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GENESIS EXTENDED MISSION TRAJECTORY DESIGN

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Abstract

As the fifth Discovery mission, the Genesis spacecraft was launched on August 8, 2001 with a science objective to collect solar wind samples for a period of approximately two and a half years while in orbit in the vicinity of the Sun-Earth L1 libration point. On September 8, 2004, the Genesis sample return capsule returned to Earth with its collection of solar wind samples. Although the capsule did not get captured in the intended manner, from a mission design and navigation perspective the mission was a complete success. Now that the samples have been delivered, the remaining portion of the spacecraft is still fully functional, and has a generous amount of maneuver capability remaining. Various proposals for how to use this very functional and proven spacecraft have been suggested, for instance, continuing to study the solar wind using the on-board instrumentation in a libration point orbit. This paper will discuss the design of the various trajectory options for some of these proposed extended missions.

INTRODUCTION

As the fifth Discovery mission, the Genesis spacecraft was launched on August 8, 2001 with a science objective to collect solar wind samples for a period of approximately two and a half years while in orbit in the vicinity of the Sun-Earth L1 libration point. On September 8, 2004, the Genesis sample return capsule (SRC) returned to Earth with its collection of solar wind samples. The SRC was supposed to be recovered via mid-air capture by helicopter. Unfortunately, the parachutes failed to deploy and the SRC impacted the Earth. This was not a complete loss for the mission, since much of the solar wind samples has been recovered from the damaged SRC, and the initial estimates are that most of the science objectives can still be met. Although the capsule did not get captured in the intended manner, from a mission design and navigation perspective the mission was a complete success. Furthermore, the remaining portion of the Genesis spacecraft is still fully operational and could be used to support a variety of extended mission options.

Spacecraft

The Genesis spacecraft is depicted in Figure 1. The spacecraft consists of basically two parts. The first part is the bus that is comprised of the solar panels, propulsion system, attitude control system, avionics system, the telecommunications systems, low gain antennae (LGA) and the medium gain antenna (MGA). There are also two scientific instruments

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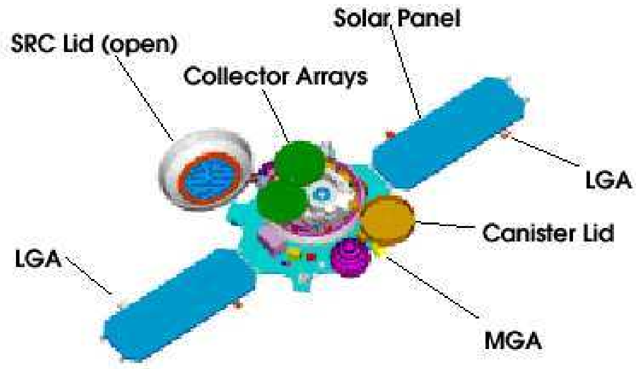

Figure 1. Top View of the Genesis Spacecraft

on board the bus, namely the electron and ion monitors that could be used to continue to study the solar wind. The spacecraft is spin-stabilized with a nominal spin rate of about 1.6 RPM. Due to the need to minimize sample contamination on the top of the spacecraft, all of the thrusters are on the aft side. This poses some operational considerations for the implementation of maneuvers, but these are well understood by the operations team.

The second part of the spacecraft is the Sample Return Capsule which contains the collectors for the solar wind samples The SRC is shown in its open configuration in Figure 1. The SRC is the portion of the spacecraft that re-entered the Earth's atmosphere on September 8, 2004.

Purpose of the Paper

Although the sample return capsule has separated from the bus, the remaining portion is still fully functional. Note that for the rest of this paper the terms Genesis and spacecraft will refer to the bus only. Furthermore, due to the decrease in mass from the SRC release, the remaining spacecraft bus actually has more maneuver capability (≈ 700 m/s) than it did at the beginning of the mission. Note that this is probably the first time this has occurred for any mission. Various proposals have been submitted for extended missions for the Genesis spacecraft. The various trajectory options that were examined to support potential extended missions are detailed in this paper.

METHODOLOGY

The Genesis trajectory was designed using Dynamical Systems Theory[1–3] coupled with a two-level differential corrections process.[4] This combination of techniques allows all of the various top-level mission constraints to be satisfied. Other references provide details on the procedures employed to determine suitable trajectory solutions.[5,6]

To begin examining potential extended mission orbits, a highly flexible design tool called LTool is used. LTool was developed at JPL and Purdue University specifically for the Genesis mission and incorporates Dynamical Systems Theory and the two level differential corrector to allow broad examinations of the design space in a relatively short amount of time. The mission designers used LTool in designing and re-designing the nominal mission, and it has proven to be a very effective design tool. See previous papers on Genesis mission design for extensive details on the methodology.[5,6]

Once a potential trajectory has been determined using LTool, higher-fidelity optimization is performed using another JPL program called CATO to further reduce the cost of the solution and to refine the propagation models. Once a final solution has been determined, the trajectory is verified using the JPL navigation software DPTRAJ. After a trajectory is generated and verified, various sanity checks are performed to detail potential geometric difficulties such as eclipses or telecom ranging. Finally, the feasibility of the solution is assessed by how well it meets the design constraints, as well as the operational complexity of implementing such a design.

