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TRAJECTORY DESIGN FOR THE GENESIS BACKUP ORBIT AND PROPOSED EXTENDED MISSION

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Abstract

In September 2004, the Genesis spacecraft will return to Earth with its collection of solar wind samples. If for some reason there are any difficulties with returning the samples to Earth during the nominal entry sequence, a contingency plan is in place for a second chance at a successful recovery. This paper will detail the processes used to generate possible backup trajectories, and provide a comparison of the contingency options examined through the course of the study. After the sample return capsule separates from the rest of the spacecraft, the remaining portion is still fully functional. It has been proposed to use this very functional spacecraft to continue to study the solar wind, albeit in a different regime than about the libration points. The remainder of the paper will discuss the design of the trajectory for this proposed extended mission.

INTRODUCTION

Genesis is the fifth mission selected by NASA under its Discovery program. The primary purpose of the Genesis mission is to collect solar wind particles and return them safely back to Earth. As such, Genesis will be the first mission to return extra-terrestrial samples since the Apollo missions. After years of development, Genesis was successfully launched from the Kennedy Space Center on August 8, 2001. The trajectory of the spacecraft takes it out to the vicinity of the Sun-Earth L_1 libration point. After insertion into a large amplitude Lissajous (or halo) orbit about the L_1 point,[1,2] the collection arrays were deployed to begin sampling the solar wind. Genesis will collect samples for five revolutions about the L_1 libration point (for approximately 2.5 years). After completing its science collection near L_1, Genesis will follow a free return trajectory home via a looping path about the Sun-Earth L_2 point to set up a daylight entry and a mid-air capture by helicopter over Utah in September 2004. After a safe return to Earth, the samples will be curated at the Johnson Space Center and then made available for study to scientists throughout the world.

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Mission/Science

The scientific objective for the Genesis mission is to more precisely determine the elemental and isotopic composition of the solar wind. The solar wind is thought to be compositionally identical to the Sun's photosphere, which in turn, is believed to be representative of the solar nebula from which the solar system was formed. By studying the solar wind then, scientists can, in essence, study the very material that formed the Sun and all the planets, moons, asteroids, and comets. A portion of the samples collected will be made available to scientists immediately, while the rest is held in reserve for future, perhaps as yet undeveloped analyses.

The solar wind is not constant in speed or composition. It is affected by activity on the surface of the Sun, such as coronal mass ejections. The Genesis spacecraft has electron and ion monitors to help classify the nature of the solar wind at any given instant. Based upon this classification, the spacecraft deploys various collector arrays that have been specifically designed for each particular solar wind regime. Genesis also has an electrostatic concentrator instrument that is tuned to optimize collection of oxygen, nitrogen, and carbon ions, while rejecting most of the hydrogen ions that make up the bulk of the solar wind.

Because the science objective is to collect pristine solar wind samples, it is critical that the collection be performed away from any interaction with the Earth's electromagnetic environment. This requirement drove the selection of an orbit about the Sun-Earth L_1 point. Trajectories in this region remain in front of the bowshock interaction of the solar wind with the Earth's magnetic field and suffer no eclipsing issues.

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, communications systems, and the electron and ion monitors. 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, as shown in the left portion of Figure 1.

The second part of the spacecraft is the Sample Return Capsule or SRC. The SRC is the portion of the spacecraft that is designed to re-enter the atmosphere and separates from the spacecraft bus prior to re-entry. The SRC consists of a hinged capsule that contains the sample collection canister, the collection arrays, and the electro-static concentrator. The SRC is visible in its open configuration in the right portion of Figure 1. The SRC

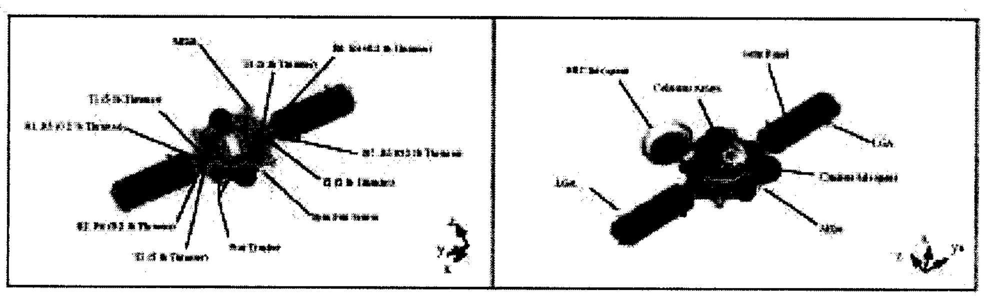

Figure 1. Fore and Aft Views of the Genesis Spacecraft

forebody is covered with an ablative material to protect the capsule during re-entry, while the backshell contains a parafoil to help slow down the capsule during the atmospheric portion of its descent. After the SRC has reached its terminal descent velocity, it will be captured in mid-air by a helicopter and then taken safely to the processing facility.

Purpose of the Paper

Since the scientific results of the mission are predicated on the safe return of the solar wind samples to Earth, it is crucial that every effort be made to facilitate a safe re-entry. In the unlikely event that the SRC is unable to return safely at the end of the nominal mission, a robust contingency plan is necessary to permit a second chance to bring the samples back to Earth. This paper details the processes used to generate possible backup trajectories for this contingency planning. In the more likely event that the SRC successfully re-enters the Earth's atmosphere, the remaining portion of the Genesis spacecraft would still be fully functional. Furthermore, it would in fact have more propulsive capability than at launch due to the decrease in mass from the SRC. One proposal for an extended mission is to send the spacecraft into a distant retrograde orbit (DRO) to continue to study the solar wind. The design of this DRO is detailed in the final section of the paper.

