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PLANETARY MISSIONS FROM GTO USING EARTH AND MOON GRAVITY ASSISTS*

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ABSTRACT

A general method has been developed which permits the transfer of a spacecraft from a highly elliptic and generally oriented Earth orbit to a specified escape direction and energy. The permitted relationship between the ellipse size and orientation with respect to the escape vector is flexible enough to allow a several month launch opportunity, for GEO launches for example. Applications currently being considered by JPL are launches of secondary payloads to Mars and other destinations from the Ariane geosynchronous transfer orbit (GTO). This GTO orbit is highly constrained to have a low inclination to the Earth's equator, and an apogee arrival at noon local time. The velocity cost to provide this enabling capability is 100 to 300 m/s above that required if the same elliptic orbit were ideally oriented and the maneuver were done at perigee, for the escape direction required. This paper describes the Ariane launch and orbital constraints, the proposed multi-burn escape strategy, the computer program modeling, and application of this program to Mars and Venus missions.

INTRODUCTION

The current thrust of technology is to move the NASA space program into using smaller and more capable spacecraft, and at a lower cost. Launch costs, on the other hand, are difficult to lower because most deep space missions require specific launch dates and Earth escape directions. Therefore, they are usually assigned dedicated launches. This no longer needs to be the case. Launch costs can now be reduced by up to 80% by flying small planetary spacecraft as secondary payloads, and this service will be offered by the French on the Ariane 5 using the Ariane Structure for Auxiliary Payloads (ASAP).[1]

Although the escape procedure presented here is quite general, it will be applied specifically to the Ariane 5 GEO launches to commence in 2001. About 8 to 12 such launches are expected to be flown each year. Specifically, the ASAP ring can accommodate eight independent 100 kg payloads, or up to 200 kg if adjacent slots are used. After release of the primary payload(s) at GEO altitude, these piggyback payloads will be released into the GTO to fend for themselves. For the purposes of this paper, the GTO is assumed to be equatorial, with perigee at 200 km, apogee 35900 km, and the major axis to be pointed at the sun on the day of launch. In addition to the high orbit eccentricity to be dealt with, the problem is further complicated by the fact that the Ariane launch could be at any time within a 2-3 month period.

Initial attempts at solving this problem were made by this author more than a year ago[2], but none of the procedures were completely satisfactory, let alone general. This reference, however, does provide some insight into the characteristics of the problem and hints at the solution. Specifically, the following were learned:

1. In general, a single burn solution is unsatisfactory, especially if a 2-3 month GEO launch period is desired.

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2. Major propulsive maneuvers need to be made in the Earth gravity well for best performance (e.g., at GTO perigee).
3. A high ellipse, reaching beyond the Moon, and possibly a lunar flyby, are necessary.

4. For the high noon GTO ellipse, escape outward (e.g., towards Mars) is easier than escape inwards (towards Venus), since a burn near GTO perigee would send the spacecraft in the direction of Earth's velocity vector.

5. Solar perturbations need to be considered for the high ellipse option.

All of these considerations eventually led to the general 3-burn high ellipse Moon-Earth Gravity Assist (MEGA) process for GTO escape.

THE MEGA 3-BURN STRATEGY

Before describing the 3-burn method of transferring from GTO to escape, it is useful to define a few simple terms (GTO has already been defined):

Launch Date: This date will always be considered the Ariane liftoff date, or the date when the GTO apogee is at high noon.

Escape Vector: To reach a target body, a spacecraft must escape Earth's gravity and have a residual velocity and direction (a vector, which gets added to Earth's velocity vector) which will cause it to encounter the desired body after coasting for a period of months.

Departure Period: Applied to planetary missions and other bodies beyond the Moon, this is the length of time, usually 2 or more weeks, when Earth and the target body are in favorable alignment (producing the lowest total escape velocity requirement) to launch the spacecraft, and begin its journey to the target body. For Mars, the next favorable departure period is January 6-26, 1999.

Departure Date: A specific Earth escape date within the departure period. For two body transfers, the departure date and the target body arrival date determine the Earth escape vector and the body arrival vector.

The goal of the 3-burn strategy, in terms of minimum propellant usage, is to perform all major burns at LEO altitude to ensure maximum use of Earth's gravity well. Specifically, with no consideration of radiation belt sensitivity, or finite burn losses (these are discussed later), the sequence of events are the following (the numbers are approximate):

1. At some predetermined time after GTO launch, perform the first burn at GTO perigee in the orbit plane to enter a high ellipse of between 700,000 km to 1,500,000 km apogee distance. This will require 720 to 750 m/s delta-v out of the GTO.

