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Spacecraft Autonomous Navigation for Earth Ground Track Repeat Orbits Using GPS

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This paper describes a simple empirical strategy for computing maneuver times and magnitudes autonomously for Earth orbiting spacecraft with a ground track repeat requirement. Ground track maintenance requires frequent in-plane drag make up maneuvers. Out-of-plane maneuvers are required less frequently, thus are not addressed by this simple approach. Using the Global Positioning System (GPS), tracking and orbit determination functions are consolidated in the GPS receiver to produce real-time position estimates. With these estimates, ground track drift behavior is examined with a simple empirical model to deduce required maneuver times and magnitudes. Thus, this technique does not require the conventional tools of orbit determination (i.e., numerical integrator for state/state partial propagation and Kalman filter for observation noise filtering). The simplified empirical model also reduces the complexity of the orbit propagation/prediction task required by the maneuver decision and design functions.

INTRODUCTION

Simplifying and automating routine spacecraft operations is increasingly desirable for reducing space science mission costs. This paper presents a simplified approach to autonomously navigate Earth orbiting spacecraft flying in fixed ground track repeat orbits. Examples of current and proposed missions are: TOPEX/Poseidon ocean topography

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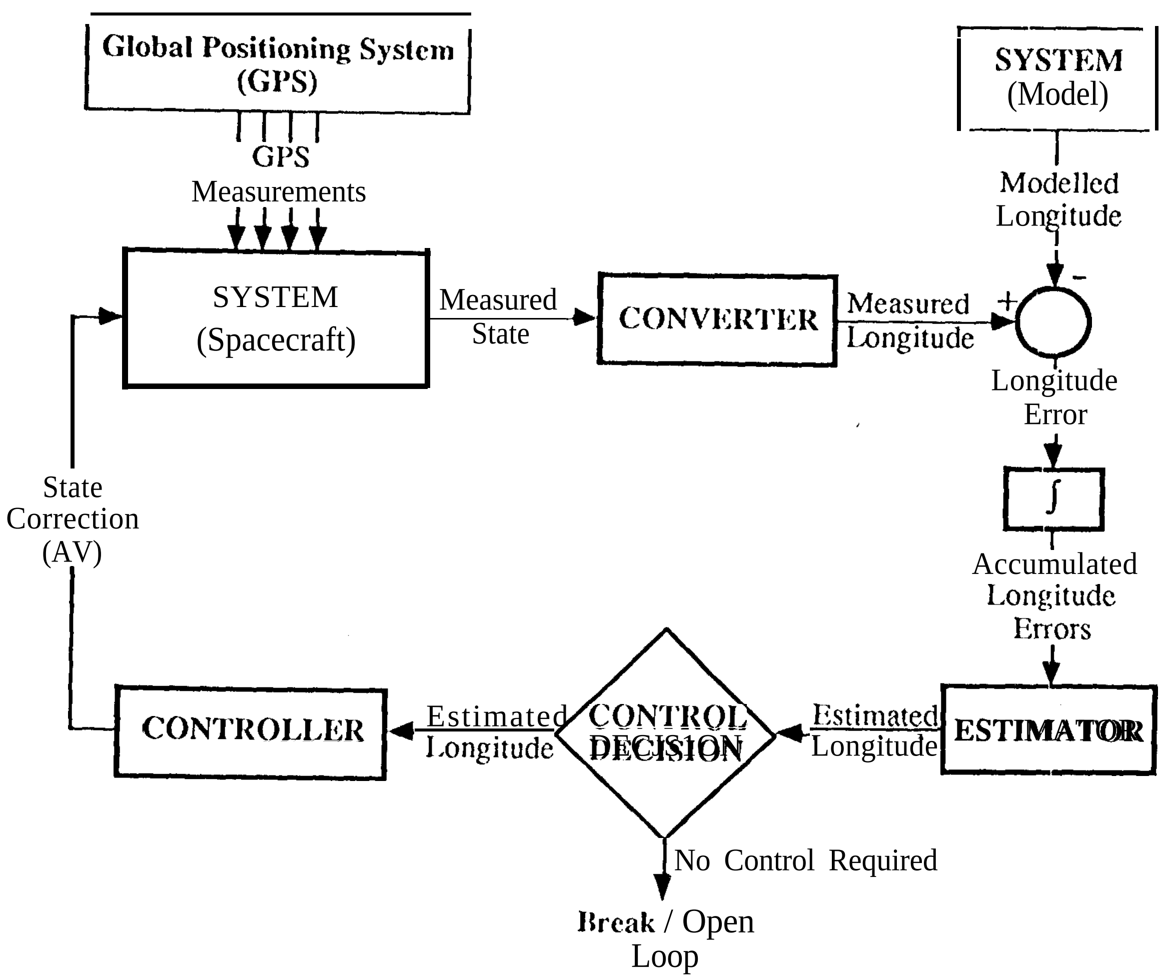
mission [1], LANDSAT land imaging missions [2], and Synthetic Aperture Radar (SAR) land imaging missions [3]. Concepts presented in this paper can be extended to non-ground track repeat and formation flying orbits. However, for clarity this paper is limited to ground track repeat orbits with other orbit profiles planned for future publication.

