TOPEX/POSEIDON ORBIT ACQUISITION MANEUVER DESIGN*

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TOPEX/POSEIDON is a mission jointly sponsored by NASA and Centre National d'Etudes Spatiales (CNES) of France for oceanographic studies, scheduled for launch in mid-summer 1992 by Ariane. The current baseline injection orbit is near-circular, $\sim$ 30 km below the desired operational orbit altitude and at the operational orbit inclination. A baseline maneuver sequence to retarget from this injection orbit to the desired operational orbit has been designed based upon the expected worst-case 3σ injection and maneuver execution errors. This sequence requires seven maneuvers, including an initial calibration burn and achieves the operational orbit with the desired ground track pattern in 30 days. A "shoot-short" strategy was applied to the maneuver design to limit penalties due to maneuver execution errors. A delay sensitivity analysis has been conducted to estimate the allowable operational delay for each maneuver without increasing the total orbit acquisition period. The baseline sequence provides back-ups for a one-revolution delay for each maneuver and one-day delay for most maneuvers. The back-up philosophy has been standardized to simplify operations. It is also demonstrated that a higher injection orbit allows the maneuver sequence to achieve the operational orbit in 26 days under a worst-case scenario. This injection orbit has been recently adopted by Project.

INTRODUCTION

The TOPEX/POSEIDON operational orbit is a near-circular frozen orbit (eccentricity e and argument of perigee ω remain nearly constant) at 1336 km altitude and an inclination of $\sim$66 deg. This orbit provides an exact repeat ground track every 127 revolutions in 10 sideareal days and overflies one NASA and one CNES ground verification site.[1,2] The NASA verification site is the Harvest oil platform located offshore from Pt. Conception, California; the CNES site lies between Lampedusa and Lampione islands near Sicily in the Mediterranean Sea.

The TOPEX/POSEIDON satellite will be launched in mid-summer 1992 by an Ariane 42P launch vehicle from Kourou, French Guiana. The current baseline injection orbit is near-circular, $\sim$ 30 km below the operational orbit altitude at the required inclination. The primary purpose of biasing the injection orbit is 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 Ariane with the satellite.

The immediate post-launch maneuver design task is to achieve the operational orbit with a specified reference ground track pattern. The mission requirements are that this task be completed in minimum time, consistent with all operational constraints.[3] The main operational constraint is to schedule all maneuvers around the same time every four days to be consistent with the activity timeline of operational teams. Another major constraint is to provide back-ups for one-revolution and one-day operational delays without increasing the total orbit acquisition period.

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The orbital mechanics to reach the operational orbit altitude are straightforward. However, maneuvers have to be implemented in such a manner that the resulting ground track pattern coincides with the desired reference pattern and overflies the verification sites. In addition, the maneuver sequence must achieve the operational orbit in the presence of injection and execution errors while satisfying all operational constraints.

An analytic propagation model developed for the maintenance maneuver design[4] has been modified for the orbit acquisition maneuver sequence design strategies. The model assumes that maneuvers are impulsive with deterministic error sources.

An operationally feasible baseline maneuver sequence has been designed based on expected worst-case 3σ injection and execution errors. This paper describes the sequence and demonstrates consistency with all operational constraints. Sequence modifications with back-ups to handle one-revolution and one-day operational delays are illustrated. A study of a possible higher injection orbit is also summarized and its advantages are highlighted.

ORBIT REQUIREMENTS

The mean orbital parameters of the operational orbit and the current injection orbit are summarized in Table 1. The injection orbit is below the operational orbit with an inclination identical to the reference orbit. The equatorial distance between two consecutive ascending nodes of the reference orbit is 3156 km compared to 3139 km for the injection orbit. There are 10 reference tracks between any two consecutive equatorial ascending nodes; one track every 315.6 km. The injection orbit ground track drifts eastward at the rate of 216.7 km/day, providing a synchronizing opportunity once every 1.5 days. The initial configuration of a set of 10 reference tracks and the injection orbit tracks is illustrated schematically in Fig. 1.

Table 1. Baseline Orbit Parameters

Assumed Injection Time – June 1, 1992 22*hrs* 00*m* 00*sec*

Parameter	Injection Orbit	Operational Orbit	Required Change	Injection[5] Errors(3σ)
Semi-Major Axis (a, km)	7686.510	7714.428	27.918	4.0
Eccentricity (e)	0.002,57	0.000,075	0.002,495	0.000,5
Arg. of per. (ω, deg)	33.0	90.0	57.0	6.5
Inclination (I, deg)	66.039	66.039	0.0	0.08
Nodal Period (τ_n, sec)	6709.15	6745.72	36.57	--
Equatorial distance between two ground tracks (D, km)	3139	3156	8	--

The orbit acquisition maneuver sequence must accomplish several objectives. In addition to raising the semi-major axis a, both the eccentricity e and argument of perigee ω

must be carefully adjusted to the desired frozen values. Any inclination error imparted by the launch vehicle must be removed. The maneuvers used for acquiring orbital parameters must also acquire the desired reference ground track pattern. The orbit acquisition process ends when the ground track is within the ±1 km control band of a reference track.

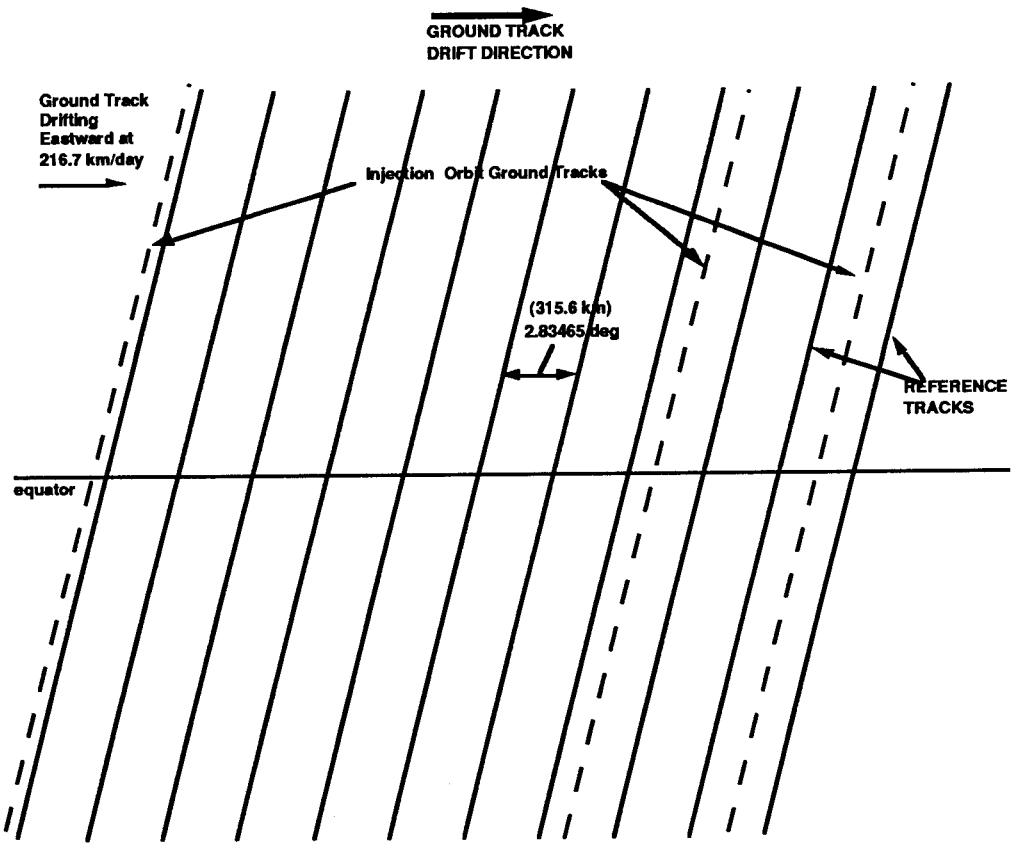

Fig. 1. Configuration of the injection orbit and reference tracks.

