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NAVIGATION STRATEGY FOR THE GALILEO JUPITER ENCOUNTER AND ORBITAL TOUR

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At Jupiter arrival on December 7, 1995, the Galileo Orbiter will have a close Io flyby (1000 km target altitude), record up to 75 min of science data transmitted from the atmospheric Probe, and then perform a large (-650 m/s) Jupiter Orbit Insertion maneuver. The ensuing 2-year "orbital tour" includes ten close satellite flybys at altitudes between 250 and 3100 km. Four Jupiter-approach maneuvers and three maneuvers per orbit arc planned to achieve accurate delivery of the Orbiter at each satellite encounter. Precise orbit determination is accomplished with S-band Doppler-data and optical navigation pictures as the primary data types. At each orbit trim mane.uvcr, the remaining orbital tour trajectory will be rc-optimized to minimize total AV by varying satellite aimpoints within allowable bounds. Satellite delivery errors for the orbital tour arc generally less than about 40 km (fl-plane) and 2s (closest approach time). 'l'he velocity charge (AV) required 10 navigate the orbital tour is about 70 rids, and the end-of-mission propellant margin is 20 kg (both 90% probability values).

INTRODUC'I'1ON

Interplanetary Trajectory

The scientific objective of the Galileo Mission is to carry out an intensive investigation of Jupiter's atmosphere, satellites, and magnetosphere (Ref. 1). On October 18, 1989, the Galileo spacecraft departed Earth bound for Jupiter. Galileo's 6-year-long journey to Jupiter is now nearly complete, and preparations and planning for the Jupiter encounter and orbital tour arc proceeding on schedule (Ref. 2).

Figure 1 shows the Earth-to-Jupiter interplanetary trajectory. The Venus-Earth-Earth gravity assist (VEEGA) phase of the trajectory was completed in December 1992 with the second of the two Earth gravity-assist flybys (Ref. 3). In August 1993, Galileo completed a successful flyby of the asteroid Ida. The results of this encounter included the discovery of Dactyl, the first natural satellite of an asteroid ever directly observed, (The Ida flyby was Galileo's second asteroid encounter; the first occurred in October 1991 when Galileo flew by Gaspra for the first-ever close-up observations of an asteroid.)

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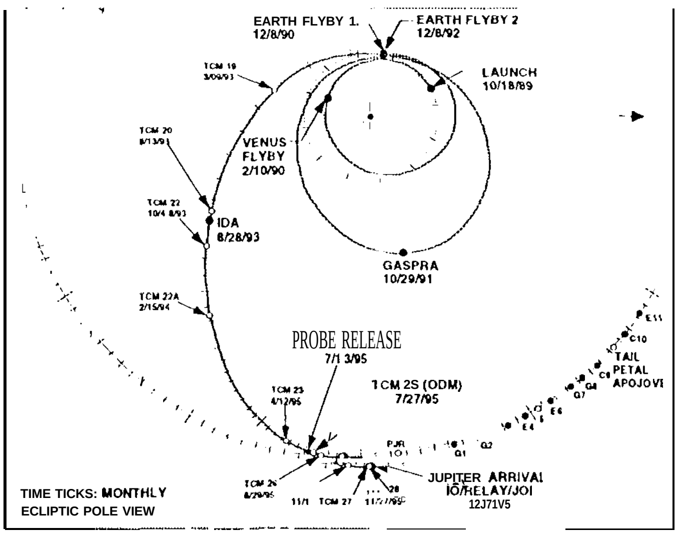

Figure 1 Galileo larth-Jupiter VEEGA Interplanetary Trajectory

In October 1993, about 1 month after the Ida flyby, the Galileo spacecraft, consisting of an Orbiter and an attached atmospheric entry l'robe, was retargeted to Jupiter. Since the Probe has no onboard propulsion system, the Orbiter/Probe spacecraft must bc placed on a ballistic trajectory targeted such that, once released, the Probe will achieve the desired atmospheric entry aimpoint (defined as an altitude of 450 km above the reference Jupiter oblate spheroid). Two subsequent small trajectory correction maneuvers (TCMs) were performed in February 1994 and April 1995 to remove small errors in the Probe entry conditions prior to Probe release.

Also shown on Figure l arc the TCMS between the Earth 2 flyby and Jupiter arrival and the locations of the satellite encounters during the orbital tour.

Jupiter Approach and Encounter

Figure 2 shows the final Jupiter approach portion of the interplanetary trajectory. On July 13,1995, the Probe was successfully released from the Orbiter. Probe entry is scheduled to occur on Dccembcr 7,1995 at 22:04 UTC. Two weeks after]'robe release, on July 27, 1995, the Orbiter deflection maneuver (ODM) was completed. This maneuver established the Orbiter on a trajectory that is targeted first to fly by Io at an altitude of 1000 km for a gravity-assist to reduce thevelocity charge (AV) for Jupiter orbit insertion (JOI) and then to overfly the Probe during its descent to enable the Probe-tcJ-Orbiter radio link. There are four TCMs scheduled between ODM and the 10 fl yb y; the first TCM corrects ODM errors, and the fi na] three (see Figure 3) are used to fine tune the **10** aimpoint.

Figure 4 shows the Orbiter and l'robe trajectories at Jupiter arrival. The Io flyby occurs about 4 hours before closest approach (4.4 hours before the start of Probe data acquisition). This gravity-assist flyby slows the Orbiter, reducing the JOI AV by 175 m/s and also provides the only opportunity during the orbital tour for close-ui~ observations of 10. As an added bonus (which accrues from the selection of December '),1995, as the Jupiter arrival date), the Orbiter will have a 32,000-km altitude fl yby of Europa 4.6 hours before the Io fl yby. The Orbiter passes through Jupiter closest approach at a distance of 4.0 R_J (1 R_J = one Jupiter radius = 71,492 km), Starting a few minutes later, the Orbiter will receive and store data transmitted from the Probe for 75 min as it descends through the Jovian atmosphere. About

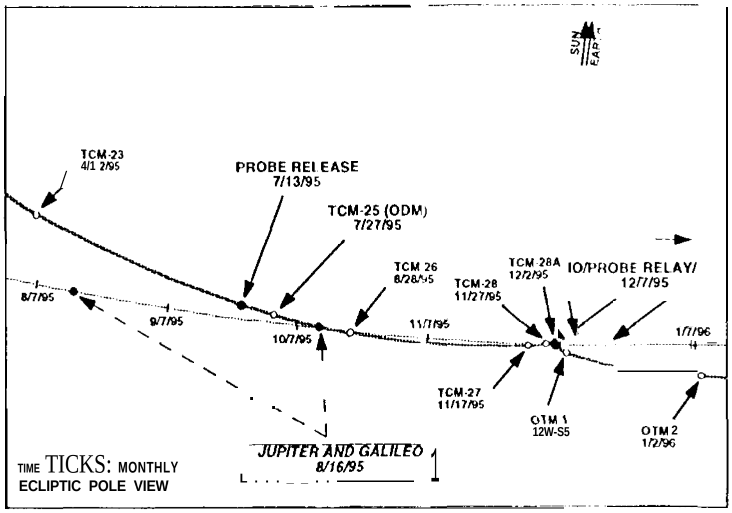

Figure 2 Interplanetary Trajectory Showing Final Jupiter Approach Through Jupiter Orbit Insertion

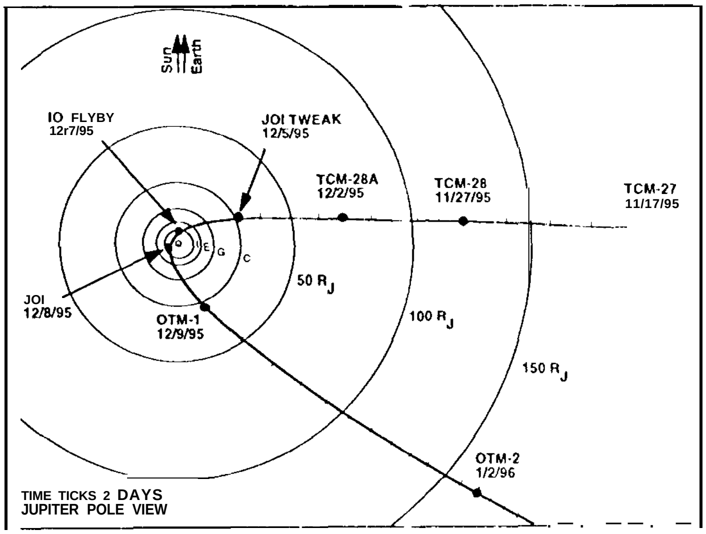

Figure 3 Jupiter Encounter: Orbiter Trajectory Showing Maneuvers During Final Approach and Orhit Insertion

I hour after the end of Probe data acquisition, the orbiter performs the JOI maneuver slowing the spacecraft in order to establish the initial 21 O-day orbit about Jupiter. The first two orbit trim maneuvers (OTMS) arc used to correct Io flyby altitude errors and JOI execution errors (see Figure 3): onc day after JOI and the second about a month later, after the Orbiter passes through solar conjunction. The times of the Jupiter encounter events arc given in '1'able 1.

