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TOPEX/POSEIDON ORBIT MAINTENANCE FOR THE FIRST FIVE YEARS*

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The TOPEX/Poseidon orbit maintenance strategy was changed following launch to include the effects of observed unmodeled, and hence anomalous, along-track accelerations. The anomalous force causes the semi-major axis, a, to either increase (called "boost") or decrease ("deboost" or "decay") depending on the satellite attitude and solar array pitch angle offset. Although this force is the most uncertain parameter in ground track prediction, it has been used as a passive technique for orbit maintenance, thereby reducing the number of propulsive maneuvers, enhancing maneuver spacing, and to place maneuvers at convenient times. This passive technique was first demonstrated in May 1993. The TOPEX/Poseidon orbit has been uniquely maintained using both passive (non-propulsive) and active (propulsive) maneuvers. Furthermore, the orbit has been maintained using only the passive technique since the ninth orbit maintenance maneuver on January 15, 1996.

Only nine orbit maintenance maneuvers have been required to maintain the ground track, including verification site over flights, since achieving the operational orbit on September 21, 1992 (mission requirement: 95% within ±1 km). During this period, a has varied within 7714,429±7 m, while the inclination i periodically fluctuated in the range 66.0408° ± 0.0040°. The frozen orbit (required $e < 0.001$ and $\omega \approx 90°$) has been maintained without any dedicated eccentricity maneuvers. The frozen eccentricity vector has completed two periodic cycles and it is currently tracing its third cycle (period ≈26.7 months).

INTRODUCTION

Since its launch on August 10, 1992, TOPEX/Poseidon†† has precisely mapped the topography of over 95% of the earth's ice-free seas. The wealth of scientific information provided by its very high quality ocean-altimetry data prompted NASA and CNES to further extend the TOPEX/Poseidon mission through 2001 to overlap with the successor Jason-1 mission. To facilitate high quality altimetry data acquisition, the satellite is maintained in a nearly-circular, frozen orbit ($e \approx 0.000095$, $\omega \approx 90°$) at an altitude of ≈1336 km and an inclination of i = 66.04° (Ref. 1). This orbit provides an exact repeat ground track every 127 orbits (≈10 days) and over flies two verification sites: a NASA site off the coast of Point Conception and a CNES site near the islands of Lampione and Lampedusa in the Mediterranean Sea.

After launch, six orbit maintenance maneuvers (OMMs) were implemented to acquire the operational orbit from the injected orbit.[2] These maneuvers achieved the frozen orbit, removed inclination errors induced by the launch vehicle, and synchronized the ground track with the reference grid and two

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†† TOPEX/Poseidon is a joint mission of the US National Aeronautics and Space Administration (NASA) and the French Centre National d'Etudes Spatiales (CNES). The primary mission lifetime was 3 years and the extended mission an additional 2 years.

verification sites. The operational orbit was achieved on September 21, 1992 and altimeter data acquisition started on September 23, 1992.

The Jet Propulsion Laboratory (JPL) of the California Institute of Technology is responsible for conducting all mission operations including operational navigation. Operational orbit determination (OD) using radiometric data acquired via the NASA Tracking Data Relay Satellite System (TDRSS) is provided by the Flight Dynamics Facility (FDF) of NASA's Goddard Space Flight Center(GSFC).

Prior to launch, orbit maintenance maneuver (OMM) design[1] was expected to depend primarily on atmospheric drag and the uncertainty of its prediction. The consequent maneuver targeting strategy had to be changed following launch due to the observation of unexpected along-track accelerations[3] called "anomalous forces". These forces did not influence operational orbit acquisition; however, it became necessary to accurately model and predict the anomalous force for effective ground track control. OMM1 was delayed for one ground track repeat cycle to collect additional OD data so that a reasonable empirical model could be constructed, thereby causing the ground track to leave the control band for a few days. Thus OMM1 was implemented outside the eastern edge of control band.

An empirical model[4] based on observed MOE (Medium-accuracy Orbit Ephemeris) accelerations is used for the anomalous force prediction. The MOE is based on a combination of laser ranging and GPS (Global Positioning System) data. The anomalous force model is validated using thrust parameters provided by the FDF. The magnitude of the anomalous forces is equivalent to or greater than the effects of the atmospheric drag and either raises ("boosts") or lowers ("decays") the orbit depending on the satellite attitude and solar array orientation. Its uncertainty significantly influences ground track prediction accuracy, especially during low drag conditions ($70 < F_{10.7} < 120$)*, when it is the largest uncertainty in orbit determination. The potential of using the anomalous force as a tool for ground track control was soon recognized and this passive technique[4,5] was demonstrated for the first time in May 1993 to avoid a propulsive maneuver near the western boundary of the control band, and later, in October 1995 to postpone OMM9 until Jan. 15, 1996. Since OMM9, the orbit has been maintained using *only* the passive technique, thereby greatly simplifying mission operations.

The TOPEX/Poseidon mission has been uniquely maintained utilizing a combination of both active (e.g., propulsive) and passive (e.g., non-propulsive) maneuvers. This paper describes the maneuver design and implementation strategies used for orbit maintenance in the presence of the anomalous force during the first five years of satellite operations. Maneuver performance characteristics and ground track maintenance statistics are provided. Use of the passive techniques in reducing the number of maneuvers and complexity of the mission operations are summarized.

MISSION REQUIREMENTS AND OPERATIONAL CONSTRAINTS[6]

Science objectives require that 95% of all equatorial crossings be contained within a ±1 km control band centered on a pre-defined earth-fixed reference ground track grid, and that 95% of all verification site over flights have a miss distance at closest approach of <1 km. The OMMs are constrained to occur over land at or near the boundary of the ≈10 day ground track repeat cycles (±1 orbit). Maneuver spacing must be as large as practical, with a minimum spacing of 30 days. Eccentricity must be maintained less than 0.001 throughout the mission; this requirement has been met by utilizing a frozen-orbit for the eccentricity vector (e, ω). Furthermore, maneuvers may not compromise satellite health and welfare; such requirements prevail over other mission requirements when conflicts arise. This leads to additional restrictions on the timing of maneuvers and the command sequence for maneuver implementation. The primary restrictions are due to satellite power, thermal, and star-tracker field-of-view constraints.

REFERENCE ORBIT

Mean orbital parameters[7] of the TOPEX/Poseidon operational orbit are shown in Table 1. This operational orbit provides an exact repeat ground track every 127 orbits in 10 sidereal days and over flies both the NASA and CNES verification sites once per repeat cycle. The first orbit of the 127-orbit ground

* $F_{10.7}$ is the 10.7 cm solar flux reported by the Penticton Dominion Radio Observatory. Units are 10^{-22} watts/(m^2-Hz).

track repeat cycle has an ascending node at 99.92° E. longitude. The operational orbit is referred to as the *reference orbit* and the mean elements describing this orbit are called *the reference elements*. The ascending nodal crossing longitudes of the *reference orbit* define the sub-satellite earth fixed *reference grid*. The *reference orbit* was initially designed[8] using a 17 × 17 truncation of the GEMT2 earth gravity field and was later refined using a 20 × 20 truncation of GEMT3. This orbit was again refined using a 20 × 20 truncated JGM2 (Joint Gravity Model-2)[9] during July 93. The JGM2 was derived by refining GEMT3 using TOPEX/Poseidon precision orbit determination (POD) results.

Table 1. TOPEX/POSEIDON REFERENCE ELEMENTS (EPOCH: JULY 1, 1993 00:00 UTC).

