IAC-02-Q.6.08

SERVICING AND DEPLOYMENT OF NATIONAL RESOURCES IN SUN-EARTH LIBRATION POINT ORBITS

D. C. Folta, M. Beckman, S. J. Leete, G. C. Marr, M. Mesarch, and S. Cooley

NASA Goddard Space Flight Center
Greenbelt, MD, USA

53th International Astronautical Congress
The World Space Congress – 2002
10-19 Oct 2002 / Houston, Texas

SERVICING AND DEPLOYMENT OF NATIONAL RESOURCES IN SUN-EARTH LIBRATION POINT ORBITS

D. C. Folta, M. Beckman, G. C. Marr, M. Mesarch, S. Cooley
Flight Dynamics Analysis Branch
NASA Goddard Space Flight Center, Greenbelt, MD, USA
dfolta@gsfc.nasa.gov

S. J. Leete
Space Science Missions Branch,
NASA Goddard Space Flight Center, Greenbelt, MD, USA

Abstract

Spacecraft travel between the Sun-Earth system, the Earth-Moon system, and beyond has received extensive attention recently. The existence of a connection between unstable regions enables mission designers to envision scenarios of multiple spacecraft traveling cheaply from system to system, rendezvousing, servicing, and refueling along the way. This paper presents examples of transfers between the Sun-Earth and Earth-Moon systems using a true ephemeris and perturbation model. It shows the ΔV costs associated with these transfers, including the costs to reach the staging region from the Earth. It explores both impulsive and low thrust transfer trajectories. Additionally, analysis that looks specifically at the use of nuclear power in libration point orbits and the issues associated with them such as inadvertent Earth return is addressed. Statistical analysis of Earth returns and the design of biased orbits to prevent any possible return are discussed. Lastly, the idea of rendezvous between spacecraft in libration point orbits using impulsive maneuvers is addressed.

Introduction

Satellite servicing has received a great deal of study and significant execution. Several satellites were designed for servicing using the Multi-Mission Modular Spacecraft design, including Solar Max Mission, Landsat IV & V, Upper Atmosphere Research Satellite, and Extreme Ultra-Violet Explorer. Rescue missions have been performed on geostationary satellites trapped in low earth orbit, such as WESTAR-IV, PALAPA-B, Intelsat-VI, and LEASAT/SYNCOM-IV. More routine human servicing work occurs(ed) at various space stations (International Space Station, Skylab, Salyut, and Mir). The servicing of the Hubble Space Telescope (HST) has become a successful landmark, allowing HST to become one of NASA's most productive missions. [1,2,3,4]

Some obstacles to human servicing are the orbital mechanics, the cost of lifting mass, and the problems associated with travel time and thermal and instrument environmental conditions. As more ambitious missions are planned, such as the placement of the Next Generation Space Telescope (NGST) and several other missions into a Sun-Earth L_2 (SEL_2) or SEL_1 libration orbit, servicing by the Shuttle and the use of low-Earth orbits (LEOs) will be limited. Development of robotic satellite servicing capabilities, such as DARPA's Orbital Express, NASA's Robonaut, and the University of Maryland's Ranger, may provide for the possibility of robotic satellite servicing at various orbital locations in the near- or mid-range time frame.[5]

An enabling set of circumstances for an expansion of satellite servicing would be the placement of humans and valuable robotic assets in close proximity to one another. A space architecture that includes these conditions is a servicing facility in a lissajous or halo orbit about one of the Earth-Moon L_1 (EML_1), Earth-Moon L_2 (EML_2), or Earth-Moon L_3 (EML_3) co-linear libration points[6]. From such an orbit, spacecraft have access to a wide variety of interesting orbits at a relatively lower ΔV cost. Use of the Earth-Moon stable L_4 and L_5 Lagrange regions provides additional scenarios. These servicing locations are also an excellent staging point for lunar surface and Earth-Moon orbital exploration.[7] The orbits also have ready access to geostationary orbits and transfer back to LEO orbits. Table 1 provides a brief overview of Earth-Moon libration orbit staging node characteristics.

Table 1. Earth/Moon Libration Orbit Overview

	Characteristics	***Potential Uses***
EML_1	-Unstable orbit between earth and moon -Halo:continuous view of the earth, moon, and sun, -Earth / moon always >150^0 separation	-Assembly & maintenance of s/c -Access to lunar surface -Low Δv access to SEL_n transfer manifolds
EML_2	-Unstable orbit far side of moon -Halo: continuous view of earth, moon, sun, -At full moon, sun/earth/moon/ in alignment	-Assembly & maintenance of s/c -Access to lunar surface -Comm relay to lunar far side, -Low Δv access to SEL_n transfer manifolds
EML_3	-Unstable orbit opposite moon -Halo: continuous view of earth, sun, -At new moon, sun/earth/moon/ in alignment	-Assembly & maintenance of s/c that are thermally sensitive -Low Δv access to SEL_n transfer manifolds
$EML_{4/5}$	-Stable orbit, low stationkeeping Δv, -Varying sun/earth/moon geometry	-Minimum fuel loads -Moderate transfer to SEL_n

The servicing of national resources in the Earth-Moon and the Sun-Earth regions is made difficult by the unstable dynamics of these regions. Trip times associated with achieving SEL_1 and SEL_2 orbits can vary from weeks to months while achieving EML_1 and EML_2 may vary from days to weeks, with both dependent upon many initial conditions such as energy and departure locations. The design of a trajectory used to achieve servicing mission requirements is intricate but can be facilitated by the use of unique orbits to achieve the proper flight time or to minimize the necessary fuel mass. The use of dynamical systems (such as invariant manifolds) and optimization plays a key role in designing transfer trajectories, as these tools afford the luxury of using natural dynamics where possible. Results demonstrate the viability of some unique transfers that meet the NASA Exploration Team (NEXT) goals. These goals focus on opening the human frontier beyond low-Earth orbit by building infrastructure robotically and with humans in-situ at strategic outposts - libration points, planetary moons, planets, etc. - and include a wide range of exploration tools (e.g., space planes, balloons, human-constructed and maintained observatories located at libration point, etc.).[8]

