SUMMER LAUNCH OPTIONS FOR THE GENESIS MISSION

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In November 2000, the decision was finalized to delay launch of the Genesis mission from February 2001 until sometime in the following summer. Given the nature of libration point trajectories and the unique characteristics of the Genesis mission, a complete redesign of the trajectory was required. Thus, the ensuing effort was focused on establishing potential new baseline options and clearly defining and exploring the trade space. Once a specific baseline was identified, the trade space was refined to include such things as launch period characteristics, deterministic maneuver dates (to avoid DSN conflicts), Sun-Earth-Probe angle history, eclipsing prior to end-of-mission atmospheric entry, error ellipse footprint at collection site (Utah Test and Training Range), and favorable geometry to allow automated deboost maneuver prior to entry. The design procedure along with the application to the Genesis redesign is discussed and the new baseline solution is presented.

INTRODUCTION

The Genesis mission is the fifth Discovery mission to be selected by NASA [1]. The scientific objective is the collection of solar wind particles for analysis on Earth to determine the origins of the Solar System. The Genesis mission utilizes a near halo-type Lissajous trajectory about the Sun-Earth interior collinear libration point (L_1) during the sample collection process to ensure that the collected particles are pristine and unaffected by the Earth environment. After the solar wind samples have been collected in the vicinity of L_1, on the near side of the Earth relative to the Sun, Genesis will return to Earth via a loop that extends nearly to the Sun-Earth L_2 libration point on the far side of the Earth; the spacecraft reenters directly for an early morning mid-air retrieval at the Utah Test and Training Range (UTTR). The original Genesis trajectory, designed for a launch in February 2001, appears in Figure 1. (The trajectory is plotted relative the rotating coordinates, as is typical in the three-body problem. The x-axis is parallel to the Sun-Earth line, positive in the direction from the Sun toward the Earth.) After retrieval, the samples will then be curated and made available for scientific research during the decades to come. As commonly occurs in space mission planning, NASA made the decision in November 2000 to delay the Genesis mission from its planned launch in February 2001 until sometime in the summer of 2001. Given the nature of libration point trajectories and the unique characteristics of the Genesis mission, a complete redesign of the trajectory was required.

The redesign process that was utilized follows the design paradigm initiated by Howell et al. [2-4], with some modifications to allow certain aspects to be automated. The

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trajectory design is initially seeded with an initial guess obtained using techniques based in Dynamical Systems Theory (DST). Specifically, a Lissajous trajectory is computed in the vicinity of the Sun-Earth/Moon L_1 libration point. From this quasi-periodic solution, regions along the stable manifold are evaluated to identify a suitable candidate for the transfer leg from Earth to the Lissajous. Next, regions associated with the unstable manifold corresponding to the Lissajous trajectory are evaluated to generate an initial transfer arc that returns the spacecraft to the Earth. After the timing issues have been addressed (including the appropriate times of year for launch and reentry), the trajectory segments along the stable and unstable manifolds are patched together with a pre-determined number of revolutions along the Lissajous orbit to produce a set of trajectory arcs that can be used to initiate the next phase of the design.

This next phase, then, enforces any desired constraints and generates an end-to-end trajectory that is continuous in position with the potential for some optimum number of deterministic maneuvers. This part of the process began as a means to generate quasi-periodic Lissajous trajectories [5] and has evolved in recent years to include the ability to place constraints/maneuvers at various locations throughout the trajectory [5-7]. At its most basic level, the algorithm involves discretizing the trajectory into a finite number of arcs (defined by beginning and ending state vectors) that are used in a two-level iteration scheme to, first, force position continuity along the path, while introducing a series of velocity discontinuities into the solution. The second level of the algorithm reduces the magnitude of any undesired velocity discontinuities and enforces any constraints that may be placed on the trajectory.

This procedure, combined with the heritage of the authors in applying this methodology, both in general, and specifically to the Genesis mission, provided a means to quickly investigate options for a new baseline solution. This paper discusses the process used to generate the new baseline solution, the various options that were examined through the course of the redesign effort and, finally, the new trajectory that was ultimately selected.

INITIAL APPROXIMATION USING DYNAMICAL SYSTEMS THEORY

No trajectory design or computation can be accomplished without some viable concept, that is, some strategy to connect various trajectory arcs such that the mission objectives are ultimately satisfied. Thus, these initial designs are rooted in a working knowledge of the fundamental structure of the solution space. As the potential behavior of a spacecraft in three- and four-body regimes becomes better understood, the range of options and the design flexibility expand greatly; the efficiency and optimality of the design process is also boosted significantly. Although no analytical solution exists for motion in multi-body problems, various techniques have been employed to approximate solutions and to investigate trajectories in the Sun/Earth/Moon space [8-9]. Much insight has been gained in recent years by the application of mathematical concepts from dynamical systems theory. This approach has been very successful in designing the Genesis trajectory, both for the initial winter launch and now in the computation of summer launch options.

In support of trajectory design efforts for Genesis as well as a number of other missions, many stable and unstable manifolds for various periodic halo orbits have been numerically generated in the circular restricted three-body problem to further understanding of the phase space [10]. As an initial step in the Genesis redesign process, various types of such solution arcs are put together to construct a trajectory. Such an analysis produced the fundamental trajectory concept for the original Genesis mission design [2-3]. However, two issues emerge that impact the construction process. First, the initial approximation is ultimately used to generate a result in the "real" solar system; the methodology must accommodate this transition. Second, the design constraints may significantly influence the general size, shape, and overall characteristics of the trajectory. These constraints are loosely considered for the initial guess; they are tightly enforced in a later step.

Assumption of Periodicity

One of the fundamental characteristics that is exploited in applying dynamical systems concepts to halo orbits in the circular problem is periodicity. This periodicity, however, is destroyed when moving to a dynamical model that includes ephemerides for the locations of the gravitational bodies as well as other perturbations. Nevertheless, DST can still be very useful in the more complex models. The shift to a more complex model is facilitated by the selection of a quasi-periodic solution as the baseline orbit (even in the circular restricted problem), rather than an orbit that is precisely periodic. With the loss of periodicity, however, two options are available for computations: 1) compute stable and unstable manifolds for the tori on which the quasi-periodic trajectories are confined; or 2) assume that a Lissajous trajectory is sufficiently close to periodic that the algorithm provides an adequate approximation of the stable and unstable manifolds. Because the approximation used to generate initial conditions that represent stable and unstable manifolds is only first order, and because the primary purpose in using the manifolds is to supply a first guess for some other numerical procedure, the second option will suffice.

