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ABSTRACT

The Genesis mission will launch in 2001, sending a spacecraft into an L1 halo orbit in the Sun-Earth system to collect solar wind samples. In 2003, the samples will be returned to the Utah Test and Training Range for a daylight, mid-air recovery. A parametric study of the Earth entry conditions has helped to characterize the solution space. Several perturbation and robustness studies were conducted to analyze the sensitivity of the trajectory. These studies indicate that the trajectory can be adjusted to accommodate multiple perturbations.

INTRODUCTION

Early in 2001, the Genesis spacecraft will be launched into a halo orbit about the Sun-Earth L1 point to collect solar wind samples. Two and one-half years later, the samples will be returned to the Earth for analysis that will contribute to the study of the origins of the solar system. The initial trajectory design employed many powerful design techniques including dynamical systems theory and complex trajectory optimization.[1,2] Ongoing work in support of the refinement and analysis of the trajectory continues to require new ideas and to reveal new insights into the solution. This paper summarizes the results of several studies that have been conducted in response to improvements in the trajectory modeling and changes in the mission requirements. Following an outline of the trajectory development process and a description of the current reference trajectory, a family of trajectories is described from which the current reference trajectory was selected. Next, the sensitivity of the trajectory to variations in the models is discussed. Finally, the results of a preliminary study are presented that have provided insight into the solution space in support of navigation and contingency planning.

TRAJECTORY DEVELOPMENT PROCESS

Orbits about the Sun-Earth L1 point provide a unique solution to many mission design problems. Such trajectories provide direct view of the Sun outside of the Earth's magnetosphere with relatively short communication distance of the Earth. Thus, this option is becoming increasingly popular for missions to study the Sun. Several missions such as ISEE-3 and SOHO have successfully demonstrated the feasibility of such a trajectory. Furthermore, the use of targeted lunar flybys by ISEE-3/ICE demonstrated the ability to complete a targeted flyby following departure from L1, with a final lunar flyby altitude of only 120 kilometers.[3] However, the high latitude of the Genesis landing site, tighter navigation constraints associated with targeting the appropriate entry conditions (with no way to recover from errors in the targeting, since the entry is ballistic), and the need for a daylight landing (to accommodate a helicopter recovery) present unique challenges to the Genesis mission.

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not explicitly used in the analysis, although the theory continues to guide the analysis and interpretation of the data.

Next, variations from the baseline model are simulated including more sophisticated models for the Earth and Moon gravity fields, the effects of solar radiation pressure, and the effects of additional planets, in addition to the effects of attitude maintenance maneuvers and maneuver calibrations that are required throughout the trajectory.[4] The daily precession maneuvers, for example, are incorporated in this step since modeling these maneuvers as part of the nominal trajectory design will help to improve the navigation since the predicted path will more closely match the actual trajectory.

Once a trajectory has been designed that achieves all of the trajectory requirements with appropriately refined dynamic models, the next step is for the trajectory designers to work with other sub-systems to accommodate other mission needs. In particular, several trajectory modifications, such as including the biasing station-keeping maneuvers, are being considered to help meet the navigation requirements. Thus, the trajectory design process is an iterative process that requires working with many mission sub-systems to achieve a solution that meets all mission requirements.

CURRENT REFERENCE TRAJECTORY

Figure 1 depicts the current reference trajectory, in three two-dimensional projections of a rotating coordinate system, with Earth at the origin. The Sun-Earth line defines the x-axis (positive pointing from the Sun to the Earth); the x-y plane is the ecliptic plane. (The lunar orbit is only shown in the x-y projection.) The trajectory has three primary phases: the transfer phase, which extends from launch to halo orbit insertion; the halo orbit (or science) phase, which includes approximately four revolutions in the halo orbit (about 2 years); and the return phase, which takes the vehicle from the L1 region, past the Earth, to a loop around the Sun-Earth L2 point before returning for a daylight, mid-air helicopter recovery of the entry capsule at the Utah Test and Training Range (UTTR).[5] In the early stages of the trajectory design, it was determined that trajectories that return directly to the Earth from the halo orbit, without going to L2, tend to return on the night side. Therefore, the L2 loop was incorporated into the solution in order to enable the daylight return that the helicopter recovery requires.

