DESCRIPTION OF THE TRAJECTORY FUNCTIONS OF TOPEX/POSEIDON NAVIGATION

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Abstract. TOPEX/POSEIDON is a joint American/French ocean topography experiment. It was launched by an Ariane launch vehicle on August 10, 1992 to study and map ocean circulation. The primary functions of the navigation subsystem of the TOPEX/POSEIDON project are to establish and maintain a pre-designed reference orbit, and to measure, monitor, and predict the satellite ground track continuously. To fulfill these functions, trajectory analysis is required to design and generate all trajectory related products. This paper is concerned with the trajectory functions of TOPEX/POSEIDON navigation. It describes various activities of this support function.

1. Introduction

TOPEX/POSEIDON is a joint American/French ocean topography experiment conducted by the National Aeronautics and Space Administration (NASA) and the Centre National d'Etudes Spatiales (CNES). It was launched by an Ariane 42p launch vehicle on August 10, 1992 to study ocean circulation and its interaction with the atmosphere, to improve our knowledge of climate change and heat transport in the ocean, and to study the marine gravity field. These objectives are accomplished through accurate mapping of the ocean surface with a dual-frequency on-board radar altimeter and precision orbit determination. The satellite is currently in the last year of its three year nominal mission and will then enter a three year extended mission.

Science requirements and constraints on the mission design led to the establishment of a reference orbit for the TOPEX/POSEIDON mission. This orbit gives a fixed grid of sub-satellite ground tracks which is repeated after 127 revolutions in about 10 days.

The primary functions of the navigation subsystem of the TOPEX/POSEIDON project are to establish and maintain this reference operational orbit, and to measure, monitor, and predict the satellite ground track continuously. This is accomplished such that 95% of all equatorial crossings are contained within a 2 km longitude band at each orbit node. To satisfy these navigational requirements, maneuver analysis to design and evaluate the performance of all maneuvers as well as trajectory analysis to fulfill trajectory functions are contained within the navigation subsystem (see Figure 1). This paper is concerned only with the trajectory functions of TOPEX/POSEIDON navigation. It describes various activities of this support. All navigation activities are accomplished by elements at NASA's Jet Propulsion Laboratory (JPL) and Goddard Space Flight Center (GSFC).

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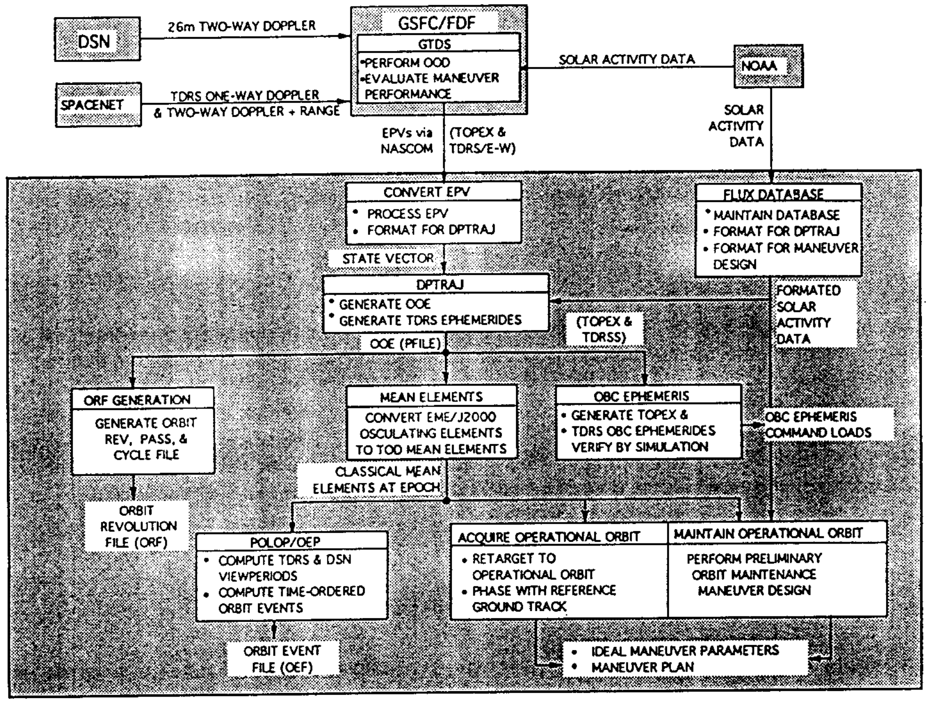

Fig. 1. The navigation subsystem.