Orbital Tour

The orbital phase of the mission, referred to as the "orbital tour," lasts 2 years (see Figure 5). Near apojove of the initial orbit, a perijove raise maneuver (PJR) is performed; this maneuver occurs nominally on March 18, 1996. PJR increases the speed of the Orbiter in

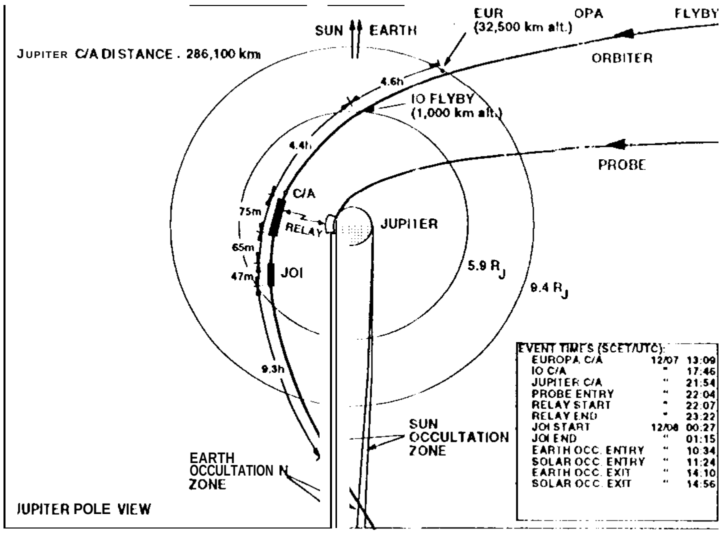

Figure 4 Jupiter Encounter Close-Up: Probe and Orbiter

Table 1
JUPITER ARRIVAL EVENT TIMES

Event	Date	Time (hh:mm) UTC/SCET	PST/ERT	ΔTime (hh:mm)
Europa Flyby	7-Dee-95	13:09	06:01	-08:56
Io Flyby	7-Dee-95	17:46	10:38	-04:19
Jupiter Closest Approach	7-Dee-95	21:53	14:45	-00:11
Probe Entry	7-Dee-95	22:04	14:56	0
Probe Relay Start	7-Dee-95	22:07	14:59	00:03
Probe Overflight	7-Dee-95	22:34	15:26	00:30
Probe Relay End	7-Dee-95	23:22	16:14	01:18
JOI Start	8-Dee-95	00:27	17:19*	02:23
JOI End	8-Dee-95	01:15	18:07*	03:10
Enter Earth Occultation	8-Dee-95	10:34	03:26	12:30
Exit Earth Occultation	8-Dee-95	14:10	07:02	16:05
OTM-1	9-Dee-95	15:40	08:32	17:36
OTM-2	2-Jan-96	20:00	12:52	21:56

*Date is 7-Dee-95. ATime = time from Probe entry.

order to raise the perijove distance and target the Orbiter to the first encounter of the satellite tour: Ganymede 1 on Jul y 4, 1996. It is necessary to raise the perijove distance at the start of the orbital tour to avoid excessive exposure to the intense radiation environment close to Jupiter. (After the first 4 R_J perijove passage, the spacecraft has already received about one-third of the total permissible radiation dosage.)

During the orbital tour, the Orbiter completes eleven orbits about Jupiter, ten of which contain a close flyby of one of the three outermost Galilean satellite.s: Europa, Ganymede, and Callisto (Figure 5). These "targeted" satellite encounters are at altitudes between 200 and

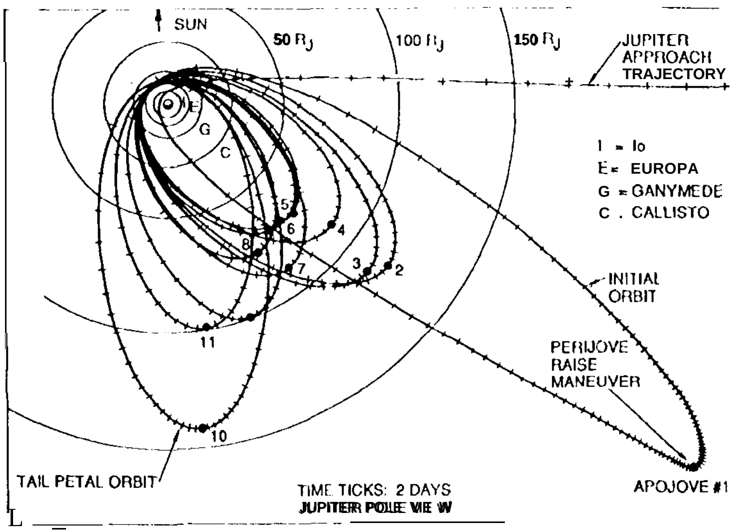

Figure S Galileo Orbital Tour Trajectory

3000 km. In addition, on four of the eleven orbits, there is a second, more-distant satellite flyby. These "nontargeted" encounters range between 23,000 and 80,000 km. The satellite encounters arc listed in Table 2. Navigation of the Jupiter approach/encounter and tour phases of the Galileo mission is a challenging task. Over a time period of slightly more than two years, there arc sixteen satellite encounters and forty planned propulsive maneuvers for navigation. There arc typically three OTMs per orbit: a prc-encounter OTM at encounter (E) -3 days to fine tune the satellite aimpoint, a post-encounter OTM at E + 3 days to correct for satellite flyby dispersions, and a third OTM near apojove.

The remainder of this paper addresses the Galileo navigation strategy for the Jupiter approach and orbital tour phases of the mission in terms of the three major navigation functions: orbit determination, trajectory design and analysis, and maneuver design and analysis. In addition, a discussion of the effects on navigation of the unavailability of the high-gain antenna (HGA) is included.

EFFECTS OF LOW-GAIN ANTENNA MISSION

The original navigation plan assumed the use of the HGA. However, when the HGA was commanded to open in April 1991, it only partially deployed. After numerous unsuccessful efforts to free the stuck HGA over the subsequent 2 years, it was finally decided in March 1993 to revise the mission plan and navigation strategies for a mission based on the low-gain antenna (LGA). The resulting irregular LGA X-Band (7200 MHz) downlink pattern precluded such a mode of operation, and the radiometric configuration was thus restricted to S-Band (2100 MHz) for both uplink and downlink. The omni LGA performance is approximately 40 dB lower than the expected performance from the directional HGA. The LGA mission has affected the acquisition of radiometric Doppler and range data and optical navigation (OPNAV) pictures used for orbit determination.

Table 3 presents a summary of the changes to the navigation process because of the LGA mission. Although these changes have, in most cases, resulted in a reduction in navigation capabilities and the amount of navigation data available, they have not caused a degradation in navigation performance. The navigation process has been modified and enhanced to successfully counter the deleterious effects of the LGA mission. The following discussion deals with some of the details of the changes caused by the LGA mission.

Table 2
ORBITAL TOUR ENCOUNTERS

Encounter	Date	Satellite	Altitude (km)	Latitude (dog)	Objective
G1	4-Jul-96	Ganymede	500	25	Wake, Alfven wing, UVS, gravity, reduce period
G2	6-Sep-96	Ganymede	259	85	Alfven wing, gravity, reduce inclination
C3	4-Nov-96	Callisto	1102	' 1 3	Wake, Alfven wing, UVS counter-rotate for atmospheric coverage, Jupiter occultations (Sun, Earth)
E3A	6-Nov-96	Europa	32150	0	Coverage (232° W. Long., ϕ=34°)
E4	19-Dec-96	Europa	6 9 ;	0	Wake, Europa occultations (Sun, Earth), Jupiter occultations (Sun, Earth)
(E5A)	20-Jan-97	Europa	27332	-1	Occurs during solar conjunction interval on phasing orbit
E6	20-Feb-97	Europa	587	-17	Europa occultations (Sun, Earth), Jupiter occultations (Sun, Earth), 10 occultation
E7A	4-Apr-97	Europa	22998	2	Coverage (1 33° W. Long., ϕ=51°), distant wake
G7	5-Apr-97	Ganymede	3056	56	Alfven wing
C8A	6-May-97	Callisto	33662	-42	Coverage (72° W. Long., ϕ=44°)
G8	7-May-97	Ganymede	1580	29	Ganymede occultations (Sun, Earth), Jupiter occultations (Earth), distant Uvs
C9	25-Jun-97	Callisto	416	2	Callisto occultations (Sun, Earth), Jupiter occultations (Earth), 10 occultations, tail petal
G9A	26-Jun-97	Ganymede	80006	' o	Coverage (98° W. Long., ϕ=20°), distant wake
Tail Peta Apojove	8-Aug-97				143 R_J, ϕ=175°, 0.2° inclination
C10	17-Sep-97	Callisto	524	5	Wake, Alfven wing, Jupiter occultations (Sun, Earth), rotate, UVS, reduce period
E11	6-Nov-97	Europa	1124	66	Alfven wing