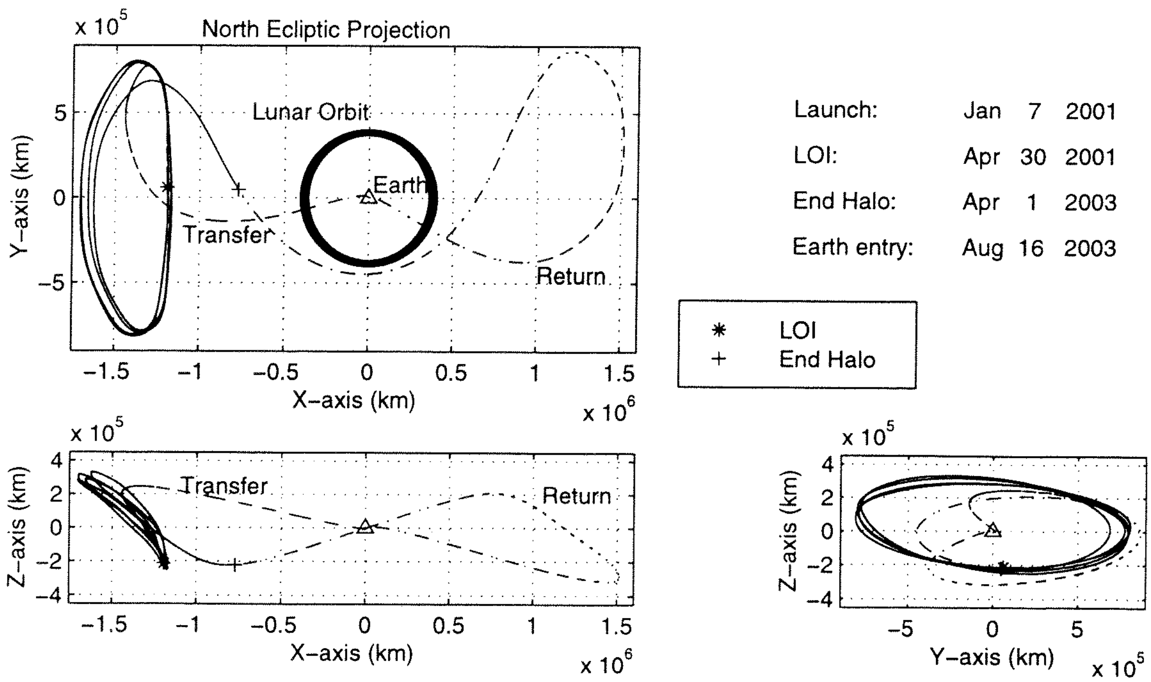

Figure 1. Current Reference Trajectory.

Each of the three phases represents a smooth leg with no delta-v's along the leg. At the end of the transfer phase, a libration point orbit insertion (LOI) maneuver inserts the spacecraft into the halo orbit. However, at the end of the halo orbit phase, the trajectory transitions to the return phase with no deterministic delta-v, using a "free-return" from the halo that was discovered by analyzing the unstable manifold associated with the halo orbit. Thus, LOI is the only deterministic maneuver required in the trajectory, although numerous station-keeping maneuvers will be necessary to satisfy navigation requirements. The LOI cost varies across the launch period from 22.5 m/s for a nominal January 7, 2001 launch to a maximum of approximately 30 m/s for launch on January 20, the end of the January launch period.

DESIGN SPACE

Since there are no deterministic maneuvers along the return leg the LOI maneuver must not only insert the spacecraft onto the halo orbit, but it must also target the entry conditions. Once the vehicle is inserted onto an appropriate halo orbit with a specific LOI delta-v, it is immediately starting its return to Earth; no additional deterministic entry targeting is required. This strategy leads to a parameterization of the design space that describes not only the entry conditions that can be achieved, but also the halo orbits that lead to those entries.

Trajectory Families

The trajectory is computed using a numerical differential corrections scheme that constrains the launch, the LOI position and time, and several entry conditions.[2] In this process, both the halo orbit and the return leg are designed simultaneously, as a single, continuous path so that both phases can adjust to respond to changes in the entry conditions, enabling the free-return. This approach to the design problem has led to the development of a family of solutions that characterize a broad range of entry conditions.