The next step requires the definition of a period associated with the motion, for approximation of the monodromy matrix. For the simply periodic halo orbit, the period is (obviously) one revolution. In the case of a Lissajous trajectory, the motion more closely repeats after two revolutions. Because the motion is not precisely periodic, it is necessary to define the "beginning" and the "end" of one period. In this study, the "beginning" of the period is defined at some specified xz-plane crossing, in rotating coordinates, that is above the ecliptic plane (i.e., positive z direction). The "end" of the period is then defined at the xz-plane crossing that is located approximately two revolutions later. Initial state vectors that are used to generate the stable manifolds are computed along the selected two revolutions of the Lissajous trajectory.

After generating the stable manifolds associated with various regions along the trajectory within the defined "period", one particular region along the two revolution Lissajous can be identified that is associated with the stable manifolds that pass closest to the Earth. Such a region along the Lissajous is visually indistinguishable from a similar region along a halo orbit [10]. In this region, there exists a trajectory on the stable manifold associated with a point very near the "beginning" of the Lissajous trajectory that serves as the initial guess for the transfer from Earth to the Lissajous orbit.

The return portion of the trajectory, i.e., the transfer from the L_1 Lissajous trajectory to Earth reentry, can be considered in much the same way. Recall that, since a day-side reentry is required, a direct return from L_1 is not feasible. The spacecraft must approach reentry from the side of the Earth opposite the Sun. Therefore, an *unstable* manifold must be generated that approaches the L_2 region before returning to Earth. It must also depart the vicinity of L_1 only after sufficient time to perform the science investigations has elapsed. Thus, further downstream along the same quasi-periodic Lissajous orbit, two revolutions are defined to represent the new "period" for computation of an appropriate unstable manifold. Investigation of the unstable manifolds along different regions of the appropriate revolutions reveals a region where the corresponding unstable manifolds have the characteristics necessary for the return. Specifically, an unstable manifold is required that reaches L_2 (comparable to a heteroclinic type motion [2]); the trajectory then must pass close to the Earth. Combining the stable manifold as the launch segment, the Lissajous trajectory, and the unstable manifold as the return segment provides the first guess for an end-to-end solution in the real system (that is, the model using ephemerides) for the Genesis mission. Note that the full model may also include additional perturbations, such as solar radiation pressure, as well as the daily precession maneuvers that are required during science collection to orient the spacecraft into the solar wind (for Genesis this is over 800+ maneuvers on the order of 2.5 mm/s per day).

REFINEMENT USING A TWO-LEVEL DIFFERENTIAL CORRECTIONS PROCEDURE

After the initial guess is determined by incorporating elements of DST, it is necessary to refine the trajectory to include any neglected force models, as well as to apply any desired constraints to the trajectory. This step in the design process closely mirrors the two-level differential corrections process as detailed in Howell and Pernicka [5]. A number of modifications to the original two-level scheme further generalize the procedure and allow constraints to be introduced into the trajectory [6-7].

First, given an initial approximation of the trajectory, both conceptually and as a numerical approximation, a series of target states along the trajectory is selected that includes the desired initial and final state vectors; these target states are called patch points. In this manner, the trajectory is effectively discretized into a finite series of state/epoch pairs that are used to define the trajectory segments that comprise the solution. At this point, the selection of the patch points is based more on experience, rather than formulated in terms of a set of hard criteria within the algorithm. Targets are generally selected to coincide with various critical events, such as deterministic maneuvers, periapses, etc., as well as at certain critical locations along the path. These critical points are now usually determined through experience, and over time, the authors have become familiar with the "hot spots" throughout the Genesis trajectory. (These critical locations sometimes become locations for trajectory correction maneuvers during the mission, as is the case during the return to Earth phase.) As understanding of the dynamical behavior in this regime improves, the selection process will, no doubt, become more automated.

Once the patch points have been selected, the first level of the differential corrections process is initiated. In this step, each trajectory segment is propagated between consecutive patch points. A simple targeting scheme using variations in velocity at the beginning of each segment is employed to achieve the desired position at the end of the segment. This process is applied to each segment sequentially to ensure position continuity throughout the entire trajectory. As a result of the procedure to ensure position continuity, however, velocity discontinuities are introduced at the end of each segment (excluding the final segment). Thus, there are now a total of *N*-2 velocity discontinuities in the trajectory after application of the level-one step, where *N* is the number of patch points selected.

The next step in the two-level scheme is to determine a set of changes to the patch points that will simultaneously reduce these velocity discontinuities. This is the original functionality that was proposed by Howell and Pernicka [5]. It has also been determined that the level-two step is also a convenient stage in the algorithm to enforce any desired constraints into the trajectory, such as launch conditions, periapse targets, entry targets, etc. [4-5] The fundamental basis of the level-two step is the determination of a linear system that approximates the relationship between the velocity discontinuities plus constraints and the independent variables in the problem, namely the positions and epochs of each of the patch points. In symbolic form this can be represented as

$$\begin{bmatrix} \delta\Delta\mathbf{V} \\ \delta\alpha \end{bmatrix} = \begin{bmatrix} \dfrac{\partial\Delta\mathbf{V}}{\partial\mathbf{R}} & \dfrac{\partial\Delta\mathbf{V}}{\partial t} \\ \dfrac{\partial\alpha}{\partial\mathbf{R}} & \dfrac{\partial\alpha}{\partial t} \end{bmatrix} * \begin{bmatrix} \delta\mathbf{R} \\ \delta t \end{bmatrix} \quad . \tag{1}$$

