Abstract

for

TOPEX/POSEIDON Orbit Acquisition Maneuver Sequence*

by

R. S. Bhat**, B. E. Shapiro#, and R. B. Frauenholtz+

Jet Propulsion Laboratory,

California institute of Technology,

Pasadena, California 91109

SUMMARY

TOPEX/POSEIDON, a joint oceanographic mission of NASA and Centre National d'Etudes Spatiales (CNES), France, was launched by an Ariane 42P on August 10, 1992 into a 1322 km near circular orbit at an inclination of 66.080 deg. The Project wanted to achieve the operational orbit as soon as possible in order to begin oceanographic data acquisition through altimeters.

A sequence of six maneuvers was implemented to acquire the operational orbit following injection. These maneuvers raised and circularized the orbit, removed inclination errors imparted by the launch vehicle, acquired frozen conditions, and synchronized the ground track with the reference. grid overflying two verification sites. Initially, the maneuver sequence was generated using a pre-launch execution error model. However, the sequence was updated after each maneuver to reflect observed maneuver performance and to use updated error models. Accurate maneuver performance evaluation was done using a newly developed technique based on mean elements. Maneuver magnitude determination accuracy was better than 0.2 mm/s and precise calibration of thrusters was possible after each maneuver. The sequence design was adaptable to unexpected schedule changes and accommodated additional satellite constraints. A backup maneuver was designed for each maneuver to handle the operational delays. The back up maneuver design was, in fact, used for the first maneuver. The six-maneuver sequence was successfully completed on September 21, 1992 (42 days after launch) after placing the satellite into the operational orbit. The sequence also provided a smooth transition from the Orbit Acquisition to the Orbit Maintenance Phase.

INTRODUCTION

The TOPEX/POSEIDON satellite was launched by an Ariane 42P on August 10, 1992 and injection occurred at 23:27:05 UTC, approximately 19 min. 57 sec after the lift off. The achieved injection orbit was 1322 km near circular at an inclination of 66.08 deg. The launch vehicle placed the satellite in a biased orbit to have frequent opportunities for phasing into the reference ground track pattern, and to avoid the possibility of a collision of the third stage of the launch vehicle with the satellite.

* The work described in this paper was carried out by the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration.

** Member Technical Staff, Navigation Systems Section.

Member Technical Staff, Navigation Systems Section, Member AAS, Member AiAA

+ Technical Manager, Navigation Systems Section, Member AIAA

The desired operational orbit is a 1336 km near circular frozen orbit with nearly constant eccentricity vector ($e \approx 0.000095$, $\omega \approx 90^o$) and an inclination of 66.040 deg[1]. 'I'he operational orbit was designed to provide an exact repeat ground track every 127 revolutions in 10 sidereal days. During every ground track repeat cycle the satellite overflies single NASA and one CNES verification sites[2]. The Project wanted to achive the operational orbit in a minimum practical time to begin altimeter data acquisition.

The satellite was built by Fairchild Space (FS) company under contract to the Jet Propulsion Laboratory (JPL). JPL is also responsible conducting satellite mission operations, including operational navigation. 'l'he Flight Dynamics Facility (FDF) of the NASA Goddard Space Flight Center (GSFC) is providing operational orbit determination support using radiometric data acquired via the NASA Tracking and Data Relay Satellite System ('1'111<SS).

TOPEX/POSEIDON is a three-axis stabilized satellite with a large solar array extended along the pitch axis to one side. The solar array is kept pointed towards the sun all the time through solar array pitching and rotating the satellite about the yaw axis(yaw-steering). "l'he satellite is in yaw steering mode for most of the time. Therefore, it is necessary to stop the yaw-steering and turn the satellite to a desired orientation before a maneuver. The satellite is put back to yaw steering mode after the maneuver. in other words, the maneuver becomes a turn-burn-turn process.

The Project conducted several test and training exercises prior to the launch, involved all operational teams, to ensure smooth operation of the complex maneuver process. Initially, a maneuvr activity timeline(5-5-4-4 -4- -- daily spacings from launch)[2]. was jointly evolved by all operational teams prior to the test and training exercises. An updated timeline provided more spacing between maneuvers to allow adequate time for team activities. "l'his timeline was adopted by the Project as the baseline to acquire the operational orbit.

Based on expected launch vehicle and the satellite performance, a pre-launch orbit acquisition sequence was generated using worst-case for injection and maneuver execution errors. This sequence required seven maneuvers (7-7 -6-6-6-6-6 daily spacings from launch) and took 44 days to achieve the operational orbit.