METHODOLOGY

The nominal Genesis trajectory was designed using Dynamical Systems Theory[3–5] coupled with a two-level differential corrections process.[6] This combination of techniques allows all of the various top-level mission constraints to be satisfied, such as: no interference from the Earth's electro-magnetic field, at least 23 months of science collection, and daylight entry over Utah for mid-air capture. These top-level constraints have driven the design and re-design of the nominal trajectory, but only the entry constraint is applicable to the backup orbits. Other references provide details on the procedures employed to determine suitable trajectory solutions.[7–10]

The nominal 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 1.5 million km from the Earth along the X-axis. After launch, the spacecraft travels out toward the L_1 point for approximately three months before a Lissajous orbit insertion maneuver (LOI) places it into a large amplitude Lissajous orbit. After five revolutions in the Lissajous collecting solar wind samples, the spacecraft begins its journey home in April 2004. A five month excursion out near the L_2 point positions 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.)

Initial Backup Orbit Design

In the nominal mission time-line, a divert maneuver is planned approximately 3.5 hours prior to entry and after the SRC has separated from the rest of the spacecraft. If the SRC has successfully separated, this divert maneuver will place the Genesis spacecraft on a trajectory that will eventually escape the Earth-Moon system after about a year if no other maneuvers are performed. For operational simplicity, it is desirable to use this

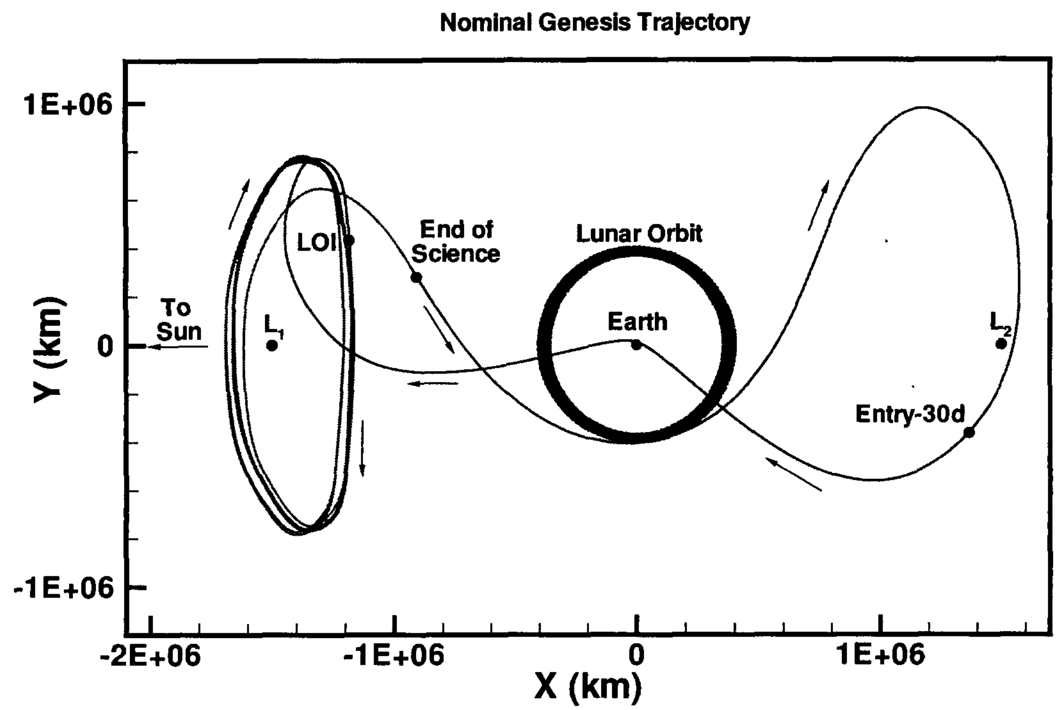

Figure 2. Genesis Trajectory in Sun-Earth Rotating Frame

same maneuver to initiate any backup orbit design. (Note that if the spacecraft is inserting into the backup orbit, the SRC will *not* have separated from the rest of the spacecraft. Thus, the mass of the spacecraft will be larger than the nominal divert maneuver, resulting in a smaller imparted maneuver.) Based on this assumption, a fixed initial state can be determined to initialize all of the backup orbit options.

If all continues as planned, the spacecraft will have approximately 330 m/s of maneuver capability after the divert maneuver, assuming that the SRC still attached. This 330 m/s is the margin available for both deterministic and statistical (corrective) maneuvers. The feasibility of any backup orbit will be judged primarily against this constraint.

Ultimately, the spacecraft has to be repositioned for a second attempt to re-enter the atmosphere. Similar to the nominal entry, this dictates a highly constrained final state that must satisfy position requirements, as well as constraints on flight path angle and velocity magnitude. In addition, daylight entry is preferred to facilitate the mid-air capture by the helicopters.

The fixed initial state, coupled with the highly constrained final entry state, and finite maneuver capability results in a difficult design space to characterize and explore. To begin examining potential backup 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.[7–10]

Verification

Once a potential backup orbit has been determined using LTool, additional 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.

BACKUP ORBITS

Based on this methodology, the following backup orbits for Genesis have been designed and validated. All solutions begin at the same initial state 3.5 hours prior to nominal entry and after the divert maneuver. All presented solutions return the spacecraft to a final state that allows for a safe re-entry of the SRC. However, not all of the solutions are feasible or operationally desirable, as will be explained.

