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Interplanetary Trajectory Optimization with Application to Galileo

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A procedure for minimizing total impulsive ΔV for constrained multiple-flyby trajectories, which was originally developed for application to satellite tours, has been modified for application to interplanetary trajectories. The modification includes adding to the cost function the ΔV required to escape from a parking orbit about the launch plant and the ΔV required for insertion into orbit about the arrival planet. The hyperbolic excess velocity vector with respect to the launch planet and the launch date have been added to the set of independent variables for the optimization. Each trajectory originates at departure from the parking orbit rather than at a fixed position in space, as is the case for the satellite tour application. The multiconic trajectory propagation techniques and the Newton optimization algorithm of the original method have been retained. Examples of the application of this new method are given for several types of Galileo interplanetary trajectory options, including Mars powered flyby, broken plane, Venus-Earth gravity assist, and ΔV Earth gravity assist trajectories.

Introduction

HISTORICALLY, trajectory design for an interplanetary mission required the solution of a deterministic boundary value problem. Given a launch date and an arrival date, the heliocentric Lambert problem is solved giving the hyperbolic excess velocity vectors at each planet. Data for many launch and arrival dates would then be presented graphically as the launch/arrival trajectory space or "pork chop" plot. Then mission analysis would typically determine a ten-day launch period, and full integrated trajectories with all perturbations would be generated for the spacecraft and mission design processes. This approach is not practical for more complex missions that may involve trajectories with one or more intermediate planet flybys for gravity-assist and/or large ΔV maneuvers between or near some of the flybys to achieve the desired flight path.

To date, trajectories for missions such as Mariner-Venus-Mercury, Pioneer 10 and 11, and Voyager 1 and 2 have been flown, which have multiple flybys but no large maneuvers except departure from parking orbit. The trajectories for these missions can be determined with some difficulty using standard targeting techniques. A method developed by Bayliss[1] solves the multiple flyby problem very efficiently when no ΔV's are required. His method segments the trajectory, which is continuous in position but has velocity discontinuities. It uses an optimization algorithm to drive the velocity discontinuities to zero, resulting in a continuous ballistic trajectory, if one exists. This method is an example of using an optimization technique to solve a targeting or boundary value problem.

For complex missions requiring trajectories with nonzero ΔV's and/or multiple flybys with constraints, a sophisiticated constrained optimization algorithm is absolutely necessary. Future missions, such as Galileo, Saturn orbiter with dual probe, and multiple asteriod rendezvous, will involve multiple planetary flybys, large deep-space ΔV's, powered flybys (large ΔV near a flyby), satellite flybys to assist in planetary orbit insertion, and satellite tours.

Previous papers by the authors[2-4] have discussed the application of constrained parameter optimization theory to the satellite tour trajectory problem. The approach involved segmenting the trajectory with one flyby per segment and with velocity discontinuities at the breakpoints, which are usually at apoapsis. These velocity discontinuities were minimized with most, if not all, becoming zero in the solution. In the optimization formulation, the satellite flyby parameters (altitude, B-plane angle, periapse time) were used as independent variables subject to range constraints. A Newton algorithm using a weighted quadratic cost function, analytic gradients, and approximate analytic second derivatives gave nearly quadratic convergence to a solution of minimum total ΔV. Multiconic trajectory models with two levels of accuracy were used for trajectory propagation.

This paper discusses the application of this constrained optimization method to the complex interplanetary trajectory options mentioned above. The same Newton algorithm is used for this new application and both high and low precision trajectory models are available. The cost function is modified in two ways. First, the departure ΔV from the launch planet parking orbit and the orbit insertion ΔV at the arrival planet are included in the cost. Second, the ΔV's must be converted into fuel expenditures since the spacecraft mass may change significantly due to staging during the course of the mission. Independent variables associated with the launch and arrival phases are also introduced. Unlike the satellite tour problem, an intermediate maneuver may be large and may occur in deep space (not necessarily at apoapsis) or quite near a flyby. Results for a number of Galileo mission options are presented.

Method

A computer program called PLATO (Planetary Trajectory Optimization) has been developed by the authors to solve constrained interplanetary trajectory optimization problems which involve intermediate flybys and/or large intermediate maneuvers. The PLATO program has been coded in Fortran V on a Univac 1100/81 computer at the Jet Propulsion Laboratory. There are two versions of the program corresponding to two different trajectory models.

Interplanetary Trajectory Optimization Problem

Figure 1 is a schematic representation of an interplanetary multiple flyby trajectory. The launch phase consists of in-