Semi-Major Axis (a)	7714.42942 km
Eccentricity (e)	0.000095
Inclination (i)	66.040°
Right Ascension of Ascending Node (Ω)	139.552°
Argument of Perigee (ω)	270.000°
Mean Anomaly(M)	0.000°

SATELLITE CHARACTERISTICS

TOPEX/Poseidon is a three-axis stabilized satellite (Fig.1) and utilizes nearly continuous yaw steering and solar array pitching for optimal solar array sun pointing. A pitch bias ψ is applied to the solar array to control battery charging, and is changed based on solar-array degradation and observed battery performance. It has changed three times during the first five years of operation; currently ψ=50.5°. There is a plan to set ψ =48.5° in April 1998. The satellite nominally flies with the solar panel in a "Lead" position (ψ>0). The solar panel is said to be in a "Lag" position when ψ<0.

To avoid excessive yaw rates, the satellite yaw angle is held fixed when -15°<β'<15°, where β' is the angle between the orbital plane and earth-sun line. Two different fixed yaw angles Y are used: Y=0° when 0<β'<15° (*flying forward*); and Y=180° when β'<0 (*flying backward*). The satellite is "flipped" (ΔY=180°) near β'=0. This ensures that the sun is kept on the correct side of the solar array, avoids shadowing of the solar array by the high gain antenna, and prevents overheating of satellite subsystems. The satellite is continuously yaw steered for all other values of β'. When β'>15° this is referred to as *positive yaw steering*, and when β'<-15° it is referred to as *negative yaw steering*.

The propulsion module is a mono-propellant hydrazine blow-down system consisting of twelve 1 N (0.2 lbf) and four 22 N (5 lbf) thrusters. The 22 N thrusters and four of the 1 N thrusters are used for orbit adjustment; the remaining 1 N thrusters are used for attitude control when required to dump excess momentum. Nominal attitude control is maintained via reaction wheels which are unloaded with magnetic torquers. The 22 N thrusters were used for large maneuvers (> 400 mm/s) during orbit acquisition.[1] The smaller maneuvers (<400 mm/s) of the orbit acquisition sequence and all orbit maintenance maneuvers, which are <10 mm/s, are performed using two 1 N thrusters. The same pair of thrusters has been used for all nine OMMs. The center of mass (CM) of the satellite does not coincide with the center of body coordinates due to one sided large solar panel. Each

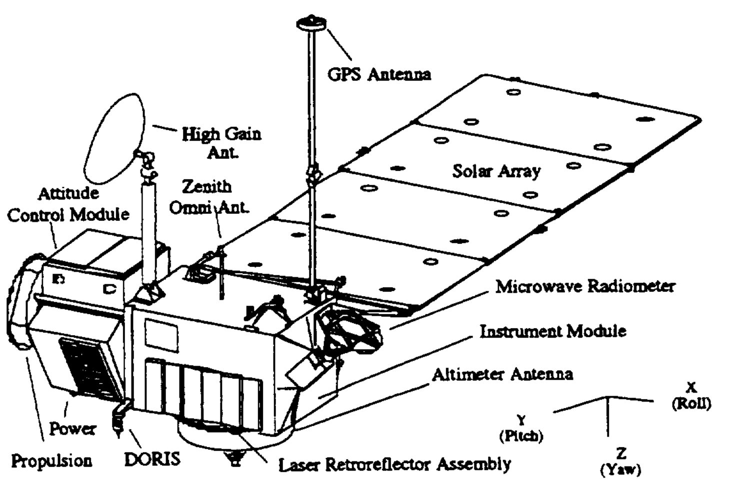

Figure 1. TOPEX/Poseidon satellite.

of the orbit adjust thrusters is oriented axially along the body roll-axis and individually canted to be aligned through the CM prior to the launch when the propellant tanks are full. The propellant tank was fully loaded prior to the launch to provide a total ΔV of ≈ 172 m/s. The orbit acquisition process used only ≈ 11.55 m/s and the nine OMMs have used ≈ 40 mm/s. Thus the satellite is still flying with nearly full propellant tanks after five years of operations.

To correctly orient the thrusters along with the velocity vector for propulsive maneuvers, yaw steering is temporarily suspended and the satellite is slewed to a fixed angle.[1] The yaw turn is accomplished using only reaction wheels. Attitude errors caused by the burn are removed with attitude thrusters. The turn or unwind duration varies depending on the yaw rate and angle. The total duration of a "turn-burn-turn" sequence varies from 20 to 90 min.

ORBIT DETERMINATION

Operational orbit determination is routinely performed by the FDF, primarily using one-way TDRS Doppler data; a small amount of two-way Doppler data is also used.[10] The orbit determination accuracy required for the orbit maintenance and maneuver evaluation was established jointly by JPL and FDF.[11] Modeling consistency between the FDF orbit determination program GTDS (Goddard Trajectory Determination System)[12] and the JPL trajectory program DPTRAJ[13] (double precision trajectory program) was established prior to the launch.[14] The critical requirement on orbit determination is to determine semi-major axis better than 1 m (3σ) throughout the mission.

The FDF supplies orbit determination results three times weekly (Monday, Wednesday, and Friday) and daily near maneuver and fixed yaw periods. The anomalous forces are estimated as an effective thrust $T=1+\tau$ μN as part of routine orbit determination. Onboard oscillator frequency bias and drift rate are also estimated during orbit determination.

Timing and polar motion data tables (UT1-UTC and polar coordinates) are provided by FDF approximately monthly. Variable Mean Area (VMA)[2] models used for atmospheric drag and solar radiation pressure are supplied by JPL to FDF. The VMA model is a function of the solar array pitch bias and is updated whenever the solar array pitch bias is changed. Furthermore, JPL supplies a fixed yaw plan to the FDF prior to any mode changes so that the appropriate models are used for OD. Current solar and geomagnetic activity data are obtained electronically from the National Oceanic and Atmospheric Administration (NOAA) Space Environment Center by both JPL and FDF. NOAA supplies observed and predicted data including a long-term outlook. Changes in other models (e.g., gravity, sun-moon ephemerides, etc.) are performed mutually as required.

OD results[15-21] have been consistently better than pre-launch requirements. The semi-major axis a has been determined to $3\sigma_a \approx 45$ cm (required: $3\sigma_a = 1$ m). Knowledge of a is a function of both OD accuracy and conversion errors in the osculating to mean value conversion process. The osculating to mean value conversion error for a consistently satisfies $3\sigma_a < 40$ cm. Thus the total $\sigma_a < 20$ cm for the mean semi-major axis. Knowledge of other orbital parameters is also much better than the specified pre-launch requirements. These improved OD results have contributed to a reduction in maneuver frequency and more precise ground track determination and control.

ERROR MODEL USED FOR MANEUVER DESIGN

All major error sources are included in the maneuver design process to ensure that 95% of all equatorial crossings are contained in the control band. These include uncertainties of the anomalous force and drag predictions, orbit determination errors, and maneuver execution errors. Drag modeling error is dominated by uncertainties in predicted solar activity. Maneuver execution errors are categorized into fixed, proportional, and pointing errors. Orbit determination error is reflected primarily as an error in semi-major axis.

Solar activity data of previous cycles was used to construct error models for solar flux and geomagnetic index data prediction.[22] High- and low-density trajectories are constructed based on the observed statistical variations over the previous 3 months and the resulting differences in the ground track with the error-free trajectory are used to calculate the drag error. The uncertainty in the anomalous force prediction is modeled from the observed statistical variations about the empirical model. Different uncertainty models are constructed for different yaw modes of the satellite. The ground track prediction error allocated to orbit determination is 225 m (3σ) of equatorial longitude after 30 days, equivalent to an initial semi-major axis error of ≈ 1 m. Maneuver execution error budgets[23] are summarized in Table 2. These error budgets were used for all OMMs. The errors due to drag and anomalous force predictions, orbit determination, and maneuver execution are propagated, converted into ground track units, and then combined to predict a total root sum square (RSS) error envelope in the ground track.[24]

Table 2. MANEUVER ERROR MODEL.*

ΔV (Proportional)	10% for CAL 5% for OMM 1 3% for Subsequent OMMs
ΔV (Fixed)	0.013 mm/s.
Pointing Error (Pitch)	2.0°
Pointing Error (Yaw)	2.0°

*All values are 3σ. CAL: Calibration maneuver. OMM=Orbit Maintenance Maneuver.

