(Preprint) AAS 19-225

END TO END OPTIMIZATION OF A MARS HYBRID TRANSPORTATION ARCHITECTURE

Min Qu,* Raymond G. Merrill† and Patrick Chai‡

NASA's Mars Study Capability Team (MSCT) is developing a reusable Mars hybrid transportation architecture in which both chemical and solar electric propulsion systems are used in a single vehicle design to send crew and cargo to Mars. This paper presents a new integrated framework that combines Earth departure/arrival, heliocentric trajectory, Mars orbit reorientation, and vehicle sizing into a single environment and solves the entire mission from beginning to end in an effort to find a globally optimized solution for the hybrid architecture.

INTRODUCTION

NASA's Mars Study Capability Team (MSCT) is continuing its architectural trade analyses to define the capabilities and elements required to field a sustainable human Mars campaign. [1] One of the architecture options currently in development is a reusable Mars hybrid transportation architecture in which both chemical and solar electric propulsion systems are combined in a single vehicle design to send crew and cargo to Mars. By using chemical propulsion near the planetary bodies and solar electric propulsion (SEP) elsewhere, the hybrid architecture has proven to be more fuel efficient than the traditional all chemical solutions, while also presenting many design challenges as the hybrid trajectories are much more complex and highly dependent on system power and payload requirements. Finding the balance between chemical and solar electric propulsions is also nontrivial. In addition, finding an optimal Mars parking orbit for the transportation system alone is a difficult task as it involves solving a complicated geometry problem for a reorientation "apotwist" in order to complete a roundtrip mission and the solution is highly dependent on the incoming and outgoing hyperbolic excess velocity vectors (V_∞) from the heliocentric trajectories.[2,3,4] Previous analyses involved optimizing the heliocentric trajectories first, feeding the V_∞ vectors into the Mars orbit reorientation solver, and then iterating the propellant masses with the vehicle sizing tool until a mass closure is found between sizing and trajectory tools. [5,6] The solutions found from the analyses are often times less optimal globally as the heliocentric trajectories are optimized independent of the Mars reorientation, resulting in costly reorientation maneuvers and higher overall system mass. This paper presents a new integrated framework that combines Earth departure/arrival, heliocentric trajectory, Mars orbit reorientation, and vehicle sizing into a single environment and solves the entire mission from beginning to end in an effort to find a globally optimized solution for the hybrid architecture. First, an integrated mid-fidelity

* Staff Scientist, Analytical Mechanics Associates, Inc., 21 Enterprise Parkway, Suite 300, Hampton, Virginia 23666-6413, U.S.A.
† Aerospace Engineer, NASA Langley Research Center, Hampton, Virginia 23681, U.S.A. AIAA Senior Member.
‡ Aerospace Engineer, NASA Langley Research Center, Hampton, Virginia 23681, U.S.A. AIAA Member.

closure model is developed to optimize a single roundtrip Mars mission for the Hybrid transportation system. The robustness of the closure model allows large trade studies to be performed in an automated fashion for different power and payload assumptions across all mission opportunities of the entire campaign. The solution from the closure model is then used as initial guesses to build the complete high fidelity end to end (E2E) trajectory for the optimized spacecraft configuration.

THE HYBRID CLOSURE MODEL

For a crewed Mars mission, the transportation system is assumed to be launched and integrated in cis-lunar space in an orbit such as the Near Rectilinear Halo Orbit (NRHO) before crew rendezvous in a lunar distance high elliptic orbit (LDHEO) for Earth departure. Earth departure is achieved through either lunar gravity assists (LGA, powered or unpowered), or an injection maneuver at perigee from LDHEO, depending on the Earth escape energy required for the trajectory. Electric propulsion (EP) is used during the heliocentric portion of the trajectory to continuously build up the orbit energy. Upon Mars arrival, a 3-burn bi-elliptic chemical maneuver sequence is performed to capture the spaceship in a highly elliptical Mars parking orbit. During the time when the crew are visiting Mars destinations, the spaceship performs an “apotwist” maneuver to reorient itself into another elliptic parking orbit that is positioned for Earth return maneuvers. For Earth return, another 3-burn bi-elliptic maneuver sequence is performed from the elliptical orbit to send the spaceship back on an Earth return trajectory, and the electric propulsion again thrusts along the heliocentric transit until the spaceship captures itself back to the LDHEO through a LGA. The crew then return to Earth from LDHEO while the transportation system returns to the NRHO.