Single Loop Options

The most straightforward backup orbit is one that simply reduces the period of the incoming trajectory and attempts to re-enter after one orbit. Backup orbits of all orbital periods within reason were examined, but two solutions were particularly interesting for various reasons. A solution with a 35-day period is shown in Figure 3. This solution requires a perigee burn of 12.630 m/s at the first close approach to the Earth, as well as a burn of 71.335 m/s near apogee for a total of 83.965 m/s. This solution is interesting because it is the longest single loop solution that still permits a daylight entry at Utah (actually right at sunrise). One of the difficulties with the 35-day single loop solution is the need to perform a perigee maneuver only 3.5 hours after the maneuver to divert into the backup orbit. This rapid maneuver sequence is an operational complexity to avoid, if at all possible.

The next logical option to try is to design a single loop option that eliminates the first perigee maneuver. The period of the resulting orbit is approximately 61 days and is shown in Figure 4. With a single maneuver of 99.829 m/s near apogee, the spacecraft can safely return to Earth. Unfortunately, the entry is now at night and the helicopters may be unable to recover the SRC before it impacts the ground. Since neither of these options, nor any other single loop solution are operationally desirable, other solutions are sought.

Multi-Loop Options

Perhaps the next simplest backup type is the multiple-loop option. For this option, of primary concern are perturbations from the Moon. To attempt to minimize these perturbations, while remaining within the maneuver budget, a 10-day period is initially examined for the multi-loop option. It is possible to return to Utah using two or three 10-day phasing loops. These solutions are shown in Figures 5 and 6. The cost to perform these backup orbits are 231.000 m/s and 241.000 m/s, respectively. For these types of solutions, there is again a large maneuver at the first perigee, which is undesirable. Furthermore, the trajectories have a number of large maneuvers and a limited time between the final maneuver

