



Mission Design for Mars Missions Using the Ariane ASAP Launch Capability

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MISSION DESIGN FOR MARS MISSIONS USING THE ARIANE ASAP LAUNCH CAPABILITY*

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Mars is now a primary NASA focus in planetary exploration. Rapid advances in microspacecraft implies that they should have a role, without the burden of expensive launches, by flying as secondary payloads. A specific launch mode, called Moon and Earth Gravity Assists (MEGA) has been developed which allows proper escape from GTO launches, with minimum constraints on launch date and time of day, over a 3 month period. Small spacecraft (200 kg) can fly piggyback on the Ariane to perform a variety of missions. Those discussed here will include penetrators, probes, planes, orbiters, balloons and gliders, for 2003, 2005, and 2007, presenting launch and arrival conditions, and mission design considerations for each. These missions are made possible with the use of the MEGA process, which will be discussed here in some detail.

INTRODUCTION

There are many ways to explore Mars. Orbiters provide a global view, and can survey the planet for follow-on missions. They can also serve as relay satellites to gather data received from the surface, and return that data to Earth. Atmospheric probes and surface penetrators, in numbers, can also be global gathering. A network system of several dozen small stations could monitor the pressure, temperature, and opacity of the atmosphere over a period of 5-10 years, to develop a climate data base for Mars weather. Balloons, small planes and gliders can explore the rugged terrain to probe Mars' geologic history. All of these functions, with the current technology thrust for miniaturization, could be performed with small instruments and spacecraft, either singly or in numbers.

What remains is a low cost means, and launch technique, of transporting small spacecraft to Mars. The means proposed here is to utilize the piggyback capability provided on the Ariane 5 during it's geosynchronous Earth orbit (GEO) launches. In this mode, up to eight 100 kg (or four 200 kg) auxiliary payloads may be carried into the geosynchronous transfer orbit (GTO) and released, after the communication satellites are released at GEO altitude (35900 km). These small payloads are attached to a ring in the upper stage of the Ariane called the Ariane Structure for Auxiliary Payloads (ASAP)¹, and when released must have their own propulsion system to depart for Mars from

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the highly eccentric GTO. The suggestion that this would be a worthwhile problem to investigate was conveyed to me by Blamont², who presented convincing arguments for possible lunar and other deep space missions launched as secondary payloads from the Ariane.

Since the Mars spacecraft would fly as secondary payloads, no conditions can be imposed on the Ariane launch date, or time of day. Therefore, an efficient technique must be devised to transfer from the GTO to the Earth escape vector necessary to get to Mars. The method proposed here to perform this transfer is shown in Figure 1.