W. Long. = West Longitude
41= phase angle (sun-satellite-spacecraft angle)

Table 3
SUMMARY OF CHANGES TO NAVIGATION PROCESS BECAUSE OF LOW-GAIN ANTENNA MISSION

	HGA Mission		LGA Mission
Two-Way Doppler Tracking:			
Cruise	12 Passes per Week		3 Passes per Week
Satellite Encounters	±5 Days (30 Passes)		± 1 Day (6 Passes)
OTMS	-3,+5 Days (24 Passes)		2 Passes Around OTM
Range Process and Accuracy:			
Process	Spacecraft Transponder		Ramped Doppler
Uplink	S-Band	S-Band	S-Band
Downlink	S-Band	X-Band	S-Band
Accuracy (1 σ)	50 m	25 m	500 m
OPNAV Processing:			
Single Picture Data Set	640,000 Pixels		2000-3000 Pixels
Accuracy (1 σ)	2–4 μrad		2–4 μrad
On Board Processing Time	None		-20 min
OPNAV Picture Budget:			
Entire Tour	-600		-180
Per Orbit	-60		10-35
Nongravitational Error Sources:			
Earth-Pointing Limit	±0.1°		±4.0°
OPNAV Total Turn Angle	-300°		0°

Radiometric Data Coverage

For the original HGA mission, two-way Doppler tracking and high telemetry data rates for science and engineering were compatible, allowing an abundance of two-way Doppler data even with other projects competing for Deep Space Network (DSN) resources. For the LGA mission, telemetry data rate capabilities have been significantly restricted, and the two-way Doppler tracking coverage has thus been reduced by about a factor of six as shown in Table 3. Since the two-way Doppler tracking requirements for the original HGA mission were somewhat conservative, it is possible to offset the loss of Doppler data for the mission based on the LGA by concentrating the reduced coverage during critical times.

The use of the spacecraft range transponder in conjunction with the DSN sequential range assembly to acquire range data has been precluded at Jupiter distances due to the low signal levels. An alternate technique of ramping the Doppler uplink signal in a sawtooth pattern (previously used during the Pioneer 10 and 11 Missions) was adapted and refined for Galileo to allow an equivalent range measurement with an accuracy of 1 km or better. This range data improves the orbit determination solutions and provides more accurate determination of the fright time to Jupiter, thereby reducing the uncertainty in the altitude of the 10 flyby (which is directly correlated with flight time).

Optical Navigation (OPNAV) Picture Processing

Pictures taken with the spacecraft camera showing target bodies (i. c., Galilean satellites) against a field of stars with known positions are used to augment radiometric tracking data in the orbit determination process. The OPNAV data improves satellite delivery accuracy by reducing the dependency on the satel 1 ite ephemerides generated based on gtound-based observations.

In order to make OPNAV picture data available with the restricted telemetry data rates associated with the LGA mission, OPNAV processing now requires a technique in which new soft ware on board the spacecraft extracts selected imaging data from each picture and then returns this data to Earth. The ncw fright software algorithm detects and extracts from the picture imaging data from only the limb and terminator regions of the observed satellite; a small box of imaging data surrounding each star is also extracted. The extracted target body and star data is temporarily stored in spacecraft memory until it is read out and transmitted to Earth in the telemetry data stream. This process takes about 20 minutes for each OPNAV picture. (Compared to the total amount of data in a single OPNAV picture, the extracted data returned to Earth represents about a 300:1 compression ratio.) Then additional ncw OPNAV ground software is used to estimate. the target body and star image locations from fragments of the imaging data returned.

This process allows acquisition of a reasonable number of OPNAV data points in the tour. The total number of OPNAV pictures currently being scheduled per orbit ranges from about ten to thirty-five; the number of OPNAV pictures is larger for the earlier orbits. For the HGA mission, it was expected that as many as six pictures would have been scheduled for each orbit. Whereas this larger picture budget would have allowed multiple picture mosaics to bc utilized to insure capture of satellite image.s, this insurance is now provided by late updates to spacecraft pointing for the OPNAV pictures.

Nongravitational Force Modeling

The spacecraft attitude is adjusted periodically to point the LGA toward Earth within a prescribed angular range to maintain telemetry performance. These attitude adjustments use a "balanced thruster" turn mode that nominally imparts no AV to the spacecraft. However, there is a small residual AV caused by these turns, and it must be accounted for in the orbit determination process as onc of the nongravitational forces acting on the spacecraft. The LGA pointing requirement is to keep the antenna pointed nominally within 4.0° of Earth, an angular limit that is 40 times larger than the 0.10 limit required for the HGA. This results in a substantial reduction in the number of attitude updates required, which fortuitously removes most of their contribution to errors in the orbit determination solutions. In addition, all turns previously required for OPNAV pictures have been deleted, which further reduces errors arising from nongravitational forces. This has resulted in an appreciable improvement in orbit determination performance.

ORBIT DI3TIJRM1NATION STRATEGY

introduction

Accurate orbit determination during both the Jupiter approach/Io flyby and orbital tour is necessary to insure successful science data acquisition and to minimize the propellant required to correct trajectory errors resulting from satellite delivery errors. Prior to each propulsive maneuver, the position of the spacecraft and the target satellite arc simultaneousl y solved for using all the radiometric and optical data available. This process consists of differencing observations, which have been calibrated for media and Earth platform effects, with predictions of the same observables computed from an assumed initial state and a precise model of the appropriate gravitational attractions and the measurement geometry. The differences (residuals) arc then processed to produce an estimate of the spacecraft state that mm-c closely compares with the observations. This state, together with updates to planet and satellite ephemerides as well as other mode] parameters, and the associated covariance matrix describing the accuracy of the estimates, arc the final products of the orbit determination process and arc used to generate the spacecraft trajectory. These solutions arc then used to calculate the required maneuver.

Data Types

Two primary data types arc used for orbit determination: (1) two-way coherent Doppler, which is a measure of the range rate of the spacecraft with respect to the tracking station and (2) optical navigation (OPNAV) images of the Jovian satellites against a background at known stars. These images are acquired using the SS1 imaging instrument onboard the spacecraft. Analysis of these images provides the right ascension and declination of the imaged satellite with respect to the spacecraft.

Two-way Doppler data during the Jupiter approach is scheduled to be at least two DSN passes per week, and continuous coverage (three passes a day) from 1 day before to 2 days after each maneuver. Onc pass will be scheduled as late as possible before the data cutoff for each propulsive maneuver.

Jupiter Approach

The two-way Doppler data schedule following ODh4 consists of continuous coverage (three passes per day) around each TCM (from 1 day before the trajectory correction maneuver [TCM] through 2 days post TCM), continuous coverage from July 11 through July 30 (ODM–9 days to ODM+3 days), two passes per week from July 30 through October 9, and four passes per week from October 9 through December 1. Doppler data is weighted at 2 mm/s for a 60-second count time up to November 20, at which time the data weight is loosened to 5 mm/s to account for the increase in data noise due to (he approach of solar conjunction. Three OPNAV images are scheduled, The first OPNAV is of Io and is acquired on October 27. Transmission of the image is completed on November 8. 'l'he next two images are of Europa and arc acquired on November 8 and November 19 respectively. Transmission of these images is completed on November 19 and December 1. The data weight applied to these images is 0.35 pixel, Range data utilizing the ramped uplink technique will be obtained every 2 weeks from August 7 to November 1.

Estimated parameters include the spacecraft state at epoch, a constant representing solar radiation pressure, the Earth and Jupiter ephemerides, Jupiter's gravitational constant plus J2 and J4 harmonics, two parameters describing the direction at Jupiter's pole, the velocity impulse on the orbiter due to Probe separation, and velocity changes from all thruster firing events, including propulsive maneuvers, attitude changes, spin rate changes and retropulsion model (RPM) maintenance events.

Although the spacecraft state prior to Probe separation is known fairly accurately in heliocentric space, the process of probe separation and ODM as well as errors in the Jupiter ephemeris, can result in an altitude error of several hundred kilometers at the 10 flyby. Due to the Jupiter gravity, the in-plane component of this error causes a detectable change in the observed Doppler signal as the Orbiter approaches Jupiter. This results in a significant improvement in knowledge of the B•T and time-of-flight components of the Io target parameters. The out-of-plane component (B•R) can be detected from the OPNAV data. Table 4 lists the 1 s B-plane uncertainties in the 10 B-plane at the data cutoff time for each of the approach maneuvers.

Table 4
ORBIT DETERMINATION UNCERTAINTIES FOR 10 DELIVERY

Data Cut Off Time (Days)	Maneuver	Number of OPNAVs	Uncertainties (1 σ) B•R (km)	B•T (km)	LTOF (s)
IO-114	TCM 26	0	258	395	28.4
10-27	TCM 27	1	64	108	8.1
10-19	TCM 28	1	60	110	8.3
10-11	TCM 28 Tweak	2	45	102	7.7
10-6	TCM 28A	3	32	54	4.1
10-3	JOI Tweak	3	32	26	1.4

Orbital Tour

The Doppler data schedule during the tour consists of three passes per week during the cruise phase between encounters and 48 hours of continuous coverage from 1 day before to 1 day after each satellite encounter. As during the Jupiter approach phase, continuous coverage is also scheduled from 1 day before to 2 days after each TCM. OPNAV data is acquired during the time between the apogee OTM and one day before the E+3 OTM. The number of OPNAV images for each orbit ranges from ten to thirly-five.