where bold indicates vectors. The vector on the left side of the equation is the target vector composed of the velocity discontinuities ($\Delta\mathbf{V}$) and constraint equations (α). This vector of dependent quantities is actually the difference between the desired targets ($\Delta\mathbf{V}$ and α) and the actual quantities evaluated from the trajectory data after the level-one propagation, hence the δ before the quantity. The vector on the right side is the set of independent variables available to achieve the desired targets. In this procedure, the independent variables are the positions and times associated with each patch point along the trajectory. The n by m matrix on the right side is termed the State Relationship Matrix or SRM and relates variations in the independent variables (positions and times) to the dependent target variables ($\Delta\mathbf{V}$ and α). The SRM is constructed from the state transition matrices for each segment of the trajectory, as well as other derived partial derivatives. (See Wilson and Howell [6] or Wilson [7] for the specific equations that are used to construct the elements of the SRM.) In general, there are more independent variables than there are target variables; this results in an underdetermined linear system. Using the smallest Euclidean norm, a set of changes to the independent variables can be determined using the relationship

$$\begin{bmatrix} \delta \mathbf{R} \\ \delta \mathrm{t} \end{bmatrix} = \mathbf{M}^{\mathrm{T}}\left(\mathbf{M}\,\mathbf{M}^{\mathrm{T}}\right)^{-1} * \begin{bmatrix} \delta \Delta \mathbf{V} \\ \delta \alpha \end{bmatrix} \quad , \tag{2}$$

where **M** denotes the SRM from Equation (1). Once the set of changes to the positions and the times corresponding to the patch points has been determined, the changes are applied to all of the patch points simultaneously. The level-one step is now repeated with the new set of patch points and a new trajectory solution is determined. This new trajectory should have smaller errors in $\Delta \mathbf{V}$ and α than the previous solution. However, due to the highly nonlinear nature of solutions in the *n*-body problem, this process is iterative. Repeated passes through the level-one and level-two steps are required until sufficient convergence is achieved.

The original implementation of these two algorithms was initiated at Purdue University as a number of separate Fortran programs written by a series of graduate students, including the authors. In the current implementation utilized at JPL, these algorithms have been rewritten in a unified programming environment called LTool. This environment (written in C++ and Python) allows the entire trajectory from initial approximation to final solution to be designed in an efficient, more automated fashion that highlights the power of these methods working in concert with each other. In particular, the implementation of the two-level differential corrector in LTool allows the entire trajectory to be considered from launch to reentry, with constraints and/or velocity discontinuities available for placement at any patch point along the trajectory. This procedure is highly robust, even incorporating poor initial guesses and difficult constraints. It should be noted, however, that the solutions that emerge from the dynamical systems portion of the design procedure tend to be excellent starting trajectories and the two-level differential corrections process rarely has difficulty converging to a solution, if one exists. Although DST is extremely useful in determining initial approximations to the trajectory from the vast array of trajectories in the solution space, the two-level differential corrections scheme is critical to developing a complete solution that meets all of the design criteria and accurately reenters the sample return capsule to reach the target site in Utah. In the next section, the application of this design philosophy to the Genesis redesign effort is thoroughly discussed.

OPTIONS FOR A SUMMER LAUNCH

The process described in the previous section for conceptualizing and generating initial "guesses" serves as the basis for seeding the actual search for a suitable trajectory using the two-level differential corrector. The results of this search process produced two distinctly different options that are presented in this section. The redesign effort begins with the original conceptualization of the trajectory design (Howell et al. [2-3]), where the impact of the scientific goals and spacecraft capabilities has already been incorporated. While a solution that is predisposed to the previous design is not a necessity, it is possible that a launch during a summer month does, in fact, result in a trajectory that inherits many or all of the same physical and/or geometrical constraints and characteristics of the previous winter launch trajectory options. Additionally, with the added requirement for a rapid development cycle (on the order of a few months), it is reasonable to consider a solution with similar geometry, if one is available, as a potential option.

Summary of Winter Launch Characteristics

Given the design parameters and the urgency demanded in producing a new launch date, it is advantageous to examine the previous baseline solution that was proposed for launch in February 2001 and returning to Utah in September 2003. The requirement to accommodate 22+ months of required science data collection translates into a libration point trajectory including approximately four revolutions as viewed in the *xy*-plane of the Sun-Earth/Moon rotating coordinates. See Figure 1. The combined impact of the various requirements, along with the timing considerations along the various arcs that comprise the Genesis trajectory, result in a mission that launches from the Kennedy space complex on February 10, 2001, and injects into a northern (class I) type Lissajous trajectory. After approximately four revolutions in the libration point orbit, the spacecraft departs the vicinity of L_1 moving outside the lunar orbit toward the region defined by the Sun-Earth L_2 point before returning to the Earth for eventual entry and capture at the Utah Test and Training Range (UTTR). The total length of this mission is approximately 31 months.

If the new baseline trajectory, one that includes a launch in the summer, can capture the same general features that are reflected in the winter solution, then, many of the inherent constraints and requirements are readily accommodated. However, due to the project constraint that the spacecraft return to UTTR during the months with favorable weather conditions (May-October), the total duration of the new baseline mission required a time of flight adjustment to a value closer to three years. This can be accomplished in numerous ways, but two specific options were considered that initially fell within reasonable operational boundaries. The first option is presented in Figure 2. Note that this trajectory is computed using a full ephemeris model. While there are still approximately four revolutions about the L_1 libration point, the most obvious distinguishing characteristic between this solution and the winter month solutions is what is termed the "Earth loop," i.e., the additional loop around the Earth, yet beyond the lunar orbit, as the spacecraft approaches L_2 and subsequently returns to Earth. This behavior is due to an initial guess for the return segment that emanates from a region of the unstable manifold that is still associated with the same Lissajous trajectory, but that differs from the region of the unstable manifold that was exploited to obtain the winter solution. Additionally, because of a combination of the timing issues governing solutions in this region of space, as well as the launch and entry geometries, the second noteworthy difference between this solution and the winter solutions is that this one is based on a southern (class II) Lissajous trajectory.

The second legitimate option to be considered, as presented in Figure 3, strongly resembles the winter end-to-end solution for a February launch. Indeed, it is actually a member of the same family of solutions. Notice the return to a northern (class I) L_1 Lissajous orbit as compared to alternative summer launch in Figure 2. The difference between the February launch option and this launch in late July is the increase in time spent in the vicinity of L_1. In fact, due to the requirement of both a summer launch and a summer return, the path now includes approximately five revolutions about the L_1 point that must now be factored into the mission time line. (Ultimately, this increases the margin for science collection time by about six months—an option that was highly supported by the Principal Investigator.)