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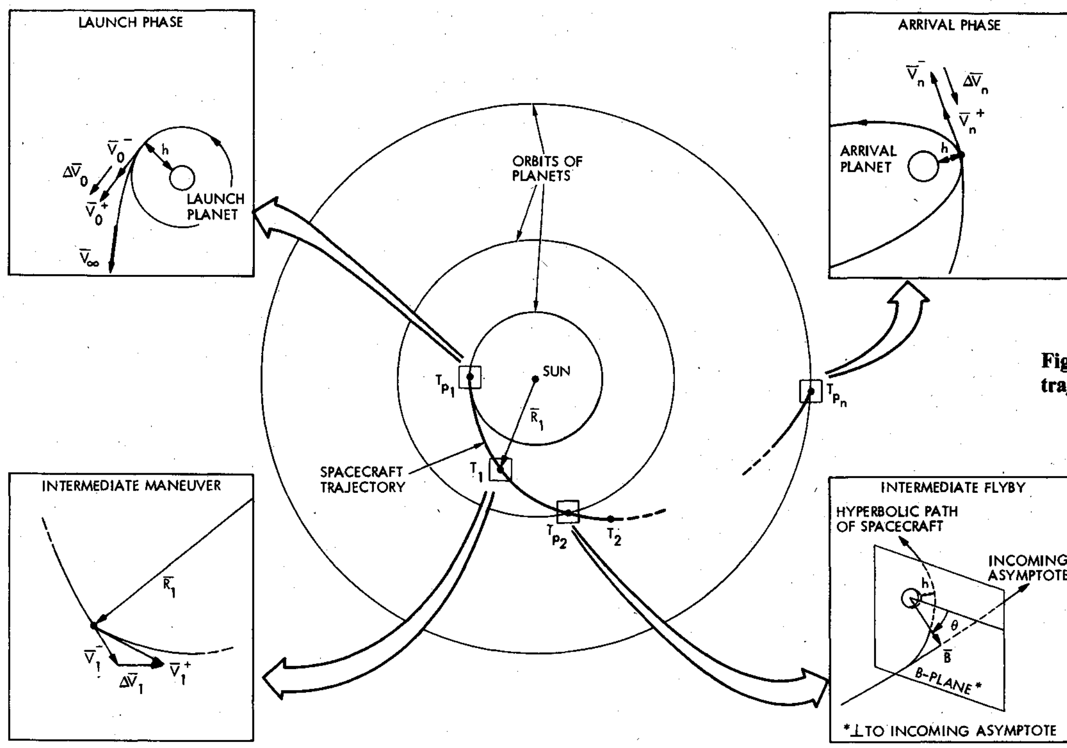


Fig. 1 Interplanetary trajectory schematic.

jection from parking orbit into the interplanetary trajectory by means of a tangential ΔV at periapse. At the first breakpoint (T_I), the trajectory is targeted to the first intermediate flyby. This results in a velocity discontinuity ($\Delta \bar{V}_I$). The trajectory is then propagated to the next breakpoint. This process continues through subsequent flybys terminating at the arrival planet, where insertion into a desired orbit is accomplished with a tangential ΔV at periapse. The cost function to be minimized is the sum of the fuel expenditures for each of the maneuvers on the trajectory. Scale factors for each ΔV, which are a function only of spacecraft mass at the maneuver and the engine being used, convert the ΔV's to fuel expenditures.

The interplanetary trajectory begins with a launch planet phase, where the spacecraft is injected onto a hyperbolic escape trajectory from a parking orbit. This is accomplished by means of a tangential ΔV at periapse. In Fig. 1, $\bar{V}_0^-$ is the parking orbit velocity, $\bar{V}_0^+$ the velocity at periapse on the escape hyperbola, and $\Delta \bar{V}_0$ the required injection impulse. The independent variables chosen to characterize the launch planet phase are the launch date, magnitude, declination, and right ascension of the V_∞ vector. These variables, along with the parking orbit periapse altitude and orientation, completely determine the launch phase trajectory.

When the launch phase trajectory is propagated to the first maneuver time (T_I), the position ($\bar{R}_I$) and arrival velocity ($\bar{V}_I^-$) are determined (see Fig. 1). The departure velocity ($\bar{V}_I^+$) is found by targeting to the altitude (h), B-plane angle (θ),‡ and periapse time (T_p) of the first intermediate flyby. This targeting problem is solved iteratively by adjusting the trajectory until errors in the target variables are sufficiently small. The resulting velocity impulse is $\Delta \bar{V}_I$. The trajectory is then propagated to the next breakpoint, and the targeting/propagation process is continued for subsequent flybys, terminating at the arrival planet. This determines all of the intermediate maneuvers.

The independent variables chosen to characterize the intermediate trajectory segments are the maneuver times and the altitudes, B-plane angles, and periapse times of each flyby.

‡The B-plane angle shown in Fig. 1 is referenced to the equator of the flyby planet.

At the arrival planet, a tangential, impulsive maneuver at periapse is used to establish an orbit of specified period with a specified periapse altitude and orientation. This maneuver ($\Delta \bar{V}_n$), shown in Fig. 1, is the difference between the arrival velocity ($\bar{V}_n^-$) and the velocity ($\bar{V}_n^+$) on the specified orbit. The only independent variable needed to characterize the arrival planet phase is the arrival time.

The formulation of the interplanetary trajectory optimization problem is analogous to that of the satellite tour problem discussed in Ref. 4, differing in the addition to the cost function of the ΔV's for the launch and arrival phases and the conversion of all the ΔV's to fuel expenditures by the use of appropriate scale factors. The set of independent variables is expanded from the flyby parameters (altitude, B-plane angle, and periapse time) and maneuver times to include the launch date and the magnitude, declination, and right ascension of the launch V_∞ vector. Range constraints (upper and lower bounds), as well as equality constraints, are allowed on any of the independent variables.