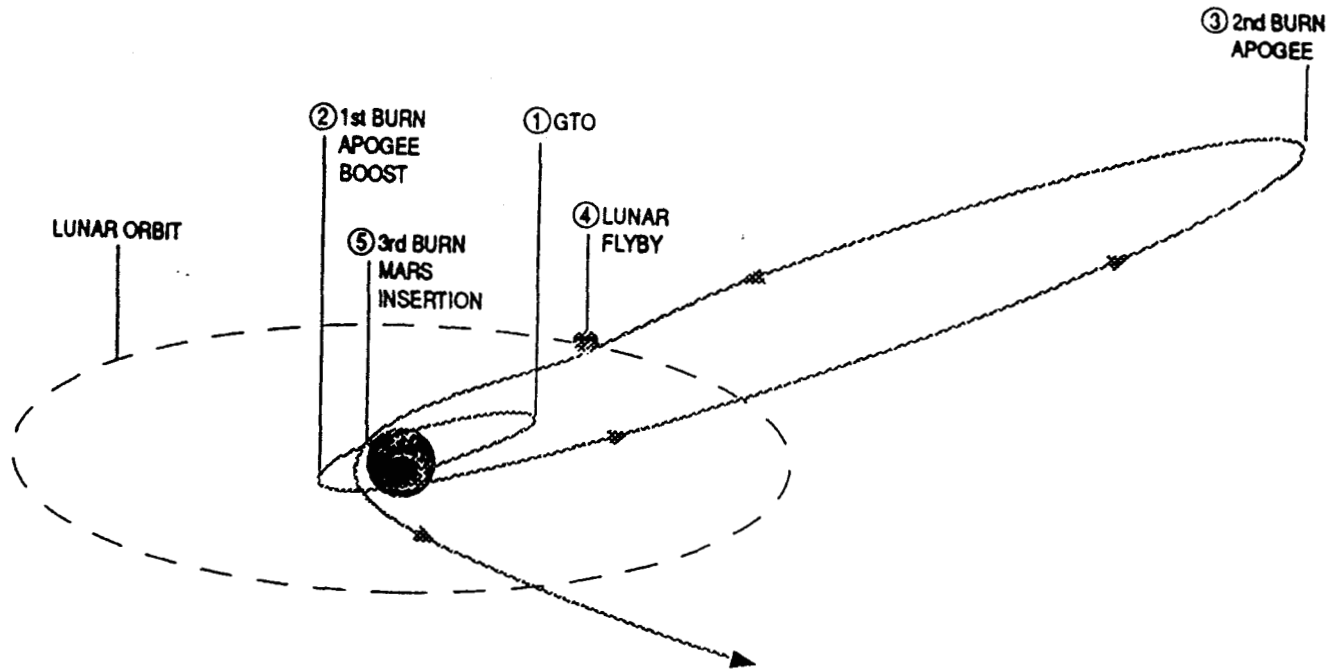


Fig. 1 GTO to Mars Using the 3-Burn Moon-Earth Gravity Assist

This process, called the Moon-Earth Gravity Assist (MEGA) requires a minimum of three propulsive maneuvers, together with flybys of the Moon and Earth, resulting in proper escape.³ As shown in Figure 1, the sequence begins with the spacecraft in the near equatorial GTO orbit (200 km by 35900 km). The first burn, performed at perigee and in the orbit plane, transports the spacecraft to some distance beyond the Moon's orbit. At the apogee of this high ellipse, a second burn targets to a lunar flyby such that the spacecraft will return to Earth with a low altitude, e.g., 300 km, where a third burn will propel the spacecraft to the required escape direction (and with the necessary velocity) to escape Earth and be on its way to Mars. The Mars craft must have an engine restart capability, and a total velocity capability of at least 1400 m/s to perform the three maneuvers. Application of this process cannot be computed based on simple conic motion alone. Nodal regression of the GTO, solar perturbation on the high ellipse, and orientation of the GTO axis with the desired escape vector must be taken into account.

THE ARIANE LAUNCH PERIOD

Unlike dedicated launches, where the optimal Earth departure and Mars arrival dates are selected based on the mission to be flown, the MEGA, or some other procedure, must deliver the small spacecraft to the desired escape vector (and on the desired departure date) with no requirements imposed on the Ariane launch. Detailed analysis of the MEGA strategy shows that a several month launch period for the GTO can be tolerated, with considerable freedom in the time of launch within each day.

The analysis begins with selecting the Earth departure and Mars arrival date space within which the mission can be accomplished. For the Mars mission opportunities, based on the minimum launch escape energy (C_3), the dates and related parameters are given in Table 1. The arrival parameters will become important when discussing specific missions.

Table 1. Launch and Arrival Conditions for GTO to Mars 2003-2007

EARTH ESCAPE DATES	C_3 (KM ² /S ²)* ENERGY RANGE	LUNAR FLYBY DATE	MARS ARRIVAL DATES	ARRIVAL VELOCITY RANGE	MARS** LATITUDE RANGE
Year 2003 (yr mo day)					
3MAY22-3JUN20	8.9/10.0(1)	3MAY28	3NOV18-4JAN20	2.7/2.8	9.5/11.5
3APR26-3MAY18	12.6/16.0(2)	3MAY01	3DEC17-4FEB05	2.8/3.2	-10.0/25.0
Year 2005					
5JUL21-5AUG30	15.9/20.0(1)	5AUG02	6JAN14-6APR20	2.4/5.0	-20.0/3.0
5JUL28-5OCT17	15.4/20.0(2)	5AUG31	6JUN24-7FEB24	2.5/6.0	0.0/40.0
Year 2006-2007					
6NOV24-7APR17	8.7/10.0(4)	7FEB14	9JAN12-9SEP22	3.2/6.0	-28.0/-1.0
7SEP01-7OCT27	12.8/16.0(2)	7SEP09	8JUL23-9JAN28	2.4/7.0	-5.0/35.0

*Numbers in parentheses are trajectory types.