in this paper, we define navigation as: the process of determining and controlling the trajectory of a spacecraft. Elements of our autonomous navigation system include (Fig, 1): Global Positioning System (GPS) tracking and orbit determination, maneuver decision, maneuver design, and maneuver implementation. This paper proposes a strategy encompassing all elements except for the constraint checking in the maneuver design and the maneuver implementation function.

Fig. 1- System Flow Diagram



Tracking and orbit determination functions are described first, followed by the maneuver decision and design strategy. Next, a brief outline of the procedure for computing the maneuver parameters is presented followed by preliminary results of fitting existing TOPEX/Poseidon GPS flight data. Finally, an appendix is provided describing detailed mathematical derivations for maneuver decision and design,

TRACKING SYSTEM AND ORBIT DETERMINATION

Tracking and orbit determination in real time have been simplified with the availability of GPS. Validation of several GPS spaceborne orbit determination experiments are complete [4-8]. This over determined system of simultaneous observations yields three dimensional position estimates with sufficient accuracy to monitor mission trajectory performance for operational activities.

For this simplified autonomous navigation strategy, only the spacecraft position is required; thus, the minimum set of observations consists of four simultaneous GPS pseudorange measurements and the GPS space vehicle ephemerides. Point position solutions (navigation solutions) are obtained with a least squares adjustment of the observations to produce estimates of the spacecraft Earth-Fixed cartesian position and GPS receiver clock offset.

GPS provides two levels of service: a Standard Positioning Service (SPS), available to all users on a continuous, worldwide basis with no direct charge and an encoded Precise Positioning Service (PPS) intended primarily for military use. SPS is intentionally degraded with a process called Selective Availability (SA) and has an advertised positioning accuracy of 100 meters horizontal and 140 meters vertical (95 percent probability) [9]. For TOPEX/Poseidon, three-dimensional accuracy of the navigation solutions using SPS is 60 to 70 meters RMS (1 σ) over 1 day sampled every 10 seconds (Fig.2).

An important aspect of the simplified maneuver decision and design functions is to use the ground track equator crossing offsets for maneuver parameter determination. Equatorial crossing offsets between the GPS navigation solutions and a near truth Precision Orbit Ephemeris (POE) [1 O] compare to approximately 50 meters RMS over 30 days sampled every five minutes (Fig.3).

The current TOPEX/Poseidon GPS flight receiver software appears to be responsible for an approximately 40 meter bias in the longitude of the ascending node. This bias manifests itself as a 16 meter bias in the along track and 38 meter bias in the ascending equator crossings. Removal of this bias results in GPS navigation solution equator crossing accuracies of 20 to 30 meters RMS.

MANEUVER DECISION AND DESIGN

Early investigations of using a simple empirical approach for TOPEX/Poseidon orbit maintenance maneuvers were performed by Kechichian and Cutting [11]. Later, a simple empirical model was identified by S ynnes [12?] to perform maneuver design for the GEOSAT mission. Most recently, as part of analysis of an autonomous navigation system for the TOPEX/Poseidon Follow On mission, elements of a simplified method were introduced by Davis, Vincent, and Boain [13].