Figure 2 includes members of one trajectory family. Each orbit begins at the same LOI position and time, and each solution ends with entry conditions that are consistent with impact at UTTR (using a preliminary model for the atmospheric effects). However, the trajectories differ significantly between the LOI and entry points, diverging immediately after the insertion maneuver. (Only one transfer path is shown on the figure since they share a common transfer trajectory as a result of the constraint that each family member must include the same LOI position and time. The transfer path is not shown in the y-z projection.)

Although all of the solutions land at UTTR, the approaches to that landing site vary considerably among the cases. Each halo orbit and return combination defines a single set of entry conditions that can be achieved with a free-return, while satisfying the UTTR impact constraints for a specified LOI and entry date combination. Some of these entry constraints are fixed parameters that must be satisfied by all of the cases (as noted in Table 1). Other parameters vary among the solutions, subject to less restrictive constraints. Several of these variable parameters are listed in **Table 2**.

One of the most noticeable differences among the solutions is the variation in halo orbit y-amplitudes, ranging from about 720,000 km for Case 3 to about 860,000 km for Case 1. The y-amplitudes of the L2 loops are also significantly different, with the trajectory with the largest halo orbit having the smallest L2 loop (Case 1), and vice-versa. Another notable characteristic is the difference in the orientation of the trajectories in the y-z projection (the view from the Earth looking toward the Sun). A line joining the minimum and maximum y-excursions in this view is rotated slightly through a positive angle about the x-axis for Case 1, while both Case 2 and Case 3 are rotated in a negative direction, with the amount of rotation increasing from Case 2 to Case 3.

Although all of the solutions utilize a free-return, requiring no deterministic maneuvers in the return leg to target the entry conditions, the delta-v cost to target the entries is reflected in the different LOI costs. For the earliest entry, the LOI delta-v is about 73 m/s, while less than 30 m/s is required for returns a few days later. However, these insertion costs are misleading in terms of the delta-v required to design a trajectory with these general characteristics. By defining a family by a fixed, common LOI position and time, each trajectory is forced to a non-optimal insertion. For example, given a specified, common value of y at LOI,

Phase	Characteristic	Case 1	Case 2	Case 3
	Line Type in Fig 1	Dot-Dashed	Solid	Dashed
Halo	Y-Amplitude	860,000 km	813,000 km	720,000 km
	LOI Delta-V (m/s)	73.0	22.2	26.1
Return	Times of Lunar Closest Approach (2003) (GMT)	April 03 00:27 April 14 02:56	April 04 18:13 April 12 11:20	April 11 13:57
	S/C-Moon Closest Approach Distance (km)	405,000 km 425,000 km	345,000 km 350,000 km	260,000 km
Entry	Date (2003) (MDT§)	Aug 13 10:54	Aug 16 10:46	Aug 19 16:34
	Inclination (EME2000**)	42.2 deg	52.5 deg	71.6 deg
	Longitude (PMD††)	42.1 deg	46.5 deg	50.6 deg
	Latitude (PMD)	231.3 deg	233.1 deg	239.0 deg
	Flight Path Azimuth (EME2000)	93.7 deg	117.7 deg	150.1 deg
	Inertial Velocity	11.0474 km/s	11.0463 km/s	11.0456 km/s
	Eclipse Duration	57 minutes	No eclipse	No eclipse

Table 2. Characteristics of Representative Trajectories.

The entry date is one differentiating parameter among these solutions. However, they all satisfy the mission requirement to return in the Fall of 2003. Thus, entry date is not a factor in the choice, but other entry characteristics do help to eliminate various options. For example, the August 13th solution has a long solar eclipse on the entry approach leg that could deplete the on-board power supply. Furthermore, the large y-amplitude of this trajectory begins to approach the telecommunication capabilities during the halo orbit. Alternatively, the inclination required for a free-return to UTTR on August 19 violates the inclination constraint that has been specified in order to minimize the over-land groundtrack. Based on these criteria, the April 16 trajectory was selected as the best candidate for further analysis since it satisfies most of the mission constraints. Optimization of Case 2, in conjunction with the requirements to satisfy various launch constraints, resulted in the current reference trajectory presented in Figure 1.