In order to standardize timelines and the description of geographical coverage, the start of every ground track repetition cycle, except Cycle 0, has been chosen to have the first ascending node at 99.95 deg east longitude.[6] The last maneuver of the retargeting sequence is used to position the ground track within the ±1 km band, and Cycle 0 begins with the last maneuver regardless of its geographic location. The maneuver performed at Cycle 0 must provide a smooth transition to the first Orbit Maintenance Maneuver (OMM), which needs to be scheduled at the transition of the ground track repeat cycles. Therefore, the last maneuver of the orbit acquisition sequence needs to be performed within the control band and prior maneuvers must guide the ground track into the control band. Cycle 1 begins with the first occurrence of 99.95 deg east longitude after Cycle 0. Thus Cycle 0 may not cover complete one ground track repeativity cycle. The first OMM may be scheduled at convenient time which may be at transition of Cycles 1 or 2.

SATELLITE CHARACTERISTICS

Although TOPEX/POSEIDON is a three axis stabilized satellite, near-continuous yaw steering about the local nadir and solar panel pitching are required to maintain the dominant solar panel area pointed near the sun for required power generation (Fig. 2). Therefore, the yaw steering mode is the Normal Mission Mode (NMM) for the satellite[6]. It is necessary to stop yaw steering and re-orient before a maneuver as part of a "turn-burn-turn" sequence. The yaw turn is accomplished using only reaction wheels at a maximum slew rate of 0.06 deg/sec. The turn time depends on the initial yaw rate and the turn angle. For some maneuvers the turn time may exceed one hour. The satellite is in Orbit Adjust Mode (OAM) "while thrusting" and its orientation is controlled by attitude jets.

The propulsion system is designed to provide sufficient thrust and directional control to meet all orbit adjust and maintenance maneuver requirements, including related attitude control. The propulsion module is a mono-propellent hydrazine blow-down system consisting of twelve 1-nt (0.2-lbf) and four 22-nt (5-lbf) thrusters. The 22-nt and four of the 1-nt thrusters, directed nearly along the roll axis, are used to perform orbit adjust maneuvers. The other 1-nt thrusters are used for attitude maintenance when satellite in OAM. The propulsion system is designed to implement a maneuver magnitude in the range of 1 mm/s to 15 m/s which satisfies the requirements[6] of both Orbit Acquisition and Maintenance Phases.

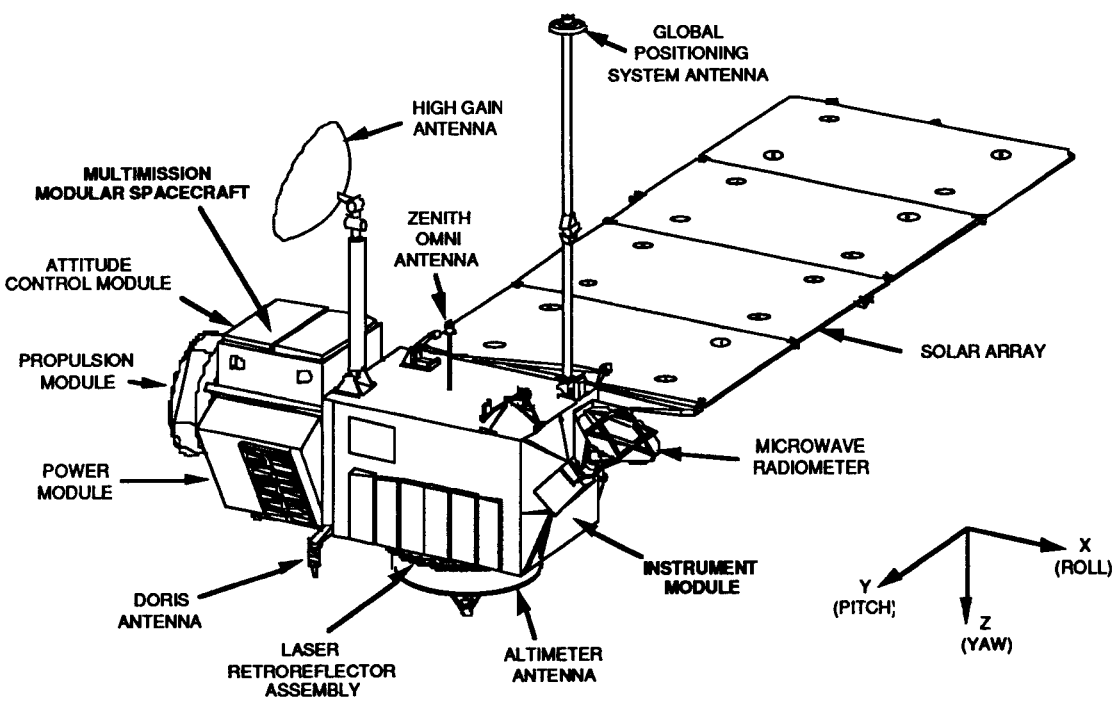

Fig. 2. Configuration of the satellite.

The Center of Mass (CM) of the satellite does not coincide with the center of body coordinates. Therefore, orbit adjust thrusters are canted to make the thrust vector of any

pair pass through the CM at the Beginning Of Life (BOL). The uncertainties in the determination of CM, thruster imbalance, thruster alignment accuracy, and thruster repeatability will generate a torque during thruster firing. Use of an Open Loop Firing Pattern (OLFP) is expected to partially compensate for this torque.[7,8] Therefore, the rate of torque build-up depends on the OLFP used. When the angular build-up due to the torque reaches a pre-set outer value, the orbit adjust thrusters are disabled and attitude thrusters are automatically enabled to reduce the accumulated attitude and rate errors to within the set inner deadband. This process of interrupting orbit adjust thrusters firing may occur several times during a maneuver even using an OLFP.[7,8] The oscillation of the system in and out of the deadband is called a limit cycle. Each interruption increases the overall time required to achieve the desired ΔV magnitude, and reduces the maneuver efficiency, defined as the ratio of the average thruster on-time to the time required to accumulate the total ΔV. The maximum ΔV that can be implemented without an interruption is expected to be $\sim$1.5 m/s using four 22-nt thrusters, and within 100 mm/s using a pair of 1-nt thrusters.[7,8] It is planned to implement a large ΔV magnitude ($\geq$2 m/s) with a single maneuver even though several interruptions may occur.

The duration of an orbit adjust maneuver may exceed one hour due to the limited turn rate and the duty cycle limitation of thrusters. During the turn-burn-turn sequence, the solar panel is not pointed towards the sun. Similarly, the orientations of several other systems with respect to the sun change during this time. From a power and thermal point-of-view, it is necessary not to schedule two maneuvers within a single orbit or during consecutive orbits. Therefore, it is not possible to apply a two-impulse maneuver technique to acquire the required values of e and ω.