The data spans utilized to support the maneuver designs arc different for each maneuver in a typical orbit. For the E+3 day OTM, the data arc begins 5 days before the encounter and ends 8 hours after the encounter. For the apojove OTM, the data arc extends to 7 days before the maneuver and data arc for the E–3 day OTM ends 1 day 3 hours before that maneuver. For the next orbit, a new data arc is started 5 days before the next encounter. Simulations have shown that including more than one encounter in the same data arc can result in numerical problems in integrating the Orbiter trajectory.

As in the Jupiter approach phase, the orbit determination process consists of utilizing the data in the current data arc to estimate all the parameters needed to determine the orbiter trajectory. The trajectory and the associated uncertainties arc mapped forward to the satellite closest approach time and transformed into target-centered B- plane coordinates. The differences between these coordinates and the desired encounter conditions (target errors) arc then used to compute the required OTM.

The parameters estimated during the tour are essentially the same as the parameters for the Jupiter approach phase with the addition of the ephemerides of the Galilean satellites. The orbiter trajectory following a satellite encounter is quite sensitive to encounter conditions. Therefore, knowledge of Orbiter epoch state will not be propagated forward from orbit I to orbit I+1 in order to avoid numerical instabilities. in contrast to this approach (for the spacecraft ephemeris), satellite ephemeris improvements accruing from previous encounters will be propagated forward to subsequent orbits. Thus, it is anticipated that the satellite ephemeris uncertainty will progressively decrease throughout the tour. This strategy should result in a significant improvement in the target accuracy for the later encounters. If we believe that the data is not adequate to estimate the effect of uncertainties in accurately modeled model parameters, we will include the effect of uncertainties through a considered analysis.

The new onboard optical editing technique developed for the current mission will be installed on the Orbiter prior to the G 1 encounter. This will enable the return of an OPNAV image in about 20 minutes, compared to the several days required to return a full image. Ideally, the satellite and at least two stars should be visible in the field of view. Then with a priori knowledge of the star positions, the direction to the satellite can be inferred in both right ascension and declination. Due to the use of the new data editing algorithm and a dearth of bright stars, most (90%) of the available OPNAVS will incorporate only a single star, resulting in incomplete directional information. This problem can be alleviated simply by scheduling more OPNAVs. However, the number of OPNAV images that can be scheduled for a given orbit is necessarily restricted. Too many OPNAVS can cause significant loss in the amount of science data returned. Too few OPNAVS can result in large navigation errors, resulting in large propellant expenditure. Another complication is the fact that, due to possible boom obscuration, the probability of a successful OPNAV is expected to be about 0.55. At least 180 OPNAVs must be shuttered to insure that about 100 will be successful] y ret urned.

Table 5 lists the 1-0 orbit determination uncertainties in the satellite B-plane expected at the times of the tweak cutoff, assuming the data collection strategy outlined above (including the number of OPNAVS currently scheduled).

Table 5
ORBIT DETERMINATION UNCERTAINTIES FOR PRE-ENCOUNTER OTM TWEAK

	Uncertainties (1 σ)			
Encounter	B*R (km)	B*T (km)	LTOF (s)	Number of Successful OPNAVS'
GI	15	21	1.0	18
G2	9	3	0.2	13
C3	19	42	1.6	16
E4	10	5	1.6	13
E6	13	3	0.7	10
G7	20	16	0.6	8
G8	10	17	0.2	5
C9	7	8	0.8	17
Clo	11	7	0.4	16
EII	20	8	2.5	20

TRAJECTORY OPTIMIZATION STRATEGY

Jupiter Approach and Initial Orbit

The Jupiter approach phase of the Galileo Mission includes optimally designed Probe and Orbiter trajectories that result in the Orbiter being at an altitude of about 3 R_J and essentially above the Probe as the Probe descends through the atmosphere. As depicted in Figure 4, the Orbiter will pass about 32,500 km above Europa's surface about 9 hours before Jupiter closest approach. This flyby will provide an initial opportunity to view Europa even before the orbital tour begins. In contrast, the closest approach of either Voyager spacecraft to Europa was a distance of 206,000 km, At about 4.1 hours before Jupiter closest approach, the Orbiter trajectory is designed to pass 1000 km above the surface of Io (at 17:45:44 UTC on December 7, 1995), providing both excellent science return as well as a gravity assist to reduce the required JOI AV by about 175 m/s or about 20%.

As a result of earlier TCMS during the interplanetary phase, the Probe trajectory is designed to reach an altitude of 450 km above the reference 1-bar pressure level in the atmosphere of Jupiter at 22:04:26 UTC on December 7, 1995. The Probe will enter the sensible atmosphere of Jupiter with a speed of about 47.4 km/s (atmosphere-relative) at a flight path angle of-8.6 ^{0} (also atmosphere-relative). Also, as result of earlier TCMs and the gravitational effects of the Europa and 10 flybys, the Orbiter trajectory is designed to pass through Jupiter closest approach at21 :53:32 UTC on December 7, 1995 or about 11 minutes before the Probe enters the atmosphere. About 3 minutes after Probe entry the relay of data from the Probe to the Orbiter begins and is scheduled to last for about 75 minutes. After the acquisition of Probe data is complete, the Orbiter requires 65 minutes to reconfigure itself for the JOI maneuver. JOI is performed with the main 400-N engine and lasts for just over 47 minutes. This results in the Orbiter being captured into a Jupiter orbit with a period of about 210 days.

The initial Jupiter orbit is designed to accomplish several competing objectives. Since the total time from Jupiter arrival to the completion of the orbital tour is limited to 2 years, the more time spent in the initial orbit, the less is available for the tour. The greater the period of the initial orbit however, the smaller is the JOI AV. Additionally, since the Orbiter can only survive one passage through the. Jovian radiation environment at a distance of 4 R_J, the PJR maneuver near apojove of the initial orbit is required to raise the subsequent perijove to a safe level. The larger the initial orbit, the smaller PJR will be. As a consequence of these considerations, an initial period of about 210 days was chosen.

PJR will be performed sometime within a window between March 13 and March 23, 1996 (102 ± 5 days after JOI). The optimal time of PJR (for minimizing overall propellant consumption) will be selected after JOI. PJR will not only raise the perijove distance to about 11 R_J, but will also accomplish the phasing necessary for Galileo to encounter Ganymede on July 4, 1996, to begin the orbital tour.

Orbital Tour

The process of designing the Galileo orbital tour, including the various constraints involved in the trajectory optimization process, is discussed in detail in Ref. 4, which also includes a thorough description of the resulting baseline tour. Table 2, which shows the sequence of satellite flybys, and some corresponding geometrical parameters and science objectives, is an updated version of the corresponding table presented in Ref. 4. This baseline orbital tour has been established as the reference for any changes caused by reoptimization of the tour trajectory as the mission progresses through each of the satell itc encounters of the tour.

From a trajectory design point of view, the basic process of orbital tour navigation is to continual y reopt imize the tour in order to cou nteract traject ory dispersions that arise from many sources. The orbit determination process produces a best estimate of the spacecraft state, with an associated uncertainty, at any given point in time; these state estimates differ from the state of the reference tour. Similarly, the ephemerides and masses of the Galilean satellites are being continually updated based on radiometric tracking data and optical navigation images. Thruster firings for OTMs produce AVS that are slightl y different than the design values. Given these various perturbations that cause the spacecraft to deviate from the reference tour, a new, slightly different trajectory for the remainder of the orbital tour will be determined regularly throughout the tour at specific OTMS (as discussed in the maneuver strategy section below) in order to minimize the AV (and thus propellant) required to complete the tour.

It is necessary to allow reoptimization of the orbital tour since Galileo has a limited supply of propellant remaining at the start of the tour for OTMS and for controlling the attitude and spin rate of the spacecraft. If each OTM were to target the Orbiter as closely as possible back to the baseline tour, the propellant expenditure would far exceed the propellant available onboard the spacecraft. The remainder of this section will describe the 'method used to accomplish the tour reoptimization and the operational limitations under which the tour reoptimization must operate.