Lunar Influence

Although it is not targeted, one of the most important parameters in the trajectory is the distant lunar passage on the first portion of the return leg (during April 2003). Trajectories that are not significantly influenced by the Moon during this portion of their departure from the halo orbit do not return freely to the Earth; trajectories that experience too large of a lunar gravity assist also do not return with the required conditions. The Moon is necessary to bend the unstable manifold sufficiently close to the Earth to produce a return trajectory. Establishing the range of trajectories that appropriately utilize this distant lunar passage, and understanding how those characteristics relate to mission design constraints, is one of the most challenging aspects of the trajectory design.

Figure 3 is a plot of the early portion of the return trajectory, from April 1, 2003 through about April 20, 2003. Time ticks at three-day intervals (labeled by the date in April 2003) identify corresponding positions along the lunar and spacecraft trajectories as the trajectories pass near the lunar orbit. This figure also helps to identify interesting alignments among the bodies during this time frame. For example, for Case 2, the baseline for the current reference trajectory, the Sun, Earth, Moon, and spacecraft are nearly aligned as the spacecraft crosses the x-z plane (y=0), while the Earth, Moon, and spacecraft align on other dates for other cases. The action of the Moon, leading the spacecraft through this region, is critical to establishing the proper timing and conditions to achieve the designed return conditions.

§ Mountain Daylight Time

** Earth-Mean-Ecliptic of J2000, Inertial Coordinates

†† Earth fixed, Prime Meridian of Date Coordinates

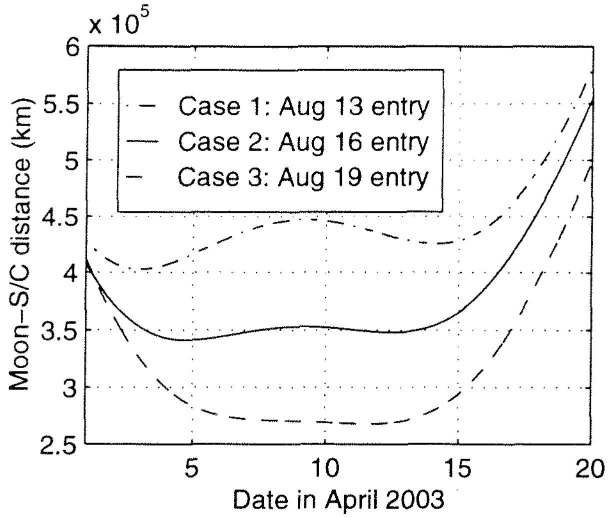

Figure 4. Lunar Closest Approach Distance During Early Part of Return.

SENSITIVITY STUDIES

Based on the August 16 entry reference solution, several perturbation studies were performed to evaluate the effect of changes in the force model imposed on the trajectory. The current reference trajectory (Figure 1) and the baseline from which it was developed (Case 2) were both computed with a force model that included only point mass models for the Sun, Earth, and Moon. To provide a more realistic prediction of the flight path, models for solar radiation pressure (SRP), 8x8 gravity fields for the Earth and Moon, and the gravitational effects of all of the additional planets in the solar system were included in the design. Also, models for all attitude adjustments, thruster calibrations, delta-v's from spin-changes, and other deterministic impulsive events were simulated. Finally, biasing maneuvers at the station-keeping opportunities in the halo orbit were modeled in order to verify the ability to design a trajectory that includes these biases and to evaluate their effect on the solution

To evaluate the effects of these perturbations, several trajectories were designed, each with a single type of perturbation (in addition to the baseline Sun, Earth, and Moon point-mass gravitational effects). The differences in the Earth range between each of these individual solutions and the reference solution are plotted in **Figure 5**. (In all cases, the difference is computed as the range on the perturbed path minus the reference path's range.) The "Oblateness" curve represents the difference between a trajectory that includes higher order gravity terms for the Earth and Moon, and the reference trajectory that models those bodies only as point masses. (Both trajectories model the Sun as a point mass.) The "Planets" curve indicates the difference between the reference trajectory and a trajectory that adds the gravitational effects of the other planets to the baseline model. The "SRP" and "Small Mnvrs" curves represent the differences between trajectories that include preliminary models for solar radiation pressure and small maneuvers, respectively, and the reference trajectory. Finally, the "Biases" curve is the difference between the reference path, in which LOI is the only maneuver, and a trajectory that adds 1.5 m/s biasing maneuvers to the model, at 60 day intervals in the halo orbit. Both the effects of the additional planets and the effects of the small maintenance maneuvers are shown with dot-dashed lines; however, the two curves are distinguished by the fact that the effect of the planets is very small throughout the trajectory (almost zero), while the effect of the maneuvers is significantly larger. The effect of oblateness is too small to be visible on this scale. Vertical dashed lines are included in the plot to highlight the times of various events such as the maximum y-excursion of the transfer leg, LOI, the end of each halo orbit revolution, and the time of an important return station-keeping maneuver.

trajectory be designed to return the vehicle to Utah from the perturbed path within the mission's delta-v capability?