**This column has the latitude range of approach vector relative to Mars equator.

Within each set of escape dates shown, there will be a minimum of one date when the Moon is in a favorable position in its orbit for the flyby, relative to the outgoing escape vector. This date is shown in Table 1, and must lie in the range of launch dates, allowing for the 2-3 days for the return to Earth for the third escape burn.

Now, with the lunar flyby date fixed, the first (high ellipse) burn, and the second lunar targeting burn may be addressed. The second, or lunar targeting burn, is easy, since it has to be the time when apogee is reached on the high ellipse, and is computed by knowing the Earth centered lunar and apogee position vectors. This is Lambert's problem, modified by solar perturbations. This burn is constrained to be in a plane perpendicular to the apogee position vector.

Then, when is the first burn to the high ellipse done? There are upper and lower limits on the time between this burn and the lunar flyby. First, this time can't be much below 20 days, since this results in a short ellipse, only out to about 700,000 km, where the apogee velocity will be high, making it expensive in delta-v to change the velocity to target the Moon.

On the other hand, the transfer time can't be much greater than about 80 days, with apogee out to about 1,400,000 km, because the solar perturbations will cause the spacecraft to escape the Earth, or drift unpredictably. The difference in these limits, or 60 days, would represent the flexibility in the date of the first burn, and hence provide a 2-month launch period for the Ariane.

The launch period can be extended even further by launching earlier, and remaining in the GTO orbit until the 80 day opportunity arrives. One could even remain in the GTO (or other orbit) 60 days longer and wait until the 20 day opportunity arrives. There is a tradeoff between stay time in GTO and travel time chosen on the high ellipse (from the first burn to the lunar flyby) and, in fact, there is an optimum pairing which will minimize the total delta-v requirements. An initial exercise in developing the optimum pairing is presented in Table 2, for the 2003 launch opportunity. The delta-v requirements and the flyby distance at the Moon are also listed.

Table 2. MEGA 3-Burn Summary for Mars 2003 Type 1 Trajectory*

LAUNCH DATE (mo. day)	GTO WAIT TIME (days)	FIRST BURN DATE	HI-ELLIPSE APOGEE (10 ⁶ km)	HI-ELLIPSE TIME (days)	FLYBY** RADIUS (km)	TOTAL DELTA-V (m/sec)
FEB. 2	50	MAR. 24	1.48	65	5570	1357
FEB. 17	40	MAR. 29	1.28	60	8170	1234
MAR. 4	35	APR. 8	1.12	50	11770	1226
MAR. 19	30	APR. 18	0.96	40	18770	1210
APR. 3	25	APR. 28	0.81	30	43280	1231
APR. 18	5	APR. 23	0.86	35	45590	1291
MAY 3	0	MAY 3	0.73	25	12190	1383

*Earth Escape: May 31, 2003 ($C_3 = 9 \text{ km}^2/\text{s}^2$)

Mars Arrival: Dec. 17, 2003

**The lunar flyby date is May 28, 2003.

The GTO orientation for these computations assumes that it is launched with the sun overhead (high noon local time) when it reaches apogee, and that it lies in the equatorial plane (actually, launched from French Guyana, its inclination would be 7 deg).

The total delta-v of the three burns is given in the last column, but the magnitude of each burn lies in a fairly narrow range. In the above example, the first burn to enter the high ellipse beyond the Moon requires from 720 to

750 m/s. The second burn at apogee varies the most, usually from 10 to 250 m/s. The third, or escape burn depends on the escape C_3 desired, and ranges from 425 to 490 m/s. This value will increase about 45 m/s for each unit of C_3 above the value of 9 in this example. In summary, Table 1 indicates that a three month launch period is available for a 3-burn capability of 1400 m/s.

Trajectory plots for the February 17th and the May 3rd launch cases are shown below in Figure 2. The GTO is the little blip at the center (the Moon is about 10 times as distant as GEO). As listed in Table 2, the early launch date has a high ellipse apogee about twice as far out as the late launch date. Also, because this apogee follows the sun (advances 1 deg per day), the May 3rd launch date requires the second burn to reverse the velocity vector, and to make the trajectory to the Moon retrograde. This burn is nearly 250 m/s, and includes a plane change of 108 deg. Note that the Moon's position for the flyby was carefully chose to be about 50 deg more in longitude than the required outbound escape direction.