Trajectory Optimization Constraints

During the orbital tour, uncertainties in the orbit determination process, the modeling of the satellite ephemerides and masses, and the execution of propulsive maneuvers all contribute to delivery errors at the satellite flybys. These errors cause trajectory dispersions that must be corrected in an optimal fashion to minimize the total AV (and thus propellant consumption) for the remainder of the tour. The trajectory optimization process that led to the reference tour (Ref. 5), is reapplied regular] y throughout the tour at specific OTMS. For the trajectory reoptimization process, however, there arc two major differences from the original design process. First, a reference. trajectory already exists and the changes to it arc expected to be relatively small. Second, the trajectory reoptimization process is subject to a number of Operational constraints. Due to stringent limits (to control total Project cost) on the amount of redesign of the sequences that will be loaded on the spacecraft to perform the science observations, the changes in the satellite aimpoints arc limited to ±50 km (B-plane components) and ±3 minutes (closest approach time) with respect to the values of the reference tour. The latest time at which a satellite aimpoint can be changed is at the post-encounter OTM of the previous encounter. Subsequently, the aimpoint of the upcoming flyby becomes fixed for the remaining OTMS prior to that encounter.

Additional constraints arise from maintaining certain geometrical conditions that arc characteristic of the reference tour, The "nontargeted" satellite flybys, which are denoted in Table 2 with an "A" (e.g., E3A, the Europa encounter following the Callisto encounter C3), arc also subject to constraints, -though somewhat less restrictive that those for the "targeted" flybys. The nontargeted fl ybys arc more distant fl ybys that occur on the same leg as a targeted flyby and provide opportunities for substantial science data such as global SS1 and NIMS coverage. The navigation process, however, can only control the aimpoint of a single satellite flyby on a given orbit, so after the aimpoint of a targeted flyby has been fixed at the post-encounter OTM of the previous flyby, the aimpoint of any associated nontargeted fl yby on that orbit can no longer be controlled.

Following two of the flybys during the tour, on the orbits following E6 and C9, science opportunities exist for the Orbiter to be occulted by Io as viewed from the Earth. These occultations occur when the Orbiter is several million kilometers from Io, but the situation is such that this geometry requires a critical value of the jovocentric orbital inclination. During the tour, inclination changes associated with the trajectory reoptimization process would normally cause these occultations to be lost. Therefore, the orbital inclination on these two orbits is also constrained during tour reoptimization.

Trajectory Optimization I'recess

The design of the reference orbital tour as described in Ref. 4 used trajectory optimization software developed and refined over the last decade and a half (Ref. 6). A recently developed trajectory optimization program (Ref. 5) that generates optimal, numerically integrated trajectories subject to arbitrary constraints was used to generate the updated tour described in Table 2. This new software will be used in the trajectory reoptimization process during tour operations.

The trajectory reoptimization that occurs during the orbital tour is somewhat less complex than the optimization required for the original design ol'the tour since on] y relatively small changes to the baseline are allowed. These small changes, however, arc what allow significant propellant savings to be realized over a more simplistic navigation strategy. The cost function for the trajectory optimization program is the sum of the magnitudes of the AVS for the remaining OTMS in the tour. The independent variables arc chosen from consistent sets of variables, which include satellite flyby parameters (altitude, B-plane angle, and time of closest approach) and jovocentric orbit parameters. The entire set of independent parameters describing the orbital tour is then varied by the optimization program to minimize the cost function subject to the various constraints described above.

In addition to the integrated trajectory optimization software described above, there also exists optimization software based on a linearized trajectory model (Ref. 7), which computes a first estimate of the necessary changes to the satellite fl yby aimpoints in order to reoptimize a dispersed trajectory. For small dispersions, the linearized-model results are often quite accurate; and since the results can be computed quite rapidly, this software is used extensivel y to evaluate a Monte Carlo ensemble of possible dispersions to generate tour AV statistics. This process is quite useful for determining end-of-mission propellant consumption estimates (discussed in the propellant margin status section below) under varying assumptions and for developing maneuver strategies as described in the following section.

MANEUVER STRATEGY

The navigation maneuver strategy is focused on all aspects of the mission related to the planning and implementation of propulsive maneuvers (as opposed to attitude control maneuvers) designed to remove trajectory errors. These manta vers are planned throughout the mission to ensure accurate control of the spacecraft trajectory, giving strong consideration to fault scenarios which may disrupt nominal planning. Accurate trajectory control satisfies science observation trajectory requirements while minimizing propellant consumption. The discussion that follows introduces the Galileo maneuver system and the strategies to be used during the Jupiter approach phase and orbital tour, and briefl y discusses some of the design issues relating to maneuver implementation.

Spacecraft I'repulsion System

The Galileo spacecraft is spin-stabilized using a unique dual-spin design accommodating the divergent requirements of both fields and particles instruments and remote sensing science instruments. All AVS required in the Galileo mission are performed by the RPM mounted on the spinning portion of the spacecraft. The propulsion system was provided by the Federal Republic of Germany and built under contract by Daimler-Benz Aerospace (DASA, formerly Messerschmitt-Bölkow-Blohm or MBB). It is a bipropellant system, with mono-methylhydrazine for the fuel, and nitrogen tetroxide for the oxidizer, and is fed by a pressurized helium system. The RPM includes twelve 10-Newton (10-N) thrusters and onc large 400-Newton (400-N) main engine. The 10-N thrusters are separated into two clusters of six thrusters each, and arc used for most trajectory correction maneuvers and control of both the spacecraft pointing and spin rate.

Figure 6 is a schematic drawing of thc RPM configuration with respect to thc spacecraft coordinate directions. Two sets of redundant S-thrusters (–S 1 A, –S 1 B, and S2A, S2B) arc used to maintain thc nominal spin rate and also to spin-up anti spin-down for probe release and for400-N burns. '1'here arc four 10- N-Z thrusters (–Z 1 A, –Z2A, –Z I B and –Z2B) and onc 400-N engine oriented parallel to the spacecraft spin axis. These thrusters impart ΔV in the spacecraft's –Z direction (HGA direction). 'Two L-thrusters (1.1 B and L2B) arc canted 10° from the lateral direction and implement ΔV anywhere within the plane perpendicular to the spin axis through proper timing of the thruster pulses as tbc spacecraft rotates. It is apparent from Figure 6 that there arc no thrusters positioned to effectively implement a ΔV in the spacecraft's +Z direction. Onc 10N thruster (P1 A) is canted 210 from the lateral direction to provide the largest available ΔV component (sin(21°)) in the +Z direction. This thruster's cant angle was limited to avoid plume impingement on the I] GA.

At launch, the usable propellant comprised 925 kg of [he initial spacecraft mass of 2561 kg. To date, approximate] y 210 kg of propellant has been used by the 10N thrusters. The 400N engine is planned to bc used three times in the mission, providing the required ΔV for the three largest maneuvers: ODM, JOI, and I'JR. The 400-N engine could not bc used until the Probe was released (separation date was July 13, 1995), since the Probe covered the 400-N nozzle en route to Jupiter. It is expected that approximately 600 kg of propellant will bc expended through the 400-N engine over an 8-month period beginning with the first firing of the 400-N engine which occurred in late July of 1995. Most of the remaining propellant will bc used by the 10-N thrusters to navigate and control the attitude and spin rate of the spacecraft during orbital operations. A detailed discussion of the RPM system is given in Reference 8. The current status of end-of-mission propellant margin is discussed in a later section .

ΔV Mechanization

'I'hc spinning thruster configuration allows for a wide variety of methods for implementing a particular trajectory correction maneuver (TCM) ΔV vector. In "vector mode," the spacecraft does not change orientation during the maneuver activity. An arbitrary

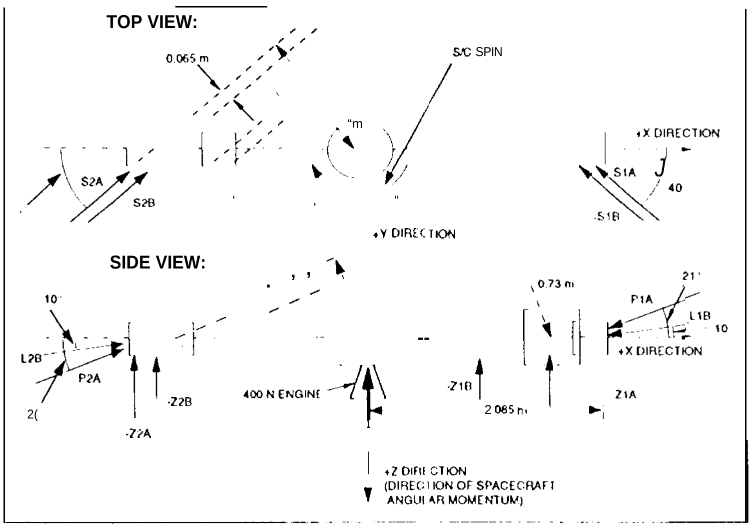

Figure 6 Galileo Propulsion Thruster Configuration and Coordinate Directions

ΔV vector is implemented through the sequential firing of the axial (–Z thrusters or P1A thruster) and lateral (L1B and L2B) thrusters. The nearly orthogonal ΔV components form the desired ΔV vector. This mode can be expensive in terms of propellant due to the sum of the components (rather than the resultant) being implemented, as well as the high cost if the axial component is in the +Z direction. ΔV in the +Z direction is approximately three times more costly in terms of propellant than the same ΔV in the --Z direction due to the 210 cant angle of the P1A thruster, Implementation constraints or the operational advantages of not turning the spacecraft usually determine when the vector mode strategy will be used. A discussion of the available Galileo spacecraft ΔV implementation modes is summarized in Reference 9.