Table 3. OMM EVALUATION REQUIREMENTS.*

a	< 1 m
e	$< 5 \times 10^{-6}$
i	0.0001°
Δa	< 20 cm
ΔV Tangential	< 0.2 mm/s
Radial	< 10 mm/sec
Out of plane	< 10 mm/sec

*All values are 3σ. Elements are osculating.

MANEUVER EVALUATION REQUIREMENT

Precise maneuver evaluation is required to calibrate the thrusters so as to reduce the effect of ΔV errors on ground track predictions and to enhance maneuver spacing. The maneuver evaluation accuracy requirements were jointly determined by JPL and FDF, as summarized in Table 3.[25] To achieve the required accuracy in maneuver evaluation the FDF performs special ODs before and after a maneuver using a 26×26 gravity field and a four day tracking arc.

MANEUVER DESIGN AND IMPLEMENTATION

The NAVT* continually monitors the ground track and provides a 30-day advance notice of all maneuvers to other mission operations teams, including geographic maneuver location and centroid time. The maneuver centroid time is chosen to allow time for a backup one repeat cycle ($\approx$10 days) later without violating the km control band. Furthermore, maneuvers are not scheduled near a fixed yaw period so that there is sufficient pre- and post-maneuver tracking data (at least 7 days) for orbit determination. This shortens maneuver spacing by one to two repeat cycles from the optimal value. The preliminary maneuver design is done using GTARG (which uses an analytical propagator) to determine maneuver magnitude (ΔV) and its direction.[24] Two maneuver design strategies were developed prior to launch:[1] (a)*longitude targeting*, which practically maximizes maneuver spacing, and (b) *time targeting*, which fixes the maneuver spacing. All maneuvers implemented so far were designed using the *longitude targeting* strategy. To ensure maneuver spacing as large as practical in the presence of various error sources, every maneuver was designed using a 95-percentile confidence envelope about the ground track.

Under low drag conditions ($F_{10.7} < 120$)** the ground track prediction is very sensitive to small variations in ΔV. The uncertainty of the anomalous force causes significant variations in the predicted ground track (and hence the maneuver spacing) under these conditions. To ensure verification site over flight requirements, as well as enhance maneuver spacing, a "shoot-short" strategy[26] is applied in maneuver design. In this strategy the targeted maneuver magnitude is updated based on a detailed sensitivity analysis conducted using both GTARG and DPTRAJ.

* The TOPEX/Poseidon Navigation Team.
** See the footnote on page 2.

The preliminary maneuver design is verified and updated (if needed) with DPTRAJ before generating and delivering the ideal maneuver parameters, consisting of maneuver centroid time and ΔV, to SPAT.† SPAT generates the maneuver commands, which may result in a slightly different maneuver magnitude or centroid time due to thruster pulse quantization and on-board computer (OBC) constraints. These updated values are verified again using DPTRAJ before they are loaded into the satellite OBC. The maneuver is normally designed seven to ten days in advance to provide sufficient time for command preparation and TDRS scheduling. The maneuver magnitude is then "tweaked," whenever needed, 8-24 hours before execution using the latest OD.[26]

Only nine maneuvers have been required during the first five years of operations. These OMMs all occurred within one orbit of the transition between the ≈10 day ground track repeat cycles, and were implemented using a complex "turn-burn-turn" sequence. The geographic location of the maneuver has been selected to accommodate satellite star-tracker field-of-view constraints, thermal constraints, and available TDRS view periods. (Fig. 2). Two maneuvers (OMM6 and OMM9) were performed over water because of these constraints, in conflict with scientific requirements.

MANEUVER PERFORMANCE

The frequency of maneuvers has been significantly lower than expected because of the use of the passive technique, the prevailing low drag, improved OD (compared to requirements) from FDF, better then predicted satellite performance, and precise maneuver evaluation. The maneuver magnitude for all maneuvers was in the range of 2-5 mm/s, except OMM1. The OMM1 ΔV was somewhat higher as it absorbed some of the residual ground track drift following operational orbit acquisition.

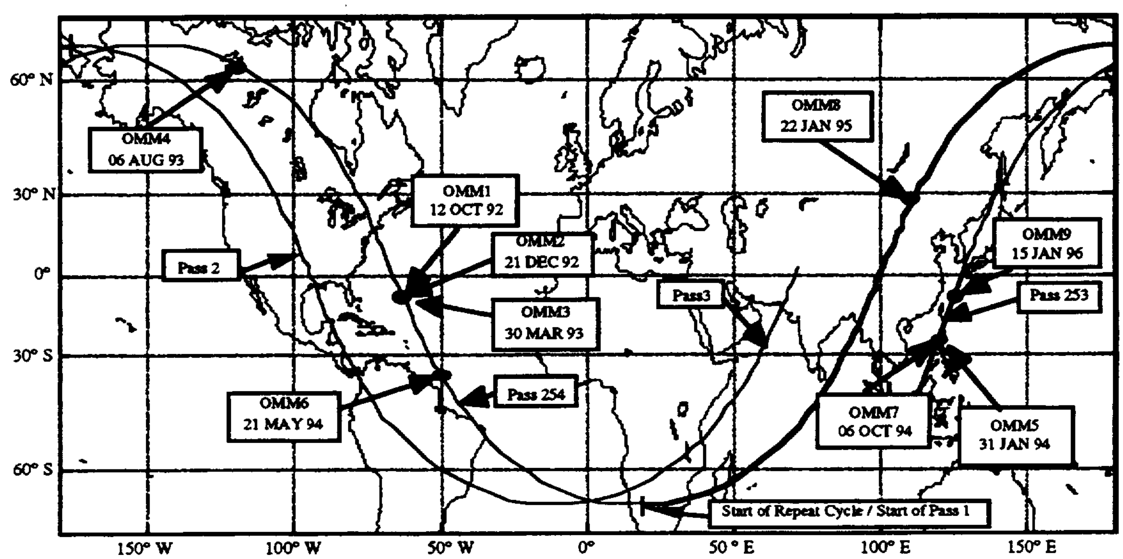

Figure 2. Geographic locations of orbit maintenance maneuvers.

Maneuver evaluation is based upon a comparison of pre- and post-maneuver OD. JPL and FDF each independently evaluate the magnitude of each maneuver using different techniques.[2] Both results have agreed to within 0.03 mm/s for all maneuvers[27] (Table 4). The close agreement between the JPL and FDF results provide greater confidence in maneuver evaluation. The accuracy of maneuver magnitude determination has been better than 0.05 mm/s for all maneuvers based on the analyses of post maneuver orbit determination and the resulting ground track behavior. The performance of all maneuvers except OMM9 was significantly better than the pre-launch expected performances. The performance of OMM1 was ≈3.6% and the thrusters were calibrated using this maneuver. Subsequent maneuvers (except for

† The TOPEX/Poseidon Satellite Performance and Analysis Team.

OMM9) showed better than 3% performance. The resultant ΔV of OMM9 was 46% higher due to unexpected attitude thruster firings during the "unwind" turn (turn after burn) of the satellite.

The presence of the anomalous force significantly altered the orbit determination strategy for maneuver evaluation. Pre- and post-maneuver ODs utilize 4-day tracking arcs. Initially (through OMM5) the thrust parameter representing the anomalous force was estimated along with state parameters. Experience (corroborated by both the JPL and FDF techniques) indicated that a minimum duration 6-day tracking arc is needed to obtain sufficiently accurate estimates of the anomalous force; shorter arcs corrupt estimates of a. To obtain a more reliable ΔV value, the anomalous force parameter is not estimated using short arcs (< 6-day), but instead use an *apriori* value based on the latest prediction model. This strategy has been used for subsequent pre- and post-maneuver orbit determinations, and a similar strategy is used for OD near fixed yaw periods. The maneuver evaluation accuracy improved further with this strategy. Pointing errors were all <1° in both pitch and yaw.[27]

Table 4. MANEUVER PERFORMANCE.