Propagation

In the first step of this analysis, the nominal state just prior to the LOI maneuver was propagated without the LOI delta-v, to simulate the effect on the trajectory of a delay in the execution of the maneuver. A plot of the rotating x-y projection of the resulting path is presented in **Figure 6,** along with the reference trajectory (starting from LOI) for comparison. Time ticks are included on both trajectories at five day intervals. This simulation indicates that the perturbed trajectory (the trajectory that does not include the 22.5 m/s LOI delta-v that corresponds to the January 7 2001 launch opportunity) remains near the nominal for a few weeks, before leaving the L1 region.

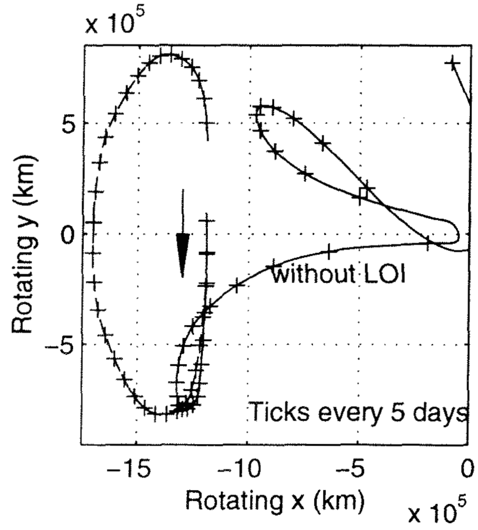

Figure 6. Trajectory Propagated Without LOI.

One important assumption in this simulation is that the initial state (the state just prior to the omitted LOI) is exactly on the nominal. That is, the only error source in the simulation was the missed delta-v; no other perturbations or initial dispersions were included. Of course, in an operational scenario, the spacecraft's state will include some error at the nominal LOI, which would result in a path with different divergence characteristics than those found in this study. However, the 22.5 m/s velocity error (that omitting the LOI delta-v represents) is considerably larger than the types of errors that would be expected as the spacecraft approaches LOI. Thus, it is expected that the perturbed path would remain relatively close to the nominal for at least a few days following an anomaly of this type.

Redesign

While this simple simulation provides a baseline for analyzing what would happen as a result of a delay in LOI, it does not address the most critical requirement for Genesis, which is to return to Earth, with appropriate entry conditions. Given the specific entry conditions that Genesis must meet, it is not sufficient to simply stay near L1 or even to be able to transition to an arbitrary halo orbit to complete the science collection. Genesis must, instead, be placed onto a path that not only satisfies the science and operational requirements but also returns to UTTR within the mission's delta-v budget. Thus, it is necessary to design a new halo orbit and corresponding return path, following a delay in LOI, that achieves acceptable entry conditions with acceptable delta-v requirements.

Although much of the trajectory design work used a simplified dynamic model, sensitivity studies indicate that the baseline trajectory can be refined with more complex models that include additional natural forces and additional deterministic maneuvers. Thus, although the design of the original reference solution required a sensitive search through a vast space yielding only limited acceptable solutions with the application of dynamical systems theory, once a nominal solution is found, the trajectory can be adjusted to accommodate reasonable perturbations to that design model.

Finally, although halo orbits are known to be inherently unstable, it is believed that a reasonable time frame exists to complete trajectory redesign efforts in response to various anomalies under operational scenarios. Potential recovery techniques for dealing with anomalies are being developed, and preliminary results are promising.

Much analysis of the Genesis trajectory remains to be done, including additional sensitivity studies, contingency planning, and operational testing of the design and analysis processes. But, the ongoing work of the Genesis trajectory design team continues to yield encouraging results for the mission and new insights for the study of this type of trajectory.

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