Nominal Trajectory

As an example of this methodology, the trajectory is shown in Figure 2 in a Sun-Earth rotating frame. The frame is defined such that the X-axis is always coincident with the vector from the Sun to the Earth. The Z-axis is perpendicular to the X-axis in the direction of the angular momentum vector of the Earth's orbit, and the Y-axis completes the orthonormal triad. All trajectories in this paper are presented in this reference frame. The Sun-Earth libration points L_1 and L_2 shown in the figure are located roughly 1.5e6 km from the Earth along the X-axis. After launch, the spacecraft traveled out toward the L_1 point for approximately three months before a Lissajous orbit insertion maneuver (LOI) placed it into a large amplitude Lissajous orbit. After five revolutions in the Lissajous collecting solar wind samples, the spacecraft began its journey home in April 2004. A five month excursion out near the L_2 point positioned the spacecraft for a daylight entry over Utah. (Note that the trajectory is actually three-dimensional with a maximum out of plane excursion of around 300,000 km.)

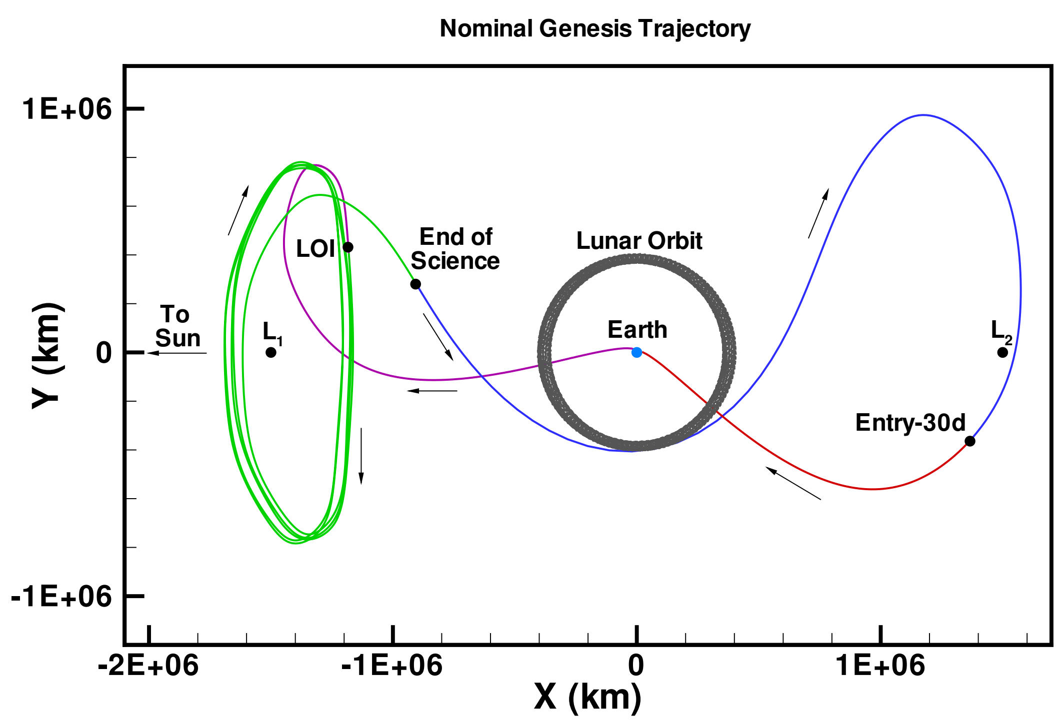

Figure 2. Genesis Trajectory in Sun-Earth Rotating Frame

DISPOSAL OPTIONS

After the SRC release sequence was executed at entry minus 4 hours, the remaining portion of the spacecraft (hereafter referred to as Genesis or just the spacecraft) performed a divert maneuver to allow the spacecraft to fly by the Earth at approximately a 200 km altitude. This divert maneuver placed the spacecraft into a highly elliptical orbit with a period of 58 days. The original decommissioning plan was for the divert maneuver to send the spacecraft out into a heliocentric orbit. However, due to constraints on how the divert maneuver was implemented, the orbit that resulted from the divert did not satisfactorily dispose of the spacecraft. The trajectory with no further maneuvers after the divert is shown in Figure 3. Without any further maneuvers, the spacecraft would have re-entered the Earth's atmosphere on October 19, 2005 at the end of the sixth loop.

Since an uncontrolled Earth impact of the spacecraft bus is unacceptable, a maneuver at the subsequent perigee of the initial 58-day orbit was planned to ensure that the spacecraft would exit the Earth-Moon system in a reasonable amount of time with no further close approaches to the Earth. Two options exist for the perigee disposal maneuver (ΔV_p), resulting in either an Earth-trailing or an Earth-leading heliocentric orbit.