OMM #	Date	ΔV, mm/sec Ideal	ΔV achieved, mm/sec JPL	FDF	Difference, % Achieved-Ideal
1	Oct 12, 92	9.100	9.431	9.425	+3.64
2	Dec 21, 92	3.200	3.153	3.151	-1.47
3	Mar 30, 93	4.640	4.692	4.688	+1.12
4	Aug 6, 93	4.620	4.611	4.611	-0.20
5	Jan 31, 94	4.000	4.089	4.065	+2.25
6	May 20, 94	3.150	3.123	3.128	-0.78
7	Oct 6, 94	3.150	3.146	3.162	-0.13
8	May 22, 95	3.860	3.832	3.832	-0.73
9	Jan 15, 96	2.500	3.652	Not Requested	+46.08

ANOMALOUS FORCE

Analysis of the OD results subsequent to launch indicated the existence of an unmodeled anomalous force.[3] The magnitude of this anomalous force is equivalent to that of a continuous thrust of a few micro-Newtons (μN). This force is believed to arise from a combination of radiative forces (including reflected radiation), solar array curling, thermal imbalances, and outgassing. The direction and magnitude are a function of the satellite attitude, solar array pitch angle offset, and β'. An empirical model (Fig. 3) was developed based on observations of unmodeled along-track accelerations.

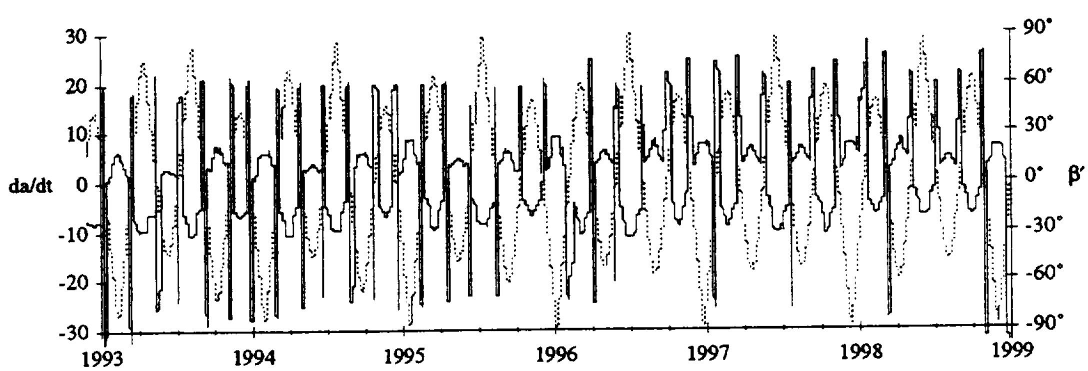

Figure 3. Change of a due to Anomalous Force. Solid line: da/dt, cm/day (left scale); Dotted Line: β′ (right scale).

The anomalous force causes a decay during positive yaw steering and when *flying backward* at fixed 180° yaw mode, and causes a boost during negative yaw steering and when *flying forward* at fixed 0° yaw mode (Fig. 3). The anomalous force results in da/dt≈3-12 cm/day during yaw steering and ≈18-30 cm/day during fixed yaw. The magnitude of the acceleration varies with ψ, β', the earth to sun distance, and thermal variations of the solar panel and some parts of the satellite bus. The uncertainty in anomalous force prediction has been $\sigma \approx 1$ to 3 cm/day during yaw steering and $\sigma \approx 1$ to 4 cm/day during fixed yaw.[4,5] The relative prediction uncertainty $|\sigma/(da/dt)|$ is consistently smaller in fixed yaw than in yaw steering.

PASSIVE TECHNIQUES FOR ORBIT MAINTENANCE

Two passive techniques were developed utilizing the anomalous force during fixed yaw for orbit maintenance.

Fixed Yaw Variation Strategy

Nominally, the length of *flying forward* or *flying backward* is around five days. The orbit may be raised or lowered by varying the nominal duration of fixed yaw periods. The fixed $Y=0°$ period is increased and the fixed $Y=180°$ yaw period is shortened to apply orbital boost and the opposite is done to apply decay. The maximum variation that is allowed is limited by satellite health and safety considerations to require a switch between fixed yaw and yaw steering (or *vice versa*). The current guideline (at $\psi=50.5°$) is that fixed yaw period can be as short as $-13° \leq \beta' \leq 13°$ or as long as $-26° \leq \beta' \leq 27°$. The upper limit of a fixed yaw period varies with time of year and solar array degradation. The yaw flip ($\Delta Y=180°$) must be performed near $\beta' \approx 0$ during all fixed yaw periods. Even with this constraint the orbit may be raised or lowered up to ≈1.5 m during a typical fixed yaw period. This strategy was been used to selectively adjust the ground track from OMM3 (March 1993) through October 1995. This strategy was also used to avoid a "micro-maneuver" (around June 17, 1993) near the west boundary of the control band.[4]

Solar Array Lead/Lag Strategy

The second passive method makes use of the fact that there is a large decay while *flying backward* and a large boost *flying forward*. The satellite normally flies with solar array in Lead position (pitch bias is positive). A positive pitch bias ("Lead Angle," $\psi>0$) indicates that the solar array normal is ahead of the sun direction. Utilization of negative pitch bias ("Lag," $\psi<0$) reverses the direction of the force and the anomalous force causes boost when the satellite is *flying backward* and decay when *flying forward*. A continuous boost can be obtained by using a "Lag" when *flying backward* and "Lead" when *flying forward* (Fig. 4); or a continuous decay can be obtained by using a Lag angle when *flying forward* and a Lead angle when *flying backward*. This technique is summarized in Table 5. In addition, fixed yaw β' limits are varied to apply extra boost or decay. The orbit may be raised or lowered up to ≈4 m, equivalent to propulsive maneuvers of up to ≈2 mm/s, with this technique. This "Lead/Lag" strategy[5] was used for the first time during the October 1995 fixed yaw period to increase the semi-major axis and postpone OMM9 to January 15, 1996. Reversing the solar array orientation (Lead to Lag) for a smaller portion of 180° or 0° yaw part of a fixed yaw period is called a "partial Lead/Lag strategy." The partial Lead/Lag strategy has been used to apply a desired amount of either orbital boost or decay. A partial Lead/Lag strategy was first applied during the March/April 1996 fixed yaw period to increase the inter-maneuver spacing.

Table 5. TERMINOLOGY AND ORIENTATION OF ALONG-TRACK FORCE IN FIXED YAW.

Fixed Yaw Angle	Solar Array "Lead" ($\psi>0$)	"Lag" ($\psi<0$)
0° "Flying Forwards"	$da/dt > 0$ ("boost")	$da/dt < 0$ ("decay")
180° "Flying Backwards"	$da/dt < 0$ ("decay")	$da/dt > 0$ ("boost")

The very first experience during October 1995 demonstrated the power of the Lead/Lag strategy to effectively control ground track. In a fixed yaw variation strategy, the limits of fixed 180° and 0° yaw periods are varied within the maximum allowable β' limits to apply the desired additional orbital boost or decay. These limits need can not be finalized a few days before the beginning of a fixed yaw period because of uncertainties in the anomalous force, violating normal mission planning constraints which require 30 days advance notice. However, with the development of the partial Lead/Lag strategy this problem was eliminated, as the fixed yaw limits can be determined several months in advance. Ground track uncertainty is absorbed by changing the times of Lead to Lag and/or Lag to Lead switch, which can be accomplished by real time commands.

APPLICATION OF PASSIVE MANEUVERS FOR ORBIT MAINTENANCE

The solar array Lead/Lag strategy[5] was used during the October 1995 fixed yaw period to postpone OMM9 until the middle of January 1996. The observed boost level during the Lag period (fixed 180° yaw) was only 75% of the expected boost. This unexplained discrepancy was used to recalibrate the model.