A primary purpose of this paper is to detail mission planning scenarios for servicing of national resources and to contrast them to other transfer options. It explores the transfer from the Earth-Moon region to Sun-Earth libration orbits. The ΔV and trip time associated with such a mission is compared to a more traditional direct injection and the relevant differences highlighted. Analysis recently completed for NGST orbit trades are used where possible.[9] The servicing options vary over a wide range, from low thrust direct transfers to transfers from elliptical orbits achieved using bi-propellant systems. This paper covers only several of the many topics analyzed, but it represents a general investigation of possibilities. The results of the analysis presented herein will be available to aid in the support of evaluating in-space servicing options for mission studies in the GSFC Integrated Mission Design Center.

Software Applications and Assumptions

For this analysis, software applications that use high fidelity perturbation modeling and full ephemeris data were utilized. The models utilized include as a minimum:

- Earth potential, 4x4 or greater
- Point mass bodies based on full ephemeris (DE405)
- Solar radiation pressure
- Jacchia-Roberts atmospheric drag
- Spacecraft area and mass characteristics
- Spacecraft engine mass depletion and accelerations
- Runge-Kutta 8/9 integrator
- Inertial and rotating coordinate systems
- Differential correctors and optimization methods

The software used consists of GSFC's Swingby and Generator tools and MATLAB. Swingby has been used operationally by GSFC to support four libration and two lunar missions. In addition to using Swingby to model and target the solution set, software that allows robust modeling and optimization of the Sun-Earth and Earth-Moon dynamics is used. This utility allows the generation of the full phase space while including full perturbation models. The Generator code developed at Purdue University was used to compute invariant manifolds for EML_2 to SEL_n transfers. MATLAB using GSFC developed m-files and a multi-level iteration scheme were used to optimize these transfer trajectories.[10]

Transfers Between The Earth-Moon And Sun-Earth Libration Orbits

Transfers between the Earth-Moon and Sun-Earth libration points have been previously investigated in varying consideration and detail.[11,12,13,14,15] This work builds upon 25 years of GSFC libration orbit experience and analysis and new optimal transfer studies. A servicing scenario using co-linear Earth-Moon libration orbits was investigated first. This scenario allows a fast transfer of human and expendable resources for deployment or servicing at the EML_2 vicinity, with a round trip servicing time of approximately a week. The transfers presented here demonstrate feasibility and are not fully optimized.

End-To-End Transfer Scenarios

The following scenarios demonstrate a cis-lunar transfer into an EML_1 or EML_2 orbit where a human deployment or servicing of a national resource can occur. A subsequent transfer to and a return from the SEL_1 via EML_2 is then designed that would provide for a second servicing opportunity. The initial cis-lunar trajectory is modeled as injection from a 28.5^0 inclined 186-km circular parking orbit based on expendable launch vehicle parameters. To obtain the trajectories in this scenario a combination of targeting goals were used including: lunar B-plane components, rotating coordinate system position and velocities, orbital energy, and epochs and propagation duration. To achieve targets, deterministic ΔVs were varied at the parking orbit injection, orbit insertion, and trajectory mid-points.

Figures 1 and 2 present a transfer from LEO to EML_1, transfer to EML_2, transfer to SEL_1, and back to EML_2. The cis-lunar trajectory as shown in Figure 1 is in an earth-moon rotating coordinate system. This trajectory transfers the spacecraft from LEO orbit injection to EML_1 orbit insertion. From there, after EML_1 insertion and any stationkeeping, a small maneuver is required to achieve the EML_2 orbit. As EML_2 is a co-linear unstable location, it is straightforward to achieve a transfer trajectory that places the spacecraft on a departing invariant manifold. Figure 2 shows the EML_2 to SEL_1 transfer in a sun-earth rotating coordinate system. Following insertion into the SEL_1 lissajous orbit, one orbit revolution about the SEL_1 point was used. A small maneuver then places the spacecraft on an inbound trajectory that results in a re-capture into the EM system. The computation of the initial transfer orbit was not constrained to achieve any SEL_1 lissajous orbit conditions that are advantageous to any return trajectory. In fact, the transfer and final SEL_1 orbit achieved may be considered a worst-case scenario since they were unconstrained and resulted in the maximum offset in timing for a return trajectory. The ΔVs and trip times associated with this transfer scenario are presented in Table 2.

As shown in Figures 3 and 4, a second return transfer was also achieved using the same initial LEO to SEL_1 outbound trajectory. The original outbound transfer results in a timing differential for a return lunar encounter of 14 days. Without a phase jump in the SEL_1 orbit, the return trajectory achieves an EML_3 intersection. Using the dynamics of the Earth-Moon region, a differential correction scheme can be used to compute a maneuver that allows a transfer to an orbit in the Earth-Moon region to achieve the EML_2 orbit. Figure 3 shows the transfer orbit in the Earth-Moon region while Figure 4 shows the continuous path from SEL_1 back to the Earth-Moon region.

