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LUNAR AND PLANETARY MISSIONS LAUNCHED FROM A GEOSYNCHRONOUS TRANSFER ORBIT*

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Considerable cost and mass savings are possible by launching small spacecraft into lunar and planetary space as secondary payloads. The Ariane, for example, provides a platform for several such payloads on each of its monthly launches of placing communication satellites into geosynchronous orbits. Specifically, the second stage injects into a geosynchronous transfer orbit (GTO), and can accommodate eight auxiliary spacecraft, each weighing about 80 kg. This paper addresses the question of how and for what propellant cost can a spacecraft be injected from a GTO orbit to the Moon or to other deep space bodies.

The analysis begins with a discussion on the restrictions due to the highly elliptic and specifically oriented GTO. The importance of target choice and wait time in GTO, perhaps for months, are noted. Only these are suitable for single burn using a solid propulsion stage. Specific application to Venus, Mars and near-Earth bodies are presented. More versitility is gained using a liquid stage where multiple burns are possible, allowing plane changes, and swingbys of the Earth and Moon. This greatly expands the mission possibilities to all targets, as well as to Mars and small bodies.

Scenarios can include direct and indirect lunar flybys, Apollo type circumlunar trajectories, the lunar flip trajectory, and Belbruno's use of the weak stability boundary beyond the Moon to aid or control the direction of escape. It is concluded that GTO secondary launches can play a significant role in solar system exploration, but that it will depend on having a small restartable propulsion system, and a small spacecraft capable of flying intricate trajectories in the Earth-Moon system.

INTRODUCTION

Deep space missions performed with small spacecraft (less than 80 kg) may be launched as secondary payloads from Ariane's geosynchronous transfer orbit (GTO), after release of the primary payload to GEO. Unlike LEO, which is circular and normally inclined to Earth's equator, GTO is highly elliptic (200 km by 35,900 km altitude, as used here) and near the Earth's equator. Although the injection magnitude will be lower, by a factor of 2 or 3, if performed near GTO perigee, the escape direction is severely limited. This is

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dealt with by waiting in GTO until injection results in a desired escape direction and the injection date resides within the launch opportunity period for that planet. The escape latitude, near zero with Earth's equator, is satisfied by choosing the appropriate arrival date at Mars or Venus.

The problem is similar, but easier, to that of injecting 3 spacecraft to independent targets, after being launched into a single LEO parking orbit (Ref. 1). In this case, these were identical spacecraft, where each injected from circular orbit at appropriate times to minimize the propellant required to achieve their respective targets. The analysis of this reference, for LEO, could also apply to other situations. The launch vehicle, for example, could have delivered a primary payload to orbit, with the other two as small secondary payloads left remaining in orbit awaiting appropriate injection times for their respective targets.

In the case of delivering a large communications satellite into GEO, a geosynchronous transfer orbit (GTO) is used, and the launch vehicle, in addition, may also have the capacity to include secondary payloads for delivery elsewhere. The question, then, is: What are the conditions required on the GTO launch date, and on the secondary payloads wait time and propellant requirements, for them to reasonably inject to the Moon, Venus, or Mars?

We restrict this analysis to the Ariane GEO launches, which is the vehicle of current interest for carrying secondary payloads, and also it offers this service to the space community (see Ref. 2). The Ariane GTO orbit is well defined. Launch is from French Guiana on the Atlantic coast of South America near the equator, so that it may be assumed that the inclination is near zero. Also, launch is in early morning, so that apogee of the GTO is at local noon when it arrived at GEO. Specifically, the orbital elements are:

Inclination	=	7.0°
Perigee Altitude	=	200 km
Apogee Altitude	=	35975 km
Argument of Perigee	=	178°
Descending Node	=	10°W

These elements are in the equatorial prime meridian system at the epoch of perigee.