The satellite entered safehold* on November 26, 1995, two days before a fixed yaw period was scheduled to begin. The recovery process took several days and the satellite remained in safehold mode throughout the period during which the fixed yaw angle is normally 0° and 10 hours into the period when the fixed yaw is 180°. The postponement of OMM9 to January 15, 1996 was accomplished by (1) reducing the solar array pitch bias† from 54° to 50.5°, (2) applying Lead/Lag strategy while flying backward during fixed 180° yaw period, and (3) extending the fixed 180° yaw period duration to the maximum allowable value.

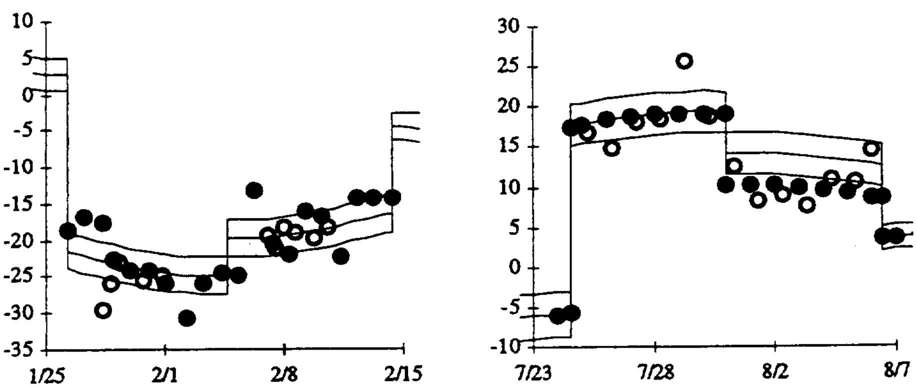

Figure 4. Variation of orbital lead/lag strategy to obtain optimal semi-major axis control. Abscissas give date in 1996; ordinates give da/dt due to the anomalous force in cm/day. Filled Circles: MOE Data; Hollow circles: FDF Data; Left: Orbit lowering when $\beta' > 0$. Right: Orbit raising when $\beta' < 0$. Solid lines: predicted da/dt ±95%.

The resultant ΔV of OMM9 was 46% higher (3.623 mm/s) due to unexpected attitude thruster firing during the "unwind." This resulted in a predicted ground trace[6] that would cross the western boundary of the control band during the last week of February 1996. This situation meant that a retrograde OMM would need to be implemented near the western boundary. This retrograde maneuver was avoided by lowering the orbit using lag during the January/February 1996 fixed yaw period.

As a demonstration for future missions, an autonomous maneuver experiment (TAME) was planned for the summer of 1996 using TOPEX/Poseidon. The NAVT was responsible for targeting the pre-maneuver orbit, including the ground track, so that TAME could occur on a specific date with a specified minimum $\Delta V \geq 1.34$ mm/sec, yet still meet all established ground track requirements and operational constraints. It was decided to achieve the required pre-TAME conditions utilizing Lead/Lag strategies only. The objective was to avoid, if possible, the use of any propulsive maneuvers prior to TAME.[6] The maneuver was designed using the minimum ΔV because of the prevailing low drag conditions. The TAME, originally scheduled for April 6, 1997, was postponed three times due to unexpected technical problems before finally being scheduled for December 19, 1997. However, during last week November 1997, the Project elected to postpone TAME indefinitely to avoid the possible loss of valuable altimeter data related to studying the El Niño conditions.

Currently, the satellite orbit/ground track is maintained using only passive techniques. The 81-day mean solar flux has been steadily increasing since November 1997 and currently the average flux varies

* An autonomous on-board safing mode controlled by analog electronics and triggered by anomaly detection software in the OBC.
† This was done for power reasons; however, the magnitude of the boost in fixed yaw increases with decreasing yaw angle.

between 90 and 145.* It is expected that uncertainties in solar flux prediction will become a significantly larger factor in the ground track prediction accuracy as the solar maximum is approached. Until then it is intended to maintain orbit control using only passive techniques as long as practically feasible. No propulsive maneuver is expected until mid-1999.

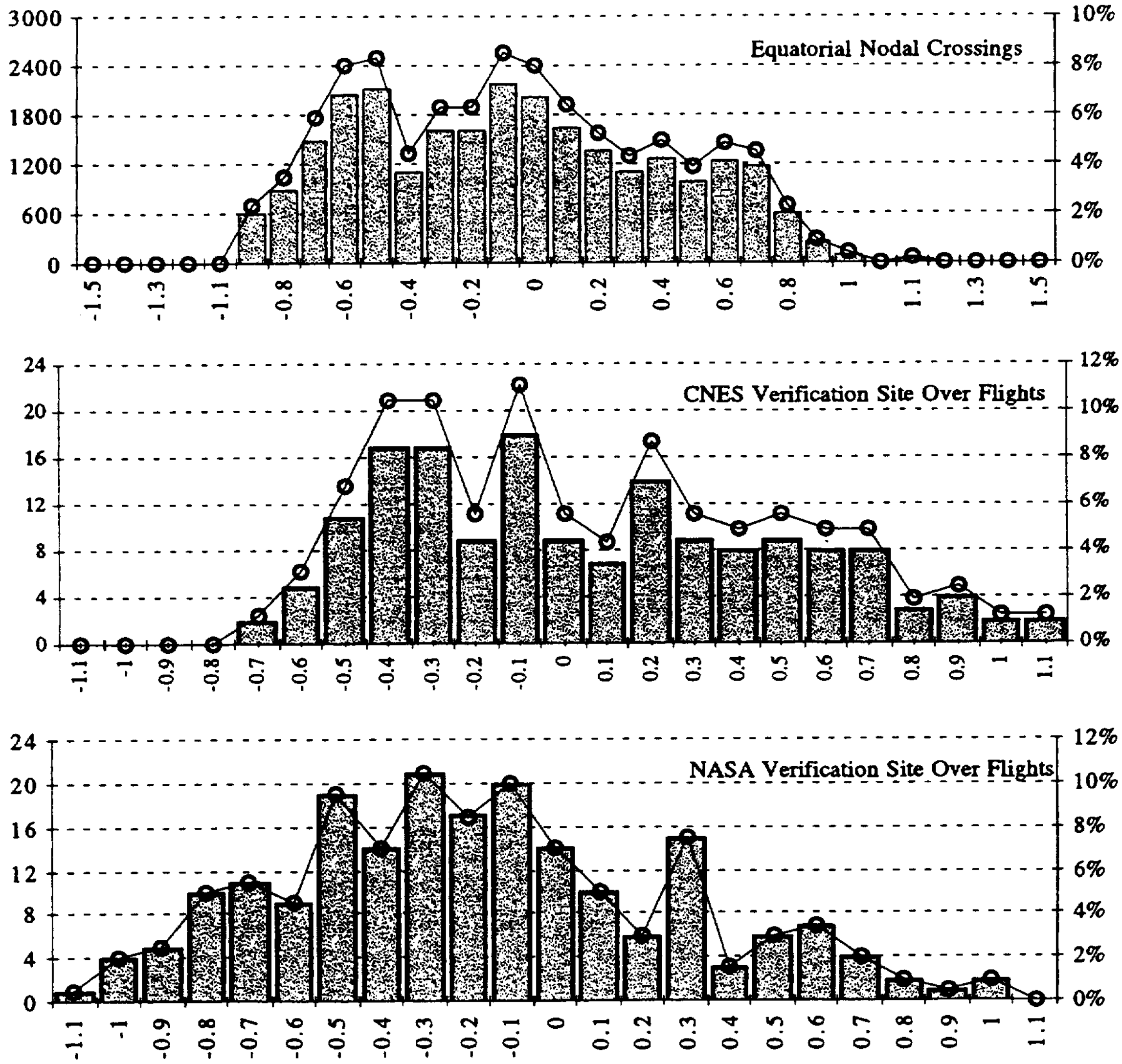

Figure 5. Ground track maintenance statistics. Left ordinates: absolute number of crossings or over flights (bars). Right Ordinate: percentages (lines). Abscissas: ground track in kilometers.