The disposal orbit to exit into an Earth-leading solar orbit via the Sun-Earth L_1 region is shown in Figure 4. The trajectories are shown for three perigee maneuver magnitudes ranging from 5.0375 to 5.3475 m/s (all with the same maneuver direction). The sensitivity of this maneuver is evident in that a change of only 31 cm/s has a large effect on the resulting disposal trajectory. The actual maneuver performed on November 6, 2004 was 5.19 m/s corresponding to the solid or middle trajectory in Figure 4. Note that the expected execution

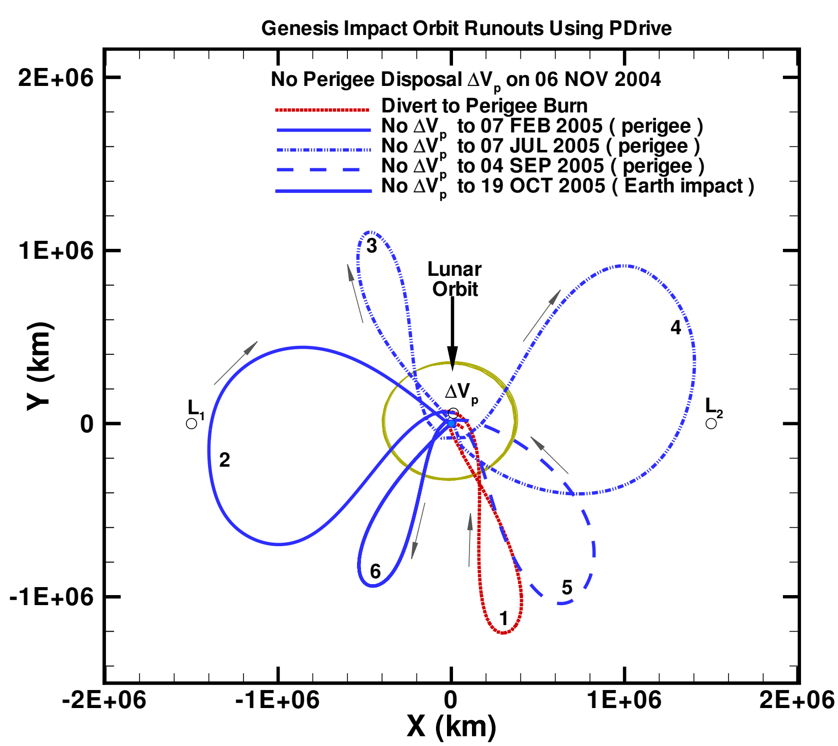

Figure 3. Impact Trajectory for No Future Maneuvers

error for the perigee maneuver was 3 percent or approximately 15 cm/s. So by comparing the three trajectories, it can be concluded that any reasonably accurate execution would have resulted in an escape trajectory via the L_1 region.

Alternatively, the disposal orbit to exit into an Earth-trailing solar orbit via the Sun-Earth L_2 region is shown in Figure 5. For this disposal, the sensitivity of the maneuver is even greater than that for the L_1 disposal due to the longer time spent in the Earth-Moon system and the distant flyby of the Earth. (Note that although the spacecraft passes the lunar orbit, there is no lunar encounter during this passage.) Again, three trajectories are shown corresponding to perigee maneuver magnitudes of 3.5610, 3.5616, and 3.6000 m/s (all in the same maneuver direction). In this case, the difference between a marginal escape trajectory (3.5616 m/s) and a trajectory that returns to a close Earth approach (3.5610 m/s) is only 0.6 mm/s. This is much less than the expected 3 percent execution error of 10 cm/s and is cause for some concern.

The "marginal escape" case is important because although the primary objective of the perigee disposal maneuver is to ensure that the spacecraft escapes the Earth-Moon system, a secondary objective is to not preclude options for an extended mission. These two objectives are in conflict. If only the primary objective needed to be satisfied, a much larger maneuver could have been executed that would have ensured beyond a doubt that the spacecraft would escape. However, this would have severely impacted any extended mission since any such mission would have to essentially undo the escape maneuver at some future time in order to remain in the Earth-Moon system. This change could be very expensive from a maneuver standpoint.

Ultimately, it was decided that the disposal via the L_1 region was superior to the disposal via the L_2 region. The primary deciding factor was the need to ensure proper disposal. The L_1 option quickly puts the spacecraft on an escape trajectory with ample margin for maneuver execution errors. Furthermore, it does not preclude the extended mission options, as will be shown in the following section. The L_2 disposal in comparison was much more sensitive to maneuver execution errors, and resulted in a longer time to exit the Earth-Moon system. If an extended mission had been selected prior to the execution of the perigee disposal maneuver, the L_2 option may have been preferable for a number of reasons that will be detailed in the next section. However, in the absence of a definitive answer on the fate of the extended mission, the L_2 option was deemed more of a risk than the L_1 disposal and was thus rejected.