After the launch, the maneuver sequence was rc-generated using achieved injection orbit and the pre-launch execution error model. The maneuver sequence and the execution error model were updated after each maneuver. Accurate maneuver performance evaluation was important for acquiring the operational orbit satisfying all requirements and constraints. A new maneuver evaluation technique was developed anti used during the Orbit Acquisition Phase. This technique used pre and post-maneuver mean elements for accurate determination of maneuver magnitudes. FDF also provided information on maneuver evaluations using position-constraint orbit solutions. The two results using independent techniques agreed to each other for all maneuvers and increased the confidence on the maneuver performance evaluations. The desired operational orbit was very accurately achieved in 42 days two days earlier than the pre-launch plan (44 days) and provided a smooth transition to the Orbit Maintenance Phase.

This paper describes the maneuver sequence design, implementation, and evaluation process carried out to acquire the TOPEX/POSEIDON operational orbit. The adaptive nature of the maneuver design to accommodate schedule changes and additional satellite constraints is highlighted. A brief description of the maneuver evaluation process is provided and the performance of each maneuver of the sequence is illustrated.

ORBIT REQUIREMENTS

The GSFC/FDF provided a series of orbit determination solutions for the injection orbit using two-way data acquired via the TDRS and the DSN. The epoch for these solutions was August 10, 1992 at 23:27:50 UTC which is approximately the satellite separation time. The injection orbit was well established by launch (l.) + 18 hours. The L+36 hour solution was used to generate the initial maneuver sequence based on the achieved injection orbit.

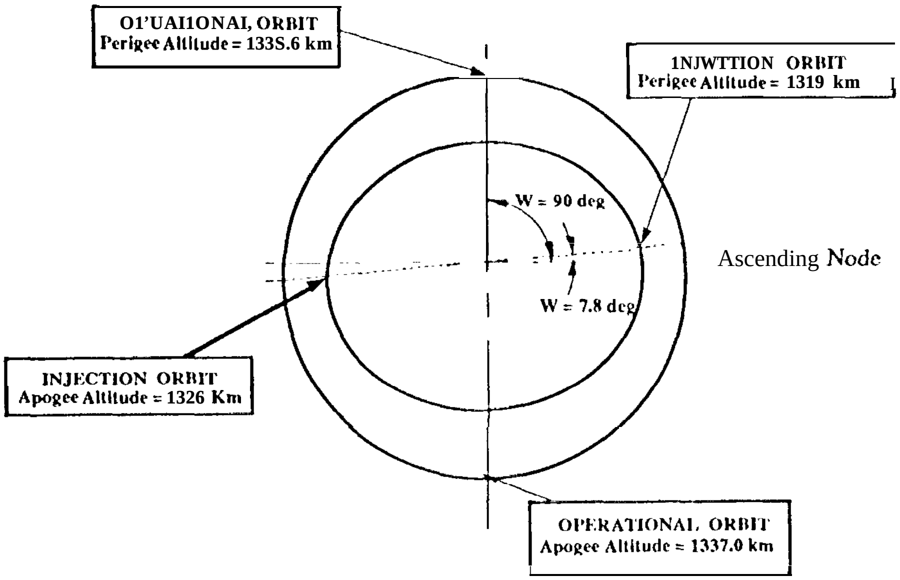
The operational orbit was designed using a semi-analytical trajectory program in the presence of 17×17 earth gravity field to meet the science requirements [1]. "l"his reference orbit was refined later with precision numerical integration software using 20× 20 gravity field 3. The reference orbit is a near circular frozen orbit at an altitude of 1336 km and an inclination of 66.04 deg. 'l'he orbit provides an exact repeat ground track every 10 sidereal days and 127 orbits, during which it overflies one NASA and one CNES ground verification site. "l'he equatorial distance between two consecutive reference orbit ascending node is 3156 km and there are 10 reference tracks between any two consecutive equatorial ascending nodes; one track every 315.6 km. The mean orbital parameters of the pre-launch nominal injection orbit, the post-launch achieved injection orbit, and the operational orbit are summarized in Table 1.

'1'able 1 Mean orbital Parameters of Injection and Operational **Orbits**
Injection Orbit Epoch - August 10, 1992 23:27:50.0000 UTC
(= 04:27:50.0000 *PM PDT*)