2. Trajectory Analysis Support

The following inputs routinely initiate activities of the trajectory support of navigation:

1. State vectors for the TOPEX/POSEIDON satellite and Tracking and Data Relay Satellites (TDRS)
2. Daily solar flux and geomagnetic data (predicted)
3. Timing and polar motion coefficients (predicted)

There are 4 support elements in the TOPEX/POSEIDON trajectory work:

1. Operational Orbit Ephemeris (OOE): generation, validation, and distribution of the OOE.
2. Trajectory Products: generation, validation, and distribution of trajectory products. Examples:
   • On-Board Computer (OBC) ephemeris command loads
   • Orbit Event File (OEF)
   • Orbit Revolution File (ORF)
   • Mean orbital elements with emphasis on the submeter accuracy of the mean semi-major axis

3. Sequence Support: Navigation involvement in review cycle and approval of sequences to ensure that navigation related activities are planned, scheduled, and executed as requested (e.g., OBC ephemeris files, maneuvers, and tracking).
4. Trajectory expertise, consulting, and special requests.

Each of the above elements is discussed below.

3. Operational Orbit Ephemeris

Two independent Orbit Determination (OD) processes are associated with the mission. A Precision Orbit Determination (POD) process which is used to support analysis of the altimeter data, and an Operational Orbit Determination (OOD) process which is used to support the daily satellite operations. This paper is concemed only with the utilization of the OOD solutions in daily operational navigation. The OOD is the responsibility of the GSFC Flight Dynamics Facility (FDF). Using tracking data from the TDRS System, the FDF produces TOPEX/POSEIDON and TDRS state vectors for transmission to the Navigation Team at JPL in NASA's Extended Precision Vector (EPV) message format. These EPV solution sets are transferred to the JPL via National Aeronautics and Space Navigation Administration Communications Network (Nascom) to be used by JPL as initial conditions for propagating the OOE and to conduct day-to-day mission operations. Operational navigation support procedures have been developed to ensure the compatibility of the FDF-estimated TOPEX/POSEIDON and TDRS state vectors with the generated OOEs (Figure 2).

Two independent trajectory software systems are used to perform the above task: the Goddard Trajectory Determination System (GTDS) at the GSFC/FDF and the Double Precision Trajectory System (DPTRAJ) at JPL. GTDS is used for operational tracking and TDRS-based OD. DPTRAJ is used for OOE generation necessary to conduct day-to-day mission operations.

Accuracy requirements on the OOE

The TOPEX/POSEIDON project has imposed several accuracy requirements on TOPEX/POSEIDON operational navigation support. The primary driver behind these requirements is a ± 1 km error tolerance on the equator crossings of the satellite orbit to maintain ground track repeatability. Orbit Maintenance Maneuvers (OMM) used to maintain this ground track must be planned, evaluated, and executed to a commensurate accuracy level based on the OOEs. More specifically, a 30-day OOE must have a 1-σ error of no more than 250 meters in equator crossing location. To ensure this level of accuracy, an error budget was prescribed (Farless 1989) that apportioned the overall error allowance for the OOEs among the identified error sources. The one error source of interest here is the prediction