For TOPEX/Poseidon, the ground track is maintained to ±1 kilometers about a fixed reference ground track. Fig.4 shows the ground track drift history for the first three years of TOPEX/Poseidon[1 4]. An important feature to note is the quadratic nature of the change in the ground track longitude offsets. Between each maneuver, a quadratic (2nd degree polynomial) fit to the longitude offsets produces an empirical model that can be used to predict the next maneuver time and magnitude.

The dominant error source in this approach is from the misfit of the quadratic to the longitude offsets rather than the orbit determination. Fig.5. shows the quadratic fit to the GPS navigation solutions. The large deviations from the quadratic curve are primarily due to luni-solar effects and, to a lesser degree, varying atmospheric drag.

Maneuver times are computed by extrapolating the quadratic and monitoring the west and east boundaries for violations. The maneuver magnitude computation depends on the which boundary is violated. The following section summarizes the computations.

MANEUVER PARAMETER COMPUTATION PROCEDURE

The steps required to determine the maneuver times and magnitudes in near real-time from raw GPS navigation solutions are given as follows:

1) For each GPS navigation solution, compute Earth-Fixed longitude.

2) Linearly interpolate longitudes at ascending equator crossing and difference with respect to fixed reference longitude.

3) Accumulate longitude offsets and fit quadratic (Fig.5).

4) Check for west boundary violation by evaluating:

$$WB \leq m_0 + m_1 t + m_2 t^2 \text{ for } t_{now} \leq t \leq t\text{+(desired look ahead time)}$$

5) Check for east boundary violation by evaluating:

$$EB \leq m_0 + m_1 t \text{ -t } m_2 t^2 \text{ for } t_{now} \leq t \leq t\text{+(desired look ahead time)}$$

6) If west boundary maneuver is required (Fig.6):

$$AV = -\frac{2V}{3\omega_e a_e}\left(m_1 + 2m_2 t_M\right) \qquad \text{(see appendix for derivation)}$$

7) If east boundary maneuver is required (Fig.7):

$$AV = \frac{V}{3\omega_e a_e}\left[2\sqrt{LO(t_M)m_2} + \left(m_1 + 2m_2 t_M\right)\right] \qquad \text{(see appendix for derivation)}$$

PRELIMINARY RESULTS

Currently, the TOPEX/Poseidon GPS flight receiver produces navigation solutions once every 10 seconds. Fig 8. shows the longit udc offsets derived from the GPS navigation solutions since Orbit Maintenance Maneuver No. 8 (OMM8) on 22 May 1995. Based on GPS observations up to 4 January 1996, the next maneuver was predicted for 14 January 199612:55:28 UTC. On that dateaAVof2.516 mm/sec would target the -700 meter west boundary assuming constant drag. For comparison the maneuver parameter designed by the TOPEX/Poseidon navigation team produced an epoch of 15 January 1996 19:41:55 UTC with a AV of 2.5 mm/sec.

CONCLUSIONS

A simplified approach to autonomously computing maneuver parameters for ground track repeat missions appears to be feasible. “l”his approach will provide for faster and cheaper development and operations costs with limited degradation compared to the conventional navigation currently used for TOPEX/Poseidon.

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Appendix - Mathematical Derivations

The effect of an orbit maintenance maneuver (AV in along track direction) is to reduce the total velocity of the spacecraft by increasing the altitude of the orbit. With constant Earth rotation and the increased spacecraft orbital period, the ground track drifts westward. The westward drift is combated by atmospheric drag and eventually the ground track drift reverses as the orbital altitude decreases. This is analogous to simple projectile motion under the influence of gravity. Derivations of the quadratic nature of this problem [15] are repeated here for completeness.