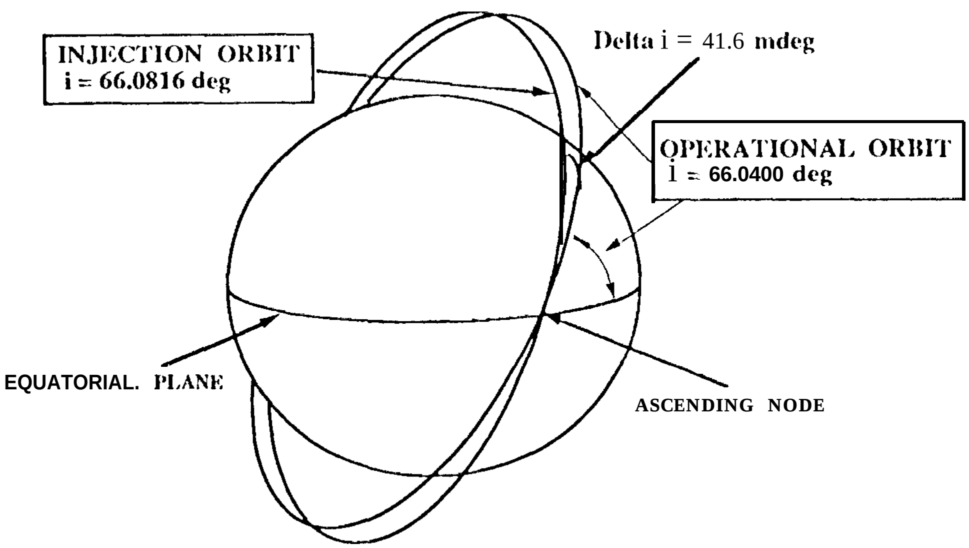
Parameter	**Pre-Launch Injection Orbit (expected)**	**Post-Launch Injection Orbit (actual)**	**Operational Orbit**	**Required Change**
a (km)	7703.056	7700.547	7714.429	13.882 (6.4 rids)
CX10^{-6}	772	485	95	390
ω (deg)	6.4	7.8	90.0	82.2
i* (deg)	66.0400	66.0816	66.0400	0.0416 (5,2 m/s)
τ_n (see)	6730.83	6727.60	6745.75	18.15

•Gravity Mean Value

The achieved semi-major axis of the injection orbit was about 2.5 km less than the pre-launch nominal value. The mean values of the apogee and the perigee altitudes were 1326 and 1319 km respectively, thus the achieved injection orbit was inside the operational orbit (Fig. 1). The nodal period was ≈ 18.5 secs less than that of the operational orbit, while the injection orbit inclination was 41.6 mdeg higher, requiring 5.2 m/s for correction (Fig. 2). The injection orbit ground track was drifting eastward relative to the reference track at the rate of 108.08 km/day and provided a synchronizing opportunity once every 2.9 days.



JUG. 1 1N.JECTION ORBIT RELATED TO OPERATIONAL ORBIT
(INPLANE ORBIT GEOMETRY)



JUG. 2 INJJW'J'1ON ORBIT RELATED TO OPERATIONAL ORBIT
(INCLINATION GEOMETRY)

The main requirement on the orbit acquisition maneuver sequence was to acquire the operational orbit including the referenence ground track pattern 2 and to provide smooth transition to the Orbit Maintenance Phase at the end of Orbit Acquisition Phase. The start of every ground track repetition cycle was chosen to have the first ascending node at 99. 92 deg east longitude[1]. Cycle O starts with the last maneuver of the sequence regardless of its geographic location and ends with the beginning of the Cycle 1. The residual ground track drift rate after the last maneuver of the acquisition sequence must be such that the ground track arrives at the east boundary of $\pm 1\,km$ when the ascending node is at the transition of ground track repetition cycles, when the first Orbit Maintenance Maneuver is planned [4].

01 WRATIONAL CONSTRAINTS

Initially, the operational teams together evolved a set of constraints [2] through mutual discussions and analyses and these constraints were modified during the pre-launch test and training exercises conducted by the Project. The modified constraints and guidelines are:

1. The first maneuver was used to calibrate the 22-nt thrusters. "l'his maneuver was an inplane maneuver and contributed to the re-targeting process. The maneuver mgnitude was constrained between 2 and 5 m/s.

2. The nominal maneuver activity timeline was 7-7-6 -6-6 -6---- This activity timeline was established during the pre-launch test and training *exercises*. The first maneuver (calibration) was scheduled 7 days after the launch, and the second maneuver was scheduled 7 days later to allow sufficient time for calibration maneuver performance analysis. Subsequent maneuvers were implemented every 6 days. A 7-day spacing may be used with advance notice (at least four days).

3. Maneuver centroid time was standardized between 10 AM and 2 PM Pacific Daylight Time (PDT). This time may vary within this window to locate the maneuver within the orbit as needed to acquire frozen values for e and ω.

4. Maneuver time was constrained to be within TDRS view periods. Large maneuvers using 22-nt thrusters were also to be consistent with Omni-Antenna obscuration periods (by HGA and GPS anttenna) and Partial coverage zones.

5. A single back-up maneuver was needed for each maneuver in response to non-satellite operational delays. This backup was to be scheduled between three and six days after the nominal maneuver time, chosen to minimize total time and/or number of maneuvers to reach the operational orbit.

6. The last maneuver in the sequence must provide a smooth transition from the Orbit Acquisition Phase to the Orbit Maintenance Phase.