The Hybrid closure model is built within a trajectory optimization tool (Copernicus), which allows additional modules to be developed by the end user and incorporated as pre-compiled plugins. The center piece of the closure model is optimization of the low thrust trajectories for both the outbound and inbound transfers. Input variables being optimized in the heliocentric trajectories include departure and arrival times, departure masses, as well as departure and arrival V_∞ vectors. Major output variables are EP propellant usages during both the outbound and inbound transfers.

While Copernicus is a high fidelity tool, the heliocentric trajectories in the closure model are being modeled at a medium fidelity, without planets’ gravities, so that the closure model can run robustly without user in the loop. Chemical propellant usages and ΔVs are calculated analytically in a plugin based on the V_∞ vectors coming in and out of the planets. For Earth departure and arrival, it is assumed that a V_∞ magnitude of 1.4 km/s can be achieved with LGAs without a chemical burn, whereas V_∞ magnitude higher than 1.4 km/s can be achieved through either powered LGA or powered Earth close approach (ECA), whichever is cheaper. [7] Powered LGA can provide a slight boost in Earth departure/arrival V_∞ with a small chemical burn at perilune during a LGA, but it is not effective for large energy boost due to Moon’s distance from Earth. A direct burn from LDHEO during its Earth close approach is needed for Earth departure/arrival V_∞ greater than 2 km/s. The ΔVs for powered LGA and powered ECA can be calculated based on Energy increases from Moon’s orbit and LDHEO respectively.

Mars arrival and departure V_∞ vectors are used to calculate capture, reorient, and escape ΔVs performed by the chemical propulsion of the hybrid system. Additional input variables such as incoming and outgoing hyperbolic b-plane angles, bi-elliptic plane change angles, and “apotwist” time are introduced in the plugin to optimize reorientation of the Mars parking orbit. Outputs from the reorientation calculation include the ΔVs from all the maneuvers and the orbital elements of both arrival and departure hyperbolic, bi-elliptic transfer, and parking orbits.

Another piece of the closure model is the parametric sizing of the transportation system. System dry mass, consumables, logistics and trash dumps can be calculated based on power, payload mass, time splits, as well as chemical and EP propellant usage. All of the pieces, Earth departure and arrival, roundtrip heliocentric low thrust trajectories, Mars parking orbit reorientation, and system sizing are integrated together in a single Copernicus model and optimized simultaneously using Copernicus' built-in optimizer SNOPT to minimize the initial mass leaving NRHO. Additional constraints can be imposed on the parking orbit targeting different Mars destinations such as orbital, Phobos, Deimos or a particular landing site on the Mars surface. Table 1 shows the crewed and cargo mission opportunities for MSCT's Mars campaign between 2035 and 2048.

The solution obtained from the closure model can be used as initial guesses to build a complete high fidelity end to end trajectory of a round trip Mars mission leaving from cislunar space and eventually returning back to ciclunar space. The end to end trajectory can in turn serve to validate the mid-fidelity closure model. The components of the end to end model and a comparison between the mid and high fidelity trajectories for the 2035 crewed mission are presented in the upcoming sections.

Table 1. MSCT's Mars campaign between 2035 and 2048.