In evaluating the two design alternatives, the Earth loop option was quickly dropped in favor of the 5-Revolution (5-Rev) option, due to the similarities in geometry with the winter baseline, as well as the increased margin for science collection. Initially, a June launch date was preferred, however, due to launch conflicts with other missions during the late June time frame, the June/July launch period was soon dropped in favor of a late-July to mid-August target date for launch. The 5-Rev option was preserved and all efforts were focused on producing a viable trajectory that met all of the design criteria. From a representative analysis across 10 days, given the 5-Rev assumption, the solution space for a summer launch appears in Figure 4. Note that this "family" of possible solutions was determined based on one known solution that is then modified to produce various characteristics as a function of the placement and magnitude one maneuver at LOI.

Trade Space Development and Final Summer Baseline

As with every mission, the trade space of design criteria can quickly become complicated and seemingly self-contradictory, possibly making it difficult to determine a solution that satisfies all constraints. As such, the ability to strategically identify the key elements, or dimensions, of the trade space provides a metric for judging the merit of any given solution. More simply expressed, it is desirable to reduce the dimension of the trade space as much as possible to eliminate those solutions that are unusable. For the Genesis mission, these dimensions include characteristics of the launch period (including total cost), geometry relative to communications requirements, shadowing penalties prior to reentry, and a reentry geometry that facilitates a contingency situation requiring an automated deboost maneuver.

The launch period is defined as the duration (typically in days) over which the spacecraft can be launched into an acceptable trajectory (not to be confused with the launch window that defines the window of opportunity around a specified time of launch on a specified day). Practically speaking, there can be only one launch opportunity each day throughout the launch period. However, in order to initially increase the range and density of solutions in this study, it was assumed that launch can occur at any time during the day. Thus, to the total cost (in terms of ΔV), as a function of launch date, can be explored and monitored with greater resolution.

A key design decision that was made for the winter options was retained for the summer months as well; that is, the decision to have a single Lissajous and return that remains fixed after LOI across the entire launch period. In adopting this approach, the only variation in the trajectories across the launch period is during the transfer phase from launch to LOI. This simplification allows a single return to be fully analyzed and flown, which greatly reduces the risk to the mission from a navigation perspective. Thus, the launch period can be characterized by a simple metric of ΔV cost versus launch date. It is desired to have a minimum of 14 contiguous days of launch opportunities. It is also desired that the maximum Lissajous Orbit Insertion (LOI) cost for any launch period should be less than 50-55 m/s. The penalty for this decision is the potential for larger LOI maneuvers and a shorter launch period than would be possible if each individual launch could be optimized through to reentry. One example of how the launch period factored into the trade space is shown in Figure 5. The selected period opens on July 30, 2001 with an LOI of approximately 53 m/s and terminates on August 14, 2001 with and LOI cost of about 14 m/s. The end of the launch period is defined by the increasing interaction of the Moon on the transfer trajectory. The design decision made previously in the winter options to avoid the Moon was retained here as well. Hence, the launch period for the baseline mission covers 16 days from July 30 through August 14. The figure shows a series of curves covering the launch period. Each curve represents a variation in the LOI target location, and thus represents a slightly different Lissajous and return. Once an LOI target is selected, the data that is represented in this figure can be generated automatically using the LTool computing environment. From this plot then, the LOI target on November 16, 2001 (lowest solid black curve) was selected to provide the longest launch period for the lowest overall cost. Note that the earlier LOI date was also selected to avoid conflicts with other missions and DSN scheduling.

Arguably one of the more critical components of the trade space is the geometry between the Sun, Earth, and spacecraft. The medium gain antenna is mounted on the aft of the spacecraft and has a boresight angle of approximately 35 degrees. Since the spacecraft will be pointing into the solar wind (4.5 degrees ahead of the Sun-spacecraft line) throughout the science collection period of the mission, the MGA will generally be pointed towards the Earth. Hence, it is important that the Sun-Earth-Probe angle not exceed 30 degrees (although toward the end of the redesign effort, this constraint was relaxed). Additionally, it is critical to avoid the solar exclusion zone so that communication is not disrupted due to geometries that require pointing from the Earth directly toward the Sun. That established minimum, while dependent upon where in the trajectory this occurred, is valued at five degrees. Although no active constraints were placed on the trajectory during the computations, the selected baseline does meet these geometric constraints during the science phase of the mission (during the Lissajous portion of the trajectory).

Also of critical importance is the amount of shadowing the spacecraft is subject to prior to reentry. This becomes an issue in particular when considering the contingency situation where the direct entry would be aborted and a significant capture maneuver at perigee would be required to drop into a phasing or backup orbit that allows a second attempt at reentry approximately three weeks later. Such a situation requires significant battery power, thus limiting the amount of permissible time in the shadow just prior to such a maneuver. A maximum of 80 minutes of shadowing limits the time spent in the umbra or penumbra. (Note that lunar eclipsing was also considered but was not an issue.) Much effort went into minimizing the eclipsing at reentry while still achieving suitable atmospheric entry conditions that allow the sample return capsule to reach the UTTR landing site. Ultimately, a solution inclined at 52 degrees to the equator was selected to eliminate all shadowing and still reach the proper atmospheric entry conditions. Note that a lower inclination is more desirable due to topographic constraints around the UTTR recovery site, however this desire could not be met without introducing eclipsing problems into the solution.

Finally, near the end of the redesign effort, a constraint was added to the trajectory that was not considered during the previous design efforts. In particular, it is desired to have a geometry that, in the unlikely scenario where the spacecraft must perform an automated deboost maneuver just prior to reentry (forcing the spacecraft to reenter and plunge into the ocean), a maneuver could be performed without significant realignment of the spacecraft, while remaining within the spacecraft capabilities. This constraint was at times in conflict with the previous shadowing constraint, which made for a difficult trade space. In the end, the selected baseline does meet the desire to have a particular automated deboost attitude, albeit marginally, without sacrificing shadow exclusion or affecting the other constraints.