7. Maneuver sequence must be updated after each maneuver to reflect the observed maneuver performance and any updates to the execution error model.

SATELLITE CHARACTER ISTICS

TOPEX/POSEIDON is a three-axis stabilized satellite utilizing sinusoidal yaw steering mode to point its solar panel towards sun [1]. The yaw steering is stopped and it is slewed to the desired orientation before a maneuver and after the maneuver the satellite is slewed back to start yaw steering again. In other words, the maneuver is part of a "turn-burn-turn" sequence. The yaw turn is accomplished using only reaction wheels, requiring one to two hours to complete the turn-burn-turn process. The burn attitude is controlled by attitude jets.

The satellite propellant tank was fully loaded few hours before the launch to provide AV equivalent to ≈ 172 m/s. The satellite is capable of implementing maneuver magnitudes between 0.013 mm/s and 15 m/s [5,6] The propulsion module is a mono-propellant blow-down syastem consisting of four 22 nt (5 lbf) and twelve 1 nt (0.2 lbf) thrusters [2]. The 22-nt and four of the 1-nt thrusters are used for orbit adjust maneuvers and rest of the 1-nt thrusters are used for attitude control requirements mainly during burn. The acquisition sequence had six maneuvers of which four were large (>500 mm/s) and other two were small (≤ 500 *mm* /s) maneuvers, Large maneuvers were implemented using four 22-nt and small with four 1-nt thrusters.

The Center of Mass (CM) of the satellite does not coincide with the center of body co-ordinates due (o the location of the solar panel (Fig. 3). The orbit adjust thrusters were carefully canted prior to the launch to align the thrust vector through the predicted CM based on pre-launch mass properties. The maneuver efficiency was expected to be less than 100 percent because of thruster duty cycling [2]. The worst-case maneuver efficiency [7] using an Open Loop Firing Pattern (OLFP) was expected to be around 60% based on dynamic simulations. However, the Project decided not to use OLFP during the operation to avoid possible satellite excitation during a large maneuver. However, the observed maneuver efficiency was >60% for almost all large maneuvers and >85% for the small maneuvers.

The solar array pitching waa stopped and parked at a 90° *or* 270° (depending on convenience) pitch position before a burn using the 22-nt thrusters. This was accomplished while yaw slewing to the burn orientation. For small maneuvers (1-nt thrusters) the solar array pitching was stopped just before the burn of a maneuver and pitching continued after the burn.

Orbit adjust maneuvers were accomplished through the following steps.

1) Stop Normal Mission Mode (yaw steering) and slew the satellite to a desired maneuver direction which accounts for thruster cant angles.
2) For maneuvers using 22-nt thrusters, High Gain Antenna (HGA) is brought to the parking position and Omni Antenna is used for all telecommunications and commanding
3) Stop the solar array pitching and rotate it to the desired parking position (90 or 270 deg). For small maneuvers using 1-nt thrusters, the solar array pitching is stopped before entering into Orbit Adjust Mode (OAM).
4) Enter OAM to execute the burn
5) Reduce the satellite attitude rates due to thrusting

6) Unwind the satellite to yaw-steering position and rotate back the solar army to pitching postion.
7) Start yaw steering mode again

Omni Antenna Constraint

The satellite communication and commanding are being carried out via one of two TDRSS, either TDRS-EAST or TDRS-WEST. The HGA used for communication with TDRSS most of the time. The onboard computer steers the HGA towards TDRS-EAST or WEST depending on the viewing geometry. However, it was planned to use the Omni Antenna in place of the HGA during a large maneuver due to expected large satellite attitude disturbances, which might have interrupted the communication [1]. "The HGA antenna was brought to the parking position before every large maneuver (Fig. 3).