Optimization Solution

Since the form of the cost function in the interplanetary problem is identical to that of the satellite tour problem, the method of solution is also identical. A Newton algorithm using the gradient of the cost function and an approximation to the matrix of second derivatives, both determined analytically, gives nearly quadratic convergence to the solution. The partial derivatives of the intermediate ΔV's with respect to the new independent variables of the interplanetary problem and the partial derivatives of the launch and arrival phase ΔV's with respect to all of the independent variables are necessary additions to the formulation of the satellite tour problem. This Newton algorithm changes the values of the independent variables on each iteration in order to converge to a zero value for the gradient.

Trajectory Propagation

Two trajectory propagation techniques are available to the optimization procedure. The more complex technique uses a multistep, multiconic method which models the sun and an arbitrary number of planets. This technique is based on the work of Kwok and Nacozy,[5] which was derived from several sources,[6-9] and produces trajectories with accuracy approaching that of precision numerical integration with significantly less computational effort. The simplified technique uses a three-body dynamic model (sun, flyby planet, and spacecraft) and a one-step multiconic propagation method. This technique was developed by Byrnes[2] and is based on the pseudostate theory of Wilson.[6] In the interplanetary case, this technique provides an excellent approximation to a three-body trajectory, eliminating 90-99% of the error of simple conic methods.

Multiconic methods are especially well suited to the generation of trajectories in the current application because the state transition matrix on each trajectory propagation is calculated analytically without significant additional computational effort.

Results

Historical Perspective

As originally planned, the Galileo spacecraft was to be launched in 1982 on a trajectory that utilized a Mars gravity-assist flyby. Problems with the development of the Space Shuttle and the IUS upper stage delayed launch first to 1984 and then again to 1985. The 1984 mission, which involved separate launches for orbiter and probe spacecraft, employed a powered Mars flyby on the orbiter interplanetary trajectory. The 1985 mission, with a combined orbiter/probe spacecraft launched by a wide-tank Centaur upper stage, uses a broken-plane trajectory. Uncertainties about congressional approval of development of the Centaur upper stage has prompted the study of several other Galileo mission options, two of which are 1984 VEGA (Venus-Earth Gravity Assist) and 1985 ΔVEGA (ΔV-Earth Gravity Assist).

To provide examples of utilization of the optimization procedure, its application to these various mission options is described below. The first two (1984 Mars flyby and 1985 broken plane) are discussed at length; the others are addressed in less detail.

1984 Mars Powered Flyby Mission

The 1984 Mars powered flyby mission utilizes a powered gravity-assist flyby of Mars in order to reduce launch energy requirements for the Earth-Jupiter transfer. The Mars flyby is powered because an unpowered flyby, which derives all of the needed energy increase solely from the gravity-assist effect, would be a subsurface flyby. A typical 1984 Mars powered flyby trajectory is shown in Fig. 2.

The PLATO program has been used to analyze the 1984 Mars powered flyby launch/arrival space with the objective of minimizing total fuel required. For this mission, the injected spacecraft mass (2600 kg) and the launch vehicle (three-stage IUS) are specified; therefore, the maximum allowable C_3 is known (51.5 km^2/s^2). Under these conditions, the fuel required for the Earth injection ΔV is not included in the cost function. The problem, therefore, is the minimization of the total fuel required for the intermediate ΔV's and the arrival ΔV at Jupiter with an upper limit on C_3 for all trajectories.

The optimization variables for this problem and their relevant constraints are: V_∞ magnitude ($V_\infty \leq 7.18$ km/s), declination ($|DLA| \leq 28.4$ deg), and right ascension; the altitude (≥ 200 km), B-plane angle, and time of the Mars flyby; and the maneuver times on the Earth-Mars and Mars-Jupiter trajectory segments. Mission operations considerations dictate that the first maneuver may occur no sooner than ten days after launch, and neither maneuver may occur within three days of the Mars flyby.

For all of the results presented below, the following conditions apply:

1) The altitude of the circular Earth departure parking orbit is fixed at 278 km (150 nm).

2) The periapse radius for insertion into orbit at Jupiter is fixed at $4R_J$ (214,194 km altitude).

3) The Jupiter arrival ΔV is computed to establish an initial orbit period of 204 days.

In order to study trajectory characteristics as a function of both launch date and arrival date, optimal trajectories were generated for various combinations of fixed values for these dates. The range of launch dates is February 19-March 2, 1984, and the range of arrival dates is May 2-August 30, 1986.

These optimized trajectories have a ΔV at Mars which ranges from 650 to 1200 m/s and which should be performed as soon as possible *after* the Mars flyby (i.e., at Mars + 3 days). Also, the altitude of the Mars flyby is always at the lower limit of 200 km. The ΔV at Mars is caused solely by this altitude constraint because the optimal trajectory with no such constraint has a subsurface Mars flyby but no maneuver.