Figures 5 and 6 show a similar transfer from or to the EML1 region without a phase jump. These return transfers demonstrate that a wide variety of transfers are available for return servicing missions. Segments of this full scenario can be used to size the requirements for a returning spcaecraft from SEL_1 and a rendezvous from Earth at the EML_2 orbit. The impact of injection from an ISS-like inclination is not significant. A similar transfer to EML_1 using a higer inclination can be computed by timing the alignment of the outgoing velocity asymptote and the addition of a small deterministic maneuver. The maneuver ΔV required for insertion into the EML_1 or EML_2 libration orbit from the incoming SEL_1 libration orbit varies as a function of lissajous amplitude and the energy of the incoming transfer trajectory. For EML_1 and EML_3 insertions, ΔVs on the order of 50 to 300 m/s were observed. A typical maneuver insertion for EML_2 was 70 m/s. In all the above scenarios, the injection energy to reach EML_1 orbit from LEO remained constant at -2.13 km^2/s^2 (ΔV ~3.14 km/s), similar in magnitude to a lunar mission such as Lunar Prospector.

Adjusting the Amplitude of the $SEL_{1/2}$ Orbit From EML_2

Achieving the above SEL_1 lissajous orbit was not dependent upon any mission constraints (such as y-amplitude). The size and shape of the orbit was unconstrained in order to show the feasibility of such transfers. The orientation and amplitude of the orbit will vary depending upon mission requirements and natural dynamics. Analysis was performed to determine the ΔV and impacts to temporal or spatial conditions for control of the lissajous characteristics in order to rendezvous with resources already in SEL_N libration orbits.

Reducing the lissajous y-amplitude

Trajectory optimization is a broad and

Table 2. Sample ΔV and Trip Times For Full Scenario

Events / Parameters	EML_1 Insertion at X-axis Crossing	Transfer to EML_2	SEL_1 Injection and Transfer	Insertion into SEL_1 and 1 Orbit	Transfer from SEL_1 Orbit to EML_2	Capture into $EML_{1/2/3}$
ΔV (m/s)	423	0.2	None	None	32 to EML_1 100 to EML_2	50-293
Flight Time Segment and *Cumulative* (days)	5 *5*	9 *29 (includes 1 orbit)*	86 *115*	185 (includes 1 orbit) *300*	78 *378*	--- *378*
Libration Amplitude (km)	X ~ 27000 Y ~ 70000 (EML_1orbit)	X ~ 27000 Y ~ 70000 (EML_2orbit)	---	X ~ 300000 Y ~ 800000 (SEL_1orbit)	---	X ~ 30000 Y ~ 70000 (EML_2 orbit)

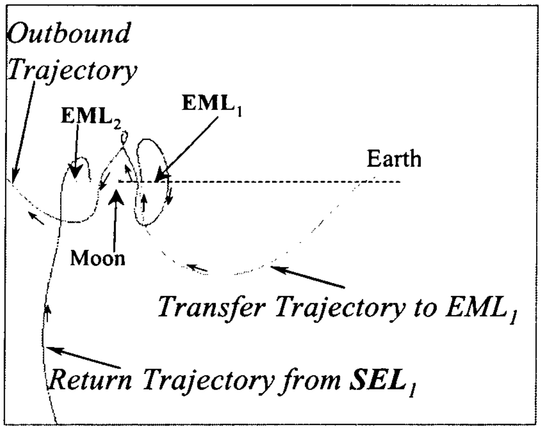

Figure 1. Transfer to/from SEL_1 via $EML_{1/2}$ in Earth Moon Rotating System

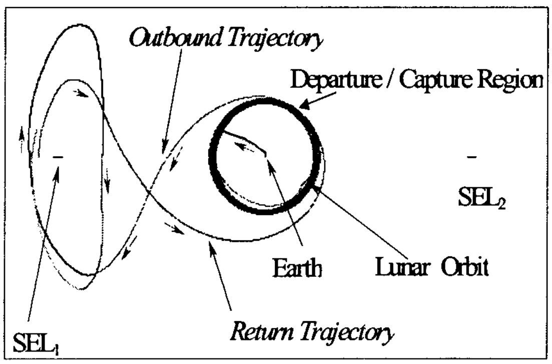

Figure 2. Transfer to/from SEL_1 in Sun Earth Rotating System

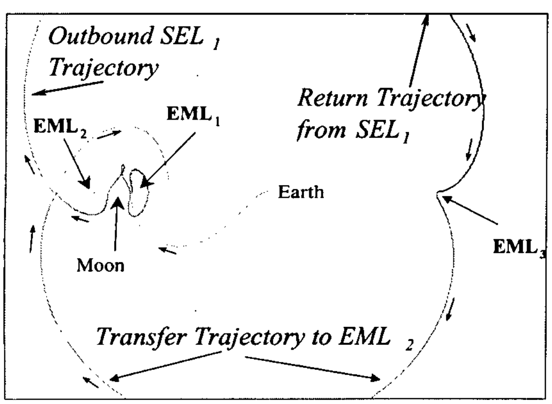

Figure 3. Transfer to/from SEL_1 via EML_3 in Earth Moon Rotating System

complicated subject that has been studied extensively in the aerospace literature.[16] Nevertheless, fundamentally any path can be discretized as a set of patch states that include time, position and velocity. At any given patch state, a before and after (-/+) maneuver state is included. In this context, impulsive