OPERATIONAL CONSTRAINTS

The TOPEX/POSEIDON operational teams evolved a set of constraints and guidelines for the orbit acquisition maneuver sequence to reduce operational complexity, ease scheduling, and meet objectives with available resources. The constraints and guidelines are:

1. The first maneuver is to be used for 22-nt thruster calibration. This maneuver is to be scheduled nominally at 5 days after launch. During this period, operational teams will complete initial check-out and become familiar with the satellite. The next maneuver will be scheduled 5 days after the calibration maneuver to allow sufficient time for performance analysis.

2. Perform maneuvers around 3 PM Pacific Daylight Time (PDT). This may vary within one orbit period ($\sim$112 min) to place the maneuver at suitable location to acquire the frozen values of e and ω.

3. Schedule maneuvers every four days after first two maneuvers. This spacing includes time required to support the design of a primary and two back-up maneuvers. Also the spacing is sufficient enough to absorb operational delay upto one-day in implementing a maneuver. The nominal maneuver activity timeline is 5-5-4-4-4-. . . .

4. There are two back-ups for each nominal maneuver to handle one-revolution and one-day operational delays. These back-up maneuver designs need to be readily available during operations, to the extent possible. If one-day back-up is not available without penalty determine maximum allowable delay.

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

6. From a power and thermal point-of-view, do not schedule two maneuvers within a single orbit or two consecutive orbits

MANEUVER DESIGN MODELING

The maneuver design process is accomplished in two steps. The first step uses analytic software to determine a preliminary maneuver magnitude and direction. The second step uses precision numerical integration software to validate the maneuver design in the presence of all force models.

The rapid and efficient analytical tool developed for the orbit maintenance maneuver design[4] was modified and used in the Orbit Acquisition Maneuver Software (OAMS). The model uses recurrence relationships for earth gravity zonal harmonics and luni-solar gravitational attraction.[4,9] The earth gravity model is consistent with that used for the Reference Orbit design[3] except that the tesseral harmonics have not been included. The effect of atmosphere drag has been not included for designing the orbit acquisition maneuver sequence because of its negligible effects on the orbit at TOPEX/POSEIDON altitude over a few days.[4] The algorithm accepts mean elements[10] for propagation and provides an exact repeat of the reference ground track when reference mean elements are used in presence of the gravity field alone.

The model uses non–singular parameters a, $\xi = e\cos\omega$, $\eta = e\sin\omega$, I, Ω, and $L = M + \omega$ in place of classical Keplerian elements to avoid singularities in eccentricity near zero[11]. These parameters are updated at every time step (h) to account for perturbations.

$$
\begin{aligned}
a_i &= a_{i-1} \\
\xi_i &= \xi_{i-1} + \overline{\Delta\xi} + \Delta\xi_g + \Delta\xi_{ls} \\
\eta_i &= \eta_{i-1} + \overline{\Delta\eta} + \Delta\eta_g + \Delta\eta_{ls} \\
I_i &= I_{i-1} + \Delta I_g + \Delta I_{ls} \\
\Omega_i &= \Omega_{i-1} + \dot{\Omega}h + \Delta\Omega_g + \Delta\Omega_{ls} \\
L_i &= L_{i-1} + nh + \overline{\Delta L} + \Delta L_g + \Delta L_{ls}
\end{aligned}
\tag{1}
$$

where

$$
\begin{aligned}
\overline{\Delta\xi}, \overline{\Delta\eta} &= \text{secular rate in eccentricity vector due to gravity,} \\
n &= \text{mean motion of the satellite,} \\
\dot{\Omega} &= \text{secular rate in right ascension of node due to gravity,} \\
\overline{\Delta L} &= \text{secular rate in } L \\
\Delta\xi_g, \ldots, \Delta L_g &= \text{long–periodic perturbations due to gravity, and} \\
\Delta\xi_{ls}, \ldots, \Delta L_{ls} &= \text{long–periodic perturbations due to luni–solar gravity.}
\end{aligned}
$$

The non–singular parameters are directly transformed to position and velocity vectors to compute earth fixed parameters. The main task is to phase the ground track with the reference grid, hence the model computes the difference between the equatorial crossing longitudes of actual and reference tracks. The program searches for the nearest reference

equatorial longitude (λ_R) to the first equatorial crossing (λ) and establishes the relation between the two tracks. Later, the difference ($\Delta\lambda$) is computed for all equatorial crossings.

$$\Delta\lambda = \lambda - \lambda_R \tag{2}$$

The variation of $\Delta\lambda$ with time provides the size of the ground track relative drift to be corrected by maneuvers to align with the reference track in a given time frame.

The maneuver design model assumes that impulsive maneuvers are impelemented at optimal locations[12] defined by their arguments of latitude u. The direction of the impulsive maneuver vector is represented in the local horizon system. The z-axis is defined by the radius vector, the y-axis by the orbit normal vector, and the x-axis completes the right-handed coordinate system. The coordinate system moves with the satellite. The x-axis is along the roll axis when yaw and pitch angles are zero. For the TOPEX/POSEIDON satellite, the x-axis is nearly along the velocity vector as both injection and operational orbits are near-circular. The maneuver vector is defined by the angles α and δ; α is the angle between the x-axis and the projection of the maneuver vector on the x-y plane (roll-pitch plane), and δ is the angle between the x-y plane and the maneuver vector. Nominally, the values of α and δ are set to zero for inplane maneuvers; for inclination maneuvers, α and δ take the values of ± 90 deg and 0 deg, respectively. For a maneuver vector in an arbitrary direction, the above angles will have different values.

The prediction model propagates to a maneuver time (at the optimal location in the orbit) and the orbital parameters are transformed to position and velocity vectors. The velocity vector is then incremented appropriately with the maneuver vector to obtain post-maneuver position and velocity vectors. Differences between the pre- and post-maneuver parameters provide the changes due to the maneuver. The orbit propagation continues with the post-maneuver orbital parameters.

MANEUVER DESIGN CONCEPT

At the TOPEX/POSEIDON altitude, the relations between the two near-circular orbits remain approximately constant as the orbit decay due to atmosphere drag is negligible over a few days.[4] The relative ground track drift rate of the injection orbit remains the same until a maneuver is implemented. The relationship between the injection and the reference orbit tracks is linear and is determined by the relative drift rate. This relation holds after every maneuver. Thus the equation of ground track phasing is:

$$\lambda_D = d_1(t_1 - t_0) + d_2(t_2 - t_1) + \ldots + d_n(t_n - t_{n-1}) \tag{3}$$

where

λ_D = equatorial distance between postioning track and the track at 1st maneuver
d_i = ground track relative drift rate after i^{th} maneuver
t_i = i^{th} maneuver time

The changes in the relative drift rates are accomplished through inplane maneuvers. Maneuvers are to be performed at pre-determined times t_i, i =1,2,. . . ,n. Initially, a reference

track is arbitrarily selected based on the current ground track position and the relative drift rate. This provides an initial value of λ_D (Fig. 3). Each maneuver is designed individually and later the entire sequence is verified for the ground track phasing using Eq. 3. However, Eq. 3 defines an over-determined system. The planned reference track (λ_D), maneuver magnitude, and the number of maneuvers are parametrically optimized. The resulting sequence has a minimum number of maneuvers and gradually reduces the relative drift rate, ending with a near-zero drift rate when the actual ground track becomes aligned with the reference track within ± 1 km.