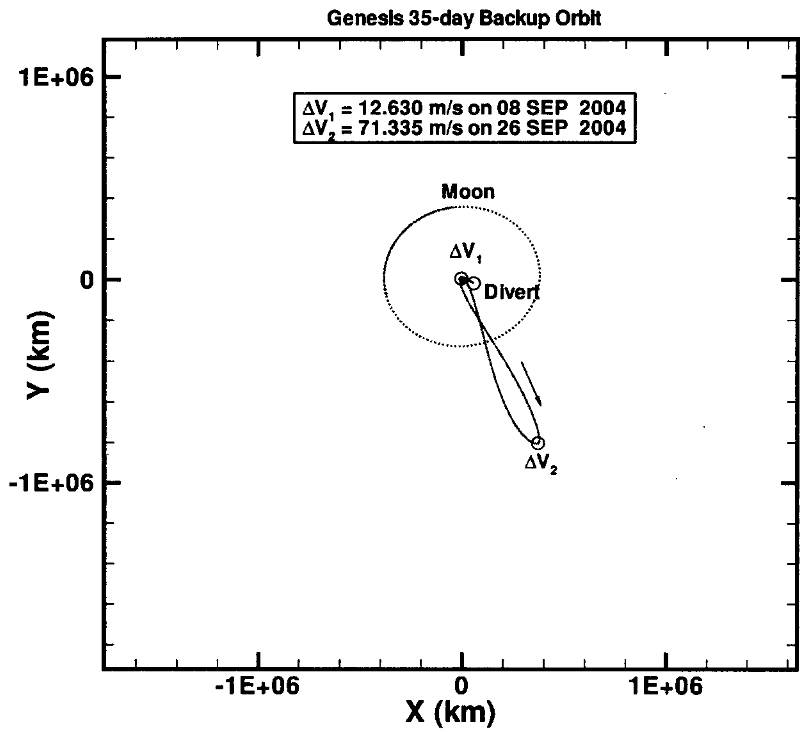

Figure 3. 35-Day Backup Orbit

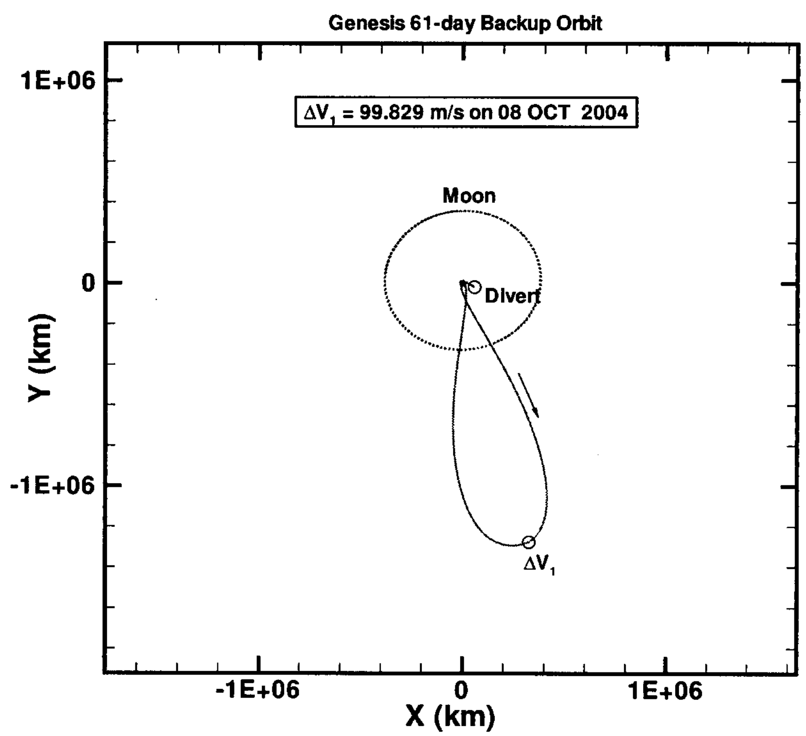

Figure 4. 61-Day Backup Orbit

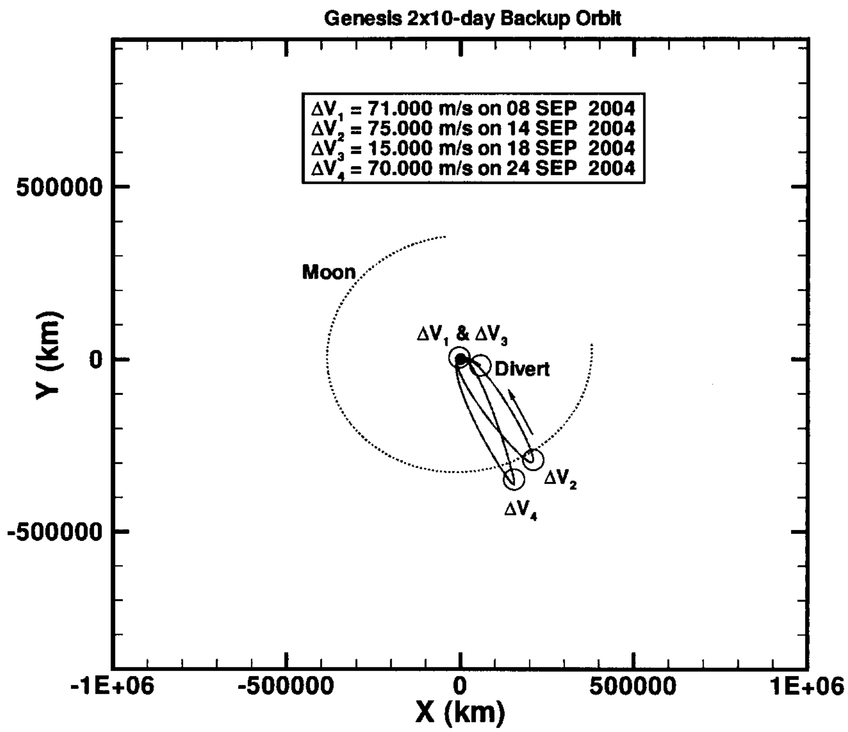

Figure 5. Two by 10-Day Backup Orbit

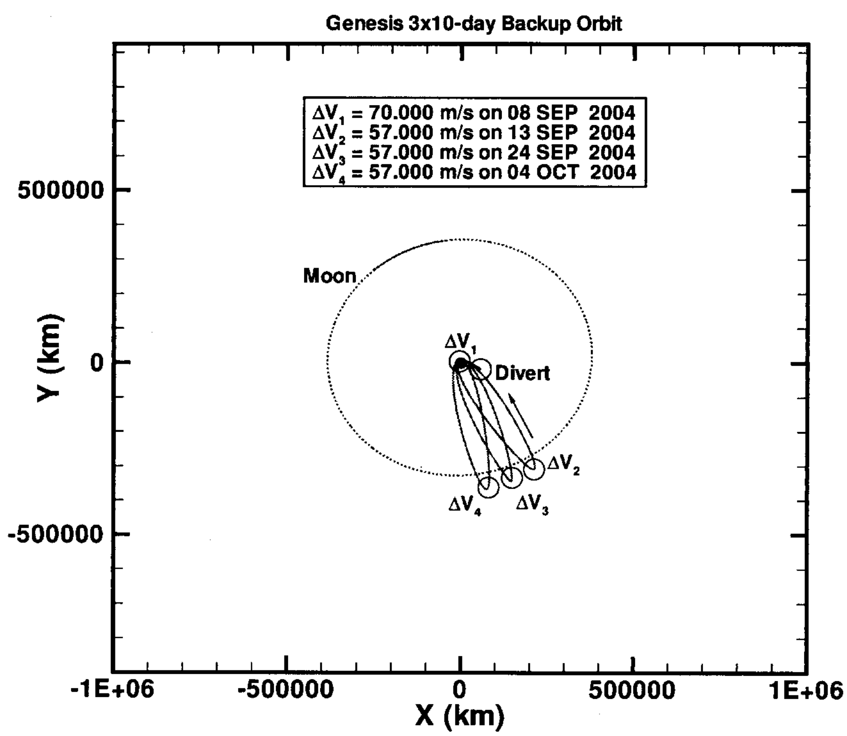

Figure 6. Three by 10-Day Backup Orbit

and entry. This situation does not permit sufficient margin to "clean up" the entry from any remaining maneuver or navigation errors, and thus ensure a safe recovery.

A solution was examined that allowed the trajectory to remain in a 7-day phasing orbit for approximately six months to attempt to alleviate the night-time entry issue. By allowing the line of apsides to precess relative to the Sun-Earth line for approximately half a year, a daylight entry would be possible. However, due to need to cancel the lunar perturbations during the 24+ phasing loops, it was not possible to return to Earth within the required maneuver budget.