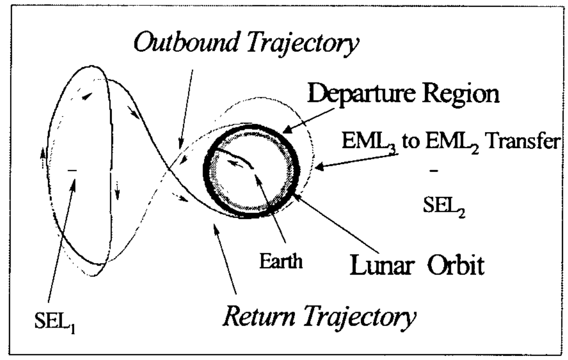

Figure 4. Transfer to/from SEL_1 via EML_3 in Sun Earth Rotating System

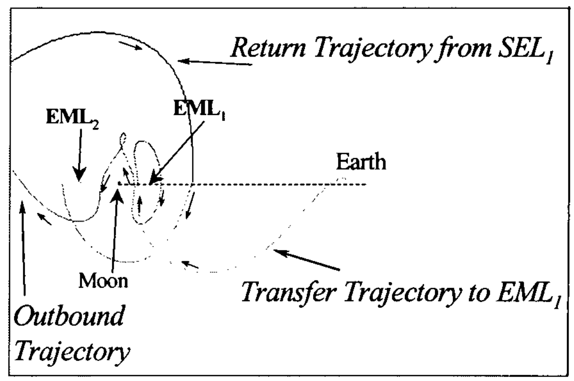

Figure 5. Transfer to/from EML_1 in Earth Moon Rotating System

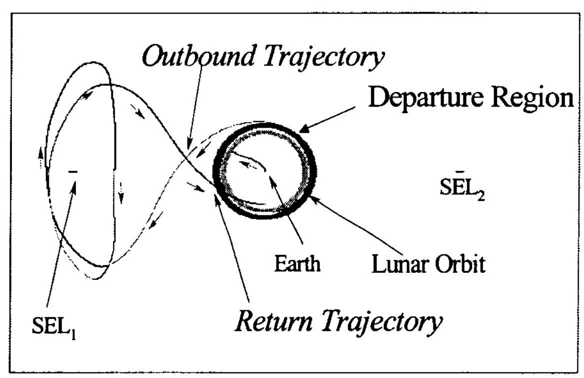

Figure 6. Transfer to/from SEL_1 via EML_1 in Sun Earth Rotating System

maneuvers — at any of the patch states — are modeled by fixing time and position while letting the velocity vary instantaneously. Once an initial guess is provided and carefully discretized, the first level of the Howell-Pernicka two level iteration scheme can be used to achieve position continuity.[17] That is, the patch states are connected in a (numerical) Lambert type of scheme. The result of this operation is the computation of the total ΔV required for the path. This total ΔV becomes the cost function. If one or more components (of one or more patch states) are allowed to vary, the total ΔV required for the path changes. As a result, a trajectory optimization scheme could vary the patch states and keep track of the total ΔV that is needed to "connect the points". This essentially allows the use of any optimization strategy.

As an example, the MATLAB optimization toolbox using the *fmincon* function is utilized. A trajectory from the vicinity of the EML_2 point to a lissajous orbit of the SEL_2 point is examined. The ephemerides model is utilized with the Earth as the central body (with J2) and with the Sun and the Moon as point mass. Solar radiation pressure is also included. An initial estimate, with a total ΔV of about 700 m/s, is computed by numerical experimentation. This initial estimate is discretized into four patch states. The patch state's positions and times are selected as independent variables. When the optimizer converges, the total ΔV decreased to 180 m/s. As a comparison to the above solution of departure from the EML_2 point, two lissajous orbits with the same epoch where constructed that yield a transfer to two selected SEL_1 y-amplitude orbits. These lissajous characteristics were chosen based on the full transfer scenario. The control of the SEL_1 y-amplitude is dependent upon the selected transfer manifold and the initial conditions at the EML_2 libration orbit. Unstable manifolds from the SEL_1 orbit back to the vicinity of the Moon at a given epoch can be constructed.[10] Also, unstable manifolds of the dynamics of the Earth-Moon region can be constructed. The intersection of these two manifolds represents a locus of transfer points.

As shown in Figure 7, a smaller SEL_1 y-amplitude lissajous orbit can be achieved by using a slightly different initial EML_2 departure condition that inserts onto a different outbound unstable manifold. Table 3 provides sample ΔVs and trip times for these transfers. The transfer trajectories to SEL_1 depart at the same epoch but have slightly different initial EML_2 states and energy. A deterministic ΔV performed several days after departure on the SEL_1 lissajous transfer manifold is required to acquire a smaller SEL_1 lissajous. Also shown in Figure 7 is a transfer into a small SEL_2 orbit. This transfer was achieved using an optimization process. The departure location was chosen to be the EML_2 libration point itself. Selection of the departing conditions as part of the overall mission design allows one to achieve any SEL_2 lissajous orbit.

Table 3. EML_2 Departure Characteristics and ΔVs Transfer

Achieved Lissajous y-Amplitudes (km)	Starting EML_2 Lissajous Amplitudes (km)	Required ΔV (m/s) and Departure C3 (km_2/s^2 - Moon centered)	Trip Time to first SEL_1 x-axis crossing (days)
SEL_1 Y~900,000	X: 20000 Y: 75000	ΔV: 0.18 C3: 0.0002	115
SEL_1 Y~700,000	X: 2000 Y: 6000	ΔV: 16 C3: -.05	88
SEL_1 Y~450,000	X: 2000 Y: 5000	ΔV: 127 C3: 0.03	82
SEL_2 Y~200,000	X: 0 Y: 0	ΔV: 180 C3: 0.75	120