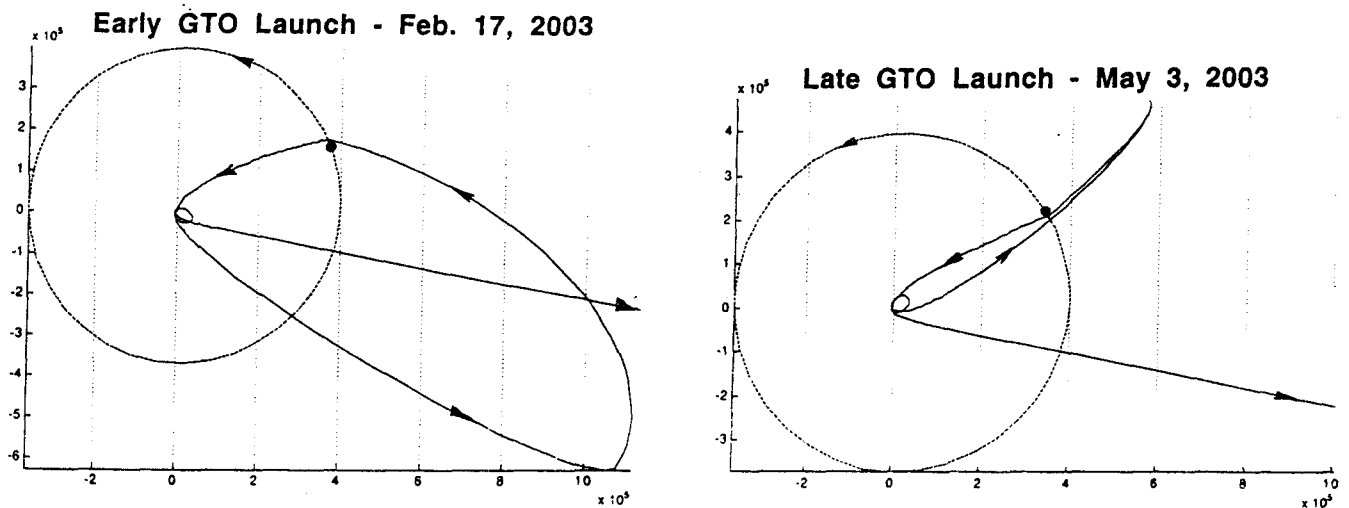


Fig. 2 Three-Burn Trajectory Plots for Mars 2003 Type 1

THE ARIANE LAUNCH HOUR

The analysis above assumes an equatorial GTO and a noon apogee arrival which, for a given day, fixes this ellipse in space. Subsequently, it was discovered that other apogee arrival times need to be considered, specifically, arrival times between 9 am and 6 pm. The GTO plane could still be considered to be in the Earth's equator. There remains one degree of freedom, then, which we will refer to as hours of apogee arrival referenced from Noon Local Time (NLT), which can range from -12 hr to 12 hr for a given day.

On the other hand, the same GTO orientation can be defined as day of launch for an apogee noon arrival (DLN), as for the Mars 2003 case above. The relationship is presented in Figure 3, and is given approximately by the equation,

$$DL = DLN - 15 * NLT$$

where DL is the GTO launch date which will provide the same GTO orientation as DLN if the apogee arrival is NLT. For example, the March 19th launch shown in Table 2 would have the same GTO orientation as a March 4th launch with a 1 pm (NLT=1) GTO apogee arrival, since in 15 days the 1 pm apogee will move into the noontime position by March 19th. In inertial space, the two orbits are the same. The earlier 1 pm launch implies waiting in GTO 15 days longer than the 30 days indicated in Table 2.