GROUND TRACK MAINTENANCE STATISTICS

As of February 28, 1998 TOPEX/Poseidon had completed 200 ground track repeat cycles in the operational orbit. A total of 99.63% of all equatorial crossings (25,422 crossings) were within the control band of ±1 km, comfortably meeting mission requirements (95% within the control band), even in presence of the anomalous force. Only 95 nodal crossings were outside the control band, and these all occurred at the very beginning of the operational mission. Cycle 1 was defined to begin at the end of the orbit acquisition process, three days before the ground track entered the control band, and the ground track was allowed to move outside the control band before implementing OMM1 to allow more time to develop

* See the footnote on page 2.

an empirical model for the anomalous force. Nearly 70% of the nodal crossings were west of the reference track (Fig. 5).

Fig. 6 shows the ground track history of the satellite. Distinct and important features in the ground track behavior are the periodic variations near the western boundary of the control band. These periodic variations are due to lunar and solar gravity and its influence on the ground track is distinguishable when the semi-major axis is within ±2m of the reference value. However, the precise nature of the variation depends on a complex combination of lunar and solar gravity, anomalous forces, and atmospheric drag. The solar activity has been relatively low ($70 < F_{10.7} < 120$)* during last five years. As a result, the influence of lunar and solar gravity has become more prominent.

VERIFICATION SITE OVER FLIGHTS

The original mission requirement was to maintain the NASA and CNES verification site over flights within ±1 km during the first six months of operations only (the "Initial Verification Phase"), but not later. However, this requirement was extended to continue throughout the mission. The closeness of the ground track to the verification site depends on the nodal crossing longitude and the mean inclination, which varies (±3.5 mdeg) due to lunar and solar gravity. A 1 mdeg variation in mean inclination causes a 70 m ground track offset at either verification site. The verification site over flight control requirement has been taken into account in the design of all orbit maintenance maneuvers. Histograms of verification site over flights are shown in the bottom two plots of Fig. 5. The CNES site was closed on February 1, 1997 and its over flight requirement was discontinued at that time; the NASA site remains in use and its over flight requirement continues to be met.

The control requirements were met for all verification site over flights except five NASA and two CNES site over flights. One NASA over flight miss was voluntary, at the beginning of first ground track repeat cycle. Three involuntary over flight misses occurred during March/April 1996 and one during May 1996. These violations were due to unfavorable inclination variations when the ground track was near the western boundary (within 150 m). Two CNES site over flights were outside the control band: one in September 1996 and the other in January 1997. During this time the effect of lunar and solar gravity was unfavorable on the inclination and the ground track was near the eastern boundary (Fig. 6). However, the mission requirement to keep 95% of all verification site over flights within the control band has been comfortably met for both the NASA and CNES sites.

ORBITAL PARAMETERS

The ground track is maintained by controlling mean semi-major axis about the reference value (7714.429 km) through periodic maneuvers or controlling its variations by passive techniques. While maintaining the ground track and verification site over flights within the ±1 km control band, the mean semi-major axis has been controlled within ±7 m of the reference value through five years of operation[15] (Fig. 7). The mean semi-major axis variations are due to a combination of atmospheric drag and the anomalous force.

The semi-major axis is raised above the reference after each maneuver and slowly decreases due to drag. The semi-major axis decreases rapidly during positive yaw steering as both the anomalous force and drag contribute to decay, whereas the semi-major axis variation during negative yaw steering period is much slower and near zero at times as the anomalous force and drag oppose each other. The semi-major axis varies by a larger amount during fixed yaw than yaw steering, and is four to seven times the effect of atmospheric drag. The semi-major axis variation has been controlled utilizing the solar array Lead/Lag strategy since OMM9 (January 15, 1996) and has stayed within ±3 m of the reference since then (Fig. 7).

The mission requirement to keep the eccentricity within 0.001 has been easily met without implementing dedicated eccentricity maneuvers since achieving the operational orbit. The selection of a frozen orbit assured that the mean eccentricity remained an order of magnitude smaller than the mission requirement. The eccentricity has varied within the range 95±50 PPM* (Fig. 8) throughout the mission. The eccentricity vector subject to only gravitational perturbations would follow a closed loop with a period

* Parts Per Million.

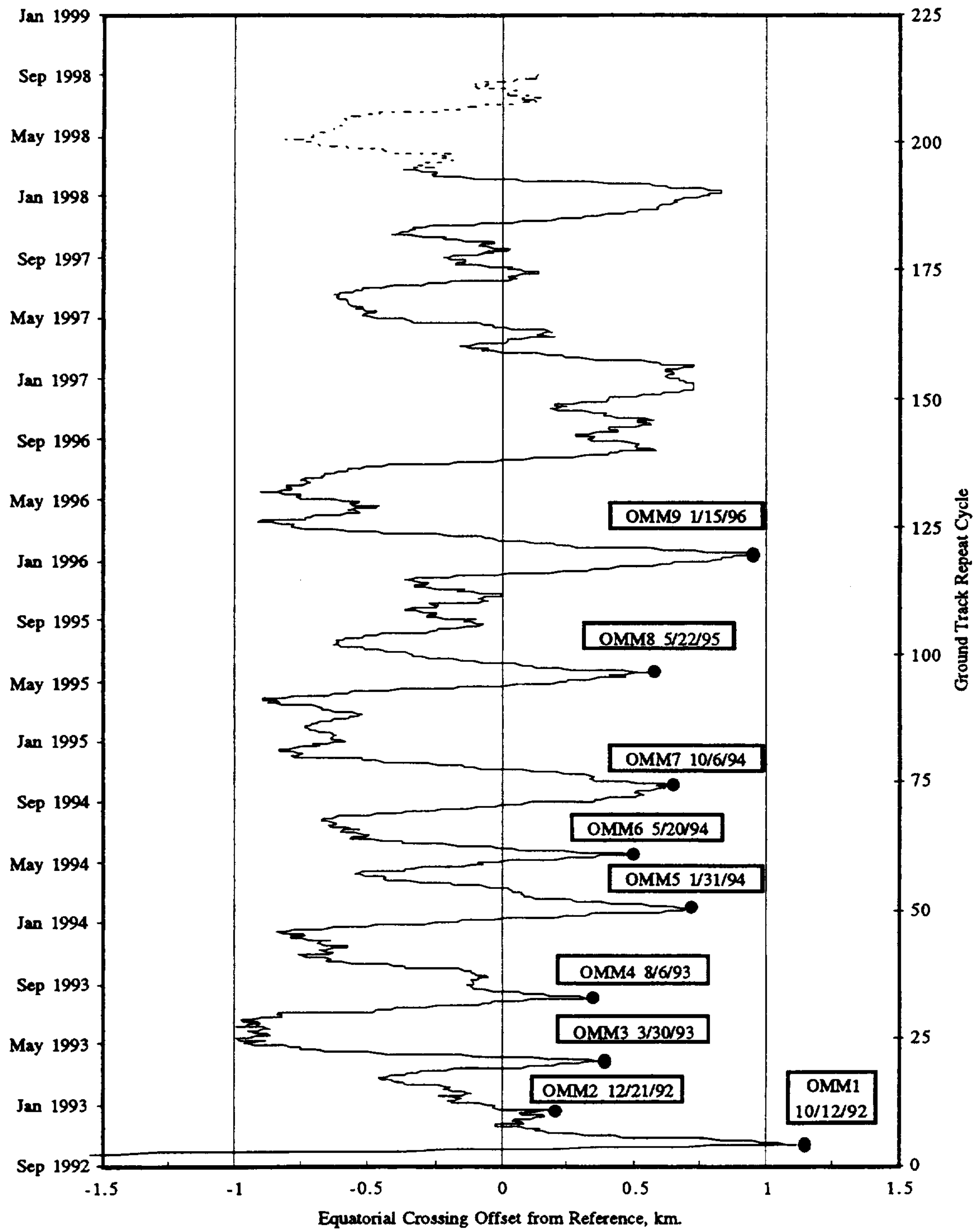

Figure 6. TOPEX/Poseidon ground track. A positive abscissa indicates an offset to the east; a negative abscissa an offset to the west. The dashed segment is the predicted continuation of the ground track at the time of publication.

of ≈26.74 months; it has completed two such loops during the mission and is currently tracing the third loop. The observed eccentricity vector varies from the loop because of (a) solar radiation pressure, drag, and anomalous forces; (b) discontinuous jumps due to propulsive maneuvers; and (c) the inherent uncertainty in the osculating to mean element conversion process and determination of the perigee for a nearly circular orbit. The variation of argument of perigee (ω) has been relatively large, as expected, varying between 48 and 120°. The variation of ω within a single ground track repeat cycle (≈10 days) is as large as 15°.