To understand how a change in the along track velocity effects the along track position we start with Kepler's equation:

$$M = E - e\sin E = n(t-t_p) \tag{1}$$

where: M = mean anomaly
E = eccentric anomaly
e = eccentricity
n = mean motion
t = time
t_p = time past periapsis

For circular, orbits and t_p = O equation 1 becomes

$$M = E = nt \tag{2}$$

The along track velocity is:

$$v = \sqrt{\frac{\mu}{a}} \quad \text{circular velocity} \tag{3}$$

where: μ = Earth gravitational constant
a = spacecraft semi-major axis

from equation 3, a change in semi-major axis yields a change in velocity as follows:

$$dV = -\frac{V}{2a}\,da \tag{4}$$

From Kepler's third law:

$$n = \sqrt{\frac{\mu}{a^3}} \text{ mean motion} \tag{5}$$

from equation 5 a change in semi-major axis yields a change in the mean motion as follows:

$$dn = -\frac{3n}{2a}\,da \tag{6}$$

The rate of change of the mean anomaly due to a change in the along track velocity can now be derived from equations 1,4, and 6:

$$AM = (dn)\ t\ =\ 3n\frac{dV}{V}\ t \text{ (no atmospheric drag)} \tag{7}$$

Adding atmospheric drag, equation 7 becomes:

$$AM = (dn)\,t + \frac{\dot{n}}{2}\,t \tag{8}$$

the time derivative of the mean motion is:

$$\dot{n} = -\frac{3n}{2a}\,\dot{a} \tag{9}$$

and for a circular orbit the atmospheric drag effect on semi-major axis is:

$$\dot{a} = \frac{C_D A}{m}\,\frac{\rho V^2}{n} \tag{1o}$$

where: C_D = spacecraft drag coefficient
A = spacecraft projected area in along track direction
m = spacecraft mass
p = atmospheric density

combining equations 7-10 the along track angular change due to an along track velocity change and atmospheric drag is:

$$\Delta M = \left(3n\frac{dV}{V}\right)t + \left(\frac{3}{4}\,\frac{C_D A}{m}\,\frac{\rho V^2}{n}\right)t^2 \tag{11}$$

The along track distance computed from the along track angular change is:

$$ALT = aAM \tag{12}$$

The change in orbital period due to a change in along track distance is:

$$\Delta T = \frac{\Delta LT}{V} \tag{13}$$

Finally, the angular change in equatorial longitude is:

$$\Delta\lambda = \Delta\lambda_0 + \omega_e \Delta T = \Delta\lambda_0 + \omega_e \frac{a\Delta M}{V} \tag{14}$$

substituting equation 11 into 14 gives:

$$\Delta\lambda = \Delta\lambda_0 + \left(\frac{3\omega_e}{V} dV\right) t + \left(\frac{3}{4}\omega_e \frac{C_D A}{m} \rho V\right) t^2 \tag{15}$$

rewriting in terms of semi-major axis rate (i.e., using equation 10)

$$\Delta\lambda = \Delta\lambda_0 + \left(\frac{3\omega_e}{V} dV\right) t - \left(\frac{3\omega_e}{4a} \dot{a}\right) t^2 \tag{16}$$

Next, convert the angular longitude offset to a length measurement as:

$$LO = a_e \Delta\lambda \tag{17}$$

where: a_e = Earth mean equatorial radius

The final expression for the ground track longitude drift is then:

$$LO = LO_0 + \left(\frac{3\omega_e a_e}{V} dV\right) t - \left(\frac{3\omega_e a_e}{4a} \dot{a}\right) t^2 \tag{18}$$

Now, the equation for a quadratic fit to LO is:

$$LO = m_0 + m_1 t + m_2 t^2 \tag{19}$$

therefore, the initial longitude offset, velocity change and semi-major axis rate in terms of the quadratic fit coefficients can be determined from equations 18 and 19 as:

$$LO_0 = m_0 \tag{20}$$

$$dV = \frac{V m_1}{3\omega_e a_e} \tag{21}$$

$$\dot{a} = -\frac{4a m_2}{3\omega_e a_e} \tag{22}$$

For west boundary maneuvers the ground track rate must be exactly reversed or given the same rate as would have been achieved at the quadratic symmetric time t_s (Fig.6), so:

$$\dot{LO}'(t_M) = \dot{LO}(t_s) = m_1 + 2m_2 t_s \tag{23}$$

thus, the change in the ground track drift rate is:

$$\Delta\dot{LO}(t_M) = \dot{LO}(t_s) - \dot{LO}(t_M) = 2m_2(t_s - t_M) \tag{24}$$