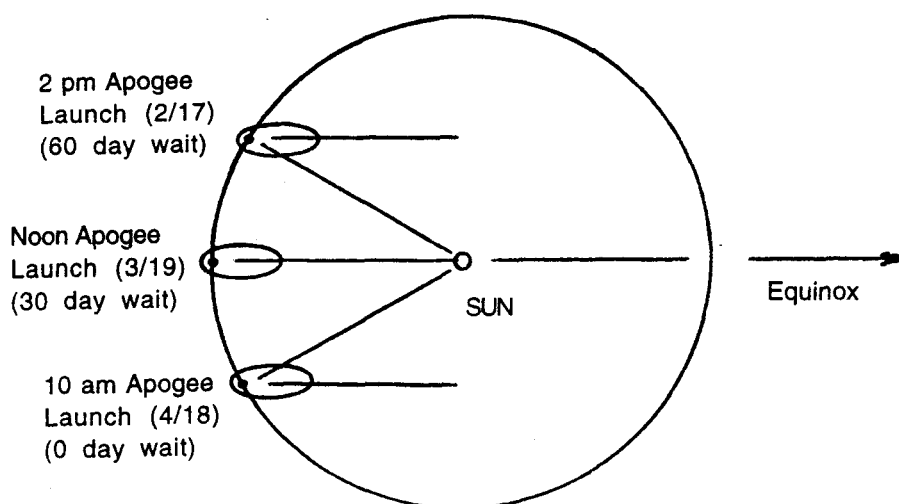


Fig. 3 Launch Day/Launch Hour Equivalence for the MEGA 3-Burn

As for the equation above, if DLN is the day of the year for March 19 (78), and NLT is 1 pm (+1), then DL is 63, which is the day number of March 4th. This means that the trajectory data we have computed for a noon time launch can also apply to other hours of the day if we adjust the GTO launch date. Note that this adjustment can apply to earlier than noon launches, if there is an assumed wait time for the noon time launch. The latest that the March 19th noon time trajectory may be used, for example, is 30 days later, or April 18th, but the NLT would have to be -2 (10 am).

With this understanding of the flexibility of the trajectories listed in Table 2, it is possible to define the GTO launch date-launch hour space, and this is shown in Figure 4. Added to this plot are values for the 5-burn and 7-burn options which will be discussed below.

In this figure, only trajectories for noon apogee arrivals for GTO have been computed, and are the points on the 12 pm line. The data in Table 2 has been interpolated to give even increments of total 3-burn velocity ranging from 1200 m/s to 1500 m/s (the higher velocities are not shown in Table 2). The DL equation above is used to extrapolate these solutions to other launch days and launch hours.

For example, the March 19th launch date is near the minimum total delta-v available for noon GTO apogee arrival. The same 3-burn trajectory is also available for a 2 pm apogee arrival launch on February 17th, 30 days earlier, or for an April 18th launch with a 10 am GTO apogee arrival. Also, since April 18th is the date of the first burn (in all cases), this 3-burn trajectory cannot be flown for a later date than this.