2. At apogee, which may take 12 to 40 days to reach, perform a second burn to target the Moon on a predetermined date for a lunar swingby back to Earth. This second burn is used to rotate the orbit plane and change the velocity magnitude only (i.e., the flight path angle is zero). This will require from 0 to 300 m/s delta-v.

3. The lunar encounter will occur roughly 10 to 38 days from the second burn, and the lunar flyby inclination and altitude will be such that the return to Earth will have a specified LEO altitude (say 300 km), and a return inclination such that the third burn performed at perigee will result in the required Earth escape vector on a date within the departure period (also predetermined). This will require from 430 to 490 m/s delta-v for an escape energy of 9 km^2/s^2.

As will be seen from a specific example, the Mars 2003 opportunity, the total delta-v requirement will range from about 1200 m/s to above 1400 m/s. This is considerably less than the 3500 m/s required to escape with the same energy from LEO.

Concerning the generality of this method, two important points should be made. First, since the third (escape) burn must be within the departure period, which may be about 3 weeks long, the variable time on the high ellipse (from 25 to 80 days) may be used to provide 60 days or more for the Ariane launch. If the launch is early, then the high ellipse may be higher to take up the longer time, so that arrival at the Moon is near the departure date. For a later Ariane launch, time to the departure date is shorter, so the chosen high ellipse will be smaller. This provides a large launch period for the Ariane. It may be made even longer by launching very early, and waiting in GTO, or an intermediate orbit, before commencing with of the high ellipse burn. Constraints on this process are discussed below.

The second point, is that the flyby of the Moon, by varying the altitude and inclination, can be such that the spacecraft will return to Earth with a desired inclination and altitude, as necessary, to perform the third burn to escape. The escape velocity magnitude will be determined by the length of the burn, and the direction by the Moon's location and the inclination with the Earth.

From this sequence, it is clear that there are four free flight segments. These are:

1. GTO perigee to the high ellipse apogee

2. High ellipse apogee to the lunar encounter

3. Lunar encounter back to Earth perigee

4. Earth perigee to escape

The first and second segments require that solar perturbations must be taken into account, but an initial calculation can be made using conic segments. The second and third segments can be conics but with a discontinuity at the Moon where the second and third segment meet. Computing these segments such that their velocity magnitudes relative to the Moon are the same, allows the flyby altitude and inclination to be computed. This approximation can then be refined by additional multi-conic calculations near the Moon.

THE MEGA3 PROGRAM

The programming choice of first computing conic segments, and then refining the calculations interactively allows homing in all four segments quickly. Then, once the complete trajectory is computed and satisfies the escape conditions, the GTO launch date may be changed by a few days, and the process repeated to get a new solution. The first solution becomes the first guess for the second and, repeating the process, the 3-burn requirements can be computed for the available GTO launch period.

There are a few preliminary computations that should be made first to ensure that MEGA3 computes a feasible initial solution. First, the set of required escape vectors should be computed for each day of the desired departure period. These vectors are usually in the form of energy (C_3), and Earth ecliptic latitude and longitude. Since these vectors are a function of the target arrival date as well as launch date, a fixed arrival date will be assumed for each launch date.

Second, looking at the vector longitudes, the Moon's longitude should be approximately 50° greater than the escape vector's value for posigrade motion about Earth after the lunar flyby, or 50° less for a retrograde flyby. A lunar ephemeris should be available so that an approximate lunar date may be chosen. The Moon, by the way, moves much faster per day in longitude than the escape vector. This choice of the lunar flyby date will then fix the departure (or escape) date to be about 3 days later.

The computer calculations can now begin by choosing, say, the GTO launch to be 60 days before the lunar encounter, and the high ellipse apogee to be one million kilometers. This is enough information to compute conic estimates of the first two segments, where the user can vary the apogee distance to get the correct lunar flyby date. The user has the option of including or not including solar perturbations.