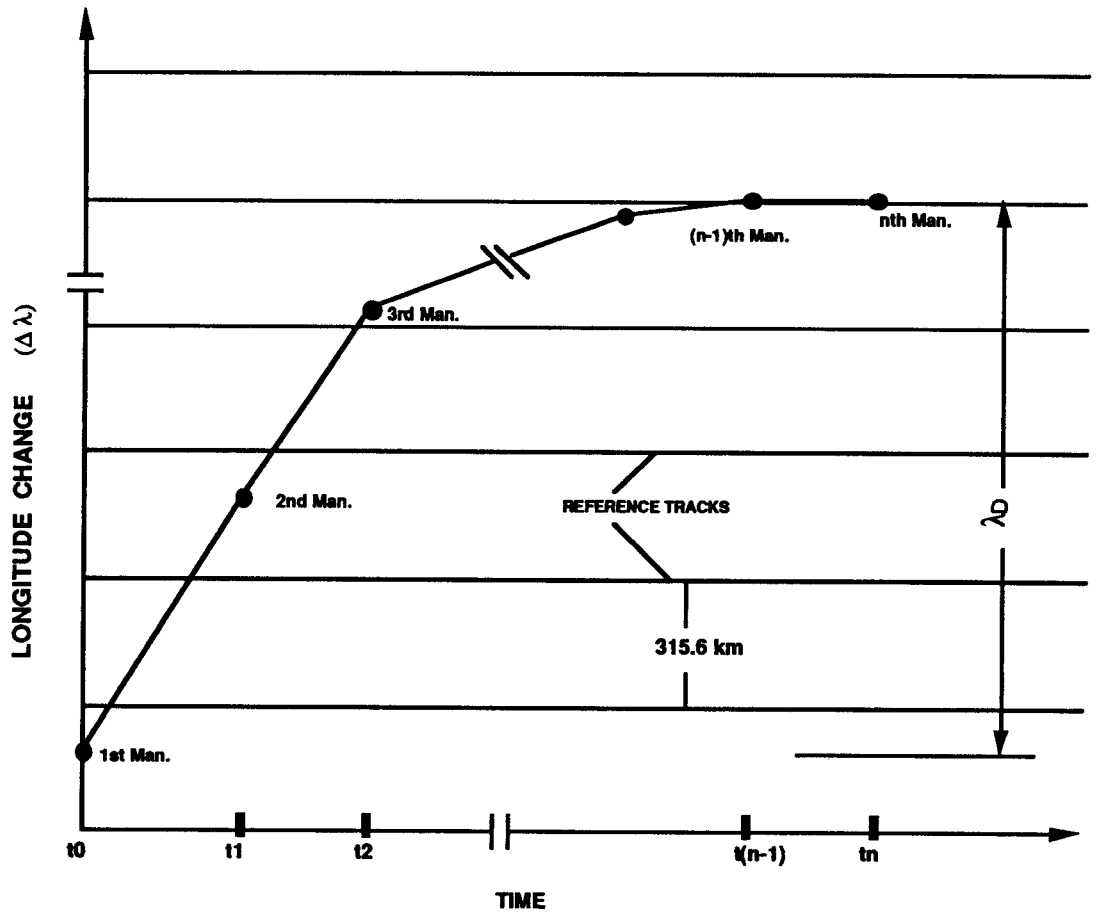

Fig. 3. Ground track phasing concept.

The OAMS generates the re-targeting sequence using the above concept. An initial maneuver sequence phases the ground track using only J_2. This sequence is refined using full models of the OAMS to achieve the required frozen values of e and ω along with ground track phasing. Remember that a two-impulse maneuver technique is not applied due to satellite constraints.

ERROR SOURCES

Maneuver design becomes realistic only when error sources are included. The error sources which affect the maneuver design are injection errors, maneuver execution errors,

and orbit determination uncertainties.

Injection Errors

The injection errors (Table 1) change the achieved injection orbit and thus influence the orbit acquisition sequence. The dominant injection error affecting the acquisition sequence design is the inclination error. A separate maneuver needs to be implemented to correct the inclination. The expected 3σ error (0.08 deg) inclination requires a corrective maneuver of 10 m/s. Errors in inplane orbital parameters a, e, and ω are easily absorbed in the sequence by adjusting inplane maneuver magnitudes.

Maneuver Execution Errors

Maneuvers are accomlished by firing either 22-nt thrusters or 1-nt thrusters along the desired orientation. Maneuver execution errors result from thruster proportional and fixed velocity magnitude errors, and satellite pointing uncertainties.

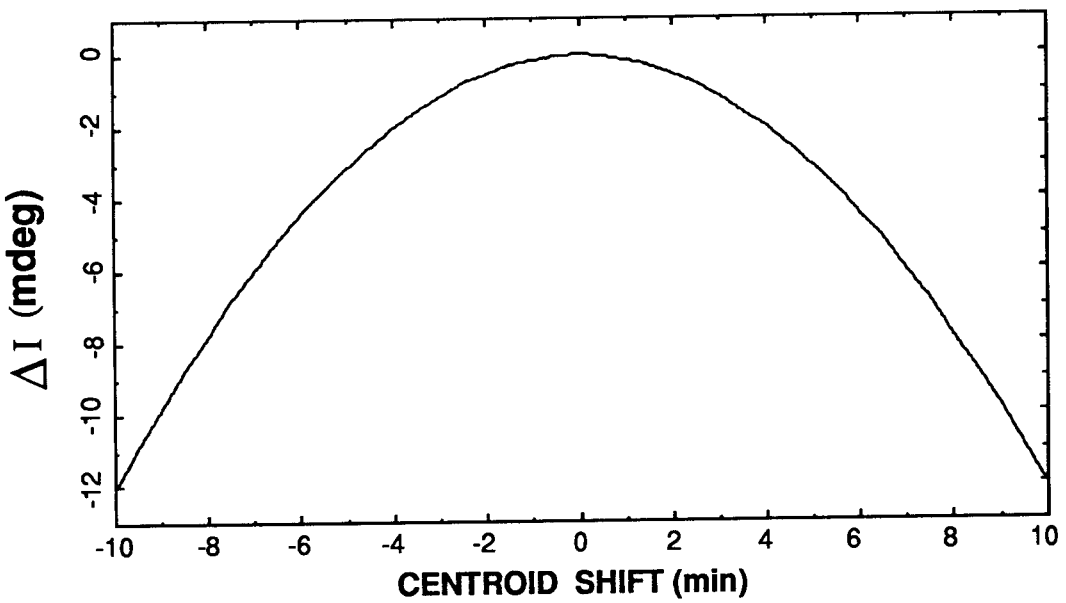

Fig. 4. Effect of Centroid Shift on Performance of a 10 m/s Inclination Maneuver.

The 3σ pointing control requirement for maneuvers is 4.5 deg, including contributions due to satellite pointing control, thruster misalignments, and uncertainties in satellite CM location.[3] The effect of this pointing uncertainty on maneuver execution accuracy is proportional to the maneuver magnitude. For example, a 10 m/s inclination maneuver applied normal to the orbit plane causes 0.8 m/s inplane component ΔV due to a 4.5 deg pointing error in the yaw direction. The above inplane component changes 'a' by 1.9 km, resulting in an unplanned large variation of the ground track drift rate. Similarly, large inplane maneuvers cause inclination errors which may have to be removed later. Therefore, pointing uncertainties play a dominant role during the Orbit Acquisition Phase.