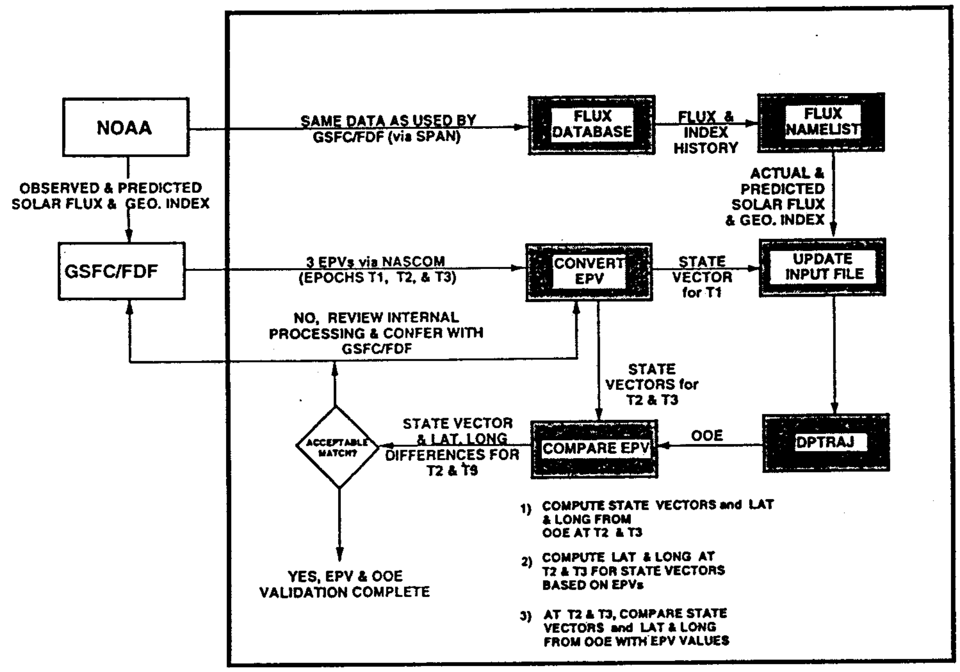

Fig. 2. OOE generation & EPV validation.

modeling errors. The 1-σ value of this error must be no more than 40 meters in equatorial crossing longitude after 30 days propagation (Table I).

PROPAGATION MODELS

The following are the major force models used in DPTRAJ and GTDS for TOPEX/POSEIDON:

- Geopotential Model

The model that had been used during the pre-launch analysis phases was Goddard Earth Model (GEM)-T3. After launch, Joint Gravity Model (JGM)-2, was derived for mission support. JGM-2 models the Earth's geopotential using an expansion of the solution to the Laplace equation, $\nabla^2\Psi(\rho, \phi, \lambda) = 0$, in spherical harmonics with respect to a body-fixed frame up to degree and order 70. A truncated 20 $\times$ 20 version is used for operational navigation after the completion of propagation accuracy studies.

- Luni-Solar Gravity

The gravitational perturbations of the Sun and Moon can be modeled adequately by considering these perturbing bodies as point masses in both systems.

- Solid Earth Tides Model

TABLE I
Predicted ephemeris error budget

Error Component	Equator Crossing Error at 30 days (1 Sigma Random or max. systematic) (m)
Definitive OD (all sources)	75
Prediction error (Nature's unpredictability after the definitive OD interval: density, UTI)	130
Prediction Trajectory Software Modeling Errors	**40**
Maneuver Execution	70
Geopotential Tuning Limitations	10
Total Error (Uncorrelated Errors)	171
Total Error	221
Allowable Error	250
Margin Available (Uncorrelated)	79
(Correlation - 1)	29

The solid Earth Tides model provides an adjustment to the quadrupole term of the geopotential model to compensate for the deformation of the solid portion of the Earth induced by the combined tidal effects of the Sun and the Moon. The model includes a lag angle between the azimuthal component of the position of the disturbing body and the stretching axis. The model also includes a Love number which serves as a proportionality constant for the effect. As implemented in GTDS and DPTRAJ, the model yields an additive adjustment to the gravitational force on the spacecraft.

- Atmospheric Drag

The greatest influence of atmospheric drag on TOPEX/POSEIDON is the orbital decay in terms of semi-major axis reduction. It is modelled as a function of atmospheric density and the velocity of the satellite relative to the atmosphere. Density is a complicated function of solar and geomagnetic activity, satellite geometric parameters, and diurnal, annual, and latitudinal-seasonal variations. Both DPTRAJ and GTDS use the same solar and geomagnetic activity data supplied by the

National Oceanic and Atmospheric Administration (NOAA). The Jacchia-Roberts atmospheric density model is used to calculate air density in both systems.