Over the launch/arrival space considered, there are three distinct classes of trajectories. All trajectories, except those in the upper left-hand region of the launch/arrival space, would have required a C_3 greater than 51.5 km^2/s^2 if not constrained and are therefore C_3-limited (see Fig. 3). Trajectories in the lower right-hand region (see Fig. 4) require a nonzero ΔV on the Earth-Mars trajectory segment, and the optimal time for that maneuver is as soon as possible after launch (Earth + 10 days). Thus the three classes of trajectories (see Fig. 5) are:

Class I	non-C_3-limited
Class II	C_3-limited/no near-Earth ΔV
Class III	C_3-limited/nonzero near-Earth ΔV

The Class I trajectories (early launch /late arrival) have an excess of launch energy available over that which is required because the C_3 optimizes to a value less than the upper limit. As expected, a maneuver is not necessary immediately after

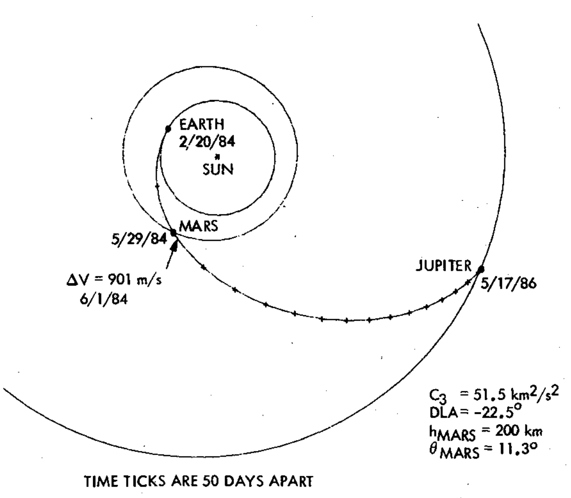


Fig. 2 Galileo 1984 mars powered flyby trajectory.

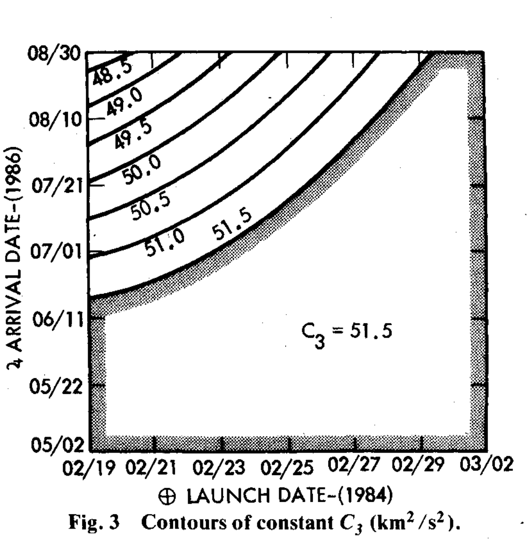


Fig. 3 Contours of constant C_3 (km^2/s^2).

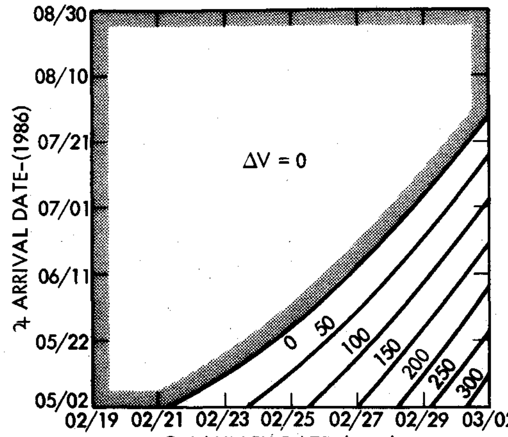


Fig. 4 Contours of constant ΔV (m/s) at Earth + 10 days.

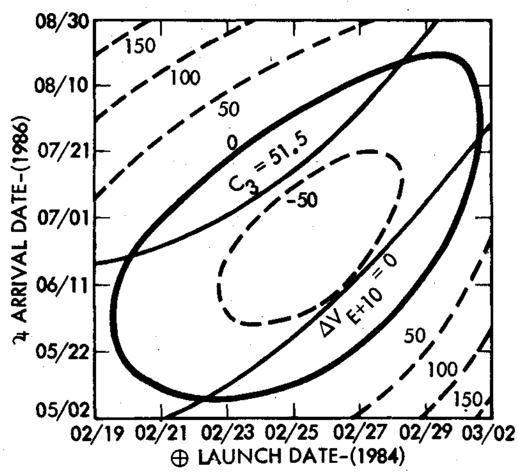


Fig. 6 Contours of constant Δ propellant loading (kg).

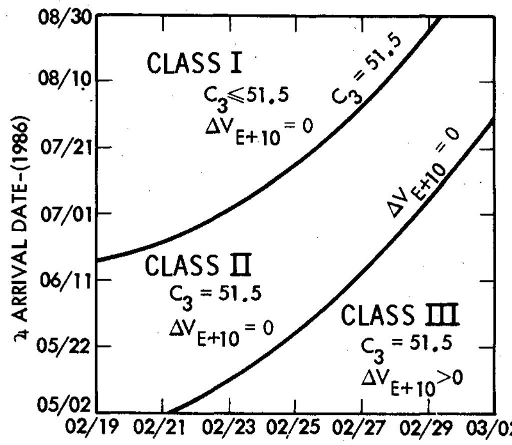


Fig. 5 Regions of the launch/arrival space for Class I, II, and III trajectories.