The satellite is required to provide velocity changes with a fixed magnitude accuracy $3\sigma_{\Delta V_f} = 16$ mm/s, and a proportional magnitude accuracy $3\sigma_{\Delta V_p} = 1.5$ % when four 22-nt

thrusters are used.[3] The fixed magnitude accuracy requirement is 0.4 mm/s when two 1-nt thrusters are used. The proportional error increases with maneuver magnitude and plays a dominant role in shaping the orbit acquisition sequence. The velocity execution errors ($\sigma_{\Delta V_e}$) are assessed by statistically adding fixed and proportional errors.

$$\sigma_{\Delta V_e} = \sqrt{\sigma^2_{\Delta V_f} + (\Delta V \sigma_{\Delta V_p})^2} \tag{4}$$

The maneuver efficiency is a function of the number and length of interruptions. Changes in the expected maneuver efficiency shift the centroid of the burn from the planned time and introduce another execution error source. This centroid shift results in a maneuver performance penalty. The penalty increases with maneuver magnitude for a given efficiency. A 1-min centroid delay for an inplane maneuver is equivalent to 3.2 deg in pitch error, which affects the resulting eccentricity vector. A 2-min centroid delay for an inclination maneuver of 10 m/s causes a 1-mdeg change in the resultant inclination (Fig. 4). Extensive maneuver simulation studies[7,8] show that the maneuver efficiency is expected to be > 60% for most cases. This translates into a 1.6-min centroid delay for a 10 m/s maneuver. Use of an appropriate OLFP is expected to increase the maneuver efficiency and reduce the centroid delay.

Orbit Determination Uncertainties

The uncertainties of Operational Orbit Determination (OOD) are not expected to affect maneuver design any time during the Orbit Acquisition Phase.

Table 2. Worst-case 3σ Execution Errors Assumed for Maneuver Design

Error Sources	3σ Values
Velocity Errors	
Fixed	16 mm/s for four 22-nt thrusters 0.4 mm/s for two 1-nt thusters
Proportional Error	8% for calibration man. 5% for first man. with 1-nt thrusters 3% for remaining maneuvers
Pointing Errors*	
yaw	4.5°
pitch	4.5°
Centroid Shift	Computed assuming 60% efficiency

* Maneuver simulation studies show that both yaw and pitch errors may reach above values in a worst-case.

MANEUVER DESIGN

There are three types of maneuvers in a typical orbit acquisition maneuver sequence.

a) calibration burn of four 5–lbf (22–nt) thrusters,

b) inclination correction (as required), and

c) in–plane maneuvers.

Calibration Maneuver

The primary objective of the calibration maneuver is to confirm satisfactory operation of 22-nt thrusters and the overall maneuver sequence (attitude control interruptions). In addition, the maneuver is used:

a) to determine the maneuver scale factor(the ratio of the achieved–to–planned maneuver magnitude),

b) to measure the unbalanced torques to update Open Loop Fire Pattern (OLFP), and

c) to get a better understanding of the execution errors.

Overall, its use is expected to eliminate systematic error sources leading to an improvement in the accuracies of subsequent maneuvers. The orbit determination results and telemetry data are used to estimate the maneuver scale factor.

The maneuver magnitude is designed to meet the above objectives and is not based on ground track phasing requirements. The operational teams recommended the calibration maneuver magnitude to be in the range of 2 to 5 m/s with a preference for 2 m/s to give a valid calibration for the subsequent maneuvers. A high relative drift rate (216.7 km/day) of the injection orbit ground track allows to select the maneuver magnitude before launch and it is not changed based on achieved injection orbit due to injection errors. Therefore, this maneuver is not adaptive for the current injection orbit. The selected maneuver magnitude is 2 m/s. The planned direction of the maneuver will be inplane along the satellite velocity, and implemented near perigee to raise the apogee altitude towards the operational orbit. Thus this maneuver will also contribute to the ground track phasing to some extent.

Inclination Maneuver

The orbit inclination must be consistent with the semi-major axis to match the required ground track pattern and to overfly the verification sites. Inclination error imparted by the launch vehicle is removed through this maneuver and the maneuver magnitude depends on size of the error. An inclination error less than ± 0.005 deg will be removed by a trim maneuver after the large (> 2.0 m/s) inplane maneuvers have been completed.

The Δi maneuver is implemented at either an ascending or a descending node and is scheduled (if required) after the calibration maneuver to take the advantage of the planned 90 deg yaw steering attitude so that the turn is minimal. An inclination raise maneuver is implemented at the *descending* node, whereas the *ascending* node is the preferred location to decrease inclination.

Inplane Maneuvers

Inplane maneuvers are implemented at specific locations in the orbit to achieve the required 'a', frozen values of 'e' and 'ω', and to phase the ground track with the reference track in the presence of expected execution errors. The number of inplane maneuvers in a sequence depends on the size of execution errors.

All maneuvers are designed together to ensure the ground track phasing with minimum number of maneuvers, accounting for the expected performance of each maneuver.

However, the implemented ΔV may be different from the planned ΔV due to execution errors. Therefore, the remainder of the sequence is redesigned after each maneuver, reflecting updated estimates of expected maneuver execution errors and to absorb the unexpected errors of the previous maneuvers. Thus the inplane maneuver magnitudes are adaptive in nature. Large maneuvers (> 2.0 m/s) are implemented early in the sequence so that later maneuvers can absorb execution errors of these maneuvers. The last maneuver is a trim maneuver ($<<100$ mm/s) used to position the ground track in the ± 1 km control band.

Shoot-Short Strategy

With every inplane maneuver, the eastward drift rate reduces such that the ground track is at the desired position at the time of next inplane maneuver to finally align with the reference track (Fig. 3). After implementing the first large maneuver, the synchronizing reference track is determined. Therefore, the drift rate after each inplane maneuver is planned to retain this reference track for synchronizing. However, due to execution errors, the achieved post-maneuver drift rate may be different from the expected value. A higher drift rate may cause the ground track drift beyond the reference track before the time of next maneuver. This would force the ground track to be synchronized with the next available track, or reverse the ground track back to the planned track with extra maneuvers. In either case the orbit acquistion period is increased. A lower drift rate causes the ground track to drift west of the planned position at the time of next maneuver. The ground track may be accelerated appropriately with subsequent maneuvers to synchronize with the planned reference track at the required time. A "shoot-short" strategy is applied while designing the maneuver sequence. The maneuver magnitudes are designed such that the resulting ground track does not cross the reference track planned for synchronizing, and to have appropriate distance between the two tracks by the time of next maneuver. This strategy avoids the penalty even under the expected worst-case 3σ execution errors.

Orbit Acquisition Maneuver Sequence

The total number of days required to acquire the operational orbit depends on the number and spacing of maneuvers in a sequence. From an orbit mechanics point-of-view, these maneuvers may be implemented every two days, but operationally such a sequence may not be feasible. Candidate maneuver sequences have been designed assuming different maneuver spacings and different maneuver epochs, based on the expected launch vehicle performance (Table 1) and the satellite performance (Table 2). The requirement to achieve the operational orbit in a minimum number of days, standardize the time required to complete the operational activities between any two maneuvers including back-ups, and operation team staffing are the principle factors in determining the interval between the maneuvers. The maneuver sequences have been iterated with operation teams to evolve a baseline maneuver activity timeline with daily spacings of 5-5-4-4-4-.... .

A set of possible maneuver sequences have been designed consistent with the baseline activity timeline assuming different levels of injection errors and execution errors (Fig. 5). Each sequence is different and the calibration is the only common maneuver among all sequences. Maneuvers are designed individually assuming no execution errors and then evaluated with execution errors to make the sequence more realistic and robust. Each maneuver maneuver is analysed for different combinations of errors. The error combination which causes a worst-case ground track phasing is chosen for providing the performance of a maneuver. For example, a nominal calibration maneuver is 2 m/s, reducing the relative ground track drift rate from 216.7 to 183.2 km/day, whereas the performance of expected

worst-case errors reduce the drift rate only to 186.4 km/day (Table 3). The subsequent maneuver sequence is then redesigned to reflect the performance.