Backup Orbit Design

An additional feature of the Genesis trajectory that required at least a preliminary consideration was the backup orbit. Specifically, in a contingency situation where it is not desirable to bring the spacecraft back to Earth with the direct entry, a small maneuver of approximately 9.0 m/s at 12 hours prior to nominal atmospheric entry can be used to raise perigee altitude to 200 km. Once at perigee, the spacecraft would then execute another maneuver to capture into a highly eccentric Earth-centered orbit. A subsequent maneuver would then be needed to retarget to the appropriate entry conditions for a second attempt at atmospheric entry. As with the nominal trajectory, the process for designing the backup orbit begins with an initial guess which is then used to initialize a differential corrections process that forces a given set of constraints. In the case of the backup orbit, the source of the initial guess appropriately comes from conic sections instead of techniques based in Dynamical Systems Theory. Otherwise, the process is exactly the same.

The basic concept of the backup orbit is a simple elliptical orbit about the Earth, giving the operations team time to assess whatever caused the abort of the primary reentry attempt and to prepare for a second entry opportunity. The primary variable in the search for an appropriate solution is the period of the orbit. Shown in Figure 6 is the current baseline option that is based on a 24-day orbit. As with previous figures, the trajectory is presented in the Sun-Earth rotating frame centered at the Earth. The divert maneuver along with the perigee and subsequent apogee maneuvers are all identified on the plot in the usual way. The total ΔV for this contingency is 102.1 m/s. Other options with shorter periods were also considered. However, with a shorter orbital period came much greater difficulty in meeting all of the entry constraints in a scenario that was within the spacecraft capabilities, i.e., often times the ΔV was prohibitively large. Other trends were identified where shorter orbital periods resulted in significantly reduced entry inclinations. This was not desirable due not only to shadowing concerns, but also because of drastic variations in the footprint at the collection site.

CONCLUSIONS

The redesign effort for the Genesis mission was deemed a success at many levels. The speed with which the myriads of options were presented to the project was crucial in providing the project and NASA with the flexibility that enabled the mission to proceed with only a six month delay. The efficiency of the process for generating options and defining and exploring the trade space is the result of years of experience of the mission designers with the project and, as importantly, a well-developed computational process. This process, as described previously, is based on obtaining an initial guess from any suitable source, then applying differential corrections techniques to enforce the mission constraints. Once a trajectory has been established that has met all mission constraints, the exploration of the trade space allows for a more informed and detailed choice for a nominal solution, using differential corrections techniques often times in an automated fashion.

Because of the numerous and changing constraints placed on the trajectory (including a precise Earth return, spacecraft limitations, and over 800+ daily precession maneuvers, as well as various maneuver biasing schemes), the Genesis mission design provided a challenging scenario to address, but one that has been worth the effort. In a very real sense, the Genesis mission design helped to define and refine the methodologies used to determine the trajectory, as evidenced by the multitude of papers and two Ph.D. dissertations related to the mission design. And synergistically, the refining and honing of these techniques allowed the redesign of the mission to be accomplished at an extreme pace to meet the pressing needs of the mission.

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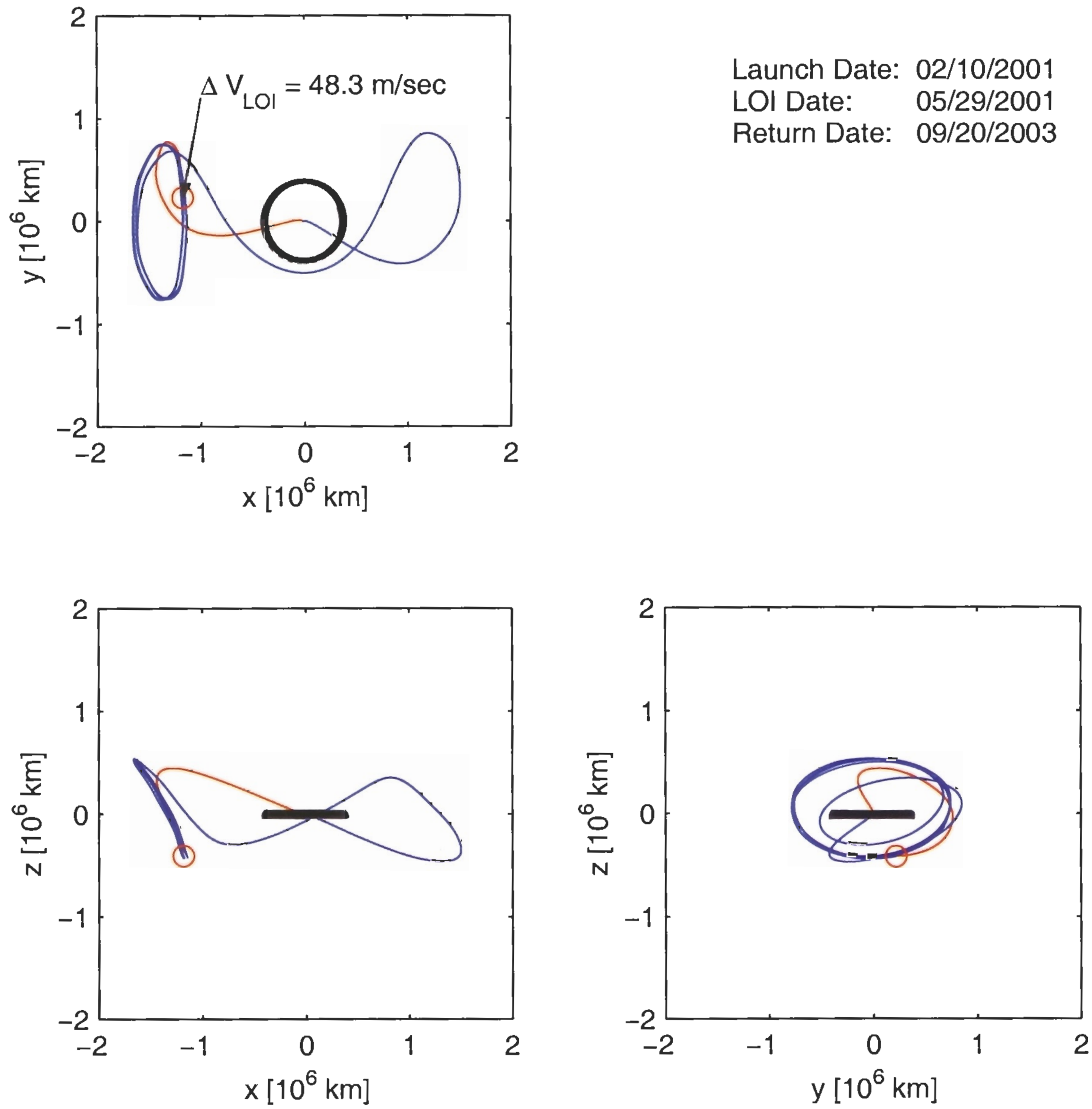