With these two segments computed, the user can input a set of Earth return inclinations, and compute the return segment which results in a 300 km perigee of Earth. An in-plane third burn will be made at this perigee for the input C_3, resulting in a specific escape latitude and longitude for each return inclination. The inclination chosen would be the one which gives the best latitude match with the escape latitude. The escape longitude depends mostly on the Moon's position, which depends on the input time of the lunar flyby, and this time can be changed by days or hours, as necessary, to get a best match with the escape longitude.

It is seen, by this interactive process, that the user has tight control of the calculations performed, which should be the case with a computation as complex as this. Further details on the MEGA3 program are beyond the scope of this paper.

MISSION AND SPACECRAFT CONSIDERATIONS

Small spacecraft missions flown piggyback using GTO could include balloons, penetrators, probes, relay satellites, and landers to Mars, Venus and near Earth asteroids and comets.[3] The preferred spacecraft design should include a bi-propellant with about 320 second specific impulse, with attitude, and possibly spin, control. For Mars and Venus, the propellant loading should provide about 1600 m/s capability, which implies a mass fraction of about 40% propellant. To perform the large burns required, the engine should be capable of about 100N thrust. This is needed to avoid large gravity losses, and loss due to fixed burn orientation, particularly for the final escape burn which must be performed in one piece.

The first burn may be divided into 3 parts, with the first getting into a one-day orbit, the second into a 3-6 day orbit, and the third going to the high ellipse. For early GTO launches, where a waiting period prior to entering the high ellipse is necessary, the longer intermediate periods will considerably reduce radiation effects on instruments due to radiation belt passages.

MARS MISSIONS

Application of the MEGA process to upcoming and new Mars missions is being actively studied here at JPL. This being the case, the MEGA3 program was used to generate data for the next 3 Mars opportunities 2001, 2003, and 2005. Typical views of the MEGA trajectory profile for these 3 years are given in Figure 1.

Mars provides a good example for illustrating the versatility of this method. The Earth in these figures is a dot (not shown) and the small ellipse within the Moon's orbit is the GTO. For the given departure dates and the Mars arrival dates, the escape latitudes are -50°, -9°, and 37°, respectively, for the three Mars launch opportunities. These are Earth equatorial latitudes of the escape vector at the time of departure. The MEGA process is sufficiently robust to handle these variations.

These high latitudes are made possible with the use of the lunar flyby. In 2001, for example, the flyby is under the Moon which causes the Earth bound trajectory to fly over the northern hemisphere and hence reach a significant southern latitude. This latitude would not be accessible for a typical Earth based launch.

In addition, there is good range in the GTO longitudinal orientation. Figure 2 shows the 3-burn velocity requirements for the 2003 opportunity, as computed by the MEGA3 program. A 3-month window easily exists for a delta-v capability of 1400 m/s. The apogee high noon GEO launches imply that the high ellipse major axis rotates prograde about 1 deg/day. Once launched, however, the GTO major axis orientation does not change. Figure 3 shows the GTO orientation, and the 3-burn profile for an early launch, where a waiting period of 40 days is required before the first burn; and for a late launch having no wait period, where only 25 days are available to get to the Moon on May 28, 2003.

The lunar flyby altitude is free to vary, and may imposes an impact constraint on the 3-burn method. Typically, the flyby will be close to the Moon for early launch and in fact may, because of impact, cut off the beginning of the launch period. Then, as the launch period progresses, this altitude increases to over 50,000 km, and decreases again. During this time, the second (apogee) burn, which is the major cause of the velocity variation shown in Figure 2, begins high, about 200 m/s, decreases, and then increases to over 200 m/s, where the burn changes the elliptic motion from posigrade to retrograde (see Figure 3).

VENUS MISSIONS

Using the MEGA process for Venus introduces a new problem. The escape vector wants to be directed opposite to that of the high noon GTO ellipse axis, so that it can counteract the Earth's velocity, lose energy, and move inward towards the inner planets. The MEGA process does not work if the angle from the GTO axis to the escape vector is too large. There are two possible solutions to this dilemma.

First, the GEO launch could be planned to be 6 months or more before initiating the burn for entering the high ellipse. This allows the GTO to establish a favorable orientation relative to the escape vector, which it will maintain since nodal regression due to the Earth's oblateness will be small. The actual departure date, then, might be 8 or more months from the GTO launch date. Putting these very early dates into the MEGA3 program will give the result shown in Figure 4 for the Venus 2004 opportunity. If the long wait time can be tolerated, this is a direct way of solving this problem.