EXTENDED MISSION OPTIONS

As mentioned, the disposal maneuver was specifically designed to allow the possibility for an extended mission in the Earth-Moon system while still ensuring a proper disposal if no extension was forthcoming. It was noted that the spacecraft after the initial divert maneuver has almost 700 m/s of maneuver capability. This vast amount of capability means that Genesis has the resources to continue as a viable spacecraft for a very long time. However, it is still prudent to preserve as much fuel as possible to maximize the extended mission options. Based on the methodology used to design the nominal and backup trajectories, the following orbits for the Genesis extended mission proposals have been designed and validated. This list is by no means exhaustive. In fact, the options presented here may best be described as just the initial phase of any extended mission. Note that all solutions begin at the same initial state after the L_1 perigee disposal maneuver on November 6, 2004.

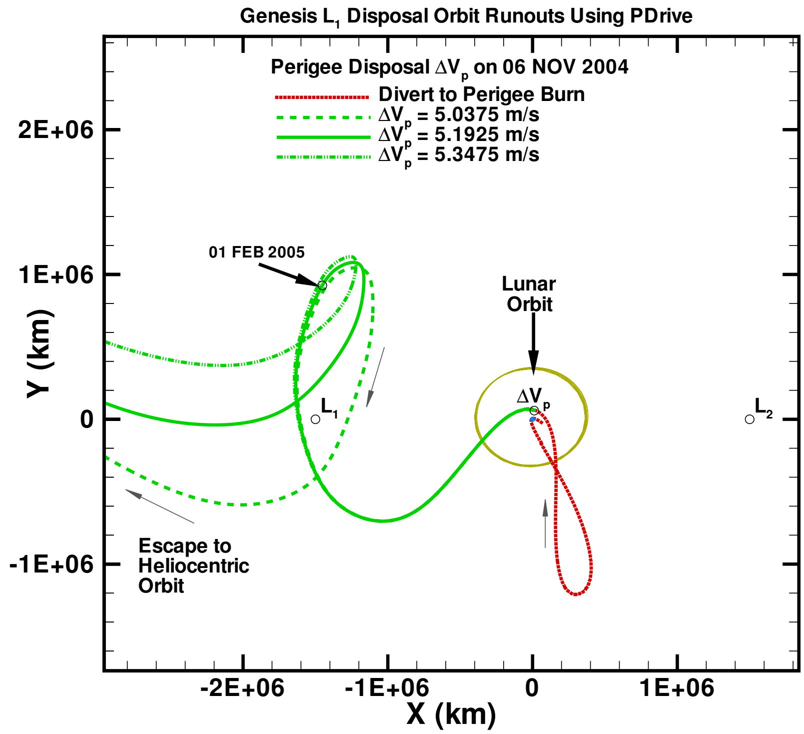

Figure 4. Disposal via Sun-Earth L_1 Region

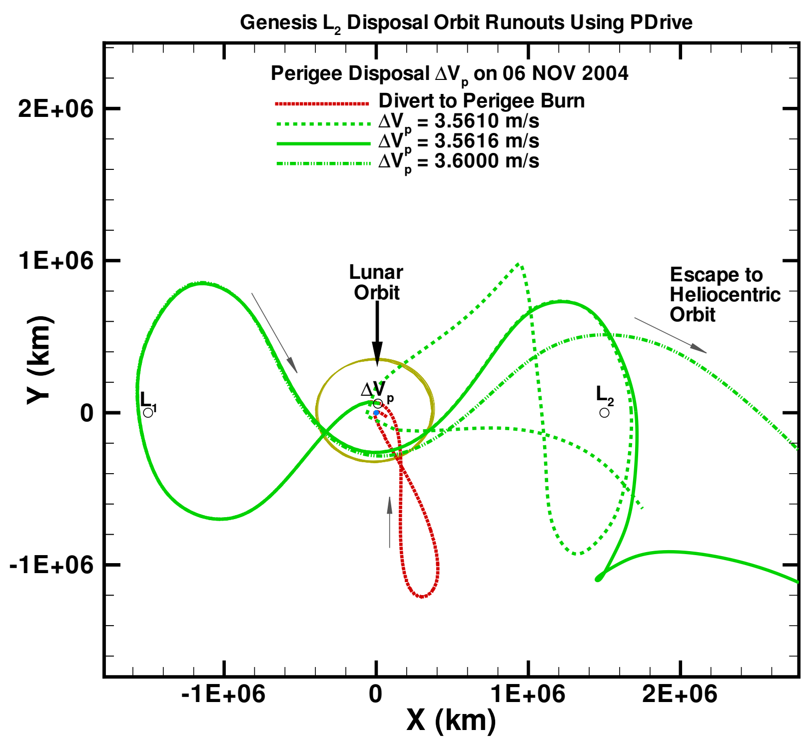

Figure 5. Disposal via Sun-Earth L_2 Region

L_1 Lissajous Orbit

Perhaps the most straightforward extended mission, given the selection of the L_1 disposal maneuver, is to place the spacecraft back into a Lissajous orbit in the vicinity of the Sun-Earth L_1 point.[7,8] One such trajectory is shown in Figure 6. In this solution, a Lissajous Orbit Insertion (LOI) maneuver is performed on January 26, 2005 to initiate two revolutions in the Lissajous. (More revolutions could be added with future maneuvers that were not designed in this study). This obviously then prevents the spacecraft from escaping the Earth-Moon system as originally designed. For the trajectory shown in Figure 6 the LOI maneuver cost is 47.87 m/s. Table 1 shows the LOI cost to insert into the L_1 Lissajous as a function of time. Clearly the later the insertion date the larger the maneuver cost. This growth is primarily due to the deteriorating geometry relative to the libration point as the disposal trajectory progresses. It may be possible to reduce the maneuver cost to remain in the Earth-Moon system by simply lowering the orbital energy without attempting to insert into a Lissajous. This would cause the spacecraft to fall back towards an Earth flyby, similar to what is seen in Figure 3 for the no future maneuver situation. A series of phasing loops would then be necessary to reposition the spacecraft to insert into another Lissajous about L_1 (or L_2), which could take up to six months. Thus, the trade represented here is maneuver size versus longer flight time, and hence higher operations cost.