The inclination remained within a ±4 mdeg band (Fig. 9). No inclination maneuvers have been required. There are several periodic perturbations in i, mostly due to lunar and solar gravity, including one ≈9.5 year component. Inclination variations are strongly correlated with β'. The peak amplitude of the inclination variation synchronizes with the peak values of β' during periods of full sun. The ground track variation near the western boundary is also strongly correlated with the inclination variation.

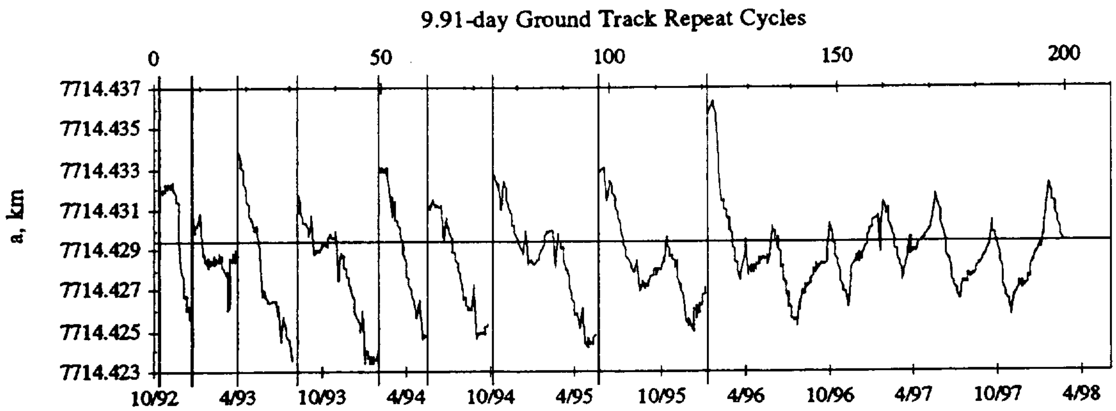

Figure 7. Mean semi-major axis, a. Vertical lines indicate maneuvers (see tab. 4). The horizontal line indicates the reference semi-major axis of 7714.42938 km.

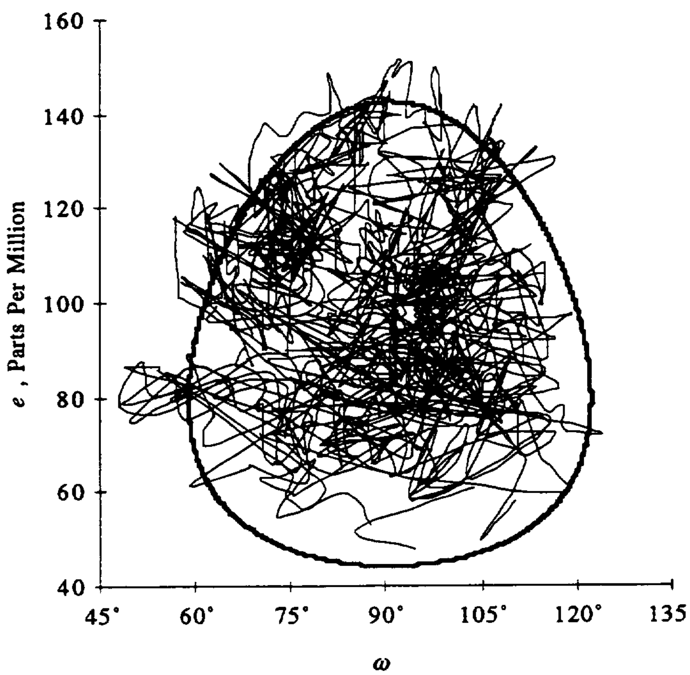

Figure 8. Observed eccentricity vector (e, ω) (thin line) and gravity-only frozen orbit (heavy line).

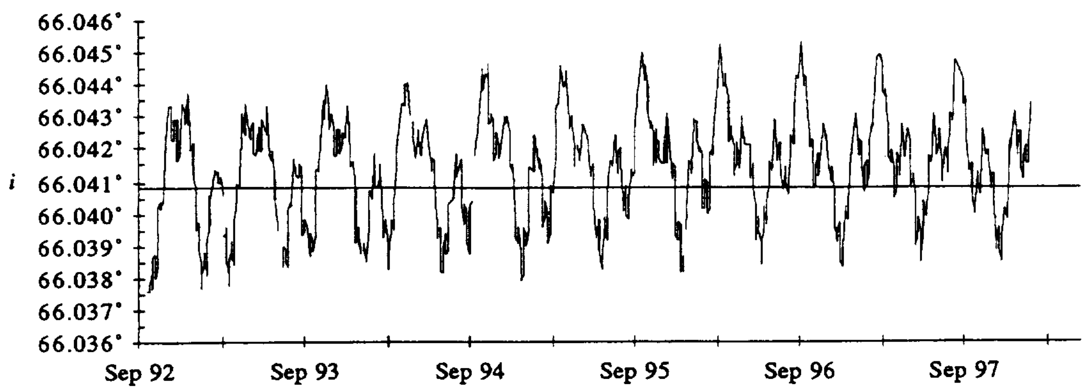

Fig. 9. Observed mean inclination. The indicated reference (horizontal line) is at i=66.0408°.

CONCLUSION

TOPEX/Poseidon orbit maintenance maneuver design, originally expected to depend primarily on effective predictions of atmospheric drag, also depends on reliable predictions of the anomalous force during the current period of low solar activity. These forces constitute the largest uncertainty to ground track prediction and maneuver design. Although the force is continuous, it causes significantly larger orbital boost or decay levels (18-30 cm/day) during fixed yaw periods. This property of the anomalous force has been used to develop a so-called "passive" maneuver technique (the "Lead/Lag" strategy) to effectively control the TOPEX/Poseidon orbit and ground track. This technique can be used to perform "passive" micro-maneuvers that raise or lower the orbit by up to 4 meters, equivalent to propulsive maneuvers of 1-2 mm/s. In fact, it was possible to avoid a retrograde maneuver near the western boundary of the control band during February 1996 using this strategy. It has also been demonstrated that the orbit can be maintained using *only* the passive techniques for a long time (> 2 years) under low drag conditions. Thus these passive techniques have eliminated the need for several propulsive maneuvers for the TOPEX/Poseidon mission.

When propulsive maneuvers were required, their performances surpassed requirements in all areas, and all aspects of satellite performance during maneuvers was excellent. Only nine propulsive maneuvers (in the range of 2-5 mm/sec, except for OMM1) have been required during five years of mission operations because of our use of the passive technique, prevailing low drag, improvements in OD, and precise maneuver evaluation. The total fuel used by all OMMs is equivalent to 40 mm/s. The satellite is using significantly less fuel compared to that expected prior to the launch (40-60 mm/s/year) and fuel tanks remain nearly full.