Another way to minimize the lunar perturbations is to inject into a larger orbit with a period much greater than 10 days. Two solutions were examined for this option. The first solution uses a phasing orbit with a one month period. In this case, the first perigee maneuver immediately after the divert maneuver is eliminated so that the first loop is approximately two months in duration, similar to the single loop option described before. This trajectory is shown in Figure 7 and is similar to the orbital "petals" (labeled A_i) used in the WIND trajectory.[11,12] In order to return in the daylight, six 1-month phasing loops are used after the initial 2-month loop. The cost of this solution is 315.431 m/s, which is close to the maximum allowable. This cost does not include any statistical correction maneuvers, which may be considerable due to the very large (> 200 m/s) maneuver near apogee on the final loop (ΔV_4). This trajectory does satisfactorily achieve the desired entry conditions and includes a daylight entry, however it is operationally risky due to the high deterministic maneuver cost.

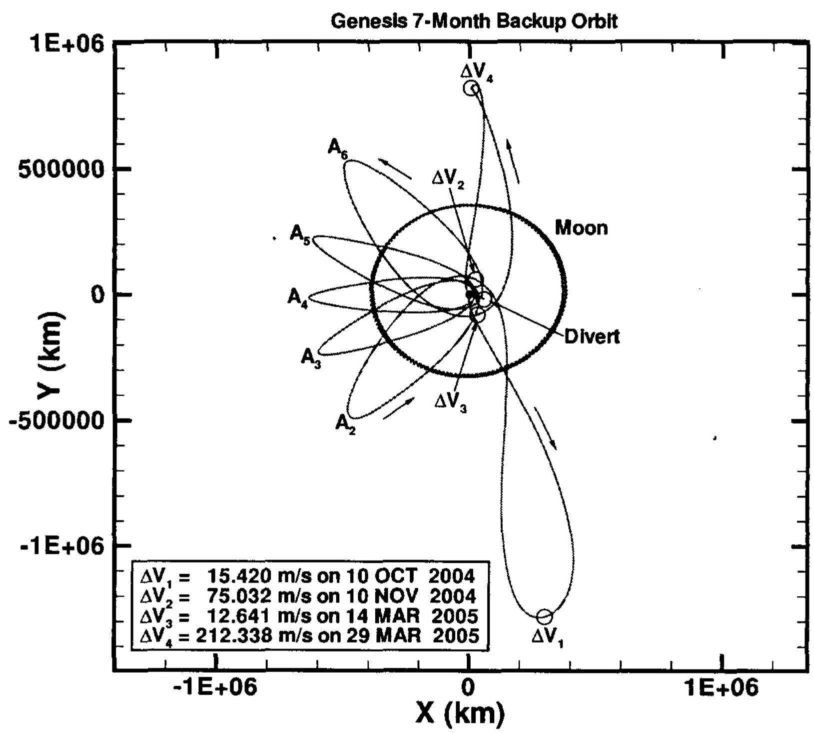

Figure 7. 7-Month Backup Orbit

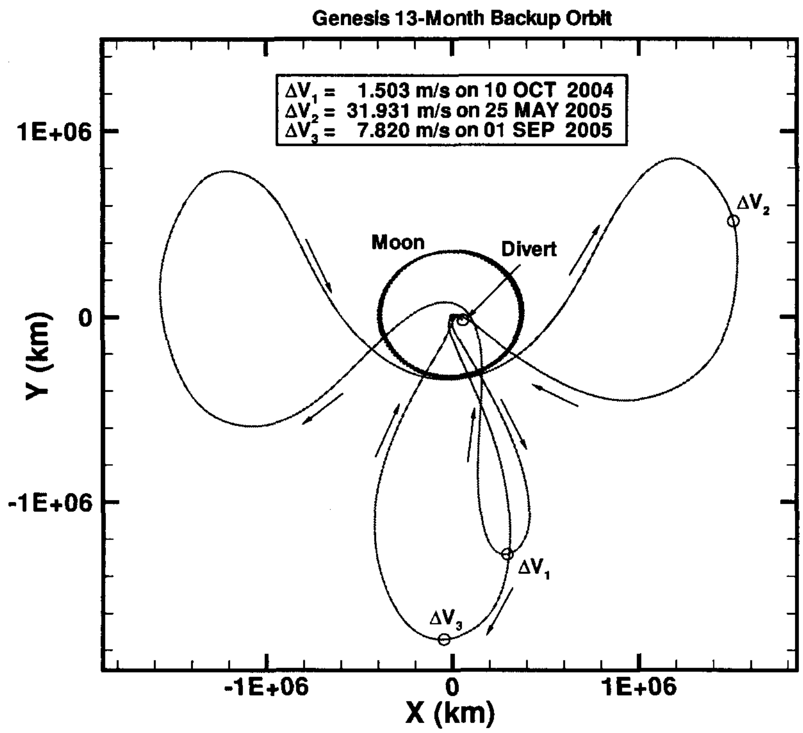

Figure 10. 13-month Backup Orbit using Sun-Earth Libration Points

A second, similar solution is shown in Figure 11. In this case, a distant lunar flyby ($> 100,000$ km) allows the spacecraft to return in just 10 months with a daylight entry, as desired. This solution is reminiscent of the type of trajectory used in the ISEE-3 extended mission to the comet Giacobini-Zinner.[18] However, the cost for this variant is 327.999 m/s, which may be too large after the statistical maneuvers are considered.