GTO LAUNCH CONSTRAINTS

It is useful to compare launch from LEO to launch from a predetermined geosynchronous transfer orbit. In the late 50s, it was discovered that low Earth parking orbits (LEO) could give essentially full access to the celestial sphere for Earth escape missions. With launch from the Cape at latitude 28.5°, the time of launch (because of the rotation of the Earth) can control the plane which will contain the Earth escape velocity vector. (This velocity vector, added on to the Earth's velocity will determine, to a first approximation, the heliocentric path necessary to get to a specific target, such as Mars.). Then, the time of injection from LEO can then control the placement of central angle through which the spacecraft must go to achieve the desired escape direction. Since launch azimuth from the Cape is limited, reaching escape latitudes greater than about 35° would require a dogleg (out-of-plane) component in the LEO injection. In summary, the escape latitude and longitude are controlled by the launch and LEO injection times.

MISSION DESIGN STRATEGY

Performing escape missions from GTO with secondary payloads does not provide the two degrees of freedom that a dedicated LEO parking orbit does. The time of launch fixes the major axis towards the sun (for Ariane), and the plane of the orbit is in or near the equator. All injections from this orbit would result in the escape V_∞ to be in the equator, and a dogleg, or multiple maneuvers, would be required to move it away.

Further, for efficiency, injection from GTO should be performed as close to perigee as possible. For example, injection to an escape energy of $C_3 = 10$ km^2/sec^2 would require 1210 m/sec at perigee, and 3780 m/sec at apogee. This apogee requirement is greater than from LEO, which is 3670 m/sec. Also, the equatorial latitudes accessible would probably be limited to about 10° using a small dogleg.

Fortunately, for the orientation of the GTO (with apogee towards the sun) the velocity vector at perigee is directed approximately in the same direction as the Earth's velocity vector (EVV). Thus, an injection here would result in an enlarged heliocentric orbit, one which could go to Mars, for example. Escape is not quite in the EVV direction because Earth's gravity turns the velocity vector through an angle of about 60°, making the injection less effective. But, as the Earth moves around the sun, the EVV also rotates one degree per day while the GTO remains inertially fixed. Then, waiting about two months before escape injection at perigee, the escape vector will be in the direction of the EVV. The wait time can be reduced by a pre-perigee injection, but with some propulsion penalty. This is shown in Figure 1, where the outbound escape vector is fixed and a burn at GTO perigee achieves this escape vector. Then, holding the GTO fixed and varying the direction of escape, by degree increments, the minimum ΔV on the GTO may be found to achieve this escape direction.

For Venus, where the escape vector should be aimed opposite the direction of the Earth, the strategy is more complex. Obviously, one cannot simply direct the injection opposite to the GTO velocity at perigee, since this will simply decrease energy and make the orbit about Earth smaller (the GTO speed at perigee is about 10 km/sec). But, just as waiting in orbit can cause Earth's velocity vector to rotate to a desired direction, the same can be done here. The wait time will be six months plus the two months already needed, for a total of eight months.

Table 1 shows the results of applying this strategy to Mars, Venus and asteroid missions. Launch and arrival dates are chosen so that the outbound V_∞ is in the Earth equatorial plane, and a required wait period in GTO is applied. These missions can be performed with a single burn, i.e., with a high energy solid propulsion stage. The wait period is less than 2 months for Mars and the asteroids, but is 8 months for Venus. The ΔV reserve may be used to perform an out-of-plane component, or compensate for slips in the GTO launch date.