Typically, in the absence of constraints, the optimum mode for maneuver implementation involves a reorientation of the spacecraft at ti t udc, followed by a burn to complete the required velocity change. Reorientation of the spin axis is accomplished through gyroscopic action induced by thruster supplied torques. If the P-thrusters (P1A and P2A) are used to supply the torque, then the two thrusters are fired simultaneously, once per revolution of the spacecraft, to induce the desired precession. Since the two thrusters point in opposite directions, no net ΔV is imparted and the turn is referred to as a balanced turn. If two of the-Z thrusters are used to suppl y the torque, then alternating thrusters are fired every half revolution of the spacecraft to induce the desired precession. Since these two thrusters point in the same direction, then in addition to precession there is a ΔV imparted to the spacecraft. This is referred to as an unbalanced turn and the applied ΔV must be considered in the design of the maneuver. Unbalanced turns are usuall y preferred overbalanced turns for the simple reason that the --Z thrusters have a moment arm about three times that of the P-thrusters and thus require less propellant for the same turn angle.

Implementation of maneuvers using the 400-N engine is different than maneuver implementations using the 10-N thrusters. Although the 400-N engine is more efficient than the 10-N thrusters in terms of the specific impulse (approximately 13% higher specific impulse), there is substantial overhead to prepare the spacecraft for a 400-N engine firing. In addition to the increased complexity of preparing the customized commands required to use the 400-N engine, extra propellant is expended to pi ace tile spacecraft in high spin mode (10.5 rpm vs 3.15 rpm) before any 400-N burn and return to low spin mode after the burn. The propellant cost for the two required spin mode transitions is approximately 2.6 kg and typically offsets any efficiency advantage of the 400-N engine.. There are aiso other propulsion system constraints that limit the use of the 400-N engine to the three planned events.

ΔV Capability

TCM implementation for Galileo is a complex and time consuming activity on the spacecraft. As mentioned previously, the thrusters used for TCMs are operated in pulsed mode. Because of thruster operation constraints and attitude and spin perturbations, only a limited number of pulses can be performed before spin and attitude corrections are required. As a result, the **ΔV** to be implemented in a TCM window on a single day (or "portion") must be broken up into "segments," each typically separated by an attitude correction and/or spin correction. Each of these activities uses memory and takes time to complete (typically about 20 minutes per attitude correction or spin correction). Any sequence which contains a maneuver activity must have a portion of the Command and Data Subsystem (CDS) memory reserved for the activity. This memory, referred to as a High-Level Module (HLM) box, limits the number of activities that can be performed within a TCM window. Because the whole TCM activity (including commands to warm-up gyros, configure heaters, perform attitude and spin corrections, etc.) must fit within one HLM box (853 bytes during cruise and the initial orbital phase), and the TCM must typically be monitored by one shift (typically 8 to 10 hours) on the ground, the amount of AV per portion is necessarily limited.

TCM-22 (imp] lemented in October 1993 to retarget Galileo to Jupiter after the Ida encounter) illustrates how these constraints affect the maneuver design. The total required AV for TCM-22 was approximately 39 m/s. The constraints specified above limited the AV that could be implemented in a single portion to just under 8 m/s. As a result, this particular TCM required 5 portions to implement the required AV. A discussion of the maneuver strategies **used** during the interplanetary phase of the Galileo mission can be found in Reference 10. The amount of AV that can be implemented in a single day is also affected by the mode selected to implement the maneuver, If turns arc required to reorient the spacecraft to and from the burn attitude, then more memory and time must be used by the turn events and less memory and time can be used implementing the AV of the maneuver.

Jupiter Approach/Orbit Insertion

The view of the trajectory in Figure 3 illustrates the Io approach maneuver strategy. After ODM on July 27, 1995 and through PJR on March 18, 1996 (nominal date), a sequence of 8 maneuvers is planned to de] i ver the spacecraft from an interplanetary trajectory to a trajectory which links to the satellite tour described earlier. The sequence of events include four 10 approach TCMs (of which three. are shown in Figure 3), a tweak (or update) of the onboard JOI burn parameters, a close flyby of 10, Probe relay, JOI, two post-JOI OTMS, and PJR.

The 10 flyby is critical to the orbit insertion success as the close 10 flyby provides a significant gravity assist, reducing the JOI AV requirement by approximate] y 175 m/s. As a result of this 10 gravit y assist, any deviation of the actual Io flyby from the planned 10 f] yby will directly result in a change to the AV required at JO I. The dominant error source contributing to a change in the AV required for JOI is an Io flyby altitude error. The goal of the pre-Io TCMs is to minimize the delivery error with respect to the desired 10 target. Delivery error predictions are such that a significant 10 flyby error is likely even after the successful implementation of TCM-28A (the last Io approach maneuver). An opportunity has been provided to detect and compensate for most of the Io flyby error by providing for a tweak to the required JOI burn magnitude, the nominal value of which at this point is already stored onboard the spacecraft. in the nominal sequence of planned events, a tweak of the nominal JOI burn parameters is planned to be uplinked to the spacecraft two days before JOI execution. Any residual flyby error that cannot be accounted for by the JOI tweak, and execut i on errors of the J 01 maneuver itself, arc p] anned to be corrected by the sequence of maneuvers beginning at OTM - 1, about one day after JOI executes. PJR is the first of this sequence of maneuvers targeting to the final aimpoint for- the Ganymede 1 encounter. The date of PJR is allowed to move (to minimize propellant usage) to any of 11 dates over the interval from March 13 to March 23, 1996 (nominal date is March 18, 1996), the actual date being selected after OTM- 1 has executed.

The planned sequence of events during the approach and insertion phases of the mission is quite complex. It is clearly desirable to minimize the number of engineering activities required during this crucial phase of the mission. This desire, however, must be balanced against the significant propellant costs anticipated for correcting trajectory errors and flying the precise tour selected well in advance of our arrival at Jupiter. The JOI tweak is a powerful tool for minimizing the effect of Jo flyby altitude errors on mission AV costs. Given the current understanding of Io delivery errors, there is a high probability of being in a situation where a tweak will be desirable. There is a large increase in AV cost for delaying the correction for 10 flyby errors from the JO] tweak opportunity to OTM-1. This correction delay results in the introduction of a deterministic AV component, or bias, in OTM- This OTM- 1 bias can easily become large and significantly reduce the AV capability available to correct for any errors in the execution of the JOI maneuver. There is a factor of three increase in AV cost for delaying the correction of 10 flyby errors from the JOI tweak opportunity to OTM- 1, and an additional factor of three cost for delaying OTM- 1 corrections to the OTM-2 opportunity. Real-time decisions will determine which activities will actually be performed. These decisions will be based on frequent orbit determination updates during the approach and insertion phases. The above strategy is robust to single event failures and provides the trajectory control opportunities necessary to ensure. a successful orbit insertion.

Orbital Tour Strategy

The orbital operations phase immediately follows the orbit insertion phase of the mission. At this point in the mission, the Probe data has been transmitted to Earth, PJR (the last 400N maneuver) has been executed, and the loading of the flight software required for orbital operations (see Refs. 2 and 11) has been completed. OTMs over the subsequent two years will control the trajectory during the planned eleven orbits about Jupiter, subject to the constraints agreed upon during the design of the science observation sequences.

The baseline tour OTM sequence, illustrated in Figure 7, typically consists of three OTMS per orbit about Jupiter. The three OTMs typical] y occur at or near apojove, 3 days prior to a targeted satellite encounter and 3 days after a targeted satellite encounter. The nominal plan is to implement the last pre-encounter OTM in “vector mode” and the post-encounter and apojove OTMS in “turn-burn mode”. The apojove OTMs arc typically statistical maneuvers. However, two apojove OTMS do have significant deterministic (or nominal design) AV components.

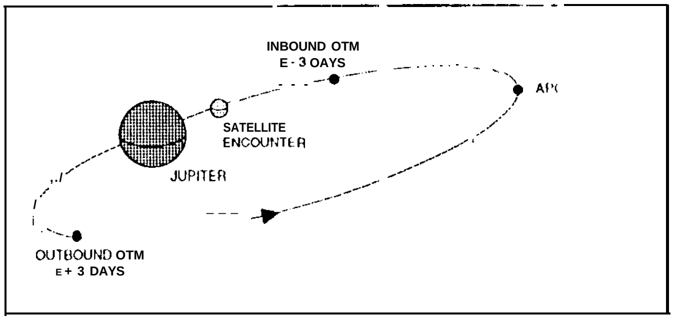

Figure 7 Typical OTM Sequence for Single Orbit During Tour

The pre-encounter OTM is planned to occur near the encounter while avoiding interference with the science observations of the encounter. Nominally the OTM is placed at E-3 days; however, this maneuver epoch may be adjusted forward or backward a day in order to accommodate science activities associated with the encounter or to minimize propellant consumption.