Crew Opportunity	ED_Date	MA_Date	MD_Date	EA_Date	Stay
2035	05/22/2035	04/19/2036	04/10/2037	04/19/2038	356
2039	08/05/2039	09/06/2040	07/03/2041	07/03/2042	300
2043	10/18/2043	10/16/2044	08/12/2045	09/15/2046	300
2048	01/27/2048	01/10/2049	01/07/2050	11/29/2050	362
Cargo Opportunity	ED_Date	MA_Date	MD_Date	EA_Date	Stay
2037	06/01/2037	10/04/2038	06/15/2039	05/26/2040	254
2039	08/05/2039	08/20/2040	06/27/2041	07/02/2042	311
2041	09/10/2041	12/06/2042	06/12/2043	09/06/2044	188
2043	10/18/2043	09/30/2044	06/25/2045	10/14/2046	268
2045	11/22/2045	12/22/2046	09/25/2047	12/16/2048	277
2048	01/27/2048	12/13/2048	12/16/2049	01/24/2051	368

EARTH DEPARTURE

Most solutions from the closure model have Earth departure C3 of 2 km^2/s^2 for the hybrid systems with sufficient array power as additional C3 requires chemical propulsion which is much more costly. Since C3 of 2 km^2/s^2 is achieved through LGA and the most energy boost can be obtained when the outgoing V_∞ vector from the Moon is aligned closely along the Moon's velocity direction, a constraint is imposed in the closure model to find the Earth departure date such that the Moon is in the proper location for LGA. The Earth departure portion of the high fidelity end to end trajectory needs to be constructed to target the outgoing Earth departure V_∞ vector on the departure date via LGA. Since this final condition is known, the trajectory is built backwards starting with a hyperbolic orbit escaping the Moon. Hyperbolic b-plane angle and flyby altitude at the Moon are used to target another LGA one lunar cycle earlier. This Moon to Moon transfer is called a “solar loop” that uses Sun’s gravity to perturb the orbit in order to increase the V_∞ magnitude at LGA for an increased energy at Earth departure. Even though bigger solar loops of multiple lunar cycles are more effective, a single lunar cycle is used in the 2035 mission to reduce crew time. At the second LGA, the b-plane angle and flyby altitude at the Moon are solved and the trajectory is back-propagated to target a LDHEO with an inclination of 28.5 degrees and perigee altitude of 400 km that is suitable for crew rendezvous. The spaceship is staying in the LDHEO for 3 lunar cycles (82 days) to allow multiple launch opportunities for the crew to rendezvous with the hybrid system, and this turns out to be the most challenging part of Earth departure trajectory to construct. A small burn at perigee is needed to phase the LDHEO to not only avoid large perturbation from the Moon during the 3 lunar cycles, but also to target another LGA at the end of the LDHEO stay. A small station-keeping burn at apogee sometime during the LDHEO stay is also necessary to keep the spacecraft from getting too close to the Earth surface. After the third LGA, the backward propagated trajectory is designed to go through a weak stability boundary (WSB) transfer and reach the NRHO, and this is achieved by finding an orbit in the unstable manifold of the NRHO and patching it up with the orbit coming out of the third LGA. A SEP thrust leg is inserted to patch up the two orbits and complete the transfer. Once the entire Earth departure maneuver sequence is built, the trajectory is then optimized to minimize the initial mass leaving NRHO. One of the main parameters that determines the shape of the weak stability boundary transfer orbit is the angle between the Sun Earth line and the Earth Moon line. For maximized Earth departure C3 via LGA, this angle is always small and has similar values at the final Earth escape LGA point across different mission opportunities, and therefore the same Earth departure sequence geometry can apply to different mission opportunities as long as the concept of operations is the same.