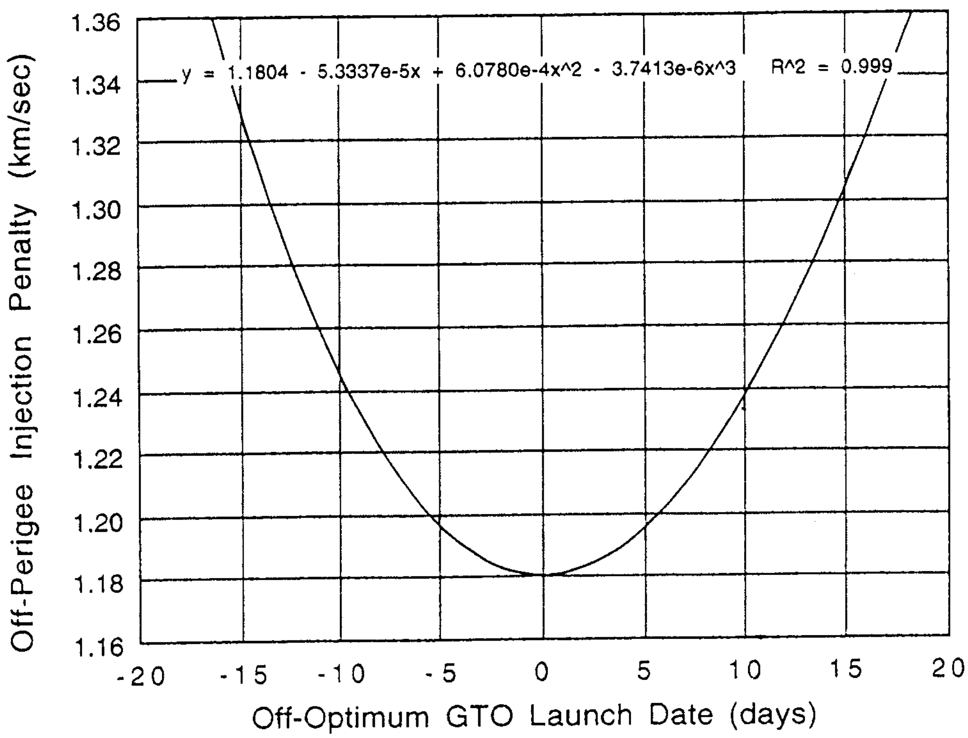


Fig. 1 Off-Perigee DV Penalty versus GTO Launch Date

	Mars	Venus	Ra-Shalom	1986 DA
GTO Launch Date	2-11-2001	5-7-2000	4-4-2000	7-12-1999
V_∞ Declination	0 deg	0 deg	0 deg	0 deg
V_∞ Right Ascension	280 deg	10 deg	350 deg	86 deg
C_3 (km^2/sec^2)	9.5	8.0	5.5	5.1
Injection Date*	3-10-2001	1-7-2001	5-20-2000	9-16-1999
Total Mass (kg)	80	80	80	80
STAR 12GV (kg)**	45	45	45	45
Delta-V (m/sec)	1190	1120	1020	990
Delta-V Reserve	180	250	350	380
Flight Time (days)	273	120	120	317
Flyby Speed (km/sec)	NA	NA	10.2	12.8
Approach Phase (deg)	NA	NA	62	145

* There is a wait in orbit to get the correct escape direction.
** Single burn solid stage

Table 1 Single Burn missions to Mars, Venus, and Asteroids

Two other strategies are possible; to reduce the wait time and to allow greater access to the escape vector direction. One is to inject from the GTO to an apogee of, say 500,000 km, where velocity is low. Then a plane change maneuver (PCM) of up to 180° can be made to rotate the ellipse so that perigee velocity will be in an alternate direction. A third maneuver at perigee can then inject to a desired escape direction (assuming that only a rotation of perigee velocity about the major axis is required). The penalty (the 2nd maneuver) would be about 290 m/s. The period of this large ellipse would be about fifteen days, and the penalty will increase if solar perturbations need to be compensated for.

The second strategy is more complex but adds considerable flexibility in escape directions. This is shown in Figure 2. It uses the plane change maneuver mentioned above plus an Apollo type circumlunar (CL) trajectory. It requires four maneuvers: the plane change maneuver (PCM) and three more at low Earth altitudes, say at 200 km.