Figure 1 February Baseline Trajectory -- Sun-Earth/Moon Rotating Frame

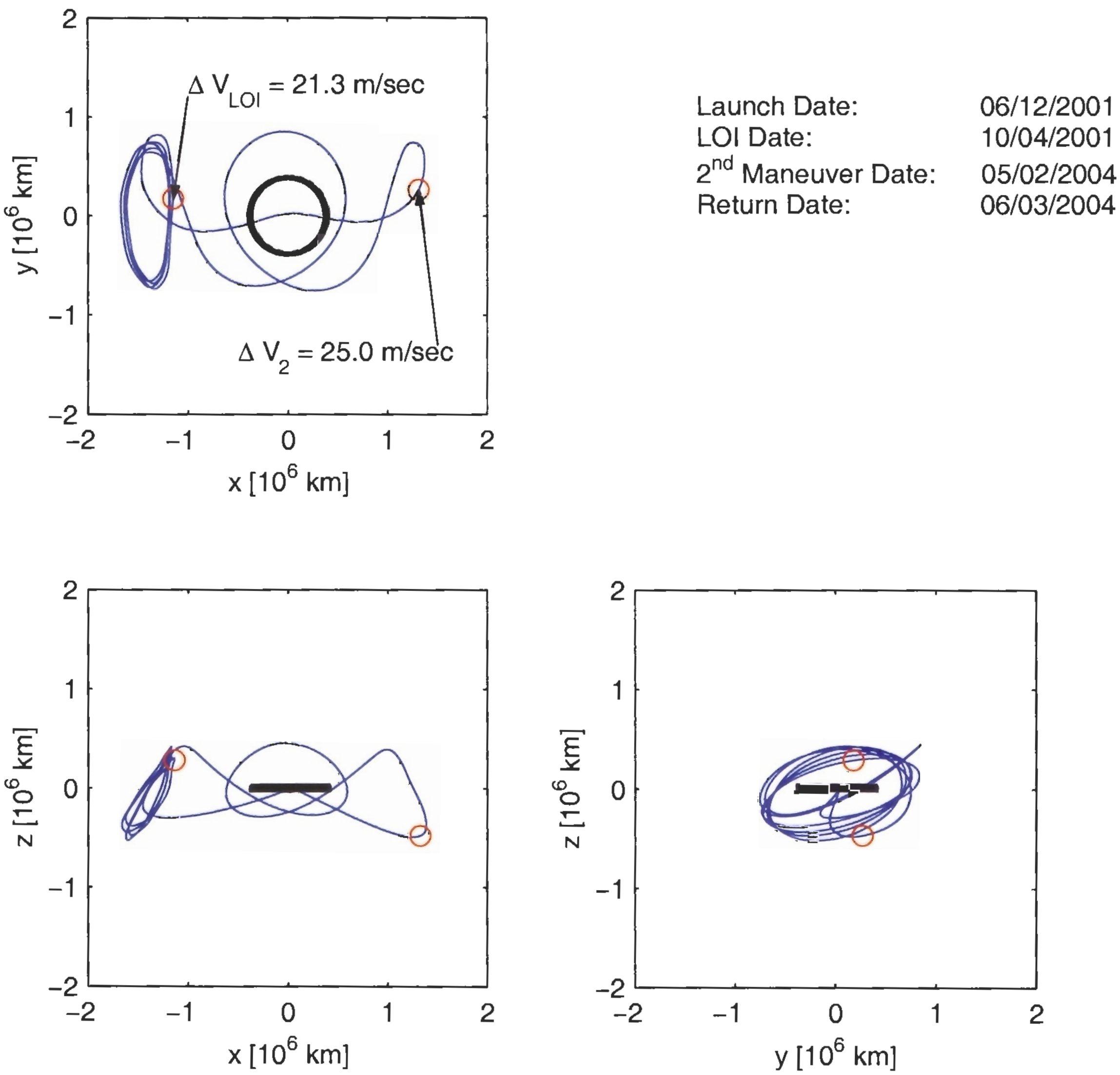

Figure 2 June Earth Loop Option

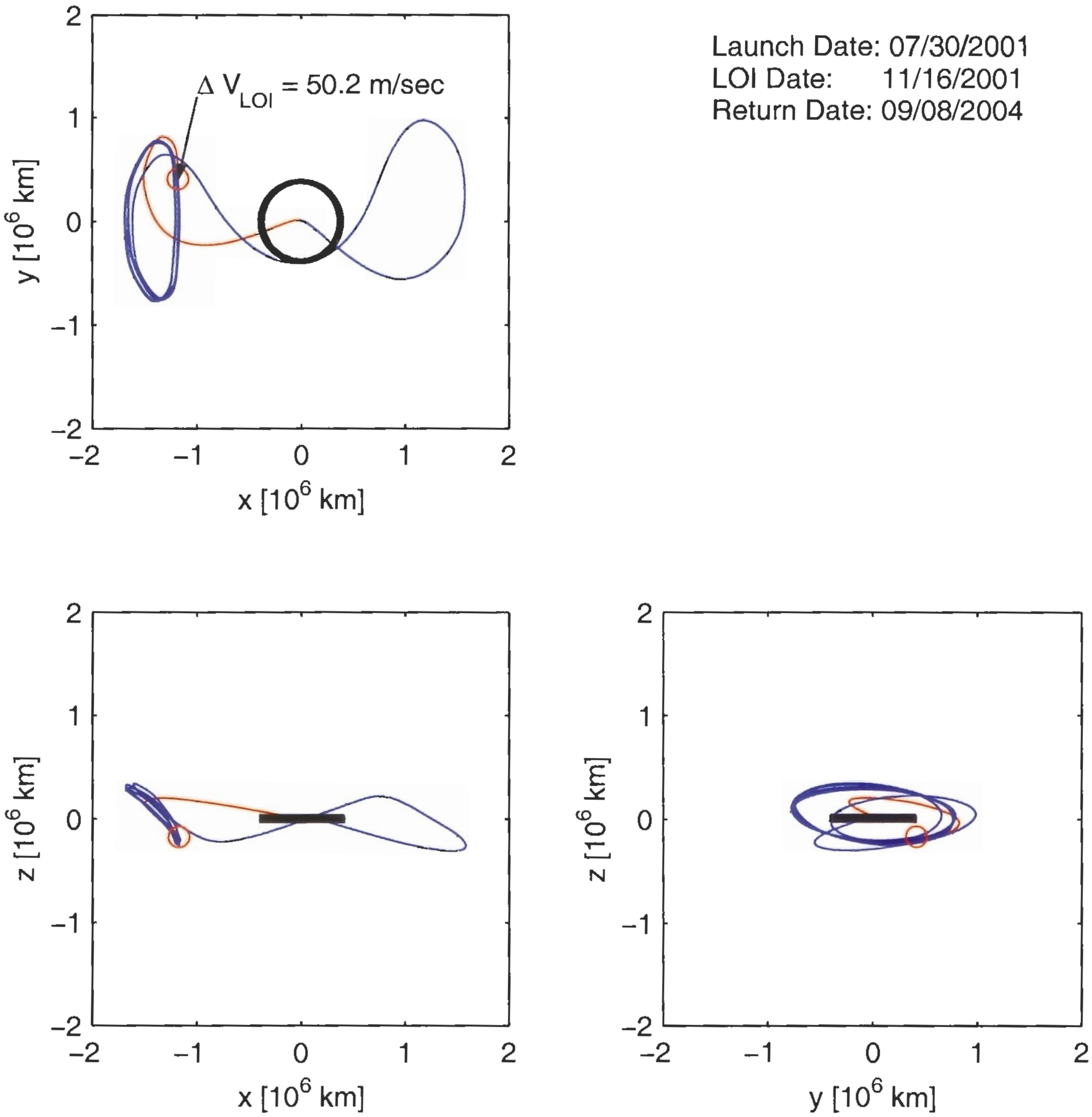