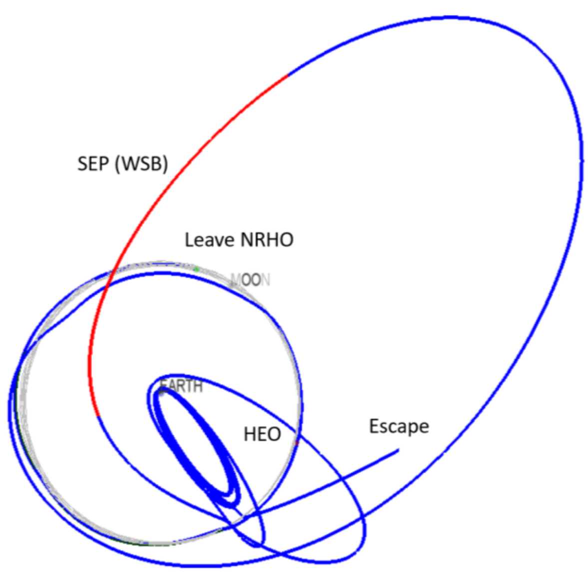

Figure 1. Earth departure sequence from the end to end trajectory.

Figure 1 shows the high fidelity Earth departure trajectory of the 2035 mission. The spacecraft leaves NRHO on 9/29/2034 with a chemical burn of 4 m/s to get into a weak stability boundary transfer orbit. A 12-day SEP burn totaling 146 m/s takes the spacecraft to the first LGA on 2/2/2035 before getting down to the LDHEO. The spacecraft stays in in LDHEO for 9 revs (82 days) where 3 phasing and maintenance maneuvers totaling 84 m/s are performed using chemical propulsion. The spacecraft leaves LDHEO via LGA on 4/25/2035 to get on a solar loop orbit before hitting the final LGA on 5/23/2035 for Earth escape.

MARS STAY

While the 2035 mission in the Mars campaign developed by MSCT is an orbital mission, the trajectory the transportation system is designed to fly is as if it is a surface mission (minus the landing and ascent). The planned landing site for the subsequent missions in the campaign is Jezero crater (18.8ºN, 77.5ºE) and it is required that the periapse of the Mars parking orbit must be directly above the landing site (for optimal landing) and the inclination of the parking orbit be the same as the landing site latitude (for due-east ascent) before Mars departure. The incoming and outgoing hyperbolic orbits, bi-elliptic transfer orbits and parking orbits have been solved by the mid fidelity closure model and the orbital elements of these orbits can be entered directly into the high fidelity trajectory model as initial guesses. The Mars arrival time must be re-adjusted within 1 day of the mid fidelity solution such that the right ascension of the arrival parking orbit periapse matches the longitude of the landing site. The parking orbit is propagated forward and a maintenance maneuver is added near apoapse of the parking orbit to keep the spacecraft from getting too close to the Mars surface. The "apotwist" maneuver is not needed in the 2035 trajectory since the maneuver ΔV is zero in the mid fidelity solution and that is the case for many of the solutions because often the plane changes can be pushed to either the arrival or departure bi-elliptic maneuver sequences where the maneuvers are performed at higher altitudes. The final Mars departure time is re-adjusted to account for the phasing of the parking orbit, and the entire trajectory for the Mars stay is optimized while holding the incoming and outgoing V_∞ vectors to be the same as those from the closure model.

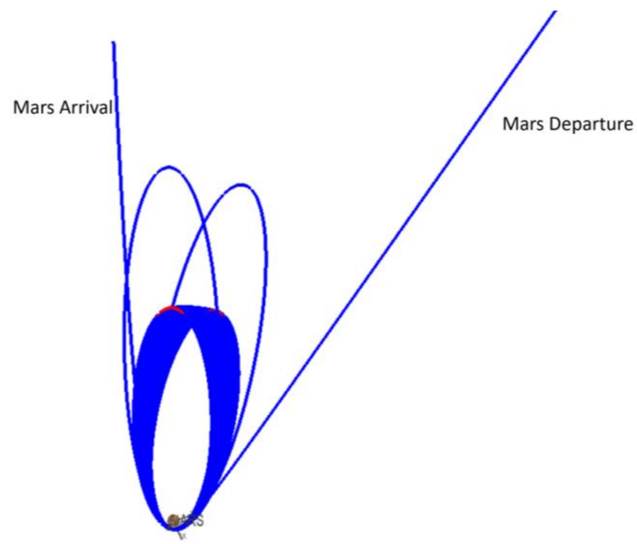

Figure 2. Trajectory during Mars stay for the 2035 crewed mission.