An L_1 (or L_2) Lissajous is suitable for solar wind monitoring using the electron and ion monitors and could be maintained for as long as desired given regular station-keeping maneuvers every three months. This is also a stable orbit from which to initiate other trajectories in the Sun-Earth-Moon system, for example a hetero-clinic connection to the L_2 point, as will be detailed next.

L_2 Lissajous Orbit

As was demonstrated in the L_2 disposal orbit (Figure 5) it is possible to design a trajectory that transfers from the Sun-Earth L_1 to L_2 regions, sometimes called a heteroclinic transfer. By also introducing an appropriate LOI maneuver, a Lissajous in the vicinity of the Sun-Earth L_2 libration point is feasible. An example of this trajectory is shown in Figure 7. For this example, a maneuver of 179.19 m/s is performed on January 26, 2005 to initiate the transfer to the L_2 region. On May 26, 2005, an LOI maneuver of 22.59 m/s is performed to place the spacecraft into a five revolution Lissajous about the Sun-Earth L_2 point. As with the L_1 Lissajous, the later the initiation of the transfer is performed the higher the cost of the maneuvers. The corresponding maneuvers required to execute this trajectory for a set of dates are shown in Table 2. The maneuver costs associated with this option are much higher than for the L_1 Lissajous. Furthermore, the time criticality of executing this option is much more severe than the previous one.

This trajectory represents the minimum time (approximately six months) to transfer to an L_2 Lissajous. It is noted that perhaps inserting into an L_1 Lissajous first, as in the previous example, and then initiating the transfer to the L_2 region after one revolution (six months) may represent a lower overall maneuver cost. Another option is to reduce the orbital energy, as mentioned previously, and then use a series of phasing loops to setup the transfer to the L_2 region. These options were not examined, but could easily be determined given the previous L_1 Lissajous solution(s).