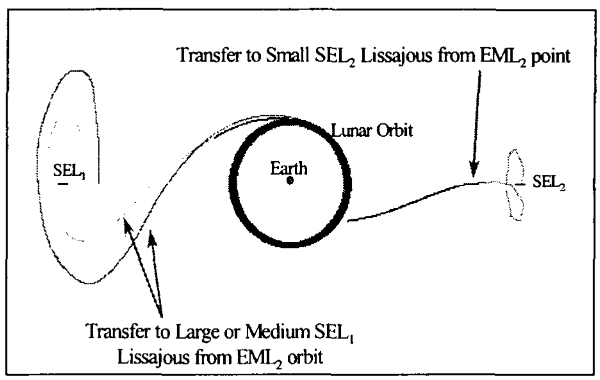

Figure 7. Changing the $SEL_{1/2}$ Lissajous Y-Amplitude

Low Thrust Options

Another option for servicing and deployment is to utilize low thrust propulsion. Low thrust propulsion has the benefit of being more efficient than an impulsive type system. A transfer from LEO into either an EML_1 orbit or into a SEL_2 orbit was analyzed. As compared to the ΔV and time associated with a high-energy direct transfer, low thrust obviously takes much longer. The amount of available mass to final EML_1 orbit is increased but is limited by the launch capability. Figure 8 shows a low thrust trajectory to the EML_1 in an Earth-Moon Rotating Coordinate system. An example of a low thrust trajectory from Earth parking orbit to the SEL_2 orbit in a Sun-Earth rotating coordinate system is shown in Figure 9. The thrust direction is along the velocity vector. While usually called constant thrust, the trajectory includes coasting arcs that are used to 'target' to an unstable manifold, which places the spacecraft on a trajectory that will achieve a lissajous orbit. Little additional thrust is required after lunar orbit distance; the majority of the post-lunar trajectory in Figure 9 is a coast phase. This trajectory is similar to a direct transfer to SEL_2. Table 4 shows a number of different direct low thrust options considered for NGST, as well as for an EML_1 mission. Obviously a high thrust to mass ratio is required to minimize flight time. The information in the table assumed a launch using the STS into a 28.5^0 LEO.

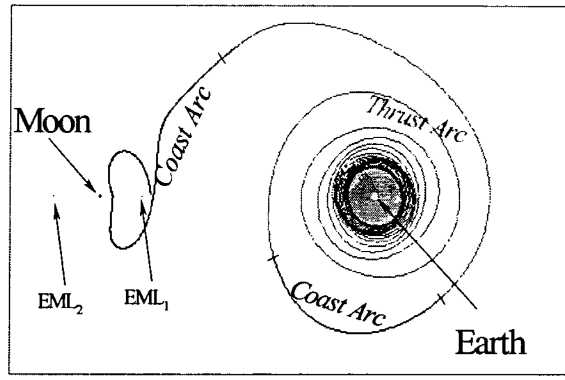

Figure 8. Low Thrust Trajectory to EML_1

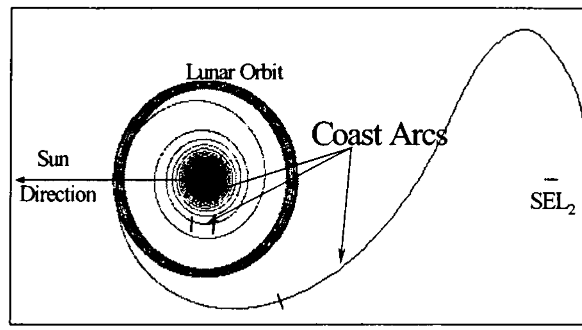

Figure 9 Low Thrust Trajectory from LEO to SEL_2

Nuclear Energy Options

An option for some missions is the use of nuclear energy as a power source for low thrust missions. Once the mission is achieved a disposal plan must be enacted and analysis performed to determine the possible reentry conditions.[18] For missions in the SEL_1 or SEL_2 region, this becomes a question of the probability of Earth impact if a perturbation places the spacecraft on an unstable manifold that returns it to the Earth environment.

No-return trajectories

The issue of an unplanned return of a nuclear power plant in a Sun-Earth libration point orbit is a valid one. Periodic orbits about the SEL_1 and SEL_2 are unstable. Loss of control of a spacecraft will cause the spacecraft to depart the periodic orbit asymptotically. The direction of this divergence is locally determined by the unstable eigenvector of the monodromy matrix, or State Transition Matrix (STM) over one period. This eigenvector is then globalized (non-linear integration) to determine the trajectory path of the departing spacecraft. The energy of the

Table 4. Sample Low Thrust Option Parameters

Propulsion Information Thrust (N), Isp(sec), Thruster # / Type			S/C Mass (kg)	Total Launch Mass (kg)	Thrust to Mass Ratio	Transfer Time to SEL_2 or EML_1 (Days)
2.02,	3100,	22 DS-1 (NGST)	5633	16421	1.230e-4	672 to SEL_2
2.78,	1732,	2 Halls (NGST)	1408	16420	1.693e-4	365 to SEL_2
2.08,	2488,	2 Halls (NGST)	4198	16420	1.267e-4	642 to SEL_2
2.31,	3800,	14 XIPS (NGST)	7200	16419	1.407e-4	610 to SEL_2
0.50,	2500,	1 Hall (non-NGST)	744	1000	1.14e-3	150 to EML_1
2.08,	2488,	2 Halls (non-NGST)	3178	4198	4.955-3	138 to EML_1

periodic orbit limits the space which the spacecraft can reach. This limit is called the zero velocity curve and is roughly a sphere about the Earth at low energy levels. At higher energy levels, specifically the energy levels of the SEL_1 and SEL_2 points themselves, the sphere opens up via corridors near SEL_1 and SEL_2 to reveal a region either inside the Earth's orbit about the Sun or outside it. The energy levels of the periodic orbits about SEL_1 and SEL_2 are higher than the points themselves and therefore spacecraft can depart the neighborhood of the Earth and SEL_1 / SEL_2 points. Once a spacecraft departs though the SEL_1 or SEL_2 corridors, its return is considered to be statistically irrelevant, although recent events indicate that an Apollo 3rd stage accomplished this.[19] However, some of the unstable manifolds remain in the region for some time and pass between the SEL_1 and SEL_2 points with multiple Earth flybys. It is these trajectories that sometimes provide Earth impacts.