Figure 2 shows the high fidelity trajectory of the Mars stay for the 2035 mission. The spaceship arrives Mars on 4/20/2036 and a 94 m/s chemical burn at 250 km altitude takes the spaceship to a 10-sol bi-elliptic transfer orbit and returns to the periapse for the Mars parking orbit insertion. The total ΔV for the bi-elliptic arrival is 122 m/s performed by the chemical propulsion and the spacecraft is in 5-sol parking orbit with its periapse directly above the Jezero crater. Using the electric propulsion, a 1-day maintenance maneuver near apoapse totaling 11 m/s keeps the parking orbit at a safe distance from the Mars surface for the entire Mars stay. Near the end of the Mars stay, another SEP maneuver at apoapse lowers the periapse altitude down to 250 km for the final bi-elliptic departure. A 32 m/s burn raises the orbit to 10-sol for the bi-elliptic departure and the final Mars departure is on 4/11/2037 and the total bi-elliptic maneuver ΔV is 127 m/s using the chemical propulsion.

EARTH ARRIVAL

The Earth arrival maneuver sequence is similar to the Earth departure sequence but in reverse order. The Earth arrival date and incoming V_∞ vector for the 2035 mission are first solved for by the closure model to achieve the Earth arrival C3 of 2 km^2/s^2 and the Moon's location is properly aligned for unpowered LGA. The high fidelity trajectory is built starting with a hyperbolic orbit arrival at the Moon, and by varying the incoming B-plane angle and flyby altitude the spacecraft goes to a solar loop and then targets a second LGA before dropping down into a LDHEO and rendezvous with a crewed capsule already in orbit. While the crew gets off the hybrid system and returns to Earth with the capsule, the hybrid system must perform phasing and maintenance maneuvers and target a third LGA. While the spacecraft could potentially get on a WSB transfer to return to NRHO after the third LGA, a more fuel efficient trajectory is found by going through a backflip orbit to the other side of Earth for a fourth LGA before getting onto the WSB transfer. An orbit in the NRHO's stable manifold is found and a SEP thrust leg is added to patch up with the WSB transfer orbit to complete the last portion of the Earth arrival trajectory.

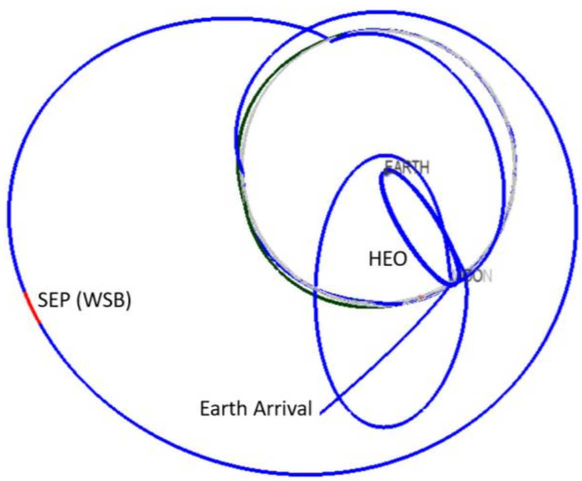

Figure 3. Earth arrival sequence from the end to end trajectory.

Figure 3 shows the Earth arrival sequence of the 2035 mission. The spacecraft arrives at the Moon on 4/19/2038 for a LGA to get on the solar loop before coming back for the second LGA on 5/16/2038. The spacecraft drops into a 28.5-degree inclined LDHEO after the second LGA and stays for one lunar cycle. The spacecraft performs 3 phasing and maintenance maneuvers totaling 56 m/s and then reaches the Moon for the third LGA on 6/12/2038. The backflip orbit after the third LGA takes the spacecraft out of the Earth Moon plane and reaches the Moon half a month later on 6/25/2038. After the 4th LGA the spacecraft goes into the WSB transfer orbit in which it performs a 2.5-day SEP burn starting on 7/26/2038 totaling 54 m/s before arriving at NRHO on 10/15/2038. The final NRHO insertion burn is 0.99 m/s.