Note in Figure 4 that the required flyby of the Moon for late launches results in surface impact, so that the GTO launch period ends in November, or about 5 months before the Venus departure date.

A second possibility is to utilize a lunar flip trajectory[4], as shown in Figure 5. Here, two lunar flybys are needed with the first lunar swingby occuring 180° from the desired second flyby. This moon-to-moon segment is easy to calculate because it is exactly the Moon's (conic) orbit except for the inclination, which can be large, and even retrograde. This double lunar flyby option effectively flips the high ellipse axis 180° and allows the GEO launches to be closer in time to the required departure date.

SUMMARY

A new multi-burn technique has been proposed for transferring from highly eccentric Earth orbit to an out-of-plane escape direction using Moon and Earth gravity assists. This method is being applied to small spacecraft carried into GTO by Ariane 5 and aimed for missions to planetary bodies. The method and a computer program have been described and applied to the upcoming Mars and Venus launch opportunities.

The impact of this technique on exploring the solar system using small sophisticated spacecraft may be considerable. Only time will tell. In any case, the methods presented here of using the gravitational influence of the Earth and Moon to shape trajectories is an indication that astrodynamics can play a very important role in enabling space missions of the future. As new technology developes smaller (or larger) spacecraft, our small community will be called upon to provide the celesial roadmap to put them in place, or on the right path.

ACKNOWLEDGEMENTS

Thanks go to Jacques Blamont for originally suggesting that I explore possible solutions to the problem of departing from GTO to planetary targets. More recently, Kim Leschly and Robert Gershman have been supportive of this work here at JPL, and have helped greatly in encouraging the technology to move in the direction which would make this technique practical. Also, working closely with those proposal teams who are including this method for their mission has been very stimulating. I wish them success.

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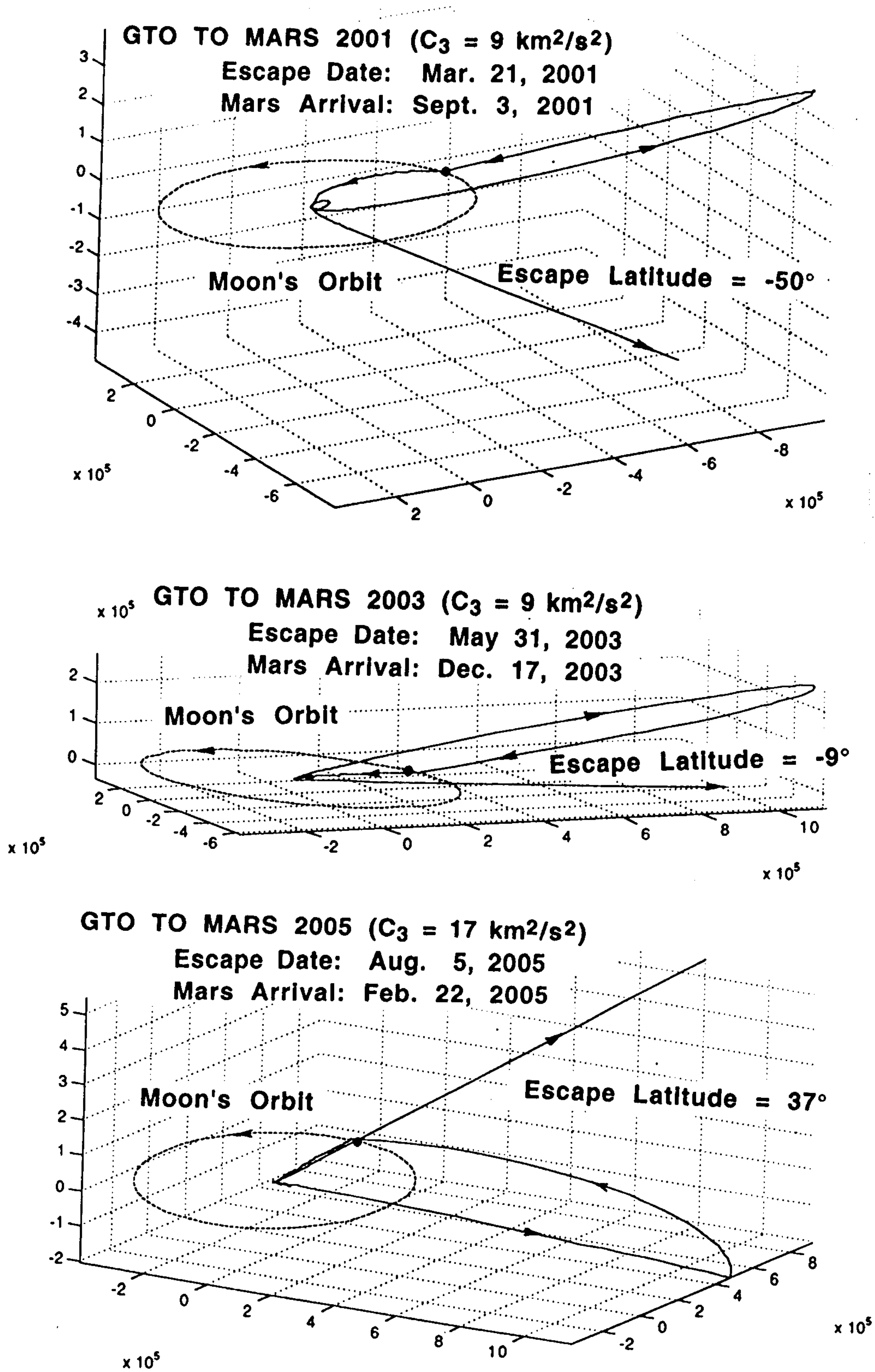