- Solar Radiation Pressure

The solar radiation pressure (SRP) has effects on TOPEX/POSEIDON that exceed those of atmospheric drag; however, this perturbation can be modeled more easily and accurately. The effect on numerical integration of the extremely rapid changes in the radiation pressure perturbation when the satellite passes through the Earth's shadow has been investigated. GTDS does not restart the integration either upon entry to or exit from its cylinder shadow model. A conical model that allows for no integrator restarts has been implemented in the JPL, DPTRAJ software to match GTDS.

- Variable Mean Area Model

The variable mean area (VMA) model was developed to provide a variable mean spacecraft cross-sectional area for computing perturbations due to atmospheric drag and SRP. The model provides for distinct SRP and atmospheric drag area profiles. Either area profile is driven by a parameter called β', which is the compliment of the angle between the Earth-sun vector and the spacecraft orbital angular momentum vector. Based on nominal attitude control, referred to as "full sinusoidal yaw steering", a table of atmospheric drag and SRP cross-sectional area values at integral values of β' has been developed as an input to GTDS and to DPTRAJ. Area values at intermediate values of β' are obtained through linear interpolation. When the spacecraft is under "fixed-yaw steering" attitude control, the VMA drag area profile is overridden with constant area values.

- Thrusting Effects

Shortly after launch, OD solutions indicated orbital decay levels about 60 times larger than could be explained by atmospheric drag. This level of decay decreased rapidly over the first few days in orbit. Later, orbit trend analysis indicated a presence of body-fixed residual along-track forces comparable to drag which caused either orbital decay or boost depending on the satellite attitude and solar array articulation mode. Consequently, plans with the FDF were made to estimate an along-track thrust parameter τ, instead of the drag multiplier, where the along-track thrust is measured in $(1 + \tau)$ micro Newtons. To ensure the compatibility of thrust modeling between DPTRAJ and GTDS, the navigation team added to the DPTRAJ force model a continuous finite burn with duration equal to the length of the OD arc and force equal to $(1 + \tau)$ micro Newton.

Performance of the OOE

The routine validation of the OOE indicates that it is well within the requirement. Figure 3 shows a one-year comparison between JPL and the FDF after 30 days of

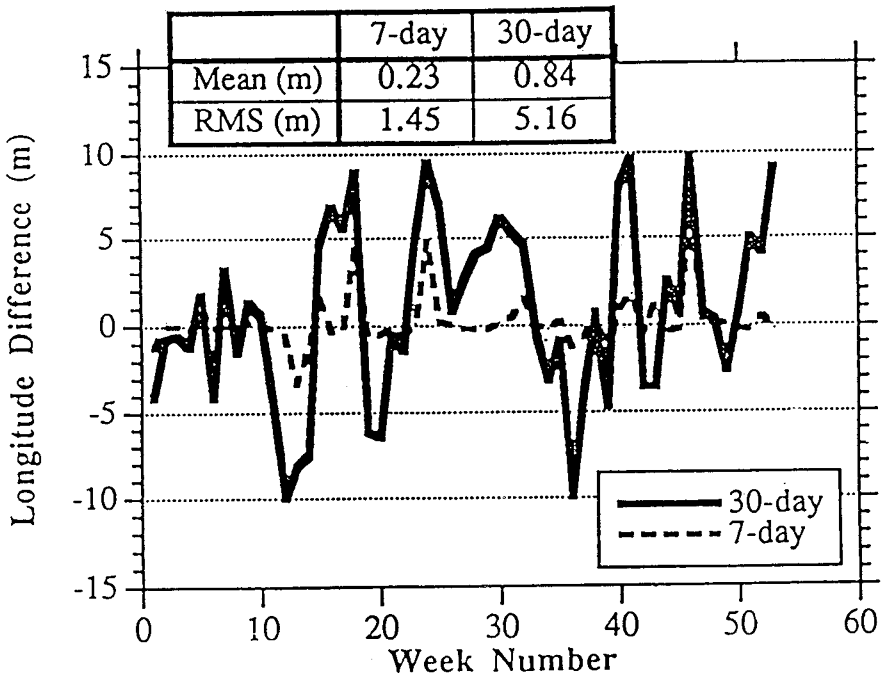

Fig. 3. One-year trajectory comparison results.

orbit propagation. Salama *et al.* 1994 addresses the selection of the force models and the accuracy requirements on the OOE and the JPL-FDF compatibility in detail. In addition to the OOE, TDRS ephemerides are also generated on a routine basis.