Generally, half of the entire manifold leaves the system directly though a corridor with no Earth encounters. The other half exhibits some sort of Earth encounter before the majority of them leave the EM system. Monte Carlo analysis has shown that approximately 0.7% of the time, a randomly perturbed spacecraft in a large periodic orbit about the SEL_1 or SEL_2 points will impact the Earth within two years. Departures from smaller quasi-periodic orbits have not shown any impacts in a similar analysis. This analysis is limited to Circular Restricted Three Body Problem modeling. Additional force modeling including lunar perturbations and solar radiation pressure would significantly affect the propagation of any single trajectory and could significantly increase the possibility of impact, such as with ISEE-1&2.[20] However, it's not clear if those forces would change the overall statistical behavior of the trajectories. Figure 10 shows the minimum Earth radial distance for 150 equally spaced displaced points on a small periodic orbit over a two-year propagation. Figure 11 shows the same data from a large periodic orbit. Figure 12 shows the trajectory in views along three axes of one of the impact scenarios.

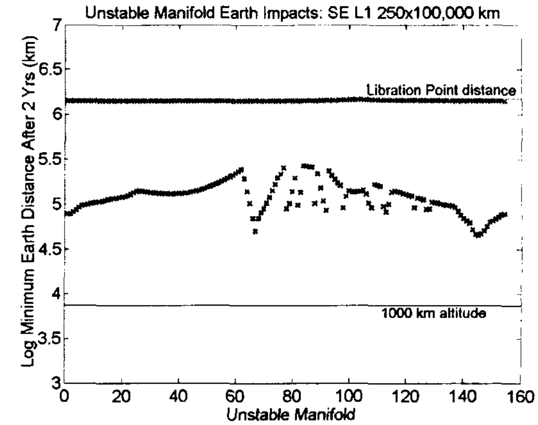

Figure 10. Minimum Earth Distance, Small Quasi Orbit

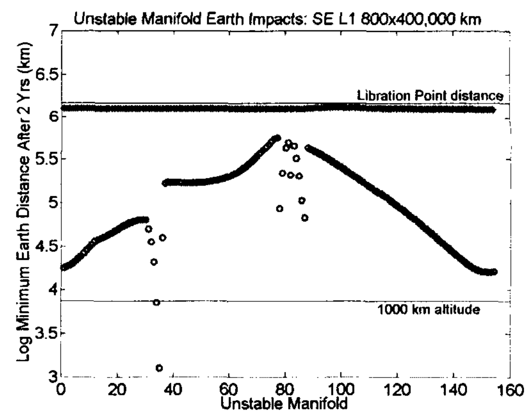

Figure 11. Minimum Earth Distance, Large Quasi Orbit

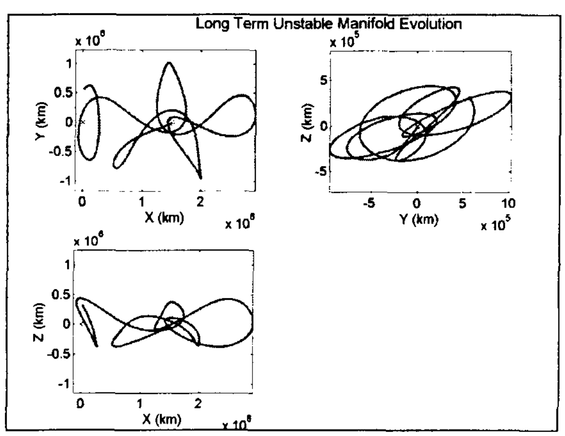

Figure 12. Return Trajectory Evolution

Biased Libration Orbits

Another method to ensure a non-return trajectory is one that models a stationkeeping strategy that incorporates a constant acceleration.[21] It necessitates the control of an unstable orbit in a constrained direction so that an onboard propulsion failure would guarantee a drift away trajectory. As with all libration orbits about the unstable SEL_1 and SEL_2 points, stationkeeping is required. One can design a biased orbit through inclusions of deterministic ΔVs and accelerations that result in a trajectory that diverges on an outbound eigenvector direction.

A strategy for a non-return libration orbit accommodation can be found by including deterministic ΔVs in the nominal SEL_1 lissajous orbit. This can be determined using the two-level differential scheme that is part of an overall dynamical systems approach. Expected gravitational and non-gravitational acceleration are modeled in the differential equations of motion. The deterministic maneuvers are pre-specified to be any appropriate value that accommodates accelerations and maintains the corrective maneuvers (i.e., stationkeeping) in a positive x direction. Without these maneuvers, the orbit departs into a drift away orbit.

Libration Orbit Rendezvous And Phase Jumps

In the above EM and SE transfer cases, the timing of the arrival into the final mission orbit was not considered. In some servicing options it may be required to rendezvous with the national resource to provide the propulsion to return it to the EML_1 or EML_2 regions, as was in the full transfer scenario.

The rendezvous problem has been extensively studied and performed in LEO for many years. Additionally, the rendezvous problem for libration point orbits has been solved and optimized using continuous thrust inputs by Marinescu, among others.[21] However, future mission concepts now include multiple spacecraft that may require rendezvous and docking. Low thrust propulsion systems are only feasible for a small number of spacecraft with large masses and extensive power systems. The libration orbit rendezvous problem using high (impulsive) thrust has not been addressed significantly. This paper addresses some of the initial investigation into this problem.