CONNECTING THE LEGS

While the mid fidelity closure model is run in a single Copernicus model to ensure a globally optimized solution, the high fidelity trajectory is built leg by leg in different Copernicus models due to the complexity and sensitivity of the maneuvers in each leg. State vectors and masses are passed between the legs and some legs need to be re-converged to ensure a complete continuous trajectory from the beginning to the end. After the Earth departure and Mars stay legs are built, the state vectors at the last LGA and Mars arrival are used as initial and final conditions for the outbound heliocentric leg in which low thrust burns and engine switching are optimized. To obtain extra performance, instead of holding all six state variables fixed at the LGA departure, the hyperbolic B-plane angle towards the Moon is allowed to be free which adds an extra degree of freedom in the Heliocentric leg that may cause a very slight change in the Earth departure leg. Earth departure leg is then re-converged easily with the updated final state vectors. The final mass from the outbound heliocentric leg is passed to the Mars stay leg to ensure mass continuation and the leg is then re-converged. Similar to the outbound leg, the state vectors for Mars arrival and Earth arrival LGA are used as initial and final conditions for the inbound leg trajectory optimization. Finally the final mass from the inbound leg is passed to the Earth arrival leg and the trajectory is then re-converged to complete the construction of the end to end trajectory.

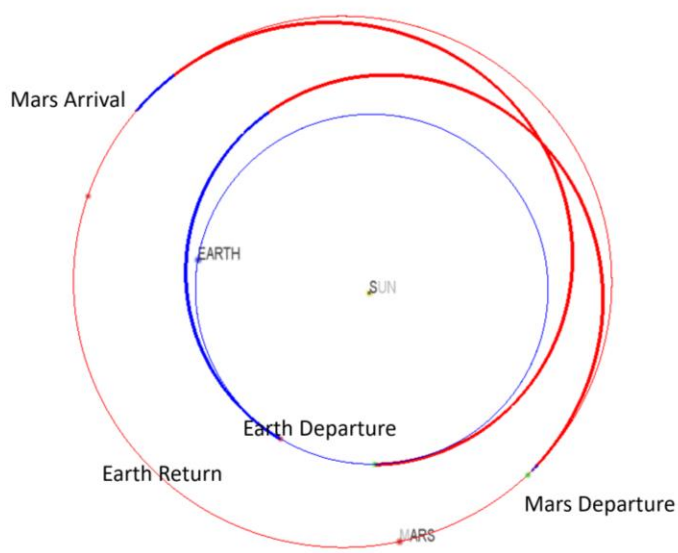

Figure 4. Heliocentric legs of the 2035 mission.

Figure 4 shows the heliocentric legs of the 2035 mission. The spacecraft leaves the Moon on 5/23/2035 and the SEP system starts to thrust immediately and continues to thrust for the majority of the outbound trip. The spacecraft arrives Mars on 4/20/2036 after a short coast and the total outbound ΔV performed by the SEP system is 3615 m/s. The spacecraft leaves Mars on 4/11/2037 and with a much lighter vehicle it has a much longer coast on the way back and arrives at the Moon for LGA on 4/19/2038. The total ΔV performed by the SEP system during the inbound trip is 3834 m/s.

VALIDATION

The high fidelity end to end trajectory can be used to validate the mid fidelity closure model which is used to perform thousands of cases of trade and sensitivity analyses for MSCT. Some assumptions and simplifications are made in the closure model to ensure the entire mission is globally optimized within a single model, and can be run in an automated fashion without user interaction. At the same time the closure model must have enough fidelity and details to realistically represent how a spacecraft would fly a mission from beginning to the end. Table 2 shows the time and mass comparisons between the mid and high fidelity models at critical events for the 2035 mission. Notice that the Earth departure mass in the high fidelity model is set to match the mid fidelity model with the Earth departure leg propagated backwards and all other legs propagated forward. All dates at critical events are within 1 day of each other and the masses match very well.