4. Trajectory Products

Trajectory products include the OBC ephemeris, OEF, and ORF. The OBC ephemeris is used by the spacecraft to perform its functions of attitude determination and control and pointing, the OEF is used in mission operations to build spacecraft sequences, and the ORF is used to process science data.

OBC EPHEMERIS LOAD

The TOPEX/POSEIDON satellite requires real-time on-board knowledge of the satellite and TDRS ephemerides for attitude determination and control, and for High Gain Antenna (HGA) pointing. The on-board ephemeris representation concept for the MMS (Multimission Modular Spacecraft) satellites has shown that compressing the predicted ephemeris in a Fourier Power Series (FPS) before uplinking, in conjunction with the OBC ephemeris reconstruction algorithms, is an efficient technique for ephemeris representation. As an MMS-based satellite, TOPEX/POSEIDON has inherited the LANDSAT ephemeris representation concept which included a daily FPS upload. During the design phase of this mission, this concept was modified by extending the ephemeris representation duration to 10 days, allowing a convenient weekly uploading without an increase in OBC memory requirements (Salama 1990). The following describes the success of TOPEX/POSEIDON, on-board ephemeris representation modified concept in achieving mission requirements. The following modeling design assumptions have been adapted:

(1) A 42-coefficient FPS is used for each of the six Cartesian state vector components (the coefficients are estimated using the least-squares method).
(2) The time span of the accurate OOE used on the ground to develop the FPS is at least 10 days and the uplinked FPS is valid for the same time span.
(3) To optimize the performance of the ephemeris representation, a grid spacing of 10 min has been chosen for the least-squares fit; the OBC recovers the ephemeris at these 10-min grid points.
(4) The residuals of the fit are computed and uploaded to the OBC for a specific 30-hour span giving increased accuracy over this limited portion of the 10 day span.
(5) Two frequencies are included in the FPS, the satellite mean orbital frequency and the earth sidereal frequency.
(6) The satellite mean orbital frequency is computed from the mean semi-major axis which is obtained by suitably averaging the osculating semi-major axis history defined by the OOE.
(7) A four-point Hermite interpolation formula is used by the OBC to compute the position and the velocity of the satellite at any request time based on the grid-point values recovered by the OBC from the FPS data.
(8) A convenient weekly uploading is adopted in routine operations, allowing 3 days for recovery if a problem develops in the up-link process.

Figure 4 shows the functional design of OBC Ephemeris Command Loads in which the interpolated ephemeris files produced from the OOE (for TOPEX/POSEIDON) and the two TDRS ephemeris files are used as input to the FPS program. The FPS program implements the truncated least-squares FPS algorithm and produces, among other data, the Fourier coefficients and a 30-hour residual span.

The altimeter electrical axis is aligned to the spacecraft z-axis which is pointed to the local geodetic nadir. Farless (1989) indicates that the overall pointing error requirement (half-cone angle) is 0.07 degrees (1-σ). Portions of this overall error have been allocated to the OOE generation and OBC ephemeris representation processes. A 0.015 degrees (1-σ) pointing error is the amount allocated to errors in ephemeris prediction over the 7-day prediction period due to operational OD and OOE generation process. A 0.022 degrees (1-σ) pointing error is allocated to FPS representation and computation and limitation of the OBC over the 10

