.407	10.407	-0.00002%
Total ΔV	21.068	21.068	-0.00007%	16.866	16.866	0.00005%
TOF	289	289	0.00000%	956	956	0.00000%
Launch Day	12/7/2015	12/7/2015		7/19/2011	7/19/2011	
Arrival Day	9/21/2016	9/21/2016		3/1/2014	3/1/2014	
Target	Venus	Venus		Jupiter	Jupiter	
Low Orbit Alt	200	500		300	150000	

4. Trade Study Results

To provide a use case for IDFS, a detailed trade study was conducted to identify the maximum PSM that could be delivered to Mars using an Atlas V-552

launch vehicle, and an RL-10D rocket engine, in the 2010 to 2020 time frame.

The trade was conducted holding all input parameters fixed except for the launch and arrival windows. Starting 1 January 2010, six months launch and arrival windows were defined and placed back to back in time. IDFS was then run for each window pair sliding by six months to the right in each case. In total, twenty cases were run and those with large PSMs were identified.

Following the initial performance of all twenty cases, 7 cases were identified for further study. For those cases the results of the initial run where evaluated, the launch and arrival windows were refined such that they were short and centered on the optimum values from the first run, and the step size minimized. The second round of runs identified Case 4b as the optimum solution for the trade space. Trade results are provided in Table 3.

Trade Parameter/Variable	Units	Window 4	
		Case 4a	Case 4b
Launch Vehicle	-	Atlas V Heavy	Atlas V Heavy
Target Planet	-	Mars	Mars
Number of LV launches	#	1	1
Minimum Total ΔV	km/s	6.5210	5.7280
Earth Departure ΔV	km/s	3.7693	3.6427
Total Mars Arrival ΔV	km/s	2.7517	2.0853
Aerobrake ΔV	km/s	0.0000	0.0000
Net Mars Arrival ΔV	km/s	2.7517	2.0853
Launch Window	Date	7/1/11-12/31/11	10/1/11-12/31/11
Launch Window Step Size	Days	10	1
Launch Date	Date	11/18/2011	11/13/2011
Arrival Window	Date	1/1/12-6/30/12	5/1/12-7/31/12
Arrival Window Step Size	Days	10	1
Arrival Date	Date	6/19/2012	7/23/2012
Time of Flight	Days	214.0	253.0
Transfer Angle θ	Deg	156.52	178.18
Low Earth Orbit	km	200.0 km	200.0 km
Low Mars Orbit	km	500.0 km	500.0 km
Total Spacecraft Mass to LEO	kg	20534.53	20534.53
Engine Type	-	LOX/H_2	LOX/H_2
Engine I_{sp}	secs	472.0 secs	472.0 secs
Departure Propellant Mass	kg	11436.43	11184.27
Spacecraft Wet Departure Mass	kg	9098.09	9350.26
Arrival Propellant Mass	kg	4076.32	3390.37
Propellant System Mass	kg	802.51	764.99
Propellant Tank Mass Fraction	kg	4.0%	4.0%
Payload System Mass	kg	4219.26	5194.90

Table 3: Trade Study Results Summary

5. CONCLUSIONS

This study has shown that MDO methodologies can be utilized to aid in the interplanetary mission design process. Further, IDFS avoids a known issue with the solution to the Lambert Problem by implementing the ESA ACT's improved Battin Method.

IDFS is an extremely useful and accurate tool that identifies optimum interplanetary mission parameters. It is easy to operate, provides flexibility to manipulate input trade parameters, offers a database on which to base input parameters, and delivers a comprehensive set of data and graphical outputs to evaluate the results. Finally, IDFS demonstrated its utility by quickly identifying a possible Earth to Mars mission.

5.1. Future Enhancements

Possible future enhancements to IDFS include:

- Aerobrake and ballistic entry software that includes a descent ΔV and shield mass calculations
- Change propellant load calculation from impulsive thrust to thrust over time so that high specific impulse/low thrust engines can be studied
- Provide the ability to calculate mission results using direct injection by the launch vehicle
- Allow LEOs other than circular so that departure burns can be performed at a transfer orbit apogee
- Calculate results for an Earth return mission

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