Table 3 shows the ΔV comparisons between mid and high fidelity models. Since the WSB transfers from NRHO to LDHEO are not modeled in the mid fidelity model, a total SEP ΔV budget of 75 m/s is assumed based on past studies. The actual WSB transfer ΔV is 145 m/s for Earth departure and 54 m/s for Earth return. The 145 m/s is significantly higher than the 75 m/s budget, but it represents a very small increase for the entire SEP ΔV budget, and it is very likely that a better solution may exist. A 90 m/s LDHEO maintenance ΔV budget is assumed in the mid fidelity model, and the actual maintenance ΔVs are 84 m/s for Earth departure and 56 m/s for Earth return. The heliocentric SEP ΔVs and chemical Vs at Mars also match very well between the two models.

Table 2. Comparison at critical events between mid and high fidelity models.

2035 mission	Closure model (Mid fidelity)		End to end model (High fidelity)	
Event	Date	Mass (t)	Date	Mass (t)
Leave NRHO		114.76	9/29/2034	115.19
Earth Departure	5/22/2035	111.67	5/23/2035	111.67
Mars Arrival	4/19/2036	95.87	4/19/2036	96.43
Mars Departure	4/10/2037	78.72	4/10/2037	78.85
Earth Arrival	4/19/2038	67.53	4/18/2038	67.83
NRHO Insert		59.15	10/12/2038	61.06

Table 3. Maneuver Comparisons between mid and high fidelity models.

Event	Closure (m/s)	E2E (m/s)
Leave NRHO	0.00	4.40
WSB Maneuver	75.00	145.56
HEO Maintenance	90.00	84.29
Outbound SEP	3624.05	3615.80
MOI	121.31	122.04
MPO maintenance	20.00	21.63
TEI	121.37	127.21
Inbound SEP	3642.58	3834.09

HEO Maintenance	90.00	56.76
WSB	75.00	54.30
Enter NRHO	0.00	0.99

LOW THRUST APOTWIST

Various trade studies have been performed by MSCT to look at impacts of system power, thruster power, and payload on the overall mass of the hybrid system. Trades were also performed on Mars landing site latitudes and it is apparent that higher latitude landing sites require more propellant and/or higher power for the roundtrip mission. This is due to the fact that the Mars parking orbits must be of higher inclinations to satisfy the descent and ascent requirements of the higher latitude landings sites, resulting in costly reorientation maneuvers which are performed by the chemical propulsion. In an effort to reduce the overall system mass, MSCT investigated the feasibility of using the low thrust propulsion to perform the reorientation maneuvers during Mars stay and it was found that, despite of the operational complexity, the Apotwist maneuver can indeed be carried out by the low thrust engines using a series of 1-day plane change burns near apoapse to reorient the parking orbit to the desired state for Earth return.

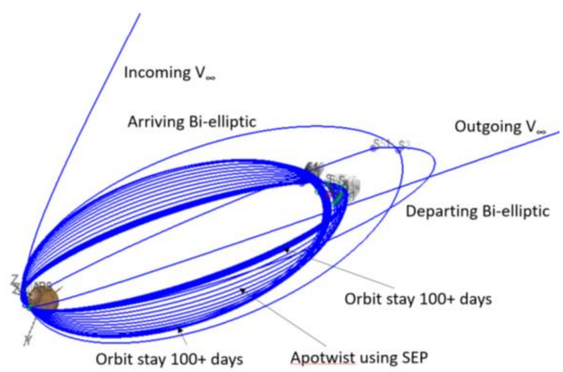

Figure 5. Apotwist using SEP in 2039 mission.

Table 4. Comparisons for Mars orbit Reorientation for 2039 mission.