The TOPEX/Poseidon orbit has been maintained using both passive and active maneuver techniques. All mission requirements have been comfortably met. The semi-major axis exhibits unique variations because of the anomalous force. It increases or decreases depending on satellite attitude. Inclination variations are highly correlated with β'. Selection of a frozen orbit has eliminated the need for dedicated eccentricity maneuvers to keep the eccentricity within 0.001. It is planned to use the passive techniques described above throughout the operational life of the satellite to minimuze the number of propulsive maneuvers.

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REFERENCES

1. Bhat, R. S., Frauenholz, R. B., and Cannell, P. E., "TOPEX/Poseidon Orbit Maintenance Maneuver Design," *Adv. in the Astronautical Sciences*, Vol. 71, part 1, Univelt, Inc., San Diego, 1989, pp. 645-670.

2. Bhat, R. S., Shapioro, B. E., and Frauenholz, R. B., "TOPEX/Poseidon Orbit Acquisition Maneuver Sequence," AAS/AIAA Astrodynamics Specialists Conference, *Adv. in the Astronautical Sciences*, Vol. 85, part 1, Univelt, Inc., San Diego, 1993, pp. 103-112.

3. Frauenholz, R. B., Hamilton, T. W., Shapiro, B. E., and Bhat, R. S., "The Role of Anomalous Satellite-Fixed Accelerations in TOPEX/Poseidon Orbit Maintenance," *Adv. in the Astronautical Sciences*, Vol. 85, part 1, Univelt, Inc., San Diego, 1993, pp. 83-102.

4. Shapiro, B. E., Bhat, R. S. and Frauenholz, R. B. "Using Anomalous Along-Track Forces to Control the TOPEX/Poseidon Ground Track," *Adv. in the Astronautical Sciences*, Vol. 87, part2, Univelt, Inc., San Diego, 1994, pp. 799-812.

5. Bhat, R. S., Shapiro, B. E., and Leavitt, R. K., "Use of Solar Array Lead/Lag Strategy in TOPEX/Poseidon Ground Track Control," Jet Propulsion Laboratory IOM FOS 97 - 018, March 20, 1997(Internal Document).

6. TOPEX/Poseidon Project, "Mission and Systems Requirements," Jet Propulsion Laboratory, JPL D-5901, April 1989 (Internal Document).

7. Shapiro, B. E., "TOPEX/Poseidon Repeat Orbit and Reference Grid Definition using a JGM2 Gravity Field," Jet Propulsion Laboratory IOM FOS 93 - 028, August 5, 1993 (Internal Document).

8. Vincent, M. A., "TOPEX/Poseidon Orbit Characteristics," Jet Propulsion Laboratory, JPL D-7511, June 1990 (Internal Document).

9. Nerem, R. S., et al.,"Gravity Model Development for TOPEX/Poseidon: Joint Gravity Models 1 and 2," *Journal of Geophysical Research*, 99(C12): 24,421-24,447, December 15, 1994.

10. Bolvin, D., "Flight Dynamics Facility Operational Orbit Determination for the Ocean Topography Experiment," GSFC Flight Mechanics and Estimation Theory Symposium, Greenbelt, MD, May 1991

11. Royer, D. A., "Support Instrumentation Requirements Document (SIRD)," Jet Propulsion Laboratory, D-8469, May 22, 1992 (Internal Document).

12. Long, A. C., et al., "Goddard Trajectory Determination System(GTDS): Mathematical Theary," Revision 1, FDD/552-89/001, Code 550, FDF/GSFC, NASA, July 1989.

13. Spier, G. W., "Design and Implementation of Models for the Double Precision Trajectory Program (DPTRAJ)," Jet Propulsion Laboratory, TM 33-451, April 15, 1971.

14. Salama, A., M. Nemesure, J. Guinn, D. Bolvin and R. Leavitt, "Compatibility of TOPEX/Poseidon Trajectory Propagation With JPL and GSFC/FDF Operational Software,"GSFC Flight Mechanics and Estimation Theory Symposium, Greenbelt, MD, 1994.

15. Frauenholz, R. B., Bhat, R. S., Shapiro, B. E., and Leavitt, R. K., "An Analysis of the TOPEX/Poseidon Operational Orbit: Observed Variations and Why", *Journal of Spacecraft znc Rockets*, Mar. - Apr. 1998.

16. Doll, C., G. Mistretta, R. Hart, D, Oza,, C. Cox, M. Nemesure, D. Bolvin, and M. Samii, "Comparison of TOPEX/Poseidon Orbit Determination Solutions Obtained by the Goddard Space Flight Center Flight Dynamics Division and Precision Orbit Determination Teams," *Adv. in the Astronautical Sciences*, Vol. 84, Part 1, Univelt, Inc., San Diego, 1993, pp. 221-236.

17. Doll, C., G. Mistretta, R. Hart, D. Oza, C. Cox, M. Nemesure, and D. Bolvin, "Accuracy Assessment of TDRSS-Based TOPEX/Poseidon Orbit Determination Solutions," *Adv. in the Astronautical Sciences*, Vol. 85, Part 1, Univelt, Inc., San Diego, 1993, pp. 123-142.

18. Doll, C.E., G.D. Mistretta, R.C. Hart, D.H., Oza, D. T. Bolvin, C.M.Cox, M.Nemesure, D.J.Niklewski and M. V. Samii, "Improved Solution Accuracy for TDRSS-Based TOPEX/Poseidon Orbit Determination," GSFC Flight Mechanics and Estimation Theory Symposium, Greenbelt, MD, 1994.

19. Doll, C. E., D. H. Oza, J. M. Lorah and D. T. Bolvin, "Accurate Orbit Determination Strategies for TOPEX/Poseidon Using TDRSS," AIAA 95-3243, AIAA Guidance, Navigation, and Control Conference, Baltimore, MD, Aug. 1995.

20. Doll, C. E., D. H. Oza, D. T. Bolvin, and J. M. Lorah, "TOPEX/Poseidon Orbit Determination Using TDRSS," MS95/059, CNES International Symnposium on Space Dynamics, Toulouse, France, June 1995.

21. Oza, D. H., D. T. Bolvin, C. M. Cox, M. V. Samii, and C. E. Doll, "Accuracy Assessment of TDRSS-Based TOPEXPoseidon Orbit Determination," *Advances in Space Research*, 16(12): 85-88, 1995.

22. Frauenholz, R. B., and Shapiro, B. E., "The Role of Predicted Solar Activity in TOPEX/Poseidon Orbit Maintenance Maneuver Design," AAS 91-515, AAS/AIAA Astrodynamics Specialist Conference, Durango, Colorado, August 19-22, 1991.

23. Bhat, R. S. "Execution Error model for TOPEX/Poseidon Orbit Maintenance Maneuver Design," Jet Propulsion Laboratory, IOM FOS 92-034, October 2, 1992 (Internal Document).

24. Shapiro, B. E., and Bhat, R. S., "GTARG - The TOPEX/Poseidon Ground Track Maintenance Maneuver Targeting Program", AIAA 93 - 1129, AIAA/AHS/ASEE Aerospace Design Conference, Irvine, CA, February 16-19, 1993.

25. "Ocean Topography experiment (TOPEX)/Poseidon Flight Dynamics Facility (FDF)/TOPEX ProjectInterface Control Document" NASA/GSFC 553-FDD-92/028R0UD0, August 1992.

26. Bhat, R. S., "Performance of TOPEX/Poseidon Orbit Maintenance Maneuver-6 (OMM6)," Jet Propulsion Laboratory, IOM FOS 94 - 064, July 8, 1994 (Internal Document).

27. Bhat, R. S., "TOPEX/Poseidon Orbit Maintenance Maneuvers Performance (OMM1-OMM5)," Jet Propulsion Laboratory, IOM FOS 94 -056, April 29, 1994 (Internal Document).