Conclusions

The viable options for the Genesis backup orbit are summarized in Table 1. The options are compared based on: number of maneuvers, initial perigee maneuver (ΔV_{P1}), initial apogee maneuver (ΔV_{A1}), maximum maneuver size (ΔV_{max}), and total maneuver cost (ΔV_{tot}). Additionally, explanations on the drawbacks to each solution are summarized in the last column. As mentioned earlier, the prime candidate for the nominal backup orbit is the Lunar Flyby option with no initial perigee maneuver. This solution meets all of the criteria set by the project and achieves the proper entry conditions, *in the daylight.*

EXTENDED MISSION

After the sample return capsule separates from the rest of the spacecraft, the remaining portion is still fully functional. Furthermore, due to the decrease in mass from the SRC release, the remaining spacecraft will actually have 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.) Since there are still two scientific instruments on board, namely the electron and ion monitors, it has been proposed to extend the mission to continue to

The second case examined for the large period phasing orbits is one that has three 2-month phasing loops. Since the first loop with no initial perigee maneuver is already two months long, this type of trajectory should have produced a relatively low-cost trajectory as compared to the previous solution. Unfortunately, this solution did not converge due to untargeted lunar encounters. Trajectories with periods longer than two months are best characterized using a three-body model and are covered later.

Lunar Flyby Options

Up to this point, lunar encounters have been avoided in the design process, where possible. In fact, the Moon has been the source of many of the difficulties in the previous solutions. If instead, the Moon is targeted during the backup orbit, other potential solutions become available. Since one of the drivers in the backup orbit selection is the ability to enter during daylight hours, a "backflip" trajectory was considered to shift the line of apsides by 180°. In the traditional backflip, a lunar encounter is used to radically change the orbital inclination out of the lunar orbit plane, while adjusting the orbital period to a multiple of the lunar period.[12,13] The trajectory then encounters the Moon half a period later to reverse the inclination and period change. This was initially looked at with little success. However, by modifying the backflip strategy to *not* encounter the Moon a second time, two potential backup orbits were found. The first solution is shown in Figure 8. This case has a perigee maneuver immediately after the divert maneuver and has a total cost of 142.066 m/s. As mentioned previously, the initial perigee maneuver is not desirable, so a second solution without the first perigee maneuver was determined. This trajectory is shown in Figure 9. This option has a total cost of 158.232 m/s, returns to Utah in the daylight and meets all of the return criteria. Note that in both cases, there is a large out of plane excursion in the -Z direction. This large -Z loop allows the trajectory to return to the northern hemisphere (Utah) during daylight hours. The lunar flyby trajectory with no initial perigee burn is currently the leading candidate for the nominal backup orbit.

For completeness it is also noted that the Double Lunar Swingby option was also examined.[11,14–17] This option uses multiple lunar flybys to alternatively raise and lower the orbital period, while shifting the line of apsides counter to the natural motion. In theory, this option should have yielded some solutions. However due to the highly constrained nature of the problem, no solutions were found that adequately met the entry criteria.

Libration Point Options

When the nominal Genesis trajectory was designed, dynamical systems theory was used to determine a low-cost solution to return to Earth using the Sun-Earth libration points. The final set of backup orbits revisits this type of solution to show that it is possible to return to the Earth using trajectories determined from dynamical systems theory. Based on the assumption that there is no initial perigee maneuver, the trajectory begins in a 61-day orbit as before. (The trajectory is shown in Figure 10.) At the next perigee, a small maneuver is performed to send the spacecraft back out to the Sun-Earth L_1 region. A heteroclinic connection to the L_2 region returns the spacecraft back to the Earth. However, unlike the nominal mission, an additional large phasing loop is necessary to achieve the proper entry conditions at Utah. This trajectory takes a total of 13 months with a cost of 41.254 m/s. Although this is the cheapest backup orbit found in terms of total cost, it unfortunately still returns in the dark, and is therefore not desirable.