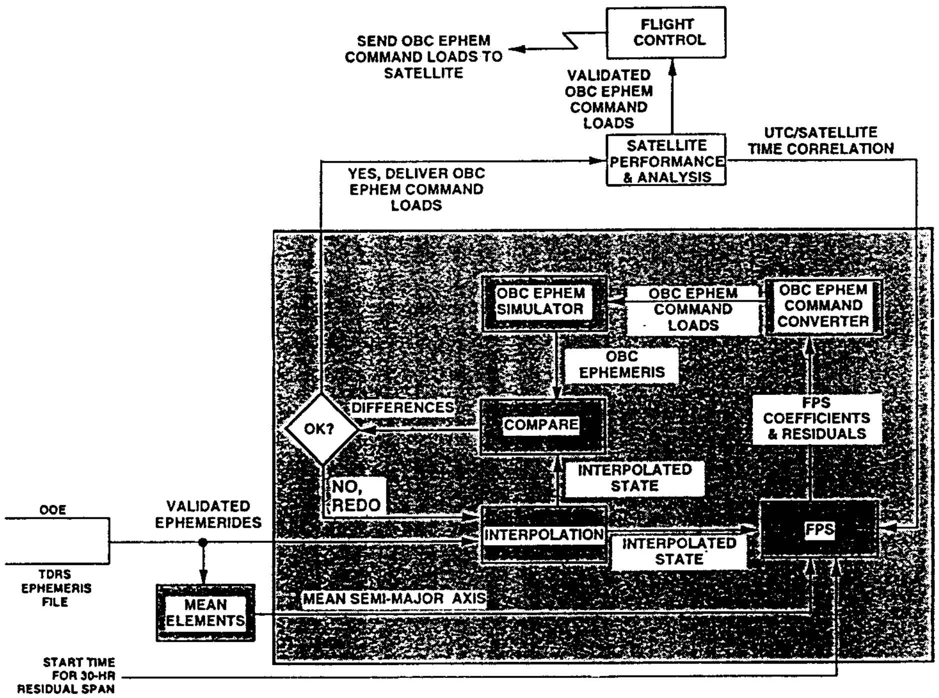

Fig. 4. OBC ephemeris generation.

days representation span. This indicates that the 1-σ along-track prediction error in the OOE after 7 days should be less than about 2 km, and the corresponding OBC representation error should be less than about 2.9 km within 10 days. To ensure that the command loads, as used by the OBC, meet mission requirements, extensive checking and review is conducted before uplinking. A software simulator (Leavitt *et al.* 1993) emulating the limitations and performance of the OBC is used to reconstruct the ephemeris from the command loads. Both the along-track position and nadir-pointing errors are well within the requirements. The OOE is also validated by comparing the state vectors extracted from it with the FDF EPVs at specified epochs. The design and performance of the on-board ephemeris can be found in more detail in Salama *et al.* (1993).

The Orbit Event File

The Orbit Event File (OEF) consists of a set of time-ordered events used in mission planning. It is generated weekly from the most recent mean elements. Two programs, POLOP (Planetary Observer Long Term Orbit Predictor) and OEP (Orbit Event Program) are used to generate the OEF. POLOP propagates the initial set of mean orbital elements using mean-averaged equations. POLOP does not model short period motion but does include gravity spherical harmonics, atmospher-

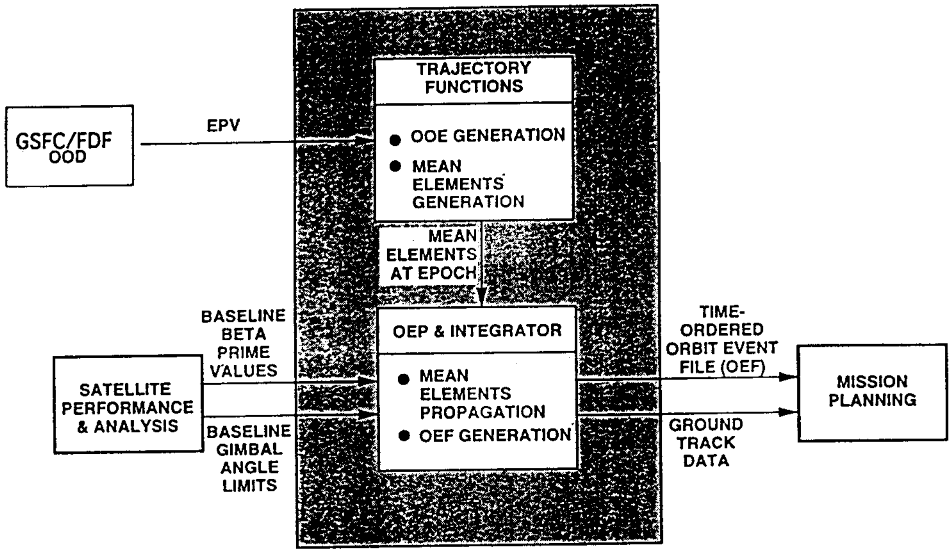

Fig. 5. OEF data flow and interface.

ic drag, solar radiation pressure, and lunisolar third body effects. OEP reads the POLOP generated ephemeris and searches for various trajectory- or attitude-related geometries (See Figure 5).