The post-encounter OTM is typically placed at E+3 days. The E+3 day epoch is chosen to allow for post-encounter Doppler tracking data to be used in the design of what is typically the first of a two-maneuver sequence designed to correct the encounter fl yby errors and achieve the desired aimpoint at the upcoming, encounter. This strategy is standard for all orbits except for the post Callisto OTM on the. ninth orbit. in this situation, there is an additional targeting constraint. Near apojove of the post-C9 orbit (also referred to as the tail petal orbit) there is an opportunity to target the spacecraft such that the radio signal to the

Earth is occulted by Io (a Radio Science experiment, approximately a 5-minute signal interruption). There is an opportunist y to achieve up to five of these distant 10 occultations near apojove if the trajectory can bc accurately controlled after the Callisto flyby. As such, the sequence of post encounter OTMS must target to the Io occultations on the outbound leg of the trajectory and then, after the occultations, begin the targeting to the upcoming satellite encounter (another encounter with Callisto).

Aimpoint Optimization Constraints

It is important to note that the final aimpoint for each encounter during the orbital tour is subject to reoptimization updates during the actual design of OTMs. These updates to the reference aimpoints incorporate the latest orbit determination information in a complete reoptimization of the tour, subject to agreed-upon constraints. The process of trajectory reoptimization during an OTM design is identical to the process used to design the original tour as was discussed in a previous section. The only difference is the brief window of opportunity to complete the reoptimization after the encounter and prior to the uplink of the post-encounter cl can-up OTM. Typically, there arc 14 hours allocated in the design templates of each of the post-encounter OTMS to accomplish this task.

As mentioned previously, the reference aimpoints at each encounter can, in general, bc changed by ±50 km about the reference aimpoint and by ±3 minutes in the time of satellite closest approach. Figure 8 is an illustration of the aimpoint constraint in the B-plane. (The definition of the B-plane is given in the Appendix and in Figure A-1.) An additional constraint on the target time of closest approach results from a desire to case the process of science sequence updates for each encounter. This desire results in the constraint that changes in the time of closest approach (with respect to the reference tour design) arc restricted to bc in increments of 60.667 seconds, or 1 RIM. RIM, Real-time Image Count, is onc of the larger fundamental units of time for the Galileo onboard computer. Changes to sequenced events in 1 -RIM increments arc simple to accommodate in the sequence update process and case the job of block shifting a series of science observations.

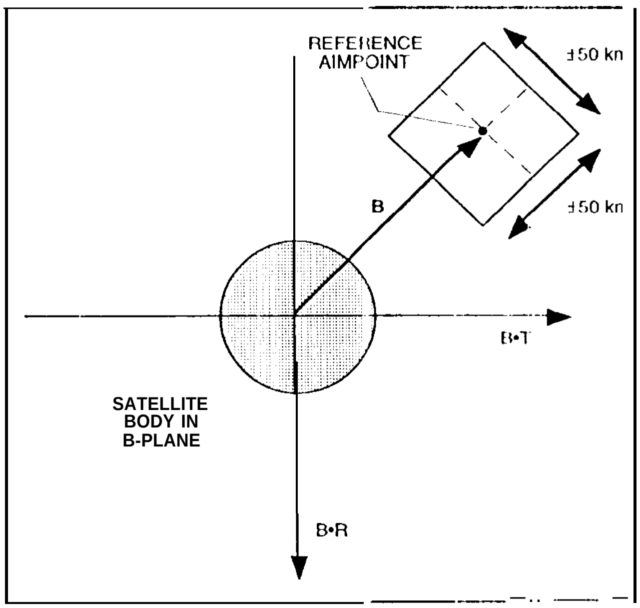

Figure 8 Constraints On 11-Plane Aimpoint Changes

The epoch for selection of each encounter's final aimpoint is restricted by the sequence design cycle. At specified epochs during the tour, the Navigation Team must specify to the Sequence and Science Teams the final satellite. flyby target for the upcoming encounter. Subsequent to the aimpoint selection, target adjustments optimizing propellant consumption arc not allowed for the upcoming encounter. The plan is that during orbital operations the aimpoint selection is made approximately 30 days before the upcoming encounter. Thus, in general, the last opportunity to effect an update to an encounter aimpoint is at the E+3 day OTM. Typically this is the latest opportunity for the trajectory reoptimization changes to be incorporated in an update to the subsequent encounter sequence, The aimpoints for non-targeted encounters are not controlled in this manner and may differ significantly from the aimpoints in the reference tour. The science observation sequences are designed to accommodate the expected range of changes to the aimpoints of the nontargeted encounters.

Maneuver Analysis Results

Based on our current understanding of the maneuver execution and orbit determination uncertainties predicted for the mission, as well as trajectory, flight and ground system constraints, estimates of ΔV requirements to contr 01 the trajectory have been determined and arc summarized in Table 6.

The AVS in Table 6 are based on linear Monte Carlo simulations of the Jupiter approach and tour that approximate the precision maneuver design strategy planned for orbital operations. These results show that the mean AV required to navigate the tour is 50 m/s and the 90% probability value is 67 m/s. These ΔV statistics arc used to predict propellant usage during the orbital tour. Given the limited nature of the propellant available on the spacecraft, propellant margin calculations are a primary metric by which various mission trades arc assessed.

PROPELLANT MARGIN STATUS

Ground Rules and Assumptions

Propellant Margin (PM) is defined as the amount of usable propellant remaining in the RPM at the end of the ten-encounter orbital tour at the 90% probability level. In order to ensure completion of the nominal mission, it is a basic mission requirement that PM be positive.

The data reported here represents the PM status following ODM which occurred on July 27, 1995. Specifically, the epoch of the PM calculation is August 1, 1995. Propellant consumption prior to this epoch is assumed to be known, whereas future propellant consumption through end-of-mission is computed such that PM represents a 90% probability estimate.

The calculation of PM is based on the following high-level ground rules and assumptions: (1) the targeted Io flyby altitude is 1000 km, (2) the targeted initial perijove radius is 4.0 R_J, (3) the nominal duration of Probe data acquisition is 75 rein, (4) the JOI maneuver starts **65** min after the end of Probe data acquisition, (5) the propellant allocation for science turns during the orbital tour is 20 kg, (6) the propellant allocation for Project Manager reserves is 13 kg.

Table 6
JUPITER APPROACH AND ORBITAL TOUR AV ESTIMATES

TCM/OTM Number	Event/TCM/OTM Name	Epoch (SCET)	Design AV (m/s)	Mean (m/s)	Sigma (m/s)	AV(90) (m/s)
	Probe Separation	13-Jul-1995				
TCM-25	ODM	27-Jul-1995	62.2	62.2	0.0	62.2
TCM-26	10-100 days	28-Aug-1995	0.0	0.3	0.2	0.7
TCM-27	10-20 days	17-Nov-l 995	0.0	1.2	0.9	2.5
TCM-28	10-10 days	27-Nov-1995	0.0	0.3	0.1	0.5
TCM-28A	10-5 days	02-Dec-1995	0.0	0.4	0.2	0.7
	Io Flyby	07-Dec-1995				
TCM-29	JOI	08-Dec-1995	643.8	643.8	5.2	650,6
OTM-1	JOI+1.5 Days	09-Dec-1995	0.0	3.0	3.8	8.1
OTM-2	OTM-2	02-Jan-1996	0.0	0,0	0.1	0.0
OTM-3	PJR	18-Mar-1 996	375.4	375.6	0.8	376.9
OTM-4	PJR + 57 Days	14-May-1996	0.0	0.7	0.4	1.2
OTM-5	G1 -15 Days	19-Jun-1996	0.0	0.1	0.1	0.2
OTM-6	G1 -3 Days	01 -Jul-1996	0.0	0.3	0.2	0.6
	Ganymede 1	04-Jul-1996				
OTM-7	G1 +3 Days	07-Jul-1996	0.0	5,0	4.6	11.4
OTM-8	G1 + Apo	05-Aug-1996	5.0	5.0	0.5	5.6
OTM-9	G2 -10 Days	27-Aug-1996	0.0	0.2	0.1	0.4
OTM-10	G2 -2 Days	04-Sep-1996	0.0	0.4	0.2	0.7
	Ganymede 2	06-Sep-1996				
OTM-11	G2+3 Days	09-Sep- 1996	0.0	1.1	0.9	2.2
OTM-12	G2 + Apo	08-Ott-1996	0.0	0.6	0.3	1.0
OTM-13	C3 -3 Days	01-Nov-1996	0.0	0.3	0.2	0.6
	Callisto 3	04-Nov-1996				
OTM-14	C3 + 6 Days	10-NOv- 1996	0.0	3.3	2.1	6.1
OTM-15	C3 + Apo	27-Nov-1996	0.0	1.2	1.1	2.7
OTM-16	E4 -3 Days	16- Dee-l 996	0.0	0.4	0.2	0.7
	Europa 4	19-Dec-1996				
OTM-17	E4 + 4 Days	23-Dec-1996	0.0	2.0	1.6	4.2
OTM-18	E4 -t Apo	04-Jan- 1997	0.0	0.5	0.7	0.9
OTM-19	Orbit 5 Apo	06-Feb-1997	0.0	0.6	0.9	1.5
OTM-20	E6 -3 Days	17-Feb-l 997	0.0	o.?	0.1	0.3
	Europa 6	20-Feb-1997				
OTM-21	E6 + 3.5 Days	24-Feb-1997	0.0	1.8	1.7	4.2
OTM-22	E6 + Apo	14-Mar-1997	16.0	16,1	1.0	17.3
0TM-23	G7 -4 Days	01-Apr-1997	0.0	1.5	0.9	2.8
	Ganymede 7	05-Apr-1997				
OTM-24	G7 + 3 Days	08-Apr-1997	0.0	2.?	1.6	4.4
OTM-25	G7 •t Apo	21 -Apr-1997	0.0	0.2	0.4	0.5
OTM-26	G8 -3 Days	04-May- 1997	0.0	0.2	0.1	0.3
	Ganymede 8	07-May-1997				
OTM-27	G8 + 3 Days	10- May-1 997	0.0	0.9	0.7	1.9
0TM-28	G8 + Apo	02-Jun- 1997	0.0	0.3	0.4	0.9
OTM-29	C9 -2 Days	23-Jun-1997	0.0	0.2	0.1	0.3
	Callisto 9	25-Jun- 1997				
OTM-30	C9 + 3 Days	28-Jun-1997	0.0	2.4	1.5	4.4
OTM-31	C9 + Apo	08-Aug-1997	0.1	0.3	0.2	0.6
OTM-32	C10 -3 Days	14-Sep-1997	0.0	0.2	0.1	0.4
	Callisto 10	17-Sep-l 997				
0TM-33	C1O + 3 Days	20-Sep-1997	0.0	0.8	0.5	1.5
0TM-34	C10 + Apo	18-Ott- 1997	0.1	1.8	1.2	3.5
OTM-35	E11 -3 Days	03-Nov-1997	0.0	0.3	0.2	0.5
	Europa 11	06-Nov-1997				