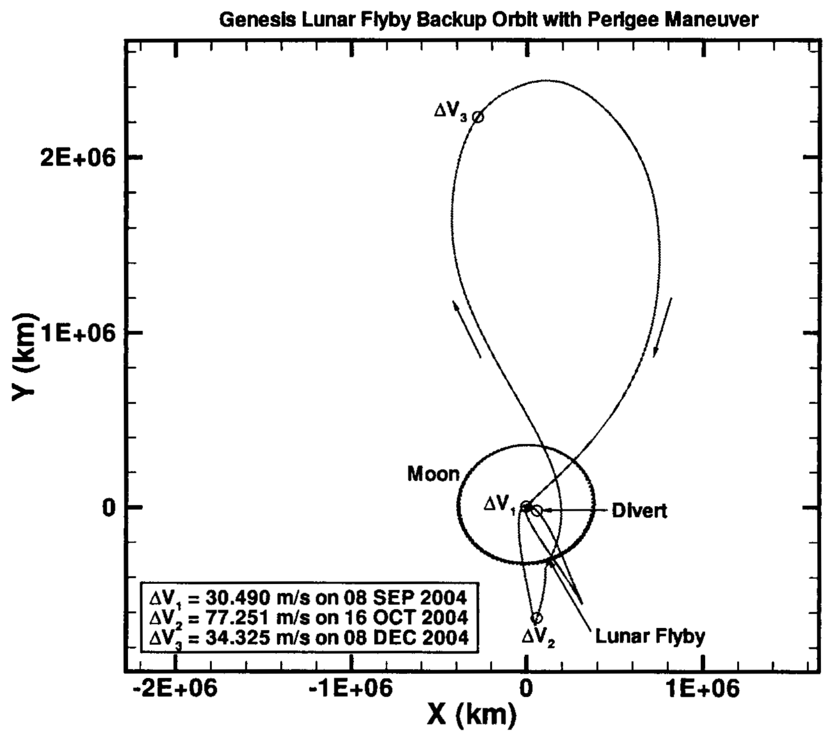

Figure 8. Lunar Flyby Backup Orbit with Initial Perigee Maneuver

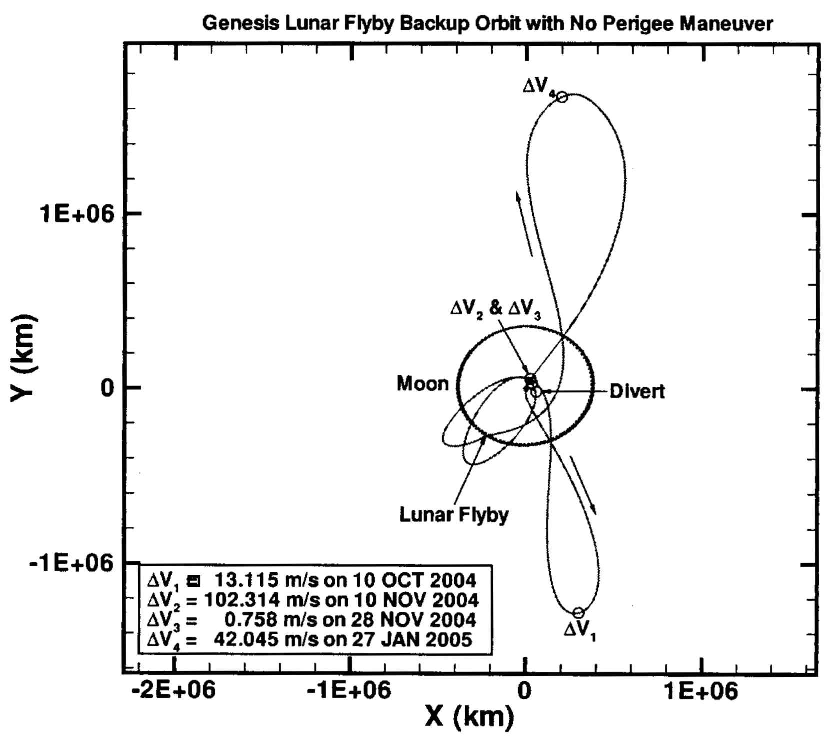

Figure 9. Lunar Flyby Backup Orbit with No Initial Perigee Maneuver

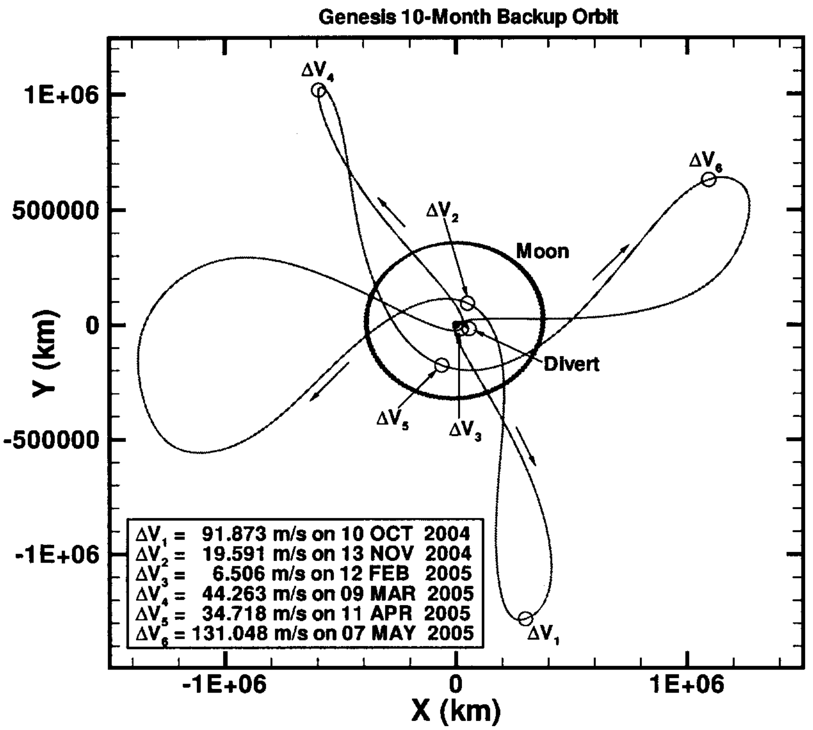

Figure 11. 10-month Backup Orbit using Sun-Earth Libration Points

Table 1. Comparison of Backup Orbit Options

Case	No. ΔV's	ΔV_{P1}	ΔV_{A1}	ΔV_{max}	ΔV_{tot}	Comment
35-day Single Loop	2	12.630	71.335	71.335	83.965	P,O
61-day Single Loop	1	–	99.829	99.829	99.829	N,O
2x10-day Multi Loop	4	71.000	75.000	75.000	231.000	P,O
3x10-day Multi Loop	4	70.000	57.000	71.000	241.000	P,O
7-month Multi Loop	4	–	15.420	212.338	315.431	C,O
Lunar Flyby w/ ΔV_{P1}	3	30.490	–	77.251	142.066	P,O
Lunar Flyby w/o ΔV_{P1}	**4**	**–**	**13.115**	**102.314**	**158.232**	*****
13-month Libration Pt.	3	–	1.503	31.931	41.254	N
10-month Libration Pt.	4	–	91.873	131.048	327.999	C