The events calculated by OEP and included in the OEF are:

Orbit Revolution Events: Times of ascending node crossings, ascending and descending pass events, and orbit cycle events.

Land/Sea Crossings: Land/sea events are determined by the spacecraft nadir point crossing an Earth land/sea boundary. The current land map data have a resolution of 0.2 degrees in latitude and longitude. These times are used to schedule altimeter calibration periods and maneuver times. Calibration and the infrequent orbit trim maneuver must take place over land.

Verification Site Closest Approach: Two verification sites have been chosen for the satellite: Harvest Platform near Point Conception, California and Lampione Rock near Lampedusa Island in the Mediterranean Sea. The closest approach occurs when the range rate of the satellite nadir point with respect to the ground site is zero.

DSN Rise and Set Times: These events occur when the satellite rises or sets with respect to three NASA's Deep Space Network (DSN) ground stations. Horizon masks, which model the local terrain relief and the DSN station mechanical constraints, are defined by elevation angle as a function of azimuth angle.

TDRSS Rise and Set Times: Two TDRSS location may be input to the OEP. Rise and set times of the TDRSS are defined by the HGA field of view constraints, HGA angular rate limits, and satellite obstructions. A satellite centered zenith-oriented cone angle is used to specify the geometric line-of-site rise and set times. The high gain antenna field of view is limited by mechanical or software stops in the antenna gimbal angles. Additionally, the solar panel may impinge on the high gain antenna field-of-view. A mask in gimbal angle alpha/beta space is included in the OEP.

The OEP also calculates the times when tracking of a given TDRS shades the HGA gimbal mechanism. This is used to preferentially select a given TDRS during full sun periods, because shading of the gimbals is desired to control their temperature.

Solar Occultation Events: The occultation of the sun by the earth as viewed from the satellite defines this event. The sun is treated as a point source. Near the times of earth-based solar eclipses, the OEP also calculates the times of occultation of the sun by the moon as viewed from the satellite.

Solar Interference: Solar interference with communication links occurs when the satellite is within 3.5 degrees of the sun when viewed from a TDRS or when a TDRS satellite is within 1 degree of the sun as viewed from the White Sands Ground Terminal.

Orbit Sun Time: Four orbit sun times are written to the OEF for each orbit revolution: orbit noon, 6PM, midnight, and 6AM.

Beta-Prime Events: A set of β' values are used by mission operations to determine transition times between regions of fixed yaw and yaw steering of the satellite and the time of a 180 degree yaw flip maneuver, usually performed at a β' value of 0 degrees.

Groundtrack Zone Events: These events are triggered when the satellite enters or exits a user defined area on the surface of the Earth, usually a radio frequency interference zone or the South Atlantic Anomaly zone of the radiation belts. Groundtrack zones are defined as polygons in latitude/longitude space.

The computation and application of the events of the OEF can be found in more detail in Spencer *et al.* 1993.

THE ORBIT REVOLUTION FILE

The Orbit Revolution File (ORF) is produced from the OOE file for the Science Date Subsystem (SDS), Centre National d'Etudes Spatiales (CNES), and Wallops Flight Facility (WFF). There are two ORF products: the weekly file and the master accumulative master data file. The weekly file spans the next 7-days starting at the end of the current definitive OD tracking data arc, and the master file is an accumulation of all the weekly files concatenated together. Additional files are produced following each Operational Maintenance Maneuver to replace the ORF data from the maneuver to the end of the corresponding 7-day period. The recipients of the 7-day files are CNES and WFF, whereas SDS receives the master file. Both files are produced and delivered weekly after generating the OOE file and, additionally, immediately following a maneuver.