Figure 3 July/August 5-Rev Solution

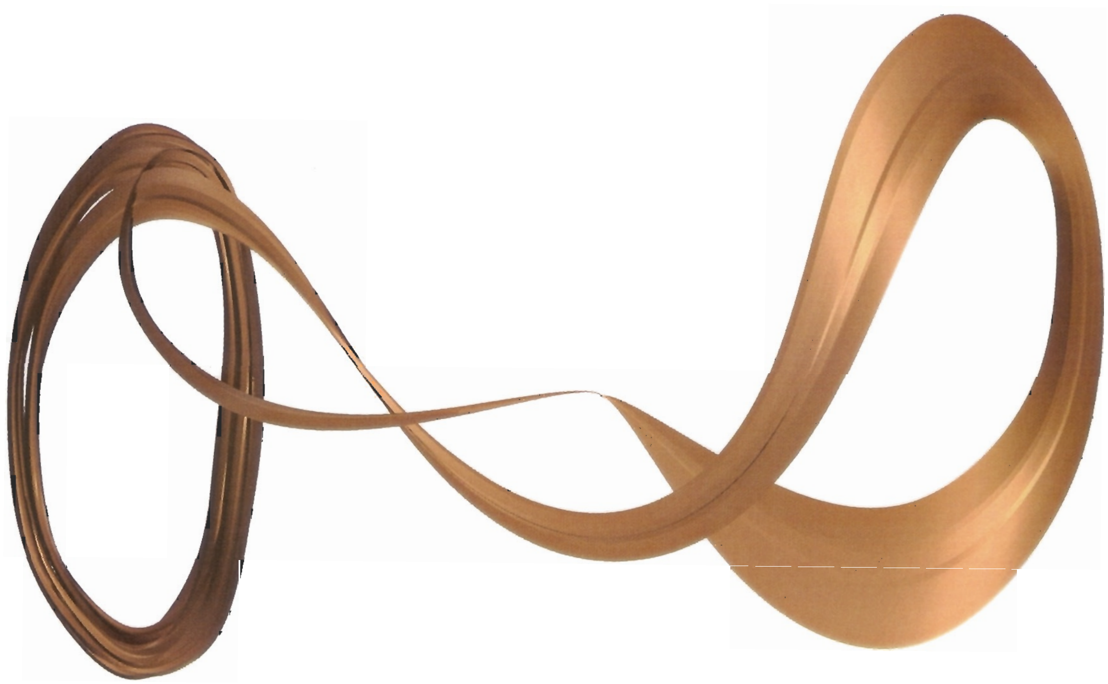

Figure 4 Solution Space for Potential Summer Solutions