Table 7
PROPELLANT MARGIN CALCULATION
Post-ODM **Epoch: 8/1/95**

Event	I_{sp} (s)	AV (m/s)	M, (kg)	M_f (kg)	AM (kg)
Completed Events:					
Drop Adapter			2717.2	2561.2	156.0
Drop Instrument Covers			2561.2	2560.4	0.8
AV Propellant (1 O N)		131.0	2560.4	2433.7	126.7
Attitude & Spin Control			2433.7	2406.5	27.1
HGA Anomaly Activities			2406.5	2355.3	51.3
RPM Line Flushing			2355.3	2349.7	5.6
Probe Release			2349.7	2010.8	338.9
ODM	**308.5**	**62.2**	2010.8	1969.8	40.9
Future Events:					
I/P Statistical AV	**270.7**	**4.5**	1969.8	1966.5	3.3
I/P Deterministic AV	**270.7**	**0.0**	1966.5	1966.5	0.0
I/P Attitude & Spin Control			1966.5	1966.2	0.3
I/P RPM Line Flushing			1966.2	1966.0	0.2
JOI	**308.5**	**643.8**	1966.0	1589.2	376.9
OTM-1 + OTM-2	**270.7**	**3.0**	1589.2’	1587.4	1.8
PJR	**308.5**	**375.4**	1587.4	1402.1	185.2
Tour AV	**270.7**	**67.3**	1402.1	1367.0	35.1
Tour Attitude & Spin Control			**1367.0**	1352.1	14.9
Tour RPM Line Flushing			**1352.1**	1350.3	1.8
Science Turns			**1350.3**	1330.3	20.0
Project Manager Reserves			**1330.3**	1317.3	13.0

End-of-Mission Mass:	1317.3
Orbiter “Burnout” Mass:	-1296.5
Propellant Margin:	20.9

Notes: I_{sp} range for completed AVS: 269.4 -274.4 s.
Values may not add because of rounding.

M, = spacecraft mass before event.
M_f = spacecraft mass after event.
AM = spacecraft mass change.

* At the time (his paper was written, the actual AV achieved at ODM had been determined, but the reconstruction of400N engine performance during the burn had not been completed. When the thrust and mass flow rate during ODM arc determined, the 400 N I_{sp} will be updated, and I’M will change accordingly.

during the orbital tour. Therefore, the total amount of propellant remaining at the end of the baseline mission, assuming Project Manager Reserves are unused, is estimated to be 34 kg (at the 90% probability level).

CONCLUSIONS

The Galileo Probe and Orbiter spacecraft have been targeted to their respective desired aimpoints at Jupiter. Atmospheric entry for the Probe occurs on December 7, 1995 at 22:04 UTC. The Orbiter closest approach to Jupiter occurs about 11 minutes earlier. After recording data transmitted from the Probe during its descent, the Orbiter executes the JOI maneuver to establish the initial orbit about Jupiter. The Orbiter then will carry out a two-year-long intensive investigation of Jupiter, its satellites, and its magnetosphere. During the orbital tour, there will be ten close flybys of the Galilean satellites Europa, Ganymede and Callisto. The gravity assist from each of these satellite encounters is used to change the Jupiter orbit parameters to achieve the next satellite encounter.

During the final Jupiter approach phase, four TCMS provide accurate delivery of the Orbiter to the desired Jo aimpoint. Orbit determination during this phase is accomplished primarily with S-Band Doppler data and three OPNAV pictures. The final delivery accuracy at 10 is expected to be about 50 km (altitude) and <2 seconds (time of closest approach).

During the orbital tour, three OTMS per orbit are planned to provide accurate delivery of the Orbiter to each satellite encounter. S-Band Doppler data and a limited number of OPNAV pictures (<20 per orbit) are used for orbit determination. The final satellite encounter delivery accuracy is generally a few 10s of kilometers in the B-plane and a few seconds in time of closest approach.

During the orbital tour, starting with the JOI clean up maneuver, the orbital tour trajectory is reoptimized (at certain OTMs) to minimize total AV by varying the satellite aimpoints within allowable bounds. The new aimpoint for the upcoming satellite encounter is specified at the time of the post-encounter OTM for the previous encounter in order to allow sufficient time to update science observations in the spacecraft sequence. This trajectory reoptimization strategy has contributed to lowering estimates for the total AV required to fly the orbital tour. Based on current ground rules, at the 90% probability level, the orbital tour AV is about 70 m/s; and the end-of-mission propellant margin is 20 kg.

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APPENDIX

Hyperbolic approach trajectories are typically described in aiming plane coordinates, often referred to as "B-plane" coordinates (see Figure A- 1). The coordinate system is defined by three orthogonal unit vectors, S, T, and R with the system origin taken to be the center of the target body. The S vector is parallel to the spacecraft hyperbolic approach velocity vector relative to the target body, while **T** is orthogonal to S and lies in the ecliptic plane (the mean plane of the Earth's orbit). Finally, R completes an orthogonal triad with S and **T**.

The aimpoint for an encounter is defined by the miss vector, **B**, which lies in the **T-R** plane, and specifics where the point of closest approach would be if the target body had no mass and did not deflect the flight path. The time from encounter (point of closest approach) is defined by the linearized time-of-fli~,ht (LTOF), which specif ies what the t i me of flight to encounter would be if the magnitude of the miss vector were zero (i. e., if the target were-the origin of the B-plane).

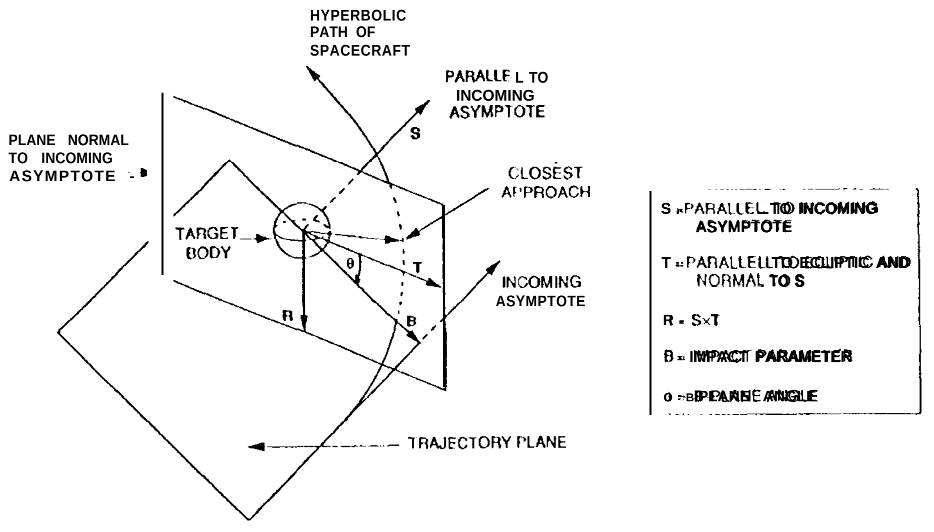

Figure A-1 11-Plane Coordinate System Definition