Identified in these products are the epochs of the satellite's orbit cycle, pass, and revolution numbers in Universal Coordinated Time (UTC) and their associ-

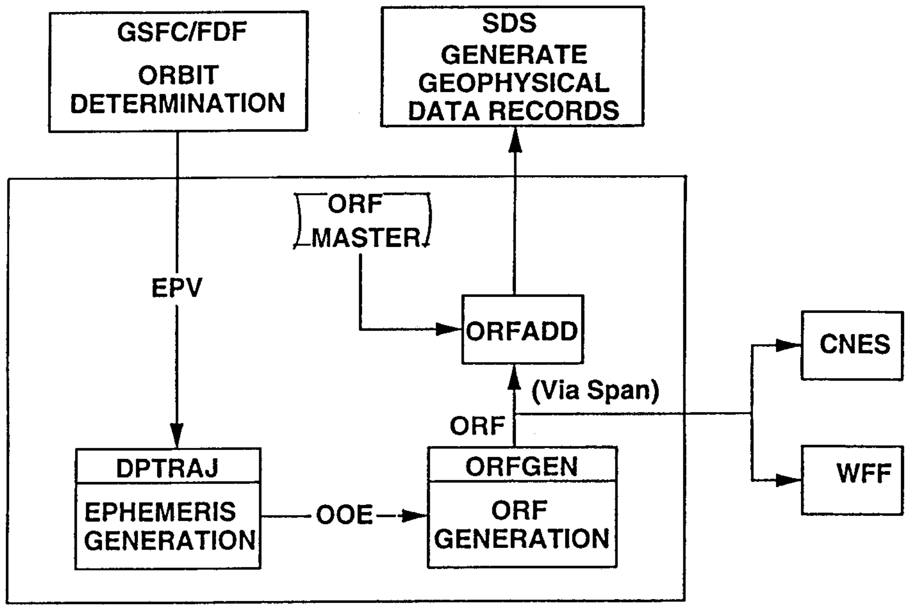

Fig. 6. ORF generation.

ated ground track values of geodetic latitude and longitude. Orbit revolutions are numbered sequentially for the life time of the Mission. Revolution 1 begins with the first northward crossing of the equator after injection. Passes are defined as half a revolution bounded between maximal ground track latitudes and are numbered from 1 to 254, a complete cycle. Ascending passes (from min to max latitudes) have odd numbers, and descending have even numbers. A cycle encompasses the 127 revolutions required before the ground track repeats itself. The first revolution of a cycle starts with the ascending pass that first crosses the equator at the reference longitude of 99.947 deg E. Cycles are numbered sequentially from zero. The zero cycle, which is incomplete, is comprised of those odd lot revolutions that occur after the operational orbit is achieved but before the start of the first complete cycle. Figure (6) is a system flow chart for generating ORF files. Software module ORFGEN produces the 7-day ORF from the OOE file generated by DPTRAJ. Module ORFADD, in turn, updates the Master ORF to include this 7-day ORF. SDS accesses the Master ORF directly from NAVS, whereas CNES and WFF use an electronic network interface.

Mean orbital elements

The accuracy of the ground track repeatability requires high accuracy in TOPEX/POSEIDON mean orbital elements. The key parameter is the mean semimajor axis which must be accurate to the submeter level. An error in this parameter will result in an error in along-track position and consequently in equator crossing

ing location. The total 1-σ error of 250 m in equator crossing location drives the accuracy of the software models and the computed orbital elements. A sufficiently accurate value of mean semi-major axis is obtained with the 20×20 truncation of the JGM-2 geopotential model. Further discussion of the calculation of the mean orbital elements can be found in Guinn 1991.

SEQUENCE SUPPORT

The OEF is used by mission planners to generate a Sequence Of Events (SOF) which is a listing of the stored and real-time commands to be uplinked to the satellite and for scheduling TDRS and DSN activities. Navigation related products such as the on-board ephemeris, maneuver scheduling, and TDRS tracking pass selection are routinely reviewed to ensure that they are implemented as requested.

TRAJECTORY EXPERTISE, CONSULTING, AND SPECIAL REQUESTS

This is an on-going process to generate, validate, and deliver non-routine trajectory products, maintain quality assurance of all trajectory data, and provide expert trajectory consultation to all elements of the project.

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