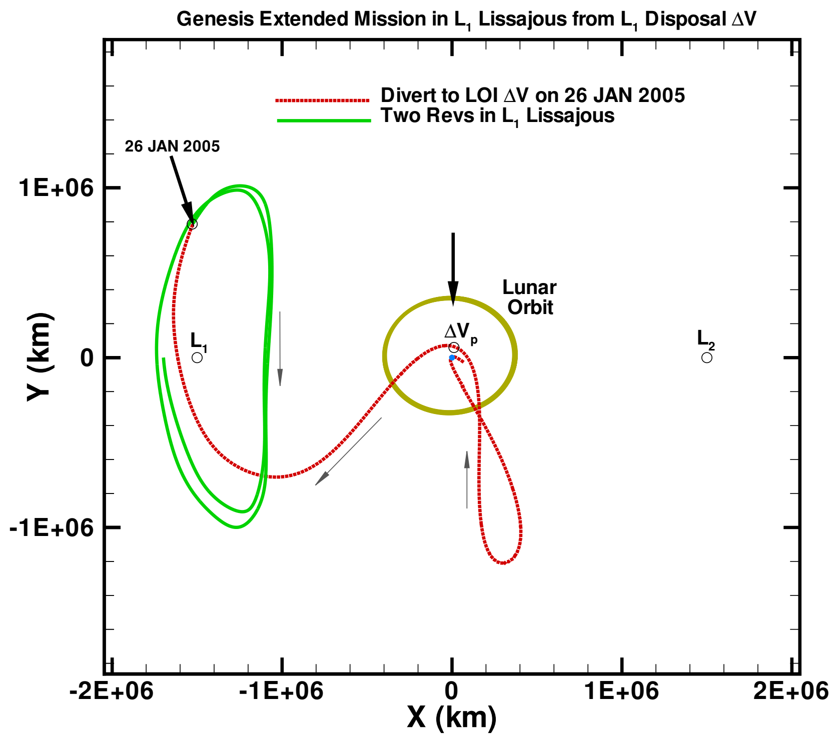

Figure 6. L$_1$ **Lissajous from** L$_1$ **Disposal**

Table 1. L$_1$ **LOI Maneuvers vs. Maneuver Epoch**

Maneuver Date	LOI ΔV (m/s)
Jan 12, 2005	29.57976
Jan 19, 2005	37.19072
Jan 26, 2005	47.86947
Feb 02, 2005	62.58218
Feb 09, 2005	82.77536
Feb 16, 2005	106.52788
Feb 23, 2005	132.20567
Mar 02, 2005	145.79297
Mar 09, 2005	160.68828
Mar 16, 2005	192.56343
Mar 23, 2005	242.05423
Mar 30, 2005	314.45544

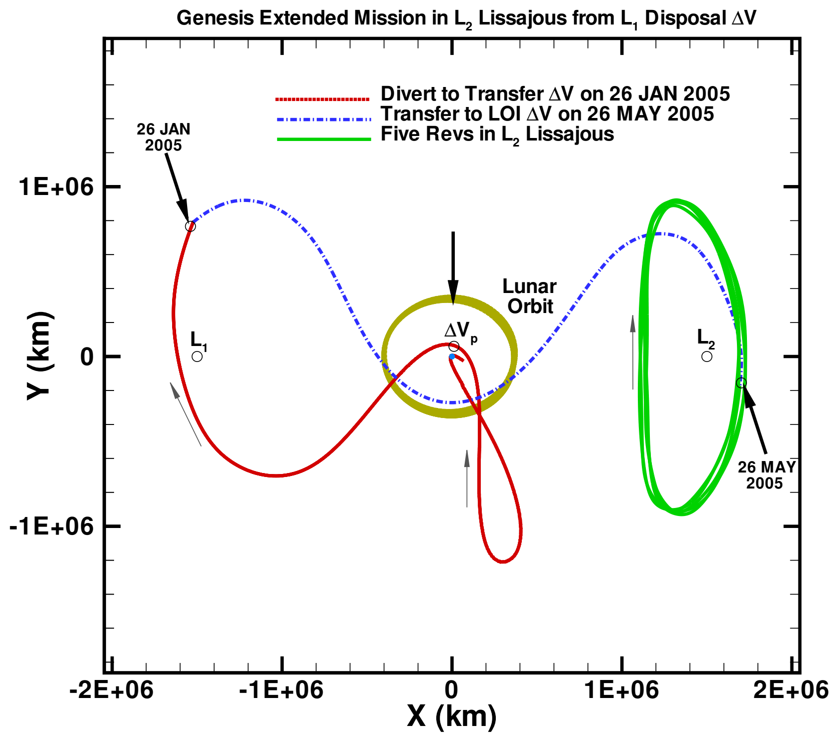

Figure 7. L_2 **Lissajous from** L_1 **Disposal**

Table 2. L_2 **Maneuvers vs. Maneuver Epoch**

Transfer Maneuver Date	Transfer ΔV (m/s)	LOI Maneuver Date	LOI ΔV (m/s)	Total ΔV (m/s)
Jan 12, 2005	94.26903	May 26, 2005	22.59123	116.86027
Jan 19, 2005	129.24693	May 26, 2005	22.31351	151.56044
Jan 26, 2005	179.19363	May 26, 2005	27.02339	206.21703
Feb 02, 2005	258.77820	May 26, 2005	33.00000	291.77820

Transfer to the Lunar Libration Points

It is possible to use either the L_1 or L_2 Lissajous orbits as a staging area to transfer to the Earth-Moon libration points. A stylized example of this is shown in Figure 8. This is loosely based on the previous L_2 Lissajous solution and supposes a 3-month transfer from the L_2 Lissajous into a lunar libration point orbital tour. This tour of the lunar libration points is only depicted here for illustrative purposes. See the papers by M. Lo et al.[9,10] for more details on these types of lunar libration point tours. Such an extended mission could be used as a validation testbed for lunar libration point orbits or in conjunction with another lunar mission.