Figure 1. MEGA 3-Burn Examples for the Mars 2001 to 2005 Launch Opportunities

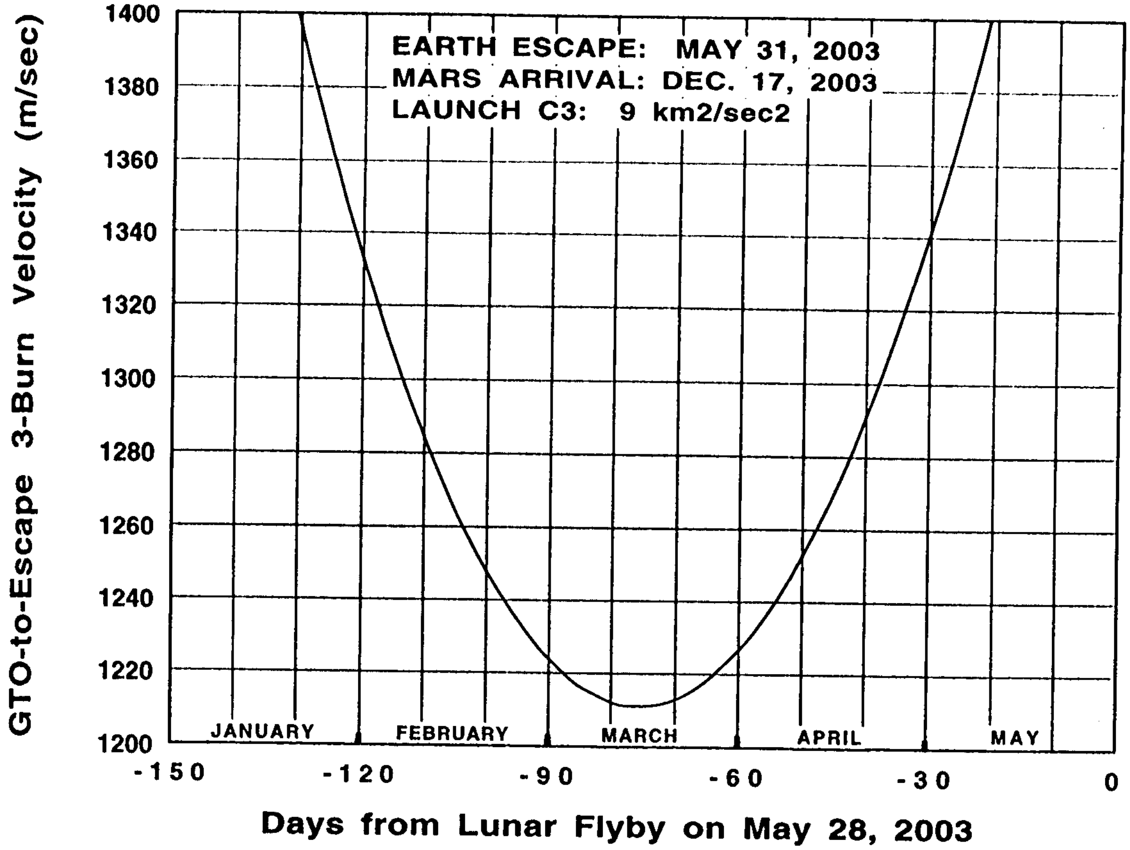

Figure 2. The GTO MEGA Delta-V Requirements for the Mars 2003 Opportunity