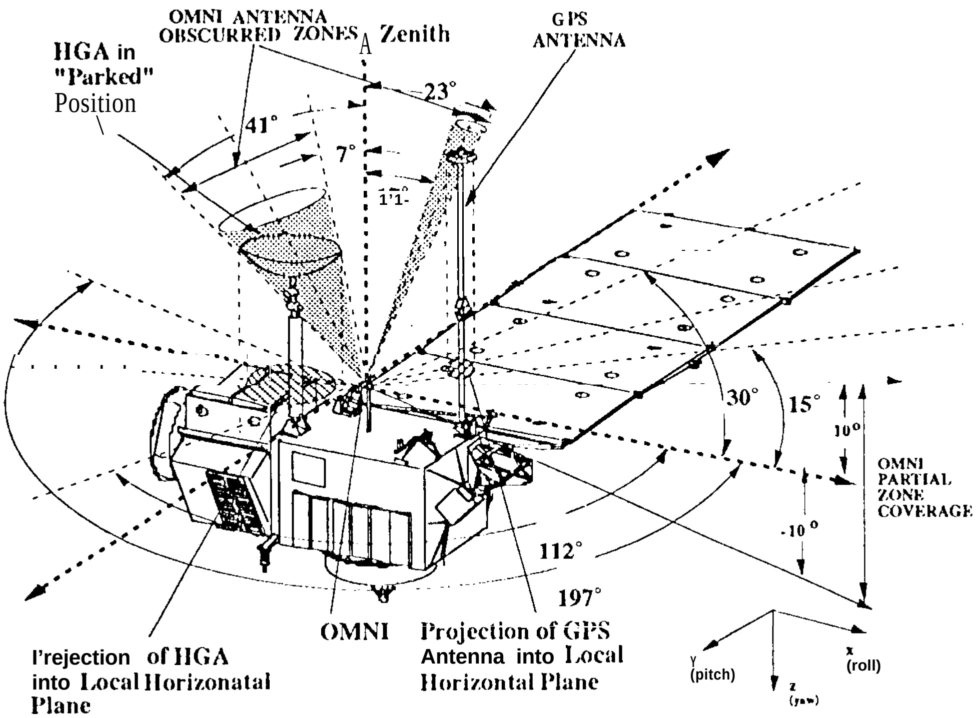


Fig. 3 ILLUSTRATION OF OMNI ANTENNA CONSTRAINTS WITH THE SATELLITE

The line-of-sight between the Omni-Antenna and TDRS-EAST or TDRS-WEST is obscred at times either by the HGA or by the GPS Antenna (Fig. 3). During this obscuration period, the communication link may be lost between the satellite and TDRSS. The obscuration period can vary between 5 and 10 min. based on the satellite - TDRSS geometry. The Project did not want to lose the communication while implementing a maneuver, maneuvers using the 22-nt thrusters were not allowed during Omni-Antenna obscuration periods.

The communication link was lost a few times while implementing the calibration (first) maneuver because the maneuver was implemented during the Omni-Partial coverage zones (Fig. 3). The Partial coverage zone is between ± 10 deg elevation with respect to the roll-pitch plane of the satellite coordinate system and forms a circular disc around the satellite. This was an additional constraint for subsequent maneuvers with 22-nt thrusters.

MANEUVER PERFORMANCE EVALUATION

The maneuver accuracy requirements were finalized jointly by Navigation at the JPL and the GSFC/FDF who provides operational orbit determination supports. The accuracy requirement was a function of maneuver magnitude and better accuracy was needed for maneuvers with magnitudes $\leq 100\, mm/s$. Both FDF and JPL studied, analysed, and jointly discussed different techniques to realize the required maneuver evaluation accuracies during operation. The FDF developed and used pre and post-maneuver position-constraint orbit determination solutions for maneuver evaluation.

The Navigation at JPL developed a new technique which determined the maneuver evaluation using pre and post-maneuver mean elements. "The pre and post-maneuver mean elements are obtained at the same time using pre and post-maneuver trajectory files [9]. Prior to the launch, the performance of the technique was verified for a null maneuver (maneuver magnitude is zero). This technique was successfully used during the Orbit Acquisition Phase and better than 0.2 mm/s accuracy was achieved for all maneuvers. The position-constrained solutions provided by FDF also provided similar accuracies. The two results based on independent techniques agreed very well with each other, thus increased the confidence on maneuver performance evaluation.

MANEUVER SEQUENCE AND ACHIEVED RESULTS

The complete maneuver sequence was established before the launch based on worst-case launch vehicle and satellite performance. The sequence had seven maneuvers including calibration, inclination, and inclination -trim maneuvers (Fig. 4). The inclination-trim maneuver was scheduled to remove residual inclination errors resulting from earlier maneuvers. All maneuvers $\geq 1.0\, m/s$ were implemented so that the expected execution errors could be absorbed by subsequent maneuvers, without adding maneuvers or delaying completion of the maneuver sequence. The complete sequence was designed to ensure ground track phasing and achieving the frozen conditions. A "shoot-short" strategy? was applied to avoid penalty (in terms of extra days required to achieve the operational orbit) under the expected worst-case execution errors. The pre -launch sequence achieved the operational orbit with the required ground track pattern in 44 days.

After launch, the maneuver sequence was re-designed using the achieved injection orbit and the pre-launch execution error model. The achieved inclination was 41 mdeg higher than the operational orbit inclination requiring 5.2 m/s for correction. This maneuver magnitude is almost half the pre-launch sequence inclination maneuver magnitude (Fig. 4). Pointing errors were neglected during maneuver design, but were accounted by the satellite team while designing the satellite yaw slew the maneuver burn orientation. The inclination maneuver was small enough to cancel the planned inclination-trim maneuver. The updated sequence required six maneuvers to achieve the operational orbit in 38 days.