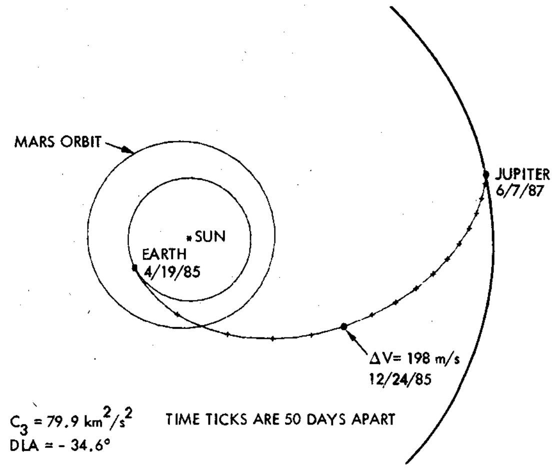


Fig. 7 Galileo 1985 broken-plane trajectory.

launch, since the launch vehicle can supply all of the required energy. For the Class III trajectories (late launch/early arrival), the launch vehicle cannot supply the necessary energy because the C_3 is at the upper limit. Not surprisingly, a maneuver is required as soon after launch as possible to increase energy in order to compensate for the limitations of the launch vehicle.

The existence of the class II trajectories is surprising. In this situation, the C_3 is at the upper limit, but no maneuver is required after launch to increase energy. Although there is insufficient energy at launch, this energy deficiency is small enough that it is optimal to increase the magnitude of the maneuver at the Mars flyby instead of adding a maneuver immediately after launch.

The selection of a launch/arrival strategy for this mission depends upon the total fuel requirement. By comparing total fuel required to the fuel available from the 2660-kg injected spacecraft mass, a quantity called "Δ propellant loading" is defined. This quantity is shown in Fig. 6 as a function of launch date and arrival date. A positive value indicates a condition of insufficient fuel (nonfeasible trajectory). A negative value indicates excess fuel capability (feasible trajectory), and the magnitude specifies how much fuel can be off-loaded from the spacecraft tank. Of course, if this excess fuel were carried through to the end of the mission, there would be an amount of fuel left in the tank which could be used for an extended mission. Note that the feasible region includes trajectories from all three classes.

The computation of Δ propellant loading takes into account the fuel required for the interplanetary trajectory and other fuel allocations for trajectory correction maneuvers, attitude control, satellite tour, etc. This computation is performed after each optimal trajectory is generated with PLATO.

1985 Broken-Plane Mission

The current baseline§ for Project Galileo is called the 1985 Broken-Plane Mission. A broken-plane trajectory includes a maneuver between launch and arrival which changes heliocentric inclination. Trajectories for this mission are Type I trajectories (transfer angle less than 180 deg), with no intermediate flybys. A typical 1985 broken-plane trajectory is shown in Fig. 7.

Launch energy (C_3) and launch declination (DLA) requirements are both very high for direct ballistic trajectories to Jupiter launched in 1985. In fact, the 1985 launch opportunity is the most unfavorable one for any year between 1980 and 1989. The reason for this is adverse Earth-Jupiter geometry. At arrival in 1987, Jupiter is almost exactly 90-deg from its node and, therefore, has a maximum out-of-the-ecliptic component. Consequently, minimum-energy ballistic trajectories have relatively large heliocentric inclinations. These can only be achieved by means of high-C_3, high-DLA launches.

With the current Shuttle/Centaur performance estimates and for a nominal spacecraft mass of 2500 kg, no direct

§Subsequent to the presentation of this paper, the Galileo mission has been reprogrammed twice: in January 1982 the 1985 ΔVEGA trajectory was selected as the new baseline due to cancellation of Centaur development, and in August 1982 the baseline was changed once again to a 1986 direct trajectory as a result of reinstatement of funding for Centaur development.

ballistic trajectories to Jupiter are possible in 1985. However, broken-plane trajectories are possible because the broken-plane maneuver allows the C_3 and DLA at launch to be significantly lower. For a fixed launch date, arrival date, and maximum allowable C_3, there is an optimal time for the broken-plane ΔV.

The PLATO program has been used to minimize fuel requirements for 1985 broken-plane trajectories in order to determine the feasible region of the launch arrival space. A feasible trajectory is one for which the total fuel required is less than or equal to the fuel available in the spacecraft tank. The mass of fully loaded spacecraft for this mission is 2500 kg. Based on current wide-tank Centaur performance estimates, the maximum C_3 available for a 2500-kg injected mass is 83.3 km^2/s^2. The cost function to be minimized includes the fuel required for the broken-plane ΔV and the Jupiter orbit insertion ΔV. The optimization variables for this problem are the V_∞ magnitude, declination (DLA), and right ascension, and the time of the broken-plane maneuver.