The velocity change required to modify the ground track rate as given in equation 24 can be derived from equations 21-23:

$$\dot{LO}(t) = m_1 + 2m_2 t = \frac{3\omega_e a_e}{V} dV + \left(\frac{3\omega_e a_e}{2}\frac{C_D A}{m}\rho V\right) t \tag{25}$$

$$\dot{LO}(t) + \Delta\dot{LO}(t) = \frac{3\omega_e a_e}{V}\left[dV + \Delta V\right] + \left(\frac{3\omega_e a_e}{2}\frac{C_D A}{m}\rho V\right) t \tag{26}$$

reducing equation 26 gives:

$$\Delta\dot{LO}(t) = \frac{3\omega_e a_e}{V}\Delta V \tag{27}$$

$$\Delta V = \frac{V}{3\omega_e a_e}\Delta\dot{LO}(t) \tag{28}$$

combining equations 24 and 28 yields:

$$\Delta V = \frac{V}{3\omega_e a_e} 2m_2(t_s - t_M) \tag{29}$$

where: t_s = symmetric time of quadratic curve = $t_0 + (t_f - t_M)$
t_0 = time of last maneuver
t_M = time of next maneuver
t_f = $-m_1/m_2$

or:

$$\Delta V = -\frac{V}{3\omega_e a_e}(m_1 + 2m_2 t_M) \tag{30}$$

Equation 30 gives the ΔV required to exactly reverse the ground track rate at any point along the quadratic curve. Therefore, west boundary maneuver AV'S should be computed using equation 30. However, for east boundary maneuvers this presents an unstable situation for continuously increasing or decreasing drag. This problem can be corrected by assuming constant drag and targeting to the maximum west boundary.

East boundary maneuver design starts by setting the time derivative of the longitude offset, equation 25, to zero and solving for the time. of the maximum western excursion of the ground track:

$$o \cdot \frac{3\omega_e a_c}{V} dV - \left(\frac{3\omega_e a_c}{2a} \dot{a}\right) t \tag{31}$$

$$t_{MAX} = \frac{dV}{V} \frac{2a}{\dot{a}} \tag{32}$$

Substituting the time of maximum western excursion back into equation 18 gives the maximum west longitude excursion:

$$LO_{MAX} = LO_o + 3\omega_e a_c \frac{a}{\dot{a}} \left(\frac{dV}{V}\right)^2 \tag{33}$$

Now for the east boundary maneuver we assume constant drag but set LO_{MAX} equal to zero longitude offset. This is a compromise since the future drag is unknown. Thus, the velocity change to maximize the westward longitude excussion is obtained by solving the following for dV:

$$0 = LO_0' + 3\omega_e a_c \frac{a}{\dot{a}} \left(\frac{dV}{V}\right)^2 \tag{34}$$

where: $LO_0' = LO(t_M)$

$$dV = V\sqrt{\frac{(LO(t_M))}{3\omega_e a_c} \frac{\dot{a}}{a}} \tag{35}$$

Equation 35 can be written in terms of the quadratic fit coefficients using equation 22.

$$dV = \frac{2V}{3\omega_e a_c} \sqrt{LO(t_M) m_2} \tag{36}$$

Equation 36 assumes a prior ground track drift rate of zero. To account for the prior eastward drift we can use one half of the AV derived in equation 30. This is possible since equation 30 assumes the drift is just reversed; thus, the prior eastward drift is equal to one half of that value.