The sequence of events are as follows:

(1) At an appropriate time, based on a future encounter with the Moon, a maneuver ΔV_1 is made at GTO perigee to send the spacecraft beyond the Moon's orbit to the PCM point. This injection will require about 700 m/sec and is made in the GTO plane.

(2) The maneuver at the PCM, ΔV_2 is then used to rotate the orbit plane for a lunar encounter, as well as maintaining the 200 km perigee on return to Earth. This maneuver may also be used to move the longitude of perigee. An average estimate of this required maneuver is 150 m/sec.

(3) On return to Earth, say 15 days later, a third maneuver ΔV_3 is made to adjust the flight time to the Moon for the circumlunar portion of the trajectory. In general, this maneuver will not be needed, unless the lunar flyby altitude required for return leg is subsurface.

(4) The return-to-Earth leg of the circumlunar trajectory is required to be in the plane of the Moon's position vector and the escape vector, and to have a 200 km perigee at Earth. This implies that the latitude of the escape vector is satisfied, but not necessarily its longitude. The 4th maneuver at Earth ΔV_4 is used to attain the correct escape energy which, for example, is about 300 m/sec for a C_3 of 5, 475 m/sec for a C_3 of 9, and 600 m/sec for a C_3 of 12. The respective total velocity requirements for these values of C_3 are 1150 m/sec, 1325 m/sec, and 1450 m/sec.

GTO TO MARS AND NEAR-EARTH ASTEROIDS

As an example of the mission design process required for GTO launch frjom the Ariane secondary payload platform, Mars, Venus, and two near-Earth asteroid possibilities are chosen. The asteroids are of interest to planetary scientists because of their spectral characteristics, and both are easy flyby targets.

The analysis begins with finding the launch and arrival dates for each target, which will minimize launch energy (C_3) and maximize launch mass. For GTO launch, the constraint that the escape vector should lie in or near the Earth's equatorial plane (to avoid a dog-leg maneuver) will not normally make this trajectory viable. Instead, the trajectory space must be scanned for those launch-arrival date combinations which do have the escape vector in the equatorial plane.

Once these two dates are selected, the other launch and arrival parameters may be calculated, including launch energy, escape direction from Earth, the asteroid approach spacecraft flyby velocity and sun phase angle, and heliocentric positions as a function of time, so that the sun, Earth, and target body distances and angles may be computed for a spacecraft design and operations. Some of these parameters, and those related to spacecraft mass are listed in Table 1 for Mars, Venus, and the two asteroids.

The 'launch date' computed above (May 20, 2000 for Ra-shalom, for example) really means the injection burn (there may be others previously done) which causes the spacecraft to escape Earth in the right direction and with the desired energy. This is computed here as if it were performed as a single impulsive burn at a GTO perigee of 200 km. For both asteroids, this is found to be about 1000 m/s. A similar burn from LEO would have required about 3500 m/s>

Spacecraft sizing can now begin, assuming a maximum total mass of 80 kg. The design chosen here assumes that about half of this mass will consist of the propulsion system, which is a Thiokol STAR 12GV motor. This motor has a specific impulse of 283. Total mass is and attachment structure assumed to be 45 kg, leaving 35 kg for spacecraft subsystems and instruments.

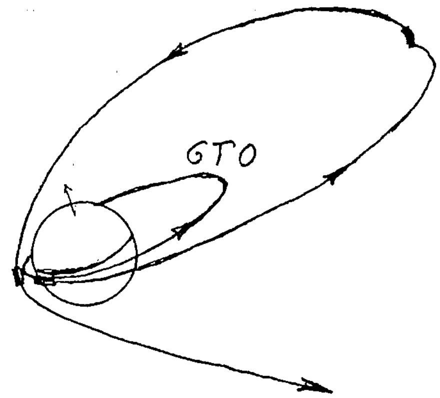
The required direction of the escape vector, about 350° for Ra-Shalom will determine the inertial location of GTO perigee. Since at GTO launch, perigee is at midnight, it is necessary to find that GTO launch date when the major axis will be parallel to the one required for the injection date of May 20, 2000. This is determined to be April 4, 2000, or 46 days earlier, which becomes the required date for GTO launch.