The optimization process is complicated by the high declinations of the broken-plane trajectories. The maximum allowable C_3 mentioned above is valid for $|DLA| \leq 28.4$ deg. If $|DLA| > 28.4$ deg, the Centaur must perform an out-of-plane injection maneuver, and the maximum allowable C_3 is reduced for the same spacecraft mass. For any value of $|DLA|$ greater than 28.4 deg, there is a corresponding maximum C_3 less than 83.3 km^2/s^2. This C_3/DLA relationship for the wide-tank Centaur is shown schematically in Fig. 8.

This curve divides the C_3/DLA space into an admissible region (available $C_3 \geq$ required C_3) and an inadmissible region (available $C_3 <$ required C_3). For the optimization procedure, this situation represents a nonlinear constraint involving two of the independent variables. The optimization formulation described above, however, only allows separate range constraints on each independent variable. This nonlinear constraint is handled with the following procedure:

1) Find the optimal trajectory with range constraints as follows: $C_3 \leq 83.3$ km^2/s^2 and $|DLA| \leq 28.4$ deg.

2) If $|DLA| < 28.4$ deg for this optimal trajectory, the optimal trajectory satisfying the nonlinear C_3/DLA constraint has been found in step 1.

3) If $|DLA| = 28.4$ deg, then a new optimal trajectory on the constraint curve is found by removing V_∞ from the set of independent variables and computing its value as a function of DLA during the optimization.

4) If the cost gradient indicates a lower cost in the inadmissible region, then the optimal trajectory satisfying the nonlinear C_3/DLA constraint has been found in step 3.

5) If the cost gradient indicates a lower cost in the admissible region, the optimal trajectory satisfying the nonlinear C_3/DLA constraint is then found by generating the unconstrained optimal trajectory.

In order to determine the feasible region in the launch/arrival space, optimal trajectories were generated for various combinations of launch date and arrival date. The range of launch dates is April 15-May 1, 1985, and the range of arrival dates is March 25-December 30, 1987.

Every optimal trajectory requires a broken-plane ΔV, the magnitude of which varies from 150 to 300 m/s. The optimal time for this maneuver can vary significantly across the launch/arrival space. For a fixed launch date and arrival date, however, the magnitude of the maneuver is insensitive to the time at which is is performed.

Two important quantities for defining the feasible region are launch margin (LM) and tank margin (TM). Launch margin is defined as the excess of launch vehicle capability over the required injected mass (2500 kg) at the optimal C_3 and DLA. The C_3/DLA constraint discussed above is exactly equivalent to a constraint of LM ≥ 0. Tank margin is defined as the propellant remaining in the spacecraft fuel tank at the end of the nominal mission, and is computed by subtracting from tank capacity (935 kg) the fuel required for the broken-plane maneuver, Jupiter orbit insertion, and other fuel allocations for trajectory correction maneuvers, attitude control, satellite tour, etc. Minimizing total fuel required on the interplanetary trajectory is equivalent to maximizing TM. A feasible trajectory is defined as a trajectory which has TM ≥ 0 and LM ≥ 0.

Contours of constant TM and the feasible region are shown in Fig. 9. All feasible trajectories have $|DLA| > 28.4$ deg, and both feasible and nonfeasible trajectories have LM = 0. This is in contrast to the results for the 1984 Mars powered flyby, where the Class I trajectories do not use all of the launch vehicle capability. The boundary of the feasible region (see Fig. 9) defines the locus of trajectories for which TM = 0 and LM = 0. All optimal broken-plane trajectories in the non-feasible region have TM < 0 and LM = 0.

It should be pointed out that the size of the feasible region is very sensitive to launch vehicle capability, which defines the C_3/DLA constraint. Relatively small changes in launch vehicle performance specifications can greatly enlarge the feasible region or, conversely, eliminate it entirely.

The three trajectories in Table 1 illustrate three methods of handling the nonlinear C_3/DLA constraint. Ignoring the constraint results in the "ballistic" solution, which has no broken-plane ΔV and a very large TM, but which is not feasible because launch vehicle capability is grossly exceeded. Satisfying the constraint in an overly restrictive manner results in the "restricted broken-plane" solution, which has a large positive LM, but which is not feasible because the broken-plane ΔV is too large. Satisfying the nonlinear constraint correctly results in the "optimal broken-plane" solution, the only feasible trajectory.

The data presented in Table 1 show that the broken-plane trajectory concept is an extremely effective method for

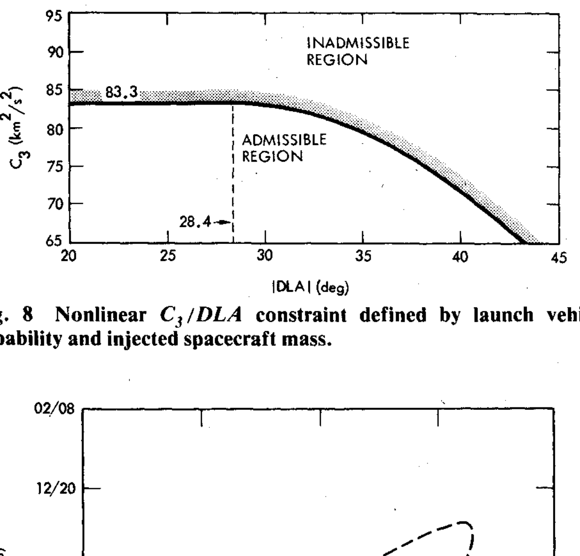


Fig. 8 Nonlinear C_3/DLA constraint defined by launch vehicle capability and injected spacecraft mass.