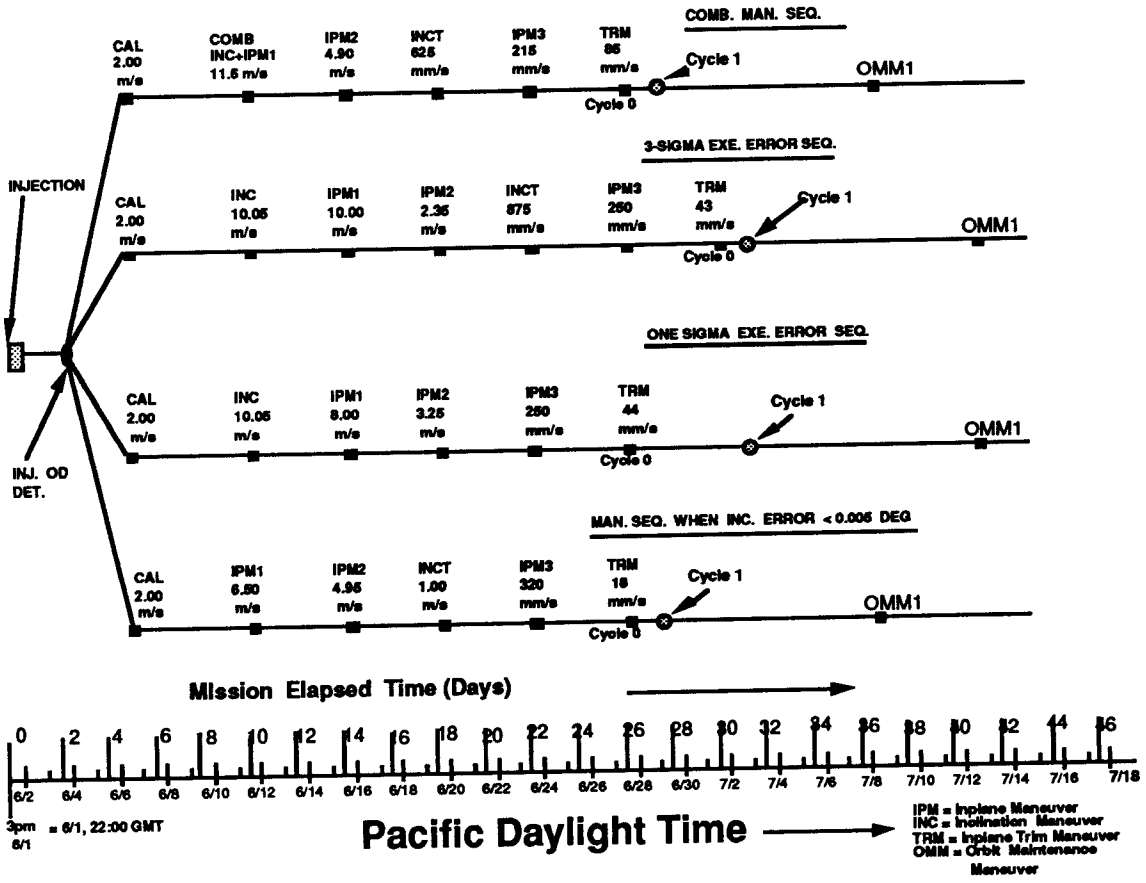

Fig. 5. Operationally Feasible Sequences for Different Error Assumptions.

The orbit acquisition period is reduced by four days if the inclination error is < 5 mdeg, or if the size of the execution error is $\leq 1\sigma$ level(Fig. 5). Use of a combined maneuver (inclination + first inplane) can eliminate a maneuver in the orbit acquisition sequence even in the presence of expected worst-case 3σ errors. However, the combined maneuvers are feasible only at one of the nodes (ascending or descending) near apogee of the orbit, as it replaces the inclination maneuver. The turn time to get the required orientation is reasonable at only one of the nodes. The turn conditions are satisfied at apogee node for the current injection orbit when an inclination increase is required.

The maneuver sequence designed with the expected worst-case 3σ injection and execution errors has been taken as a baseline sequence (Fig. 6) for further analysis. This

sequence achieves the operational orbit in 30 days requiring seven maneuvers, including an initial calibration maneuver. Table 3 provides the details about each maneuver and Fig. 7 illustrates the ground track phasing for this sequence.

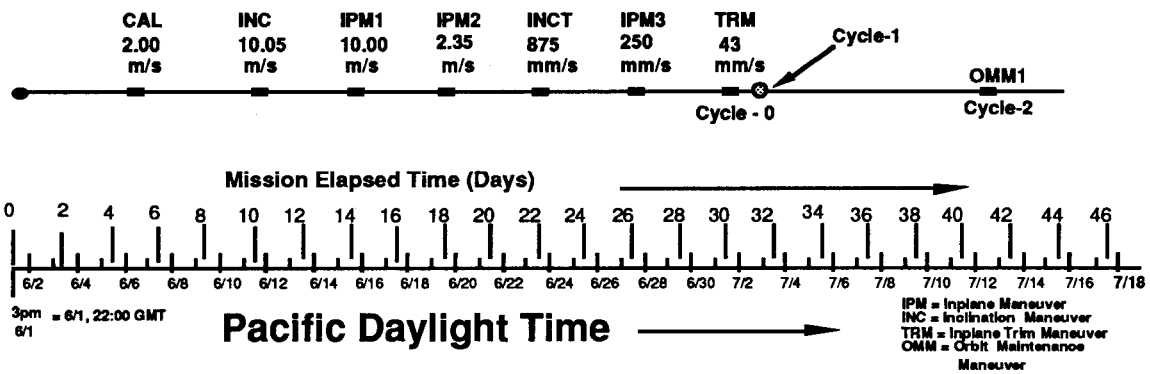

Fig. 6. Baseline Maneuver Sequence.

Table 3. Baseline Maneuver Sequence Performance Characteristics

Man. No.	Day from Injection	Rev. No.	Man. type	Targ. para.	Man. Loc. (u, deg)	G. T. Drift (East) (km/day)	Thruster used (lbf)	ΔV (m/sec)
0.						216.7		
1.	5	64	calib.	a,e,ω	50	186.4	5	2.000
2.	10	128	incli.	i	node	200.9	5	10.050
3.	14	180	inplane	a,e,ω	190	41.0	5	10.000
4.	18	231	inplane	a,e,ω	245	3.5	5	2.350
5.	22	282	incli.	i	node	4.8	5	0.875
6.	26	334	inplane	a,e,ω	225	0.85	0.2	0.250
7.	30	385	cycle-0	a,e,ω	230	0.15	0.2	0.043
8.	40	516	omm1	a			0.2	