Mission design must now address the question of how to handle GTO launch date variations. It is unlikely that the Ariane will let their launch date (or equivalently, the inertial location of perigee) be dictated by their lesser paying customer. The problem is solved by allowing an off-perigee injection at the time of Earth escape as shown in Figure 1.

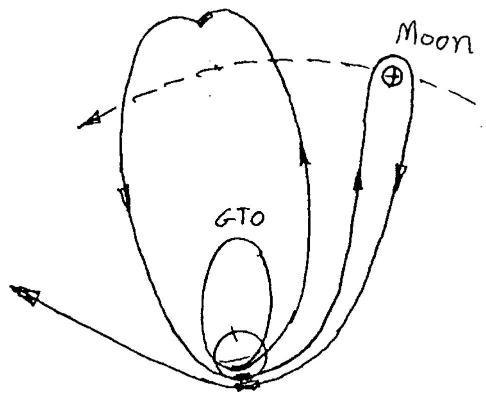
MULTIPLE IMPULSE AND LUNAR OPTIONS

Liquid propulsion and lunar and Earth flybys can greatly enhance the accessibility of many targets. Figure 2 presents three scenarios, each of which have particular benefits. The first is the deep space plane change, which requires three maneuvers useful for out-of-plane escape directions. The first enlarges the the GTO ellipse to an apogee of about 500000. km. A plane change is then made, which can include a flight path angle change also, to return to a 200 km altitude of the Earth. The third maneuver at this altitude then boosts the spacecraft to the proper C3 energy.

Deep Space Plane Change



Circumlunar



Lunar Flip Trajectory

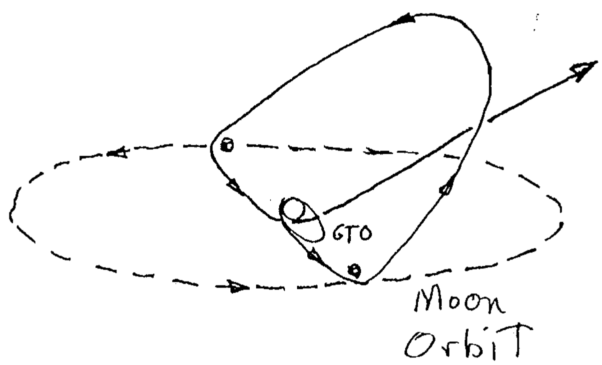


Fig. 2 Multiple Impulse and Lunar Swinbys for Greater Flexibility

The second method includes this plane change, but directs the trajectory to the moon after an Earth flyby of 200 km. A small ΔV at this altitude can modify the flight time to the moon, as necessary, so that the flyby of the moon causes the spacecraft to return to Earth at any desired inclination. The advantage here is that the plane change maneuver can be much smaller (only that necessary to get to the moon) and save ΔV (Ref. 4)

The third method (Ref. 5) called the back-flip trajectory can rotate the GTO perigee velocity vector by 180°. This requires the spacecraft to swing by the moon and enter an orbit equivalent to the moon's orbit but inclined. Then, 14 days later, it will again encounter the moon with a flyby returnig it to the Earth where a boost can send the spacecraft to a desired escape direction.

OPPORTUNITIES FOR LUNAR MISSIONS

The Moon, being in orbit about the Earth, requires considerably less energy to reach than bodies requiring Earth escape. If the GTO orbit were ideally aligned, the perigee injection ΔV required would be about 700 m/s, resulting in an elliptic transfer orbit with a C_3 of -2 km^2/s^2.