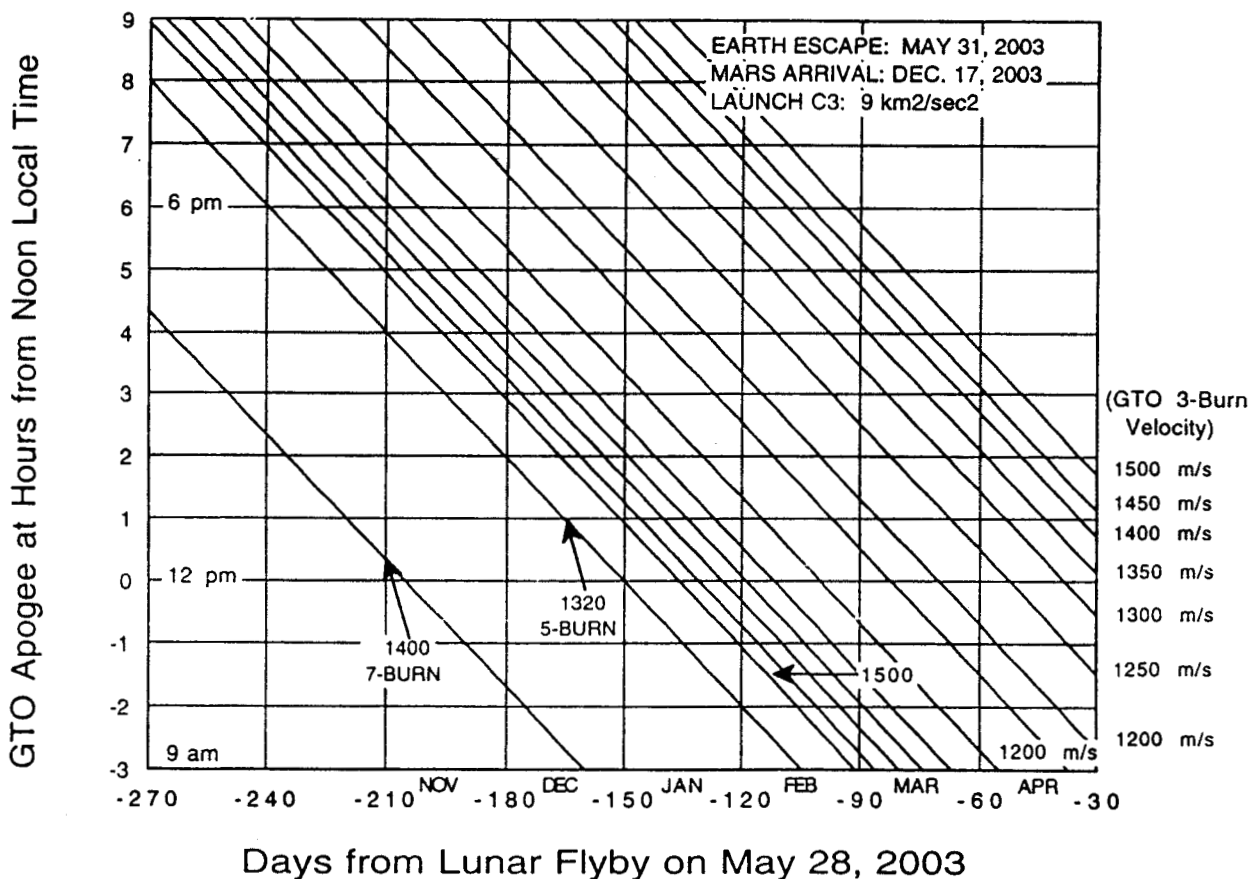


Fig. 4 GTO MEGA Launch Opportunities for Mars 2003 Type 1

THE MEGA 5-BURN AND 7-BURN OPTIONS

Note that in Figure 4, the 3-burn option cuts a swath through the GTO Day/Hour space which is almost 4 months wide. However, the swath does not cover the regions of early day-early hour cases, nor late day-late hour cases (This is the lower left and the upper right portions of the plot.)

No solution has been found for the upper right region, where the 3-burn process must begin very soon after launch, but a very early GTO launch, say in November and December of 2002, and January of 2003, allows several months during which maneuvers in the Earth-Moon system may be constructed to align the high ellipse axis for a possible 3-burn trajectory.

The method proposed here is to use this time to include additional high ellipse loops. Each loop, with the aid of a lunar flyby, will rotate the line of apsides about 60 deg in about 60 days, so that a noon time apogee launch in December, say, will look like a noon time GTO launch in February. Each loop will require two additional burns, increasing the total delta-v requirement by about 50-100 m/s. Fortunately, this apogee burn is not greater than this because of the benefit of solar perturbations in increasing the perigee altitude of each loop, allowing the spacecraft to essentially reach out to the Moon. This effect has been studied in detail in reference to the lunar capture problem (see Reference 4, for example).

For the 5-burn option, in the case shown in Figure 5, the first burn, made in-plane at GTO perigee, places the high ellipse apogee above the ecliptic. The second burn, about 30 days later at this apogee, targets the Moon so that the lunar flyby will rotate the line of apsides about 60 deg. and return the spacecraft to a low Earth altitude. The flyby of the Moon can also rotate the orbit plane, so that it lies in or near the ecliptic. The small 3rd and 4th burns are made at perigee and apogee of the 2nd loop, followed by a second lunar flyby which again returns the spacecraft to a low Earth perigee. Here, the 5th burn is made, in-plane and at perigee, to escape to Mars.

The seven burn simply adds another loop, permitting the GTO launch date to back up to November 2002, and uses this extra loop to rotate the high ellipse into the position required for the 5-burn option. A spread in the launch dates other than an even 60 days for November (7-burn) and January (5-burn) is obtained by allowing one or more of the loops to have a shorter period than 60 days. Further study is required to determine the optimum combination of periods for the loops.

MARS MISSION ANALYSIS

With the Ariane launch periods reasonably developed, it is useful to consider various Mars mission possibilities.⁵ There are essentially two classes of missions which can be accommodated by small spacecraft.