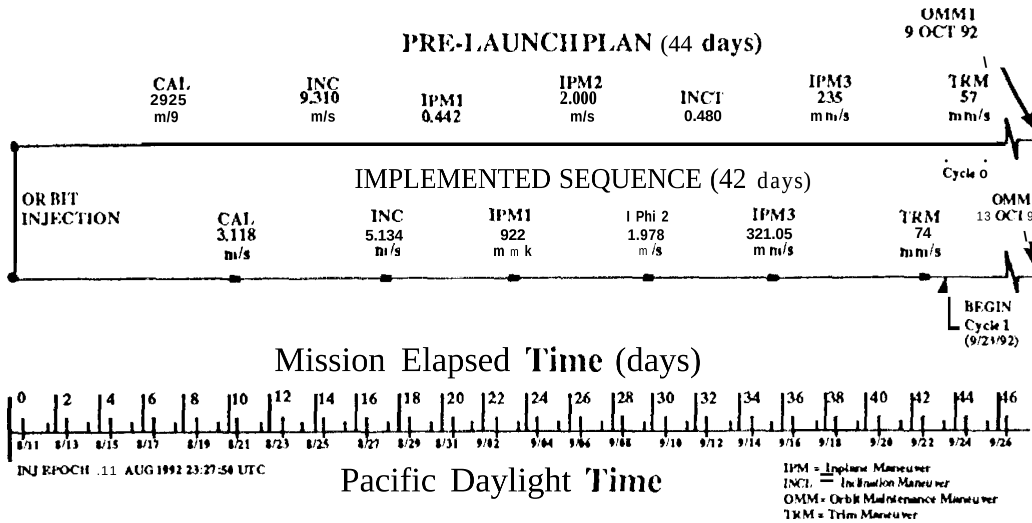


FIG. 4 MANEUVER SEQUENCE DESIGN & EXECUTION

The nominal and the back-up sequences were designed simultaneously for every maneuver. The back-up sequence includes the back-up for the current maneuver and subsequent maneuvers are implemented consistent with the activity timeline (7-7-6-6-6-6--).. The calibration maneuver was not implemented at nominally scheduled time (L+7 days) due to a problem with time correlation between the satellite and the ground. The maneuver was implemented at L+10 days using cal-backup maneuver sequence. Also, the expected t-e-targeting period was extended by three days (41 days). The large maneuvers using 22-nt thrusters were completed by IPM2 (Fig. 4) requiring 2.9 days (10-7-6-6). The satellite performance was better than expected (error model used for design) for these maneuvers.