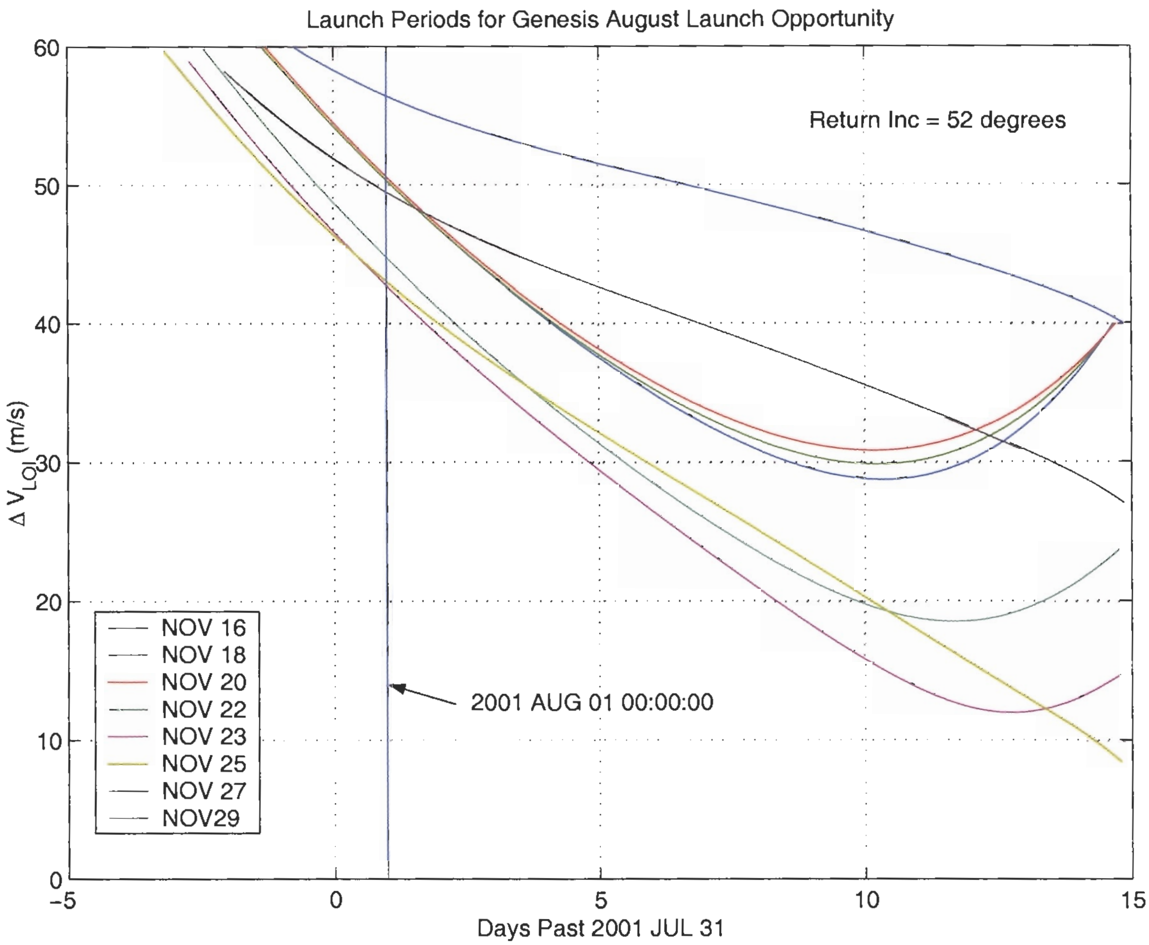

Figure 5 Launch Period Curves for July/August Solution

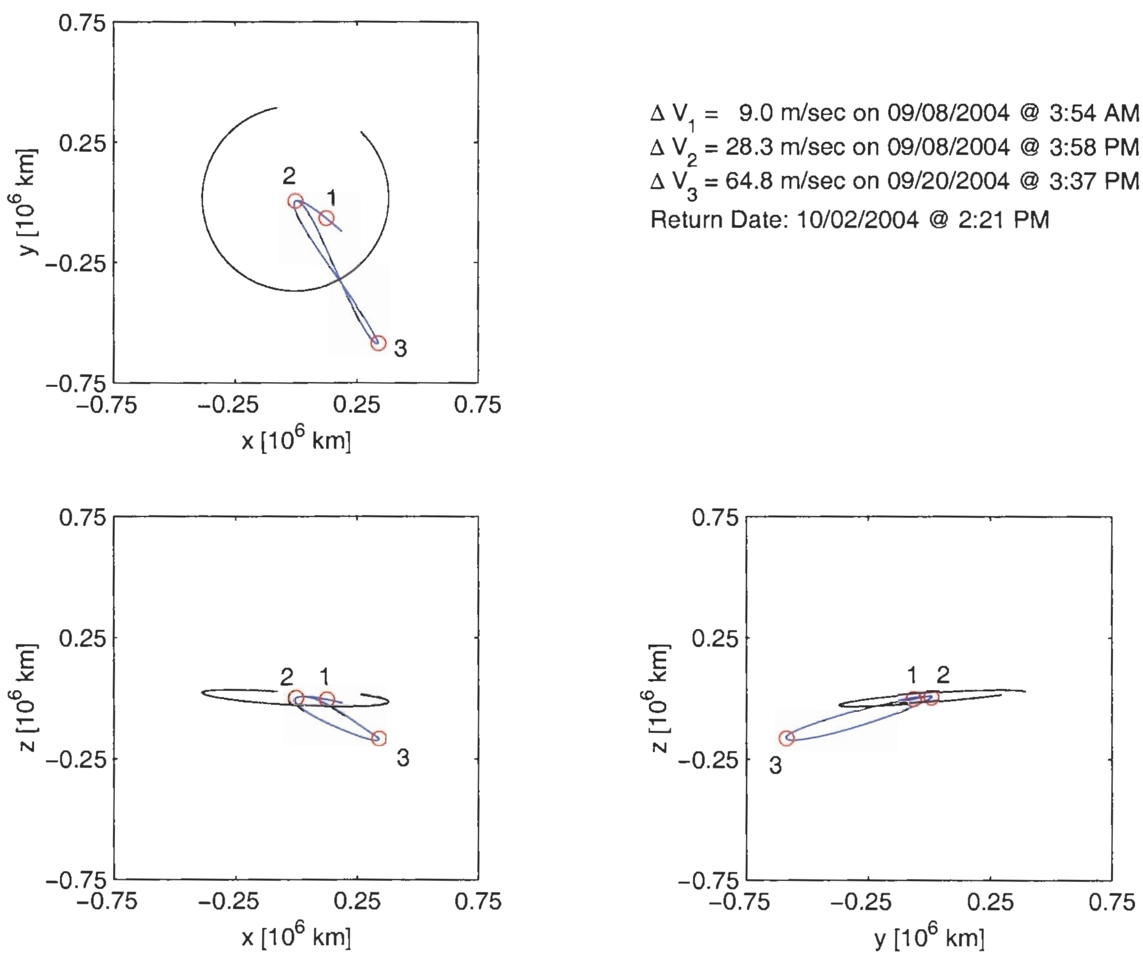

Figure 6 Backup Orbit for July/August 5-Rev Solution