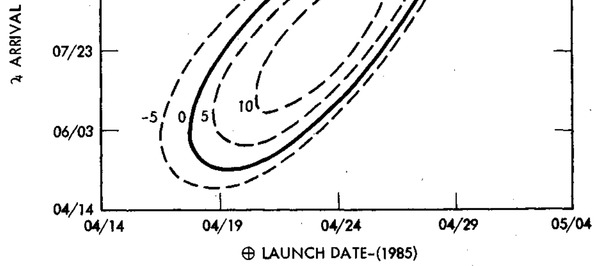


Fig. 9 Contours of constant tank margin (kg).

Table 1 Comparison of Ballistic and Broken-Plne Trajectory Characteristics

Launch Date = 4/19/85
Arrival Date = 7/23/87

Trajectory	C_3 km²/s²	DLA, deg.	Broken-Plane ΔV, m/s	LM, kg	TM, kg
Ballistic	87.5	−43.0	0	−905	110
Optimal Broken-Plane	81.1	−33.3	202	0	13
Restricted Broken-Plane*	79.5	−28.4	299	179	−32

LM = Launch Margin
TM = Tank Margin

*$C_3 \leq 83.3$ km²/s²
$|DLA| \leq 28.4°$

reducing C_3 and DLA. The optimal broken-plane trajectory, as compared to the ballistic trajectory, has much lower values for C_3 and DLA. The ballistic trajectory has LM = −905 kg and is, therefore, not feasible. On the other hand, the optimal broken-plane trajectory is feasible with TM = 13 kg and LM = 0 kg. LM is zero because the solution lies on the C_3/DLA constraint. The third row in Table 1 shows the characteristics of a broken-plane trajectory optimized with the following overly restrictive constraints: $C_3 \leq 83.3$ km²/s² and $|DLA| \leq 28.4$ deg. This set of constraints guarantees that LM ≥ 0 without considering the nonlinear portion of the C_3/DLA constraint. It excludes the admissible region of C_3/DLA values for $|DLA| > 28.4$ deg. The resulting trajectory has a broken-plane ΔV so large that TM = −32 kg, and the trajectory is therefore not feasible.

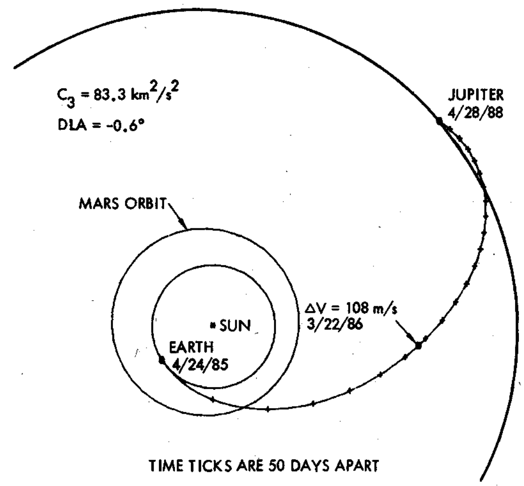


Fig. 10 Galileo 1985 type II broken-plane trajectory.

Other Galileo Mission Options

An expansion of 1985 broken-plane mission possibilities can be effected by considering Type II trajectories. The baseline mission discussed above is a Type I trajectory; that is, the heliocentric transfer angle from Earth to Jupiter is less than 180 deg. A Type II trajectory has a transfer angle greater than 180 deg but less than 360 deg. Figure 10 shows a typical 1985 Type II broken-plane trajectory. Given the current Centaur performance estimates and a spacecraft mass of 2500 kg, the 1985 Type II trajectories all require a broken-plane maneuver, as do the Type I trajectories. In this case, however, the need for the broken-plane maneuver is due entirely to high C_3 requirements, because the required DLAs are close to zero.

The analysis for the Type II trajectories is very similar to that for Type I and is simplified by not having the nonlinear constraint involving C_3 and DLA. Using the magnitude, declination, and right ascension of the V_∞ vector and the time of the maneuver as independent variables, optimal trajectories for fixed launch and arrival dates are found. The feasible region for Type II trajectories is much larger than that for Type I with correspondingly larger tank margins. However, the trajectories have much longer flight times to Jupiter, with arrival in 1988 and 1989. Thus, the 1985 Type II option is primarily a backup to the Type I baseline mission and would only be used if launch vehicle capability decreases from current specifications or if spacecraft mass estimates increase.

If the Galileo mission is delayed beyond 1985, a 1986 launch option is possible with both Type I and Type II trajectories. Since Earth-Jupiter geometry is more favorable in 1986, direct ballistic trajectories without a broken-plane maneuver are possible. However, the feasible regions for both the Type I and Type II trajectories can be expanded for near-180 deg transfers by allowing for a broken-plane maneuver. If the optimal trajectory has a zero broken-plane maneuver, then the optimization procedure will generate the solution to the standard interplanetary targeting problem.