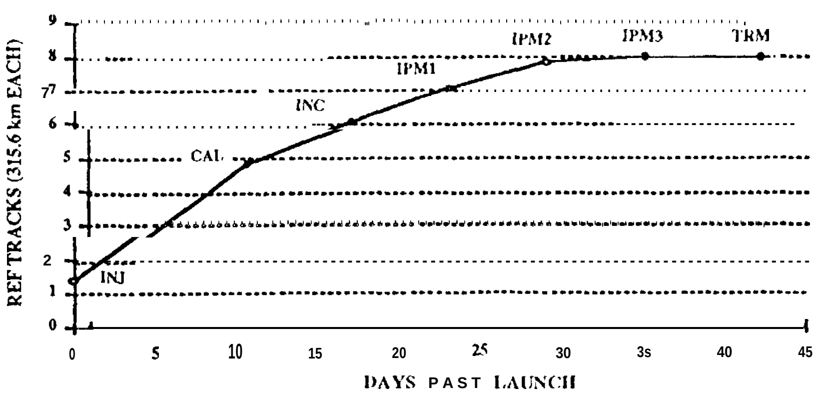


Fig. 5 SYNCHING UP WITH THE REFERENCE GROUND TRACK

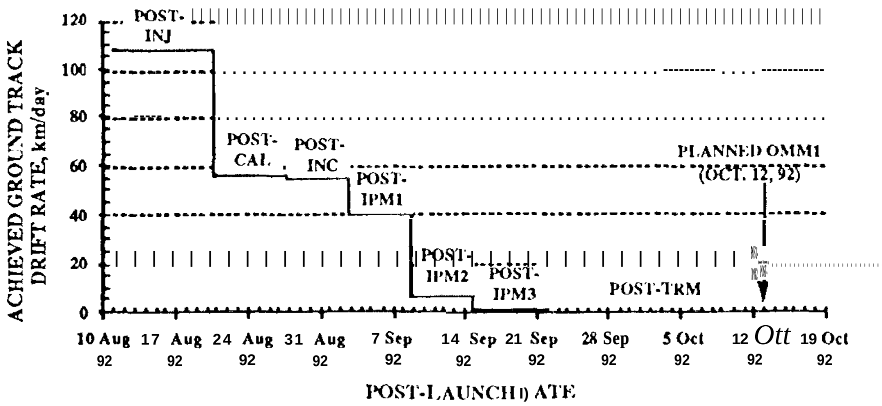


Fig. 6 GROUND TRACK DRIFT RATE HISTORY WHILE ACHIEVING REFERENCE ORBIT

The IPM3 was the first maneuver to be implemented using 1-nt thrusters. The pre-launch execution error model was used to design IPM3 and TRIM maneuvers (Fig. 4). The post-IPM3 results demonstrated that the satellite performance was excellent and the error model used for the design was conservative. The ground track drift was 1.4 km/day and it was clear that TRIM maneuver must be delayed at least by a day so that ground track could enter the control band within few days (5-7 days) after "TRIM. Therefore the TRIM maneuver was implemented 7 days after IPM3. The acquisition sequence was completed by implementing TRIM on September 21, 1992 requiring 44 days after launch to achieve the operational orbit(Fig. 4). The phasing of ground track with the reference grid was also accomplished by that time (Fig. 5).

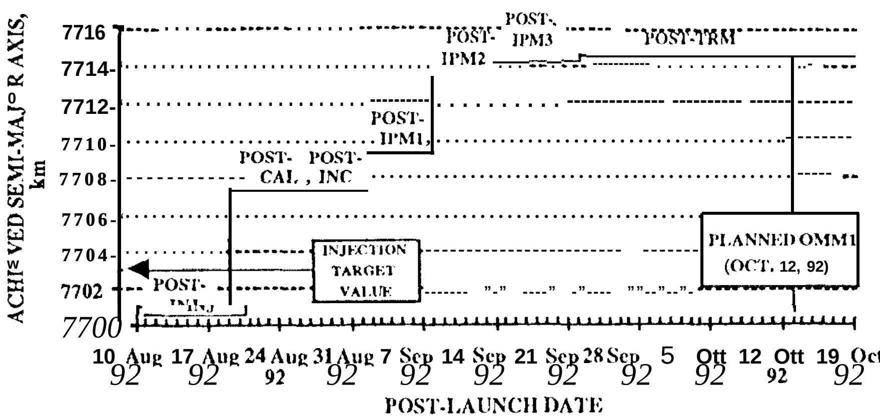


Fig. 7 SEMI-MAJOR AXIS HISTORY WHILE ACHIEVING REFERENCE ORBIT

The large in-plane maneuvers using 22-nt thrusters reduced the drift rate from 108.08 km/day to 6.7 km/day (Fig. 6). The last two maneuvers using 1-nt thruster reduced the drift rate to a near zero value (136 m/day). The inclination maneuver did not contribute to the orbit raising or ground track phasing process. All inplane maneuvers were used to raise the semi-major axis (Fig. 7) and to achive the frozen values for e and ω (Fig. **8).** The orbit was near the operational orbit by IPM2 but IPM3 and TRIM maneuvers were used to refine the semi-major axis and the eccentricity vector and to position the ground track in the control band. The achieved orbit after TRIM maneuver was withen tolerance window box of the operational orbit. (Table 2).

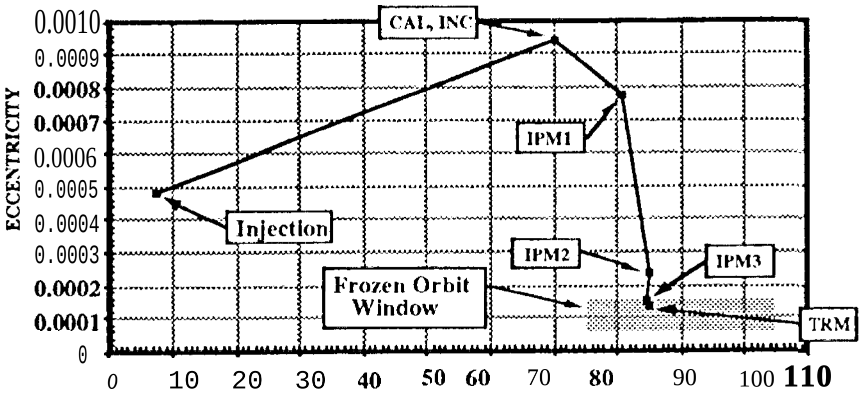


Fig. 8 ACHIEVING THE FROZEN ORBIT

Table 2 Achieved Orbit from Orbit Acquisition Maneuver Sequence

Mean Orbital Elements	Reference Value	Tolerance (plus/minus)	Achieved Value	Difference (Ach. - Ref.)
a (*km*)	7714.429		7714.412	-0.017
$e\times10^{-6}$	95	50	137	42
i* (deg)	90.0	15.0	92.3	2.3
ω (deg)	66.040	0.003	66.041	0.001

•Gravity Mean Value

The post-TRIM ground track was about 1.65 km west of the reference track and drifting slowly eastward towards the control band (Fig. 9). The post-TRIM ground track drift rate was adjusted so that it drifted slowly get into the control band and reach its east boundary at the transition of ground track repetition Cycles **2** and **3** on October 12, 1992. The ground track entered the control band on September 26, 1992 and the first Orbit Maintenance Maneuver (OMM 1)[4] was implemented on October 12, 1992 giving the proof of smooth transition from Orbit Acquisition Phase to Orbit Maintenance Phase.