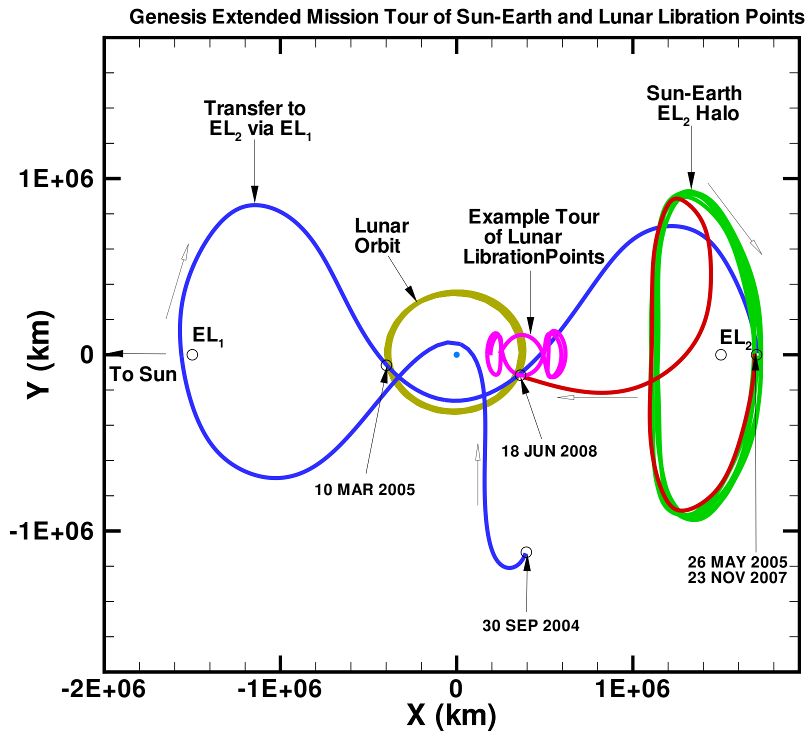

Figure 8. Transfer to Lunar Libration Point Orbits

Distant Retrograde Orbit

The final option examined is actually the original proposed extended mission. For this proposal a distant retrograde orbit of the Earth was deemed a suitable trajectory for scientific study of the solar wind due to its distance from the Earth and the orbit's long-term stability.[11,12] Using similar techniques to those for the nominal mission and the backup orbit design, a DRO with an X-amplitude of 5e6 km and a Y-amplitude of 10e6 km was designed. The final DRO is shown in Figure 9, while the initial part of the transfer out to the DRO is shown in Figure 10. In order to get into this orbit, a transfer trajectory needs to be determined with the proper phasing. For the portion of the transfer closest to the Earth, the trajectory once again starts in a 58-day period orbit. (Note that for this option, a different perigee maneuver at the end of the first loop would have been required than the one that was executed.) After five maneuvers totaling 171.873 m/s over four phasing loops,