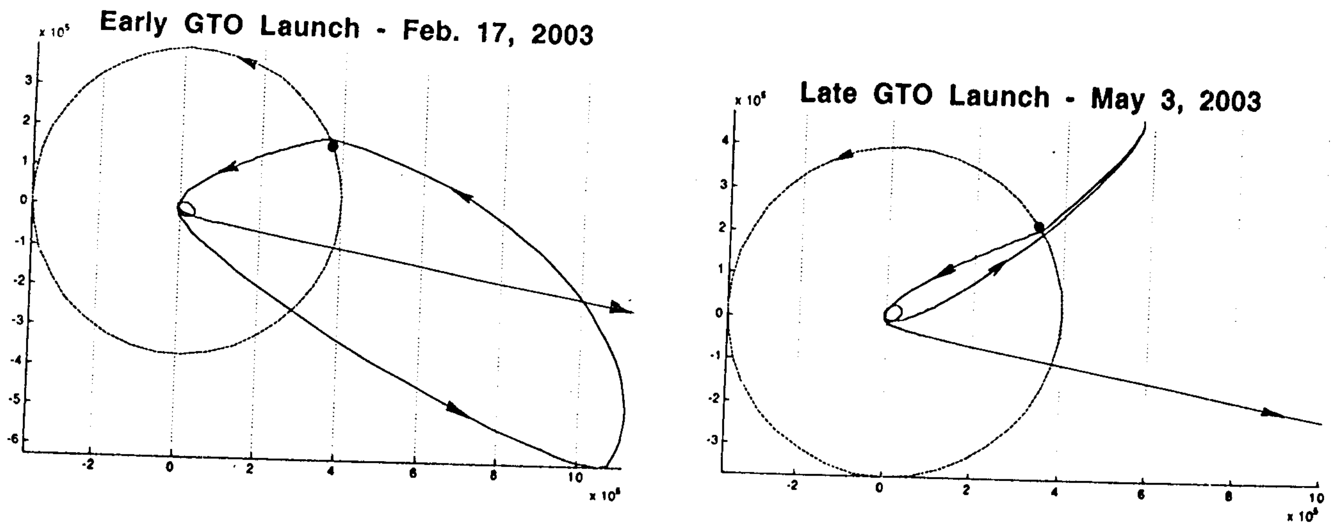

Figure 3. MEGA Trajectory Profiles for the Mars 2003 Early and Late Launches

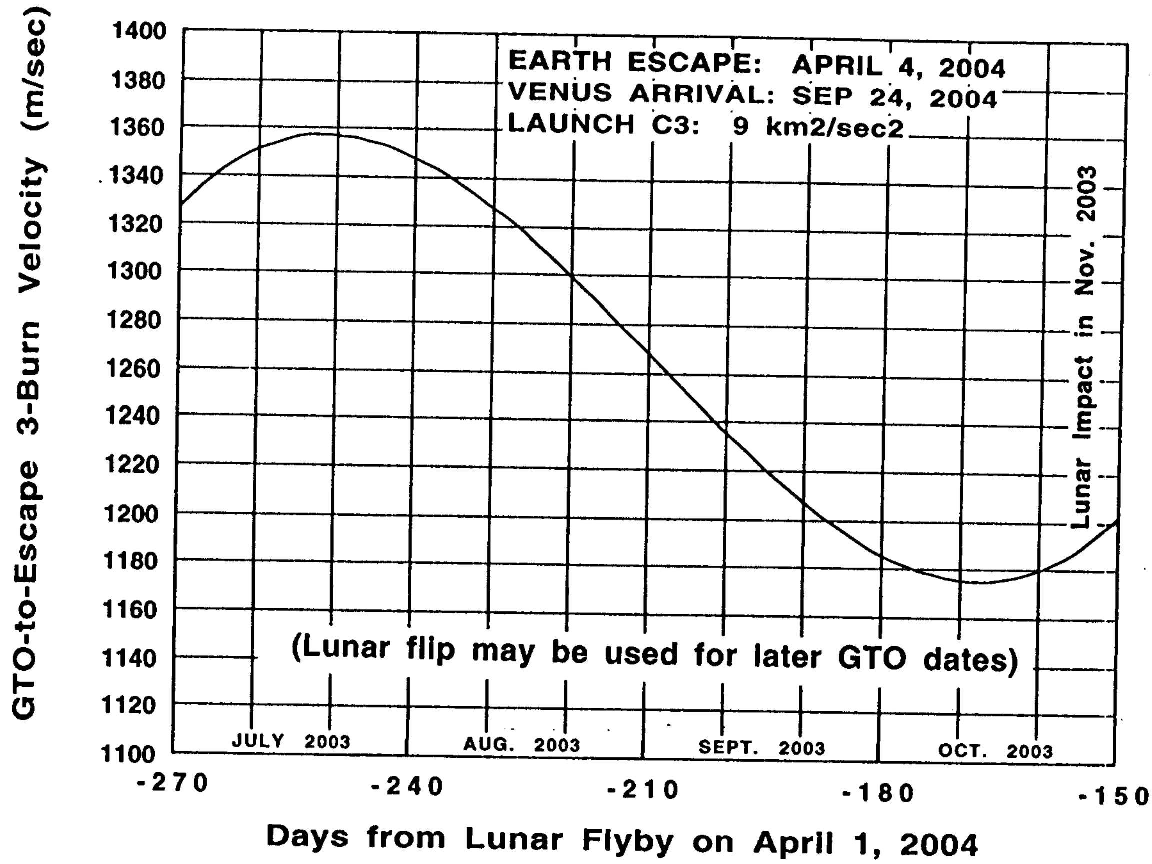