P = Initial Perigee Maneuver
N = Nighttime Entry
C = Maneuver Cost
O = Operationally Difficult
* = Best Nominal Backup Orbit Candidate

study the solar wind. Various trajectory options were examined including sending the spacecraft back out to a halo orbit about the Sun-Earth L_1 point, or a WIND-like trajectory[11] that alternates between the L_1 and L_2 Sun-Earth libration points. Ultimately, the scientists decided that a better use for the spacecraft would be to try to obtain solar wind data from a greater distance than the libration points and off of the Sun-Earth line. A distant retrograde orbit of the Earth was deemed a suitable trajectory for this scientific study due to its distance from the Earth and the orbit's long-term stability.[19]

Using similar techniques to those for the nominal mission and the backup orbit design, a DRO with an X-amplitude of about 5 million km and a Y-amplitude of about 10 million km was designed. The final DRO is shown in Figure 12, while the initial part of the transfer out to the DRO is shown in Figure 13. In order to get into this DRO, a transfer trajectory needs to be determined to get from the divert maneuver out to the DRO insertion with the proper phasing. For the portion of the transfer closest to the Earth, there is again no initial perigee burn after the divert, and thus the trajectory is once again in a 61-day period orbit. After five maneuvers totaling 171.873 m/s over four phasing 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. The total cost to get into the DRO is 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. Unfortunately however, the proposal for the extended mission was rejected and thus, the ultimate fate of the Genesis spacecraft is yet to be decided.

ACKNOWLEDGMENTS

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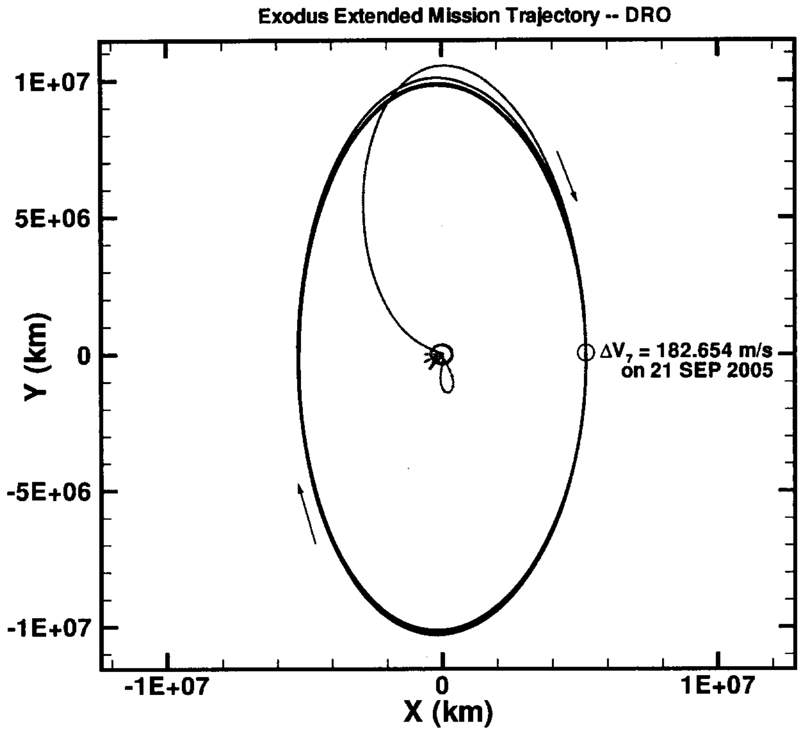

Figure 12. Proposed Extended Mission Obit - DRO

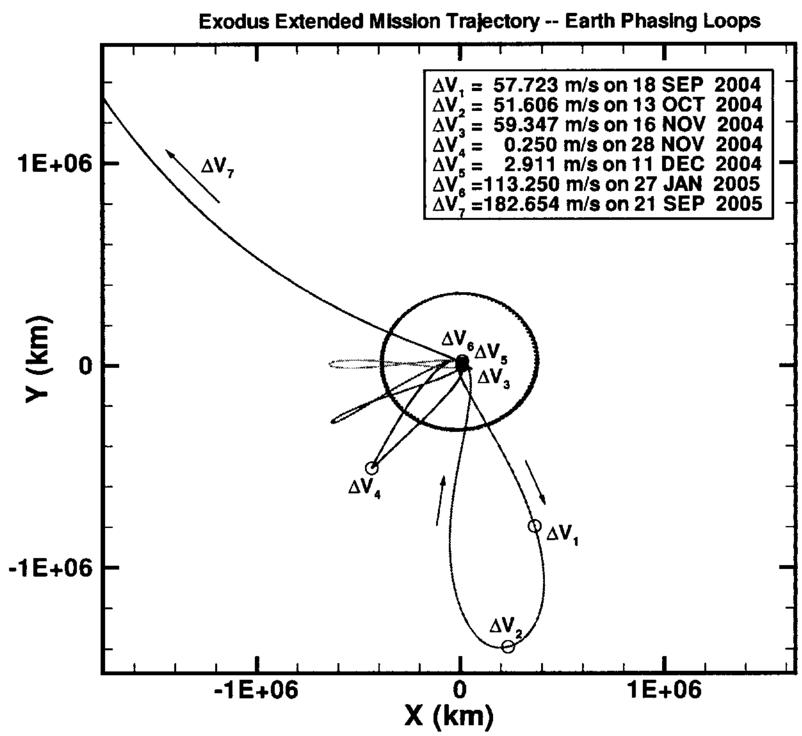

Figure 13. Proposed Extended Mission Obit - Inner Loops