The primary technique used here is a Lambert solution using a numerically calculated STM and a simple differential correction. The STM is obtained through the integration of the variational equations along with the equations of motion. A state vector is created consisting of the six state elements of the target spacecraft (B) and the relative state vector between the two spacecraft. The STM matrix provides the sensitivity of the relative position between the two spacecraft and the velocity of the maneuvering spacecraft (A): $\frac{\partial \bar{r}_r}{\partial \bar{v}_A}$. Given a time to rendezvous of T, the calculated ΔV applied to A at time 0 is obtained from: $d\bar{V}_A(0) = \Phi_s^{-1}(-\bar{r}_R(T))$ where $\Phi_s = \frac{\partial \bar{r}_r}{\partial \bar{v}_A}$. This technique will provide a one-impulse solution to this rendezvous problem. There is no expectation that this technique will work for most rendezvous problems, however the applicability of this technique will show where enhanced techniques such as multiple impulses, weak stability boundary, target points, or Floquet Modes are required.[20,21,22] An example case is shown below.

The target spacecraft is placed in a small Lissajous orbit about the SEL_1 point. The maneuvering spacecraft is placed 7 days behind the target along the same trajectory. The time to rendezvous is set for 60 days. In this case, figure 13 shows the relative position closes to zero at a cost of 24 m/s in ΔV of spacecraft A at time 0. However, the relative velocity at time T was 44 m/sec. As a comparison STS requires less than 0.3 m/s to rendezvous with ISS.

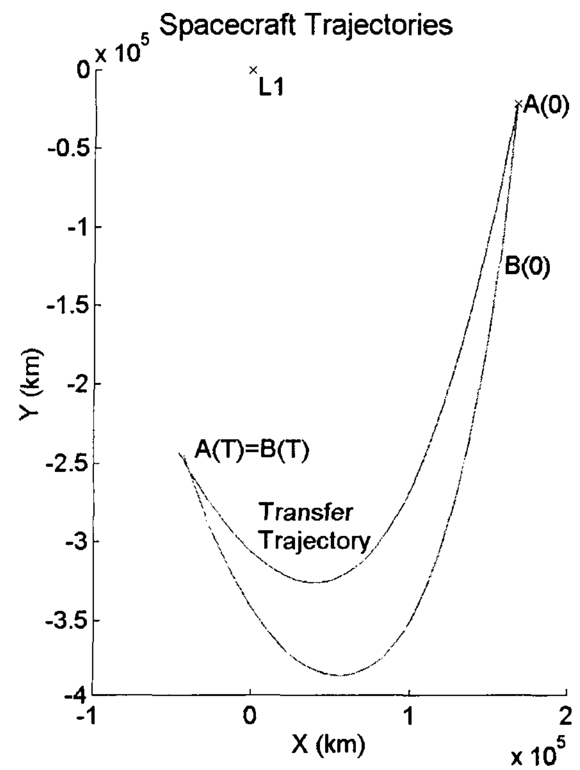

Figure 13. Cost of Phase Jump

Once the two spacecraft are within approximately a 5000 km box of each other, the above Lambert solution works quite well in closing the remaining distance with a small relative velocity on approach. A closed loop system would obviously be required for final approach. The total ΔV cost from within this box would be on the order of 10 m/s. This small distance solution is essentially a free-space approximation.

The process of jumping ahead or behind on the same periodic orbit is called "Phase Jumping". This idea has been used previously for z-axis control to avoid the solar exclusion zone. The goal is to phase jump the maneuvering spacecraft to rendezvous with the target spacecraft at a later time. This initial phase error would be due to a number of different sources including launch errors, maneuver execution errors, or phasing of other systems/orbits. Obviously, two spacecraft coming from different orbits or launches would not end up in the same phase of the same orbit. Phase jumping is critical in order to close the distance between spacecraft to within the free-space approximation defined above. The following figures show the effect on the cost of phase jumping due to initial phase angle and time to rendezvous. Figure 14 shows the 1st ΔV, 2nd ΔV, and the total ΔV for a 6 day phase jump over 60 days (time to rendezvous) versus the initial location of the target spacecraft in the periodic orbit (or phase angle). Figure 14 shows that the total ΔV is fairly insensitive to the location. Total ΔVs of 54 to 84 m/s are seen about one revolution of the orbit. Figure 15 fixes the initial location (approximately 0 deg initial phase angle) and shows total ΔV versus time of the phase jump for various T (times to rendezvous).