Figure 4. The GTO MEGA Delta-V Requirements for the Venus 2004 Opportunity

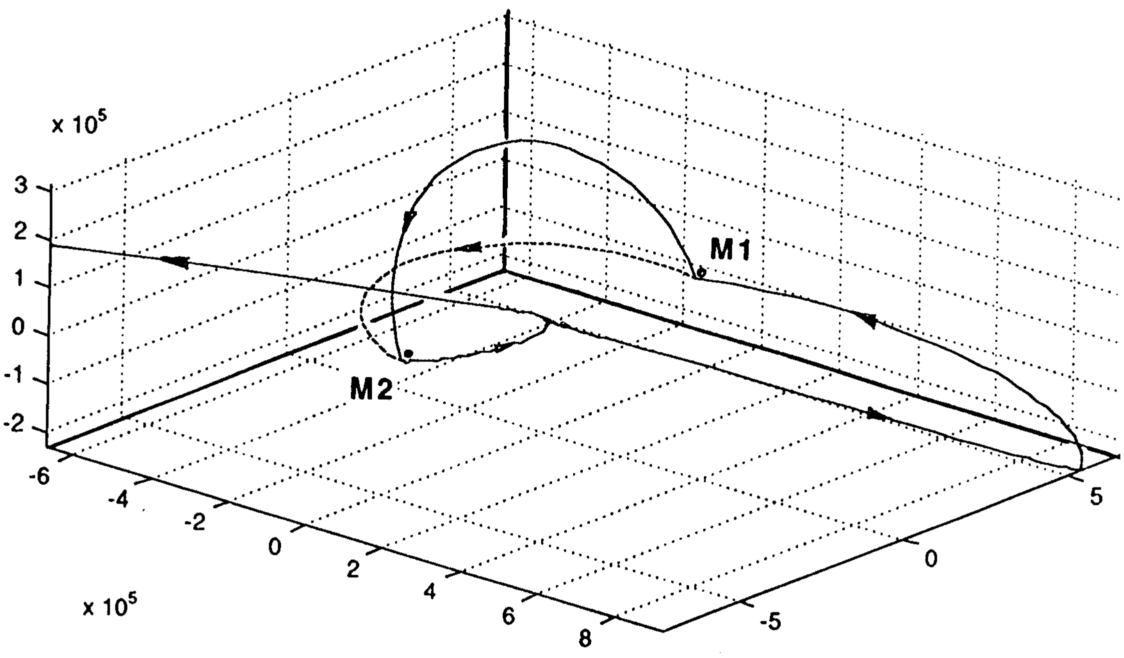

Figure 5. Example of the MEGA Lunar Flip Profile for the Venus 2004 Opportunity