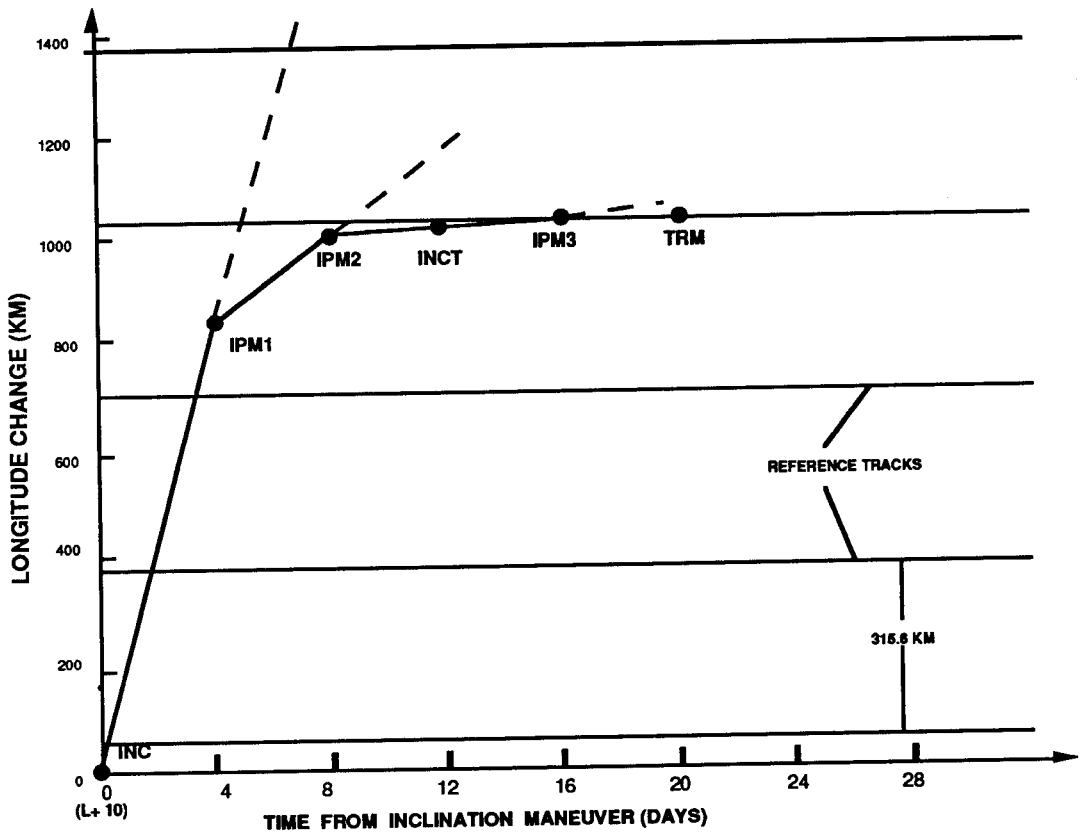

Fig. 7. Ground Track Phasing with Baseline Maneuver Sequence.

BACK-UP PHILOSOPHY

The sequence design must accommodate unexpected delays in maneuver execution. The response time may vary from hours to days. Ground track phasing demands a sequence of maneuvers implemented at designated times. If a maneuver is delayed, the ground track position changes and the remaining sequence to align with the reference track planned becomes obsolete. Delay in a maneuver implementation may lead to severe penalty in total orbit acquisition period. A back-up philosophy was evolved to avoid penalty for few operational delays (one-revolution and one-day).

One-day delay with four day spacing allow the sequence to return on the original schedule. The original schedule is resumed after a delayed maneuver. If the inclination maneuver is delayed by a day, for example, the first inplane maneuver (IPM1 in Fig. 6) is scheduled after three days instead of four days (Fig. 8). Therefore, if a maneuver is delayed by a day, the number of days required to achieve the operational orbit remains same and there is no penalty.

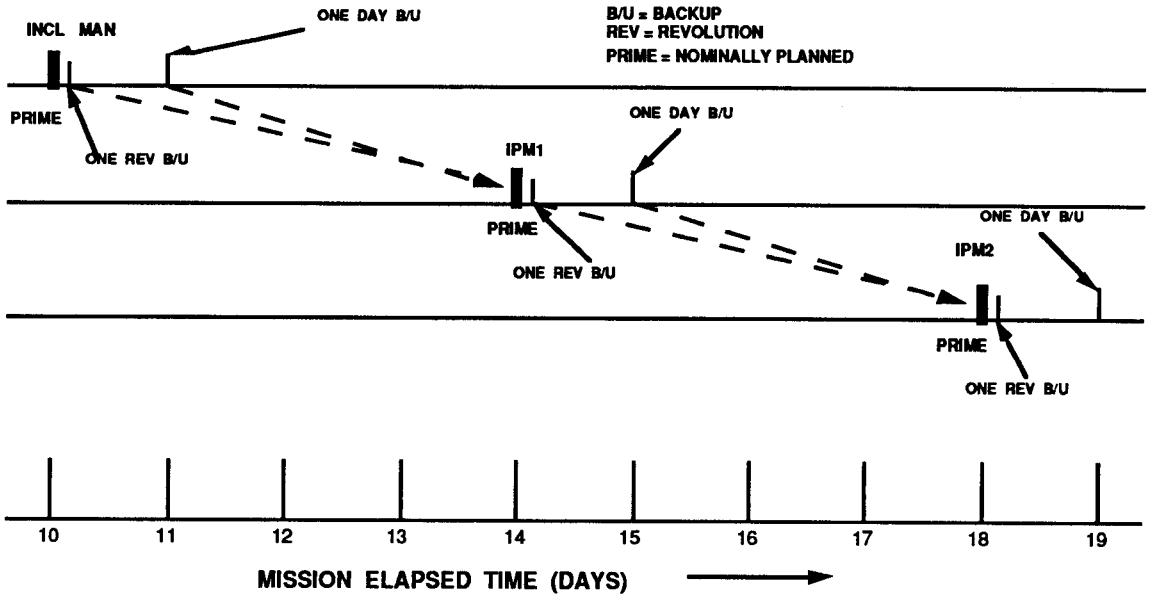

Fig. 8. Scheduling Plan with a Delayed Maneuver.

Table 4. Ground Track Phasing Sensitivity to Operational Delays.

	CAL	INCL	IPM1	IPM2	INCT	IPM3	TRM
Nom. Time (L+Days)	5	10	14	18	22	26	30
Allowable Delay Without Penalty	ONE DAY	ONE DAY	ONE DAY	**5-10 REVS**	ONE DAY	**4-8 REVS**	—
Penalty for One Day Delay	NO PENALTY	NO PENALTY	NO PENALTY	**4-DAY PENALTY**	NO PENALTY	**4-DAY PENALTY**	ONE DAY PENALTY

A ground track phasing sensitivity analysis to operational delays was conducted using the baseline sequence. The analysis was conducted by including one delay in the entire sequence. The study shows that a one-revolution delay does not affect the current maneuver magnitude, but it does affect the magnitudes of subsequent maneuvers. If a maneuver is delayed by one day, the whole sequence must be redesigned including each delayed maneuver. The results (Table 4) of the sensitivity analysis indicate that a one-day delay may be tolerated without penalty except for the second and third inplane maneuvers (IPM2 or IPM3 in Fig. 6). A one-day delay in these maneuvers results in loss of the reference

track planned for synchronizing, and requires either an extra maneuver or a long period of cruise (7-8 days) to finally re-synchronize with reference track (Fig. 7). The resulting minimum penalty will be four days in the total orbit acquisition period. The allowable delays in these maneuvers were estimated as 5-10 revolutions in IPM2 and 4-8 revolutions in IPM3, respectively. The allowable delay is identified as a window of revolutions rather than a particular revolution due to the statistical nature of the execution errors. The lower limit is for the worst-case 3σ errors. The realistic allowable delay is then determined in-flight following the performance evaluation of the last completed maneuver. For these two maneuvers, back-ups are standardized at 5-10 and 4-8 revs respectively.