Unfortunately, the Moon's orbit is near the ecliptic (inclined about 5.5°), where the GTO orbit is nearly in the Earth's equatorial plane. This means that the GTO's major axis should be near the intersection of the two planes, which would occur in March or September. The off-perigee injection to the Moon would allow a month's span for the actual GTO launch, with a penalty of about 80 m/s. An additional penalty may be necessary for a plane-change (dogleg), but the calculations have not yet been done.

Missions to the lunar libration points may also be of interest for secondary payloads injected from a GTO, and these are discussed (injected from LEO) in Reference 3. The ΔV requirements using a bi-elliptic transfer to these points are given in Figure 3, taken from Reference 3. Assuming ideal conditions, the $\Delta V1$, which would be the requirement from LEO can be replaced by 700 m/s for GTO. Then, the total ΔV for going to the L4 or L5 points would be about 1050 m/s. As indicated in the footnote of Figure 5, this may be reduced assuming the use of solar perturbations (and increasing flight times up to 3 months) but this has to be investigated further.

USE OF SOLAR PERTURBATIONS, AND LUNAR SWINGBYS

In addition to the direct approach discussed here for GTO launching of secondary payloads, there are other techniques that may be used to enlarge the range of targets available, or to ease the requirements for getting to Mars, Venus, or the lunar libration points. Only a brief indication of benefits will be mentioned here.

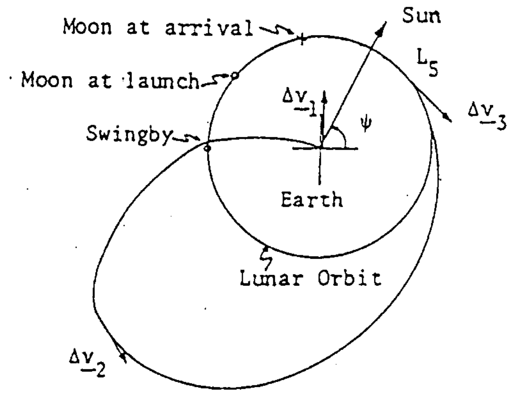
The first technique to consider is the inclusion of intermediate maneuvers to alter the conditions under which the escape ΔV must be made. Figure 5 is an example where ΔV_2 is used to raise perigee from LEO to the moon's altitude in order to enter the L5 point.

For escape missions, where it is necessary to reach higher declinations, the ΔV at GTO perigee could raise apogee, say, to 3 times the lunar distance,

• Reference: Tanabe, et.al.,"Visiting Libration Points in the Earth - Moon System Using a Lunar Swingby, "1982.

Conic results (Bi-ellipitic transfer):

Lib. Point	L1	L2	L3	L4	L5
ΔV_1	3130	3130	3130	3130	3130
ΔV_2	135	209	126	195	150
ΔV_3	319	59	215	167	196
ΔV_{23}	454	268	341	362	346
T_f(days)	50.6	50.0	65.1	45.1	55.3



Transfer Orbit to L_5.
(Numerically Integrated)

• Use of solar perturbations can decrease and almost eliminate these ΔV_{23} requirements, at the expense of longer trip times.

Fig 3. Transfers to the Lunar Libration Points from LEO

where the velocity would be only 70 m/s. Once there, only 36 m/s would be necessary to alter the existing inclination by 30°. Then on return to the 600 km perigee (or 200 km at very little cost), a third ΔV to escape to a higher declination, say to Mars, could be applied. These 2 additional maneuvers may reduce the overall ΔV requirement, as well as enlarge the GTO launch period.

At the distance assumed above, about a million kilometers from Earth, the solar perturbations will be important. The effects may be favorable or unfavorable, but will have to be taken into account. They may be favorable for shaping the trajectory to include lunar flybys. For example, Reference 4 discusses the use of multiple lunar flybys in getting to Mars, Venus and near-Earth bodies. In the ideal case, only a ΔV required to reach the Moon (700 m/s) would be needed to do these missions. For any specific mission, it will be necessary to investigate which of these techniques, or combination of them, would best apply.

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