- 1) Those missions which have no significant deterministic delta-v's after Earth escape, such as probes, landers, balloons, etc., which rely totally on the Mars atmosphere for any braking that is required.
- 2) Those missions which will require significant delta-v at Mars arrival, primarily orbiters with diverse functions, such as science remote sensing, relays for surface (or air) to Earth communications, and support craft for in-orbit rendezvous and sample return.

For generality, we will call the first class carriers, and the second class orbiters. In selecting the trajectories, mission requirements for the first class

are generally satisfied by selecting Earth-to-Mars trajectories which will minimize the total MEGA velocity, which means choosing those trajectories which will minimize the required launch energy. The orbiter class must include the orbit insertion delta-v in the minimization, which will generally change the date of Mars arrival, if not the departure date.

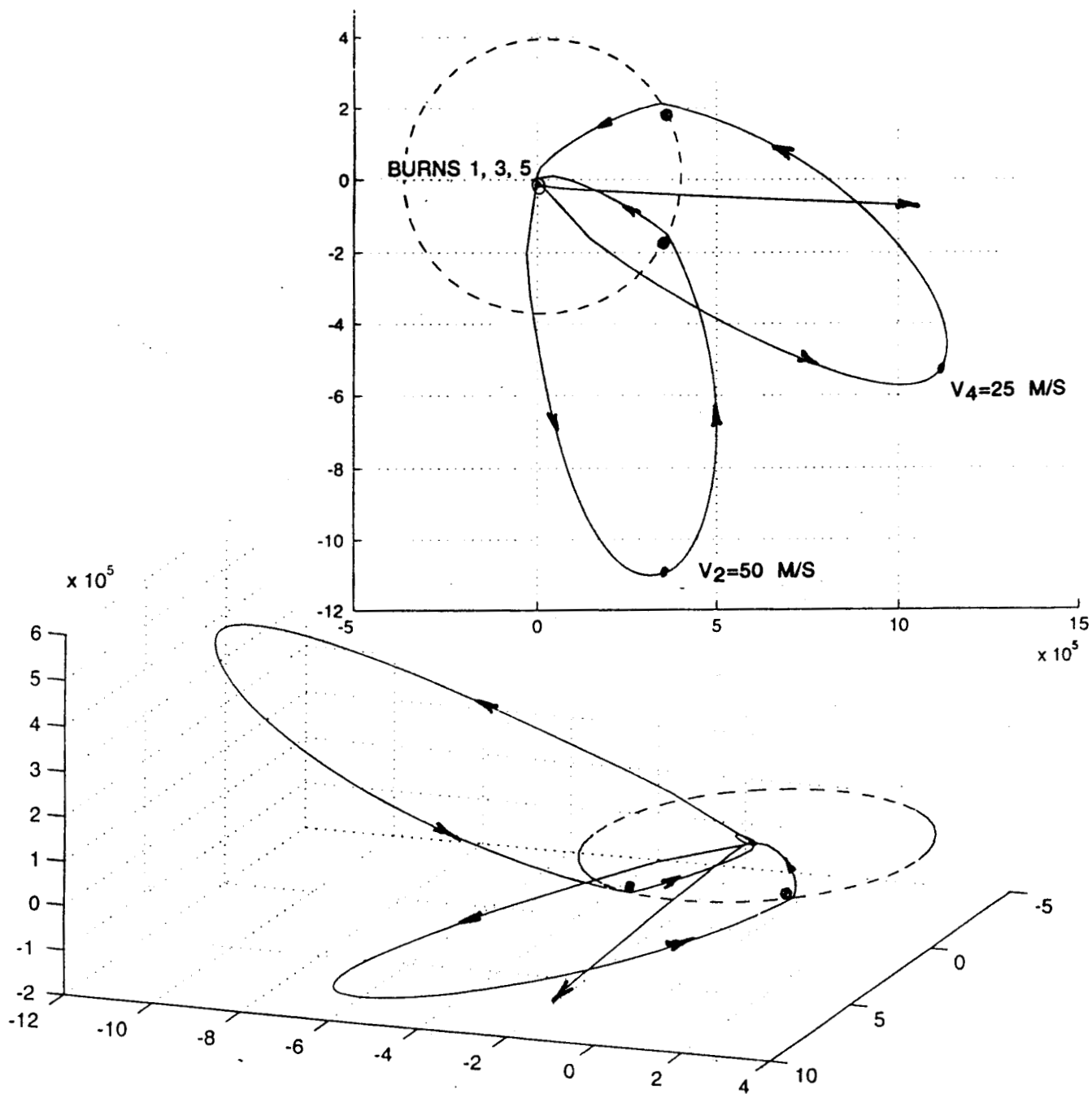


Fig. 5 Five-Burn Trajectory Plots for Mars 2003 Type 1

For the years of interest here, these dates and other parameters are shown in Table 3. It is assumed here that a three-month GTO launch period will be satisfactory, and that the 3-burn is sufficient. The 5- and 7-burn options would add 100 to 200 m/s more to the listed delta-v numbers, but would extend the launch period to earlier days and times per day. Included in the total are delta-v's for trajectory corrections, attitude control, or other small burns required at Mars, as listed in the footnotes.

Table 3. Carrier(C) and Orbiter(O) Missions for Mars 2003-2007

EARTH ESCAPE DATE	C_3 (km ² /s ²) ENERGY (TYPE)	GTO LAUNCH PERIOD	3-BURN DELTA-V (m/sec)	MARS ARRIVAL DATE	MARS* INSERTION VELOCITY (m/sec)	TOTAL** DELTA-V (m/sec)
<u>Year 2003</u> (mo day)						
(C) MAY 31	9(1)	FEB 2-MAY 3	1400	3DEC. 17	0	1550
(O) MAY 31	9(1)	FEB 2-MAY 3	1400	3DEC. 17	900	2700
<u>Year 2005</u>						
(C) AUG 4	16(1)	APR 10-JUL 3	1700	6FEB 22	0	1850
(C) SEP 2	16(2)	MAY 3-AUG 5	1700	6OCT 2	0	1850
(O) AUG 4	19(2)	MAY 3-AUG 5	1800	6JUL 12	900	3100
(O) SEP 3	21(1)	APR 10-JUL 3	1900	6MAR 27	750	3050
<u>Year 2007</u>						
(C) SEP 13	14(2)	MAY 16-AUG 15	1600	8AUG 19	0	1750
(O) SEP 13	14(2)	MAY 16-AUG 15	1600	8AUG 19	750	2750

- * Delta-V Requirements to Insert into a 40 Hour Orbit at a 250 km Altitude at Mars
- ** For Carrier: Navigational Corrections(100 m/s), Mars Maneuvers(50 m/s)
- ** For Orbiter: Nav. Corr.(100 m/s), Aerobrake(100 m/s), Orbit Raise(200 m/s)

It is interesting to note that in Table 3, the year 2005 is the most difficult. Velocity requirements decrease in 2007, and further still for the 2009 opportunity (not listed). In 2005, both the type 1 and the longer type 2 trajectories are essentially equivalent, but both are shown because the combined launch period extends from April to August, or 4 months. Also, the two provide a wider range of Mars approach velocity latitudes, -20 to 40 deg (see Table 1), which allows flexibility in probe landing sites for carrier missions, and orbit inclinations for orbiter missions.

In 2007, the type 2 longer flight time trajectory (about a year) was chosen because the type 1 is worse in all respects. The launch C_3 is 19 or higher, and the Mars approach velocities are about the same as type 2. Also, the type 2 trajectories have lower Mars approach latitudes, which allow a wider range of possible orbiter inclinations.

SUMMARY

Technology is moving rapidly in the development of smaller and smarter spacecraft which can perform planetary missions. The MEGA process described here provides a means for delivering these spacecraft to Mars as secondary payloads on the Ariane 5 launch vehicle, maintaining a balance between spacecraft, mission, and launch costs. An important element in the feasibility and acceptability of this mode of transportation is the ability to be independent of the day and time of the GTO launch, which is the primary emphasis of this paper.

It has been shown that a three month or more launch period can be developed, with significant latitude in the time of launch during the day. This has been applied to Mars launch opportunities of 2003, 2005, and 2007. It has been necessary to introduce the concept of multiple high ellipse loops to include all or most of the launch day/launch hour space. It is anticipated that additional MEGA modes will be discovered which will expand the launch space further, and add greater flexibility in the use of the Ariane ASAP launch capability, not only for Mars, but other planetary missions as well.

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