More than one operational delay in a sequence will be treated as a contingency situation. For each maneuver there are two back-ups; one-revolution, and one-day, respectively. It is planned to design these back-up maneuvers in parallel with the nominal maneuver so that they are readily available.

INVESTIGATION OF A HIGHER INJECTION ORBIT

The baseline maneuver sequence achieves the operational orbit in presence of worst-case 3σ errors in 30 days. The only way to reduce the orbit acquisition period without changing nature of the activity timeline is to reduce number of maneuvers in the sequence. The possibility of a higher injection orbit (closer to the operational orbit) was therefore investigated to achieve the operational orbit in less than 30 days and with fewer maneuvers. The main criteria for selecting the baseline injection orbit was to have a syncronizing opportunity once every 1.5 days. However, the operational activity time line (5-5-4-4-4-...) allows a synchronizing opportunity of once every four days. The orbital parameters of a high injection orbit were selected such that it provides synchronizing opportunity once every three days and also avoids the possibility of a spacecraft collision with the last stage of the launch vehicle even with 3σ-high performance.

Table 5. Orbit Parameters of a Higher Injection Orbit

Assumed Injection Time – June 1, 1992 22*hrs* 00*m* 00*sec*

Parameter	Injection Orbit	Operational Orbit	Required Change	Injection Errors(3σ)
Semi-Major Axis (a, km)	7702.0	7714.406	12.428	4.0
Eccentricity (e)	0.000,5	0.000,095	0.000,405	0.000,5
Arg. of per. (ω, deg)	33.0	90.0	57.0	6.5
Inclination (I, deg)	66.039	66.039	0.0	0.08
Nodal Period (τ_n, sec)	6729.44	6745.72	16.57	--
Equatorial distance between two ground tracks (D, km)	3148	3156	8	--

The mean elements of a possible higher injection orbit are summarized in the Table 5. The ground track of this orbit drifts eastward at a rate of 96 km/day compared to 216.7 km/day for the current baseline orbit. The maneuver sequence is more sensitive to injection and execution errors due initial smaller drift rate. The injection error in 'a' also influences the maneuver sequence design. In fact, the calibration maneuver magnitude is strongly dependent on 'a' achieved by the injection (Fig. 9) and its maneuver magnitude can vary between 2.4 m/s and 3.5 m/s depending on the achieved 'a' after injection. The calibration maneuver is therefore adaptive like other maneuvers in the sequence. Analysis also indicates that pointing errors of a large inclination maneuver (10 m/s) strongly influences the maneuver sequence (0.8 m/s inplane component at 4.5 deg yaw error).

A calibration maneuver magnitude in the range of 2 to 5 m/s has been selected by the operational teams; the actual magnitude must be determined after the injection. A reduction of at least four days in the orbit acquisition period is possible with the higher injection orbit even under a worst-case scenario (Fig. 9). In certain cases, the reduction is greater than four days by eliminating more than one maneuver. The magnitudes of the inplane maneuvers are smaller compared to the baseline sequence and hence, the sequence becomes more controllable and predictable after the large inclination maneuver due to lower execution errors.

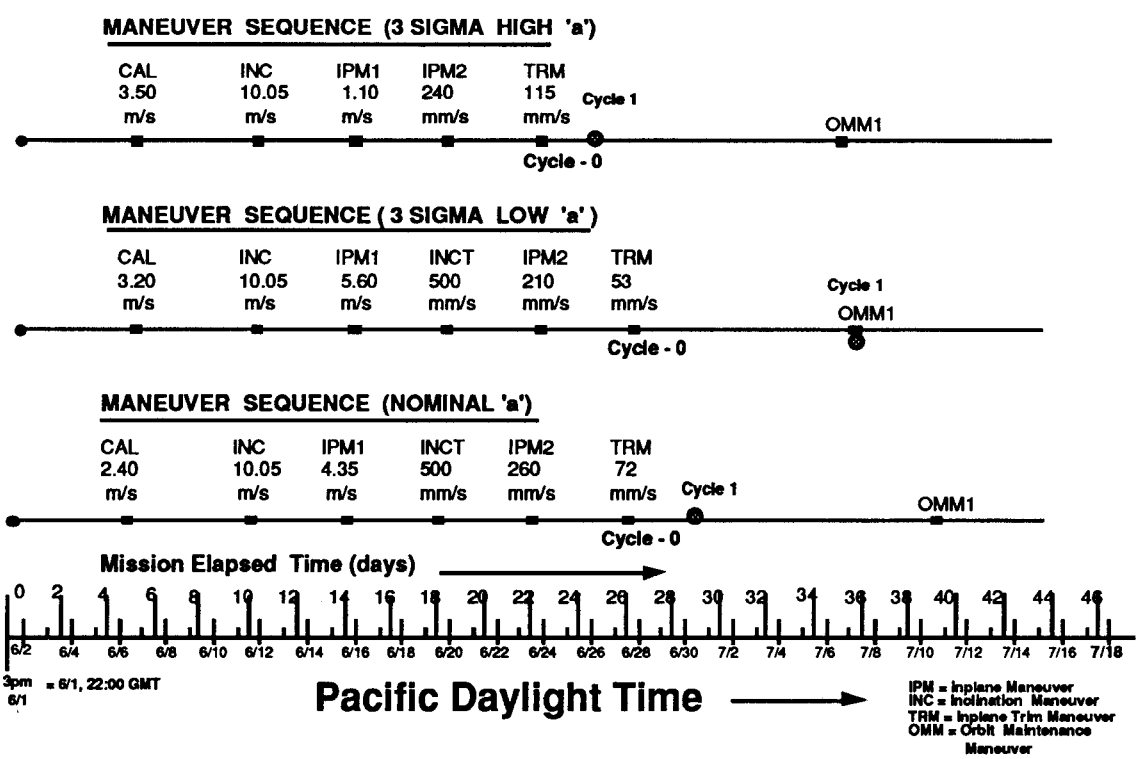

Fig. 9. Maneuver Sequences for a Higher Injection Orbit.

CONCLUSIONS

An operationally feasible maneuver sequence has been designed to achieve the TOPEX/POSEIDON operational orbit from an injection orbit based on expected worst-case 3σ injection and execution errors. The study indicates that the orbit acquisition period depends on the number of maneuvers in a sequence, a function of injection and execution errors. The orbit acquisition period may be reduced by four days if the inclination error is < 5 mdeg, or if the size of the execution error is $\leq 1\sigma$ level. The feasibility of a combined maneuver may also reduce the acquisition period by four days. A "shoot-short" strategy has been applied while designing the maneuver sequence. This strategy enables synchronization of the ground track with the reference track even under worst-case expected execution errors and thus avoids penalty. The remaining sequence is redesigned after each maneuver performance evaluation and thus the maneuvers are adaptive, except for the calibration maneuver.

The maneuver design provides back-ups for operational delays: one-revolution for each maneuver and a one-day for most maneuvers in the sequence. For two inplane maneuvers (IPM2 and IPM3 in Fig. 6), the required back-ups are less than one day and have been standardized in terms of window of revolutions. The actual back-up maneuver time for these maneuvers is determined after performance evaluation of the previous maneuver.

Minimum reduction of four days (by saving a maneuver) is possible with a higher injection orbit even under worst-case scenario. The calibration maneuver becomes adaptive and is selected after injection orbit determination. The maneuver sequence is more controllable and predictable after the large inclination maneuver. The Project has recently (August 91) adopted the higher injection orbit.

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