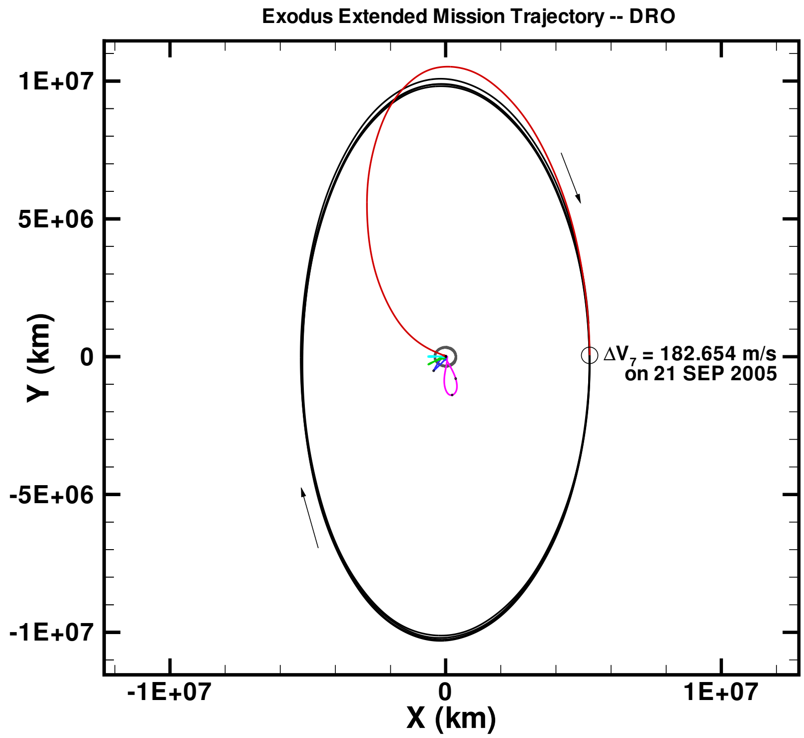

Figure 9. Distant Retrograde Orbit

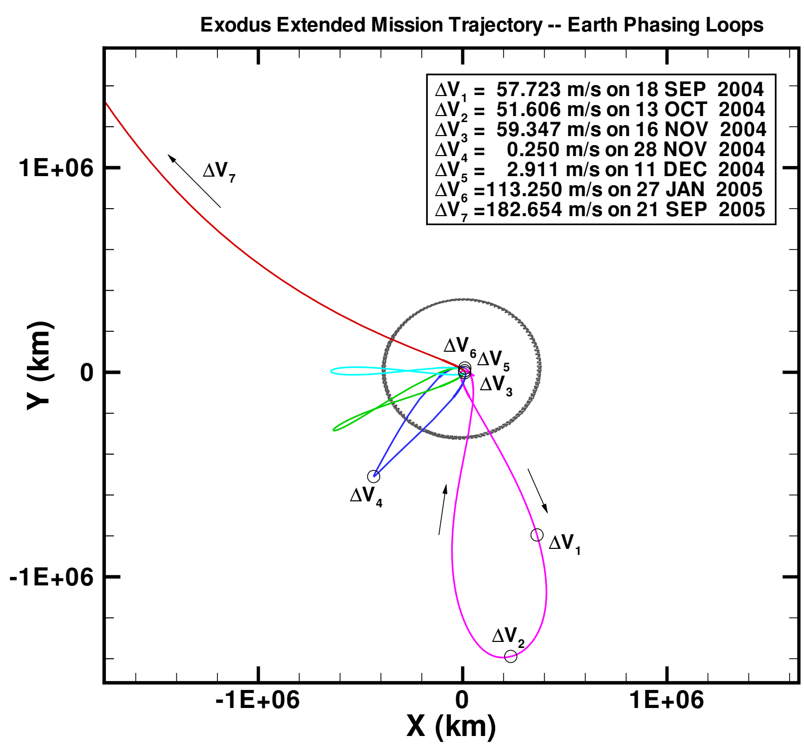

Figure 10. Distant Retrograde Orbit - Inner Loops

the timing is correct for a final perigee maneuver of 113.250 m/s to be performed to place the spacecraft on the transfer trajectory out to the DRO. A final maneuver of 182.654 m/s is required to inject into the DRO for a total cost of 467.741 m/s, which is well within the capabilities of the spacecraft. This would have been the first spacecraft to be placed into such a distant retrograde orbit.

CURRENT STATE OF THE SPACECRAFT

As of the date of this writing, no extended mission has been approved for the Genesis spacecraft. In the absence of any further maneuvers, the spacecraft will depart the Earth-Moon system and enter heliocentric orbit in roughly June of 2005. The resulting long-term propagation of this orbit is shown in Figure 11. This is the same Sun-Earth rotating frame as the other figures in this paper. In this figure, the Sun is fixed at (-1 AU, 0) while the Earth is fixed at (0, 0). A one AU orbit is indicated by the light shaded circle about the Sun. The spacecraft's orbit is clearly within the 1 AU orbit which is consistent with an Earth-leading trajectory. Note that each "loop" in the spacecraft's trajectory represents one year of propagation. Thus in a little less than 12 years the Genesis spacecraft will return to the vicinity of the Earth. However, it will be a distant return of approximately 0.1 AU. It will take multiple synodic periods of the trajectory before Genesis will even have a chance of re-encountering the Earth. No high accuracy long-term propagations were performed for this study, so the exact nature of the future encounters are unknown at this point.

Farewell, Genesis.

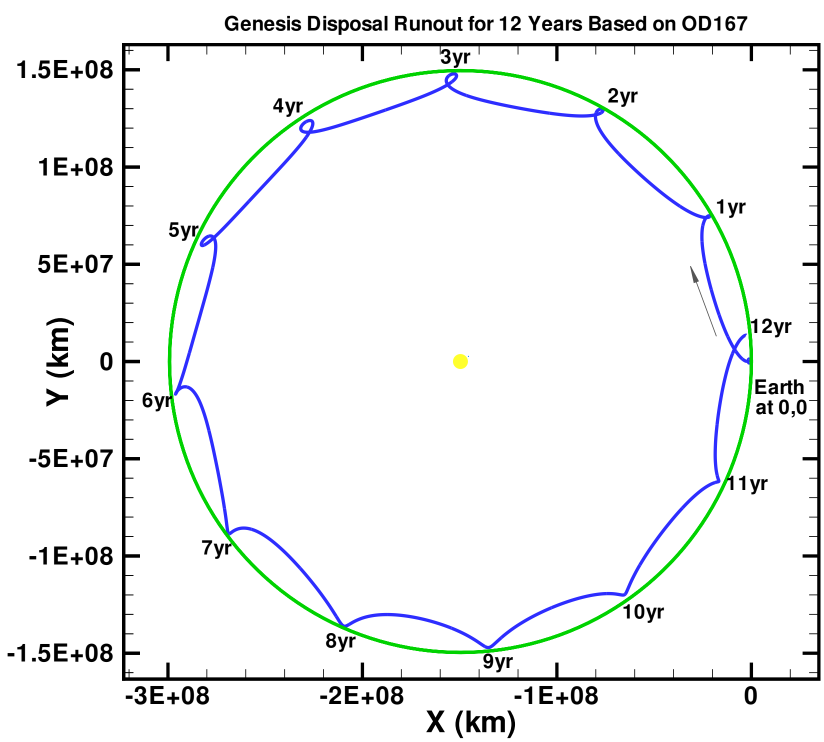

Figure 11. Runout of Current Spacecraft Trajectory

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