So, the final expression for the AV for east boundary maneuvers is:

$$AV = \frac{2V}{3\omega_c a_c}\sqrt{LO(t_M)m_2} + \frac{V}{3\omega_c a_c}\left(m_1 + 2m_2 t_M\right) \tag{37}$$

or rewritten as:

$$\Delta V = \frac{V}{3\omega_c a_c}\left[2\sqrt{LO(t_M)m_2} + \left(m_1 + 2m_2 t_M\right)\right] \tag{38}$$

Fig.2 - GPS Navigation Solution Comparisons

Near Truth Source: TOPEX/Poseidon POE

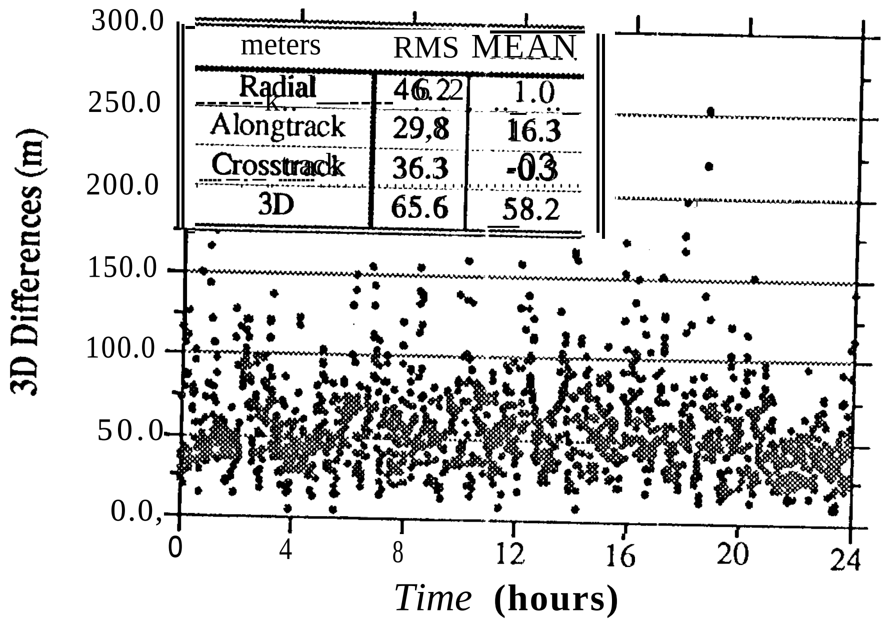

Fig.3 - Equator Crossing Comparisons
POE - GPS Navigation Solutions

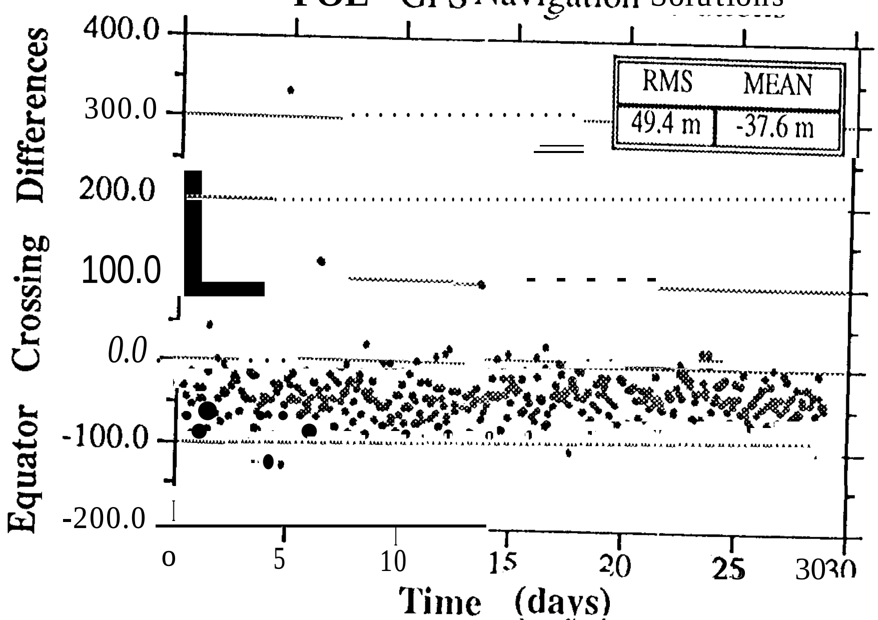

Fig. 4 - Ground Track History
(*Source:* Frauenholz,Bhat and Shapiro [14])