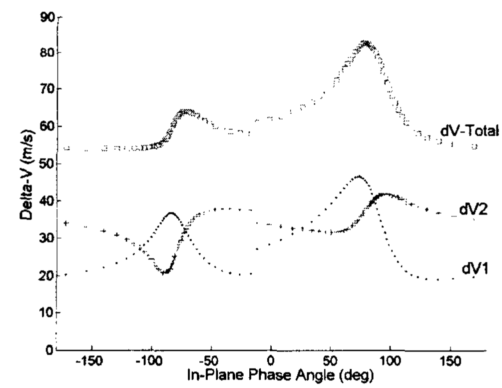

Figure 14. Total ΔV vs. Starting Phase Angle

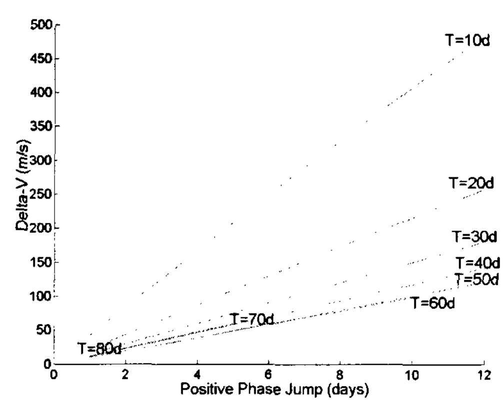

Figure 15. Total ΔV For Phase Jump

Conclusions

We have demonstrated several concepts and trajectory designs in support of servicing Sun-Earth libration missions in Earth-moon libration orbits. Several critical areas regarding ΔVs, travel time, rendezvous, and orbit dynamics were addressed. It was found that the ΔVs are in a manageable range of less than 500m/s, similar to direct transfer mission requirements with small SEL_n y-amplitudes. Timing associated with servicing is of importance and can drive the orbit mechanics and rendezvous opportunities. The timing requirement results in additional maneuvers, phase jumping, and targeting schemes. It is clear that algorithms and software tools for trajectory design in this regime must be further developed to incorporate better understanding of the solution space, and to improve the efficiency, and to expand the capabilities of current approaches.

References

1. D. Portree and R. Treviño, Walking to Olympus: An EVA Chronology, Monographs in Aerospace History Series #7, October, 1997.
2. D. Waltz, On-Orbit Servicing of Space Systems, Krieger Publishing Company, Malabar, Florida, 1993
3. D. Waltz, Supplement to On-Orbit Servicing of Space Systems, Krieger Publishing Company, Malabar, Florida, 1998
4. S. Leete, "Design for On-Orbit Spacecraft Servicing", Proceedings of the 2001 Core Technologies for Space Systems Conference, Colorado Springs, CO, 11/28-30, 2001.
5. H. Willenberg, M. Fruhwirth, S. Potter, S. Leete and R. Moe, "Site Selection and Deployment Scenarios for Servicing of Deep-Space Observatories", Proceedings of IEEE Aerospace Conference, 2002.
6. B. Kent Joosten, "Exploration in the Earth's Neighborhood" Architecture Analysis, NASA JSC, July 11, 2000.
7. M. Lo & S. Ross, "The Lunar L1 Gateway: Portal to the Stars and Beyond", Proceedings of the AIAA Space 2001 Conference and Exposition, pub. AIAA.
8. Report of the NASA Exploration Team: Setting a Course for Space Exploration in the 21st Century, June 1999 through December 2000. October 2001.
9. M. Beckman and D. Folta, NGST Servicing Trade Study, Aug. 2002
10. J.Guzman, D. Cooley, K. Howell, D. Folta, "Trajectory Design Strategies that Incorporate Invariant Manifolds and Swingby", AAS 98-349, AIAA/AAS Astrodynamics Conference, Boston, August 1998.
11. J. Breakwell, J. Brown, " The Halo Family of Three Dimensional Periodic Orbits in the Earth-Moon Restricted Three Body Problem, *Celestial Mechanics*, 20, 389-404.
12. R. Farquhar, "The Utilization of Halo Orbits in advanced Lunar Operations", NASA TN D-6365, NASA/GSFC, July 1971
13. R. Farquhar," Stationkeeping In The Vicinity Of Collinear Libration Points With Applications To A Lunar Communications Problem", AAS Preprint 66-132, AAS, July 1966.
14. R. Farquhar, D. Muhonen, and D. Richardson, "Mission Design for a Halo Orbiter of the Earth", AIAA/AAS Astrodynamics Conference, San Diego, CA, August 1976.
15. E. Belbruno, "Low Energy Trajectories for Space Travel and Stability Transition regions, Proceedings of IFAC Workshop on Lagranian and Hamiltonian Methods for Nonlinear Control, IFAC Publ, Elsevier Science LTD., Oxford, March 2000.
16. J. Breakwell, J. Brown, " The Optimization of Trajectories", *Journal of SIAM*, Vol 7, June 1959, pp. 215-247
17. K. Howell and H. Pernicka, "Numerical Determination of Lissajous Trajectories in the Restricted Three Bode Problem",*Celestial Mechanics*, Vol. 41, 1988, pp.107-124.
18. M. White and D. Stenger, "The Safe Use Of Nuclear Power Systems In Outerspace: Current Safety Standards And A Suggested Framework For The Future", Space Technology and Applications international forum, 2000, American Institute of Physics CP504
19. NASA Press Release, Media relations office Jjet Propulsion Laboratory, California Institute of Technology, September 20, 2002
20. J. Amenabar, "ISEE Orbital Decay Due to Lunar Third-Body Perturbations", Advances in Astronautical Science/ Astrodynamics, Vol. II, AIAA/AAS 87-535, August 1987
21. D. Folta, S. Cooley, K. Howell, "Trajectory Design Stratgies for the NGST L2 Libration Point Mission", AAS 01-205, Astrodynamcis Conference, Feb. 2001
22. A. Marinescu, A. Nicolae, and M. Dumitrache, Optimal Low-Thrust Libration Points Rendez-vous in Earth-moon System, AIAA 99-4050
23. E. Belbruno, "Analytic Estimation of Weak Stability Boundaries and Low Energy Transfers", Contemporary Mathematics, AMS Vol. 292 February 2002
24. K. C. Howell & T. M. Keeter, Station-Keeping Strategies for Libration Point Orbits: Target Point and Floquet Mode Approaches, AAS 95-195, AAS/AIAA Spaceflight Mechanics Meeting, Albuquerque, MN, Feb 13-16, 1995.
25. G. Gomez, et al, Dynamics and Mission Design Near Libration Points, Vol. I Fundamentals: The Case of Collinear Libration Points, World Scientific, 2001.