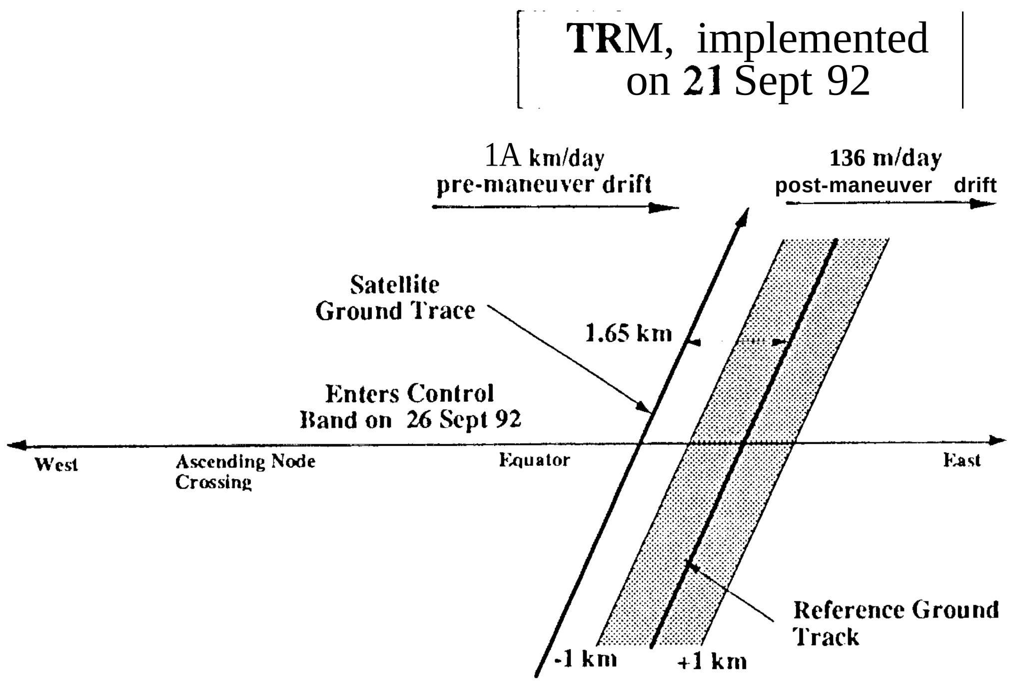


Fig. 9 POST-TRIM GROUND TRACK

REFERENCES

1, TOPEX/POSEIDON Project, "*Mission Plan*," Doc. No. 633-321 Rev. C, JPL D-6862 (internal document), August 1991.

2. Bhat, R. S., "*TOPEX/POSEIDON Orbit Acquisition Maneuver Design*," AAS 91-514, 1991 AAS/AIAA Astrodynamics Specialist Conference, Durango, Colorado August 19-22, 1991.

3. Shapiro, B. E., "*TOPEX/POSEIDON Repeat Orbit and Reference Grid Definition*" JPL IOM No. 314.5-1642, June 22, 1992 (internal document).

4. Shapiro, B.E., Bhat, R. S., and Frauenholz, R. B., "*TOPEX/POSEIDON Ground Track Maintenance Experience*," 1993 AAS/AIAA Astrodynamic Specialist Conference, August 16-19, 1993, Victoria B. C., Canada. (to be published).

5. TOPEX/POSEIDON Project, "Mission and *Systems Requirements*," Doc. No. 633-103, JPL 1>-5901 (internal document), April 1989.

6. Bhat, R. S., "*Execution Error Model for Orbit Maintenance Maneuver Design*," JPL IOM FOS 92-034, October 2, 1992 (internal document).

7. Sanneman, P., "*TOPEX/POSEIDON Orbit Adjust Mode Impact of C M Estimation Uncertainty and ACS Thruster Failures on* **AV** *Maneuver Performance*," GNC:TOPEX:90 **-079,** Fairchild Space, June 1990.

8. FDF/GSFC "*TOPEX/POSEIDON Interface Control Document*" 554-FDD-91/1 16, FDF, GSFC June 1992 (internal document).

9. Guinn, J. R. "*Short-Periodic Gravitational Perturbations for Conversions between Osculating and Mean Element s,*" AAS 91-430, 1991 AAS/AIAA Astrodynamics Conference, Durango, Colorado, August 19-22, 1991.