Event	Closure	E2E
Mars Orbit Insertion (m/s)	367.72	368.16
MPO Maintenance (m/s)	20	4.29
Apotwist (m/s)	135.27	127.11
PO Maintenance (m/s)	none	1.02
Trans-Earth Injection (m/s)	279.35	281.16

Figure 5 shows a 2039 Mars surface mission targeting a 50º landing site latitude. After a bi-elliptic maneuver sequence using chemical propulsion to capture itself into a 5-sol parking orbit that passes over the landing site (for the descent stage), the spacecraft performs a 4 m/s maintenance burn near apoapse to keep it away from the Mars surface. 106 days after the Mars orbit insertion, the spacecraft starts the apotwist maneuver sequence in the next 11 revs by performing 1-

day burns per rev near apoapse that averages 4º plane changes per rev. In each rev during the Apotwist phase, the SEP thrust direction is optimized to maximize the plane change and at the same time target a 250 km altitude at the periapse passage. The inclination of the parking changes from 63º to 50º (for due-east ascent) after the 11 revs. The spacecraft later performs another 1 m/s maintenance maneuver to lower the periapse to 250 km before the final bi-elliptic maneuver sequence for Earth return. Table 4 shows the general agreement between the mid-fidelity closure model and the high fidelity E2E trajectories for the 2039 mission targeting 50º landing site latitude. By switching from chemical to SEP propulsion for the apotwist maneuver, the Earth departure mass is reduced from 126.9 metric tons to 124.5 metric tons targeting a 40º landing site latitude. And with the same power and payload assumptions, the trajectory could not close with chemical propulsion performing apotwist targeting 50º landing site latitude but is able to close after switching to SEP performing the apotwist maneuver.

CONCLUSION

NASA's Mars hybrid architecture is a promising concept that combines chemical and solar electric propulsions in a single vehicle design. An integrated framework has been developed and it solves Earth departure and arrival, roundtrip heliocentric low thrust trajectories, Mars parking orbit reorientation, and system sizing in a single Copernicus model to find a globally optimized solution for the hybrid system. The mid-fidelity closure model provides a fast and robust solution that can be used to run through a large set of trade and sensitivity studies to analyze the impacts of power, payload, and landing sites on the overall design of the system, and the solution is then used as basis to build detailed, high fidelity end to end trajectories that leaves from cislunar space, reaches Mars destinations and eventually returns back to cislunar space.

REFERENCES

[1] D. A. Craig, N. B. Herrmann, and P. A. Troutman, "The Evolvable Mars Campaign - Study Status," IEEE Aerospace Conference, March 2015. 2015-8.0101.

[2] D. Landau, J. Longuski, and P. Penzo, "Method for Parking Orbit Reorientation for Human Missions to Mars," Journal of Spacecraft and Rockets, Vol. 42, No. 3, pp. 517–522, 2005.

[3] D. Landau, "Orbital Transfer Techniques for Round-Trip Mars Missions," Paper AAS13-449, AAS/AIAA Spaceflight Mechanics Meeting, Kauai, Hawaii, 2013.

[4] M. Qu, R. G. Merrill, P. R. Chai, and D. R. Komar, "Optimizing Parking Orbits for Roundtrip Mars Missions," AAS Space Flight Mechanics Meeting, 2017.

[5] R.G. Merrill, P. Chai, and M. Qu, "An Integrated Hybrid Transportation Architecture for Human Mars Expeditions," 2015 AIAA SPACE Conference, Pasadena, CA, 2015, AIAA 2015-4442.

[6] P. Chai, R.G. Merrill, and M. Qu, "Mars Hybrid Propulsion System Trajectory Analysis Part I: Crew Missions," 2015 AIAA SPACE Conference, Pasadena, CA, 2015.

[7] M. Qu, R.G. Merrill, P. Chai, and D.R. Komar, "Trajectory Designs for a Mars Hybrid Transportation Architecture," AAS Paper 15-522, 2015 AAS/AIAA Astrodynamics Specialist Conference, Vail, CO, August 2015.