Another mission option for Project Galileo is the 1984 VEGA (Venus-Earth Gravity Assist) mission. A VEGA trajectory includes gravity-assist flybys of first Venus and then Earth before proceeding to Jupiter. A typical 1984 VEGA trajectory is shown in Fig. 11. Note that the spacecraft completes one and one-half orbits about the sun after launch

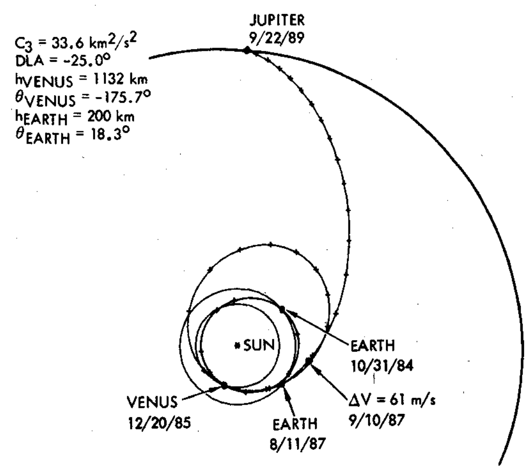
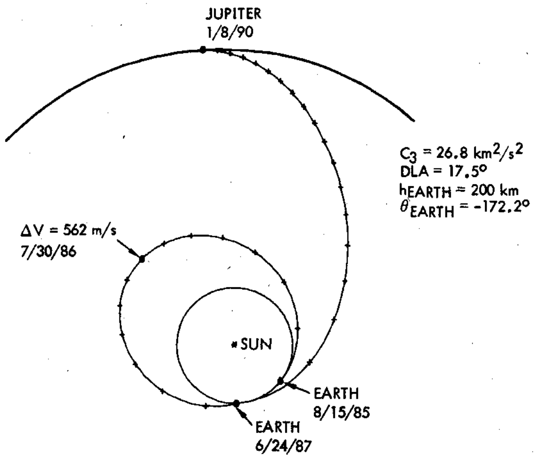


Fig. 11 Galileo 1984 Venus-Earth gravity-assist trajectory.

before encountering Venus and then one full orbit before the Earth flyby. An advantage of this mission option is the very low required C_3 (25—35 km²/s²). This mission option could thus serve as an alternative for the baseline 1985 broken-plane mission, if the wide-tank Centaur development were delayed or cancelled. However, the 5-yr flight times, with arrival in 1989, are unattractive, and the spacecraft would have to be redesigned because of increased heating loads.

In applying PLATO to 1984 VEGA trajectories, the optimization variables are the V_∞ magnitude, declination, and right ascension; the altitude, B-plane angle, and periapse times of the Venus and Earth flybys; and the times of the maneuvers on the three trajectory segments. Lower limits for the altitudes of the flybys are 200 km. Preliminary results for a range of launch/arrival dates show that the only nonzero intermediate ΔV occurs just after the Earth flyby, and this ΔV is caused by the constraint on the Earth flyby altitude. The



TIME TICKS ARE 50 DAYS APART

Fig. 12 Galileo 1985 Δ*V* Earth gravity-assist trajectory.

low C_3 requirements and the small interplanetary maneuvers result in large launch margins and tank margins.

A third mission option, which also has low C_3 requirements (27-30 km²/s²), is the 1985 ΔVEGA (Δ*V*-Earth Gravity Assist) mission. For this type of trajectory (see Fig. 12), the spacecraft is launched on a 2-yr orbit. Near apoapsis a relatively large deep-space maneuver is performed in order to target the spacecraft for a pre-perihelion Earth flyby which supplies the necessary gravity assist for a transfer to Jupiter. Total flight time is about 4½ yr.

The optimization variables are the $V\infty$ magnitude, declination, and right ascension; the altitude, B-plane angle, and periapse time of the Earth flyby; and the times of the deep-space maneuver and the maneuver after the Earth flyby. The Earth flyby altitude has a 200-cm lower limit. Preliminary results for a range of launch/arrival dates indicate that the magnitude of the deep-space Δ*V* is in the range of 500-600 m/s and the altitude of the Earth flyby always optimizes to the lower limit, but no post-Earth Δ*V* is required. This mission option has large launch margins, but requires addition of a separate propulsion module to the spacecraft due to the size of the deep-space maneuver.

Both the 1984 VEGA and 1985 ΔVEGA mission options are low-energy alternatives to the higher-energy, more direct trajectories exemplified by the 1984 Mars powered flyby or the 1985 broken plane. The main disadvantage of these alternatives is the much longer flight times.

Summary and Conclusions

A procedure for minimizing fuel consumption for constrained interplanetary multiple-flyby trajectories has been described and its application to a wide range of trajectory types has been demonstrated. This procedure is used to define a feasible launch/arrival space for use in mission planning, and has efficiently generated optimal trajectories for options ranging from direct ballistic trajectories to trajectories containing multiple flybys and intermediate maneuvers. The use of fast multiconic trajectory propagation techniques with a second-order Newton optimization method has allowed the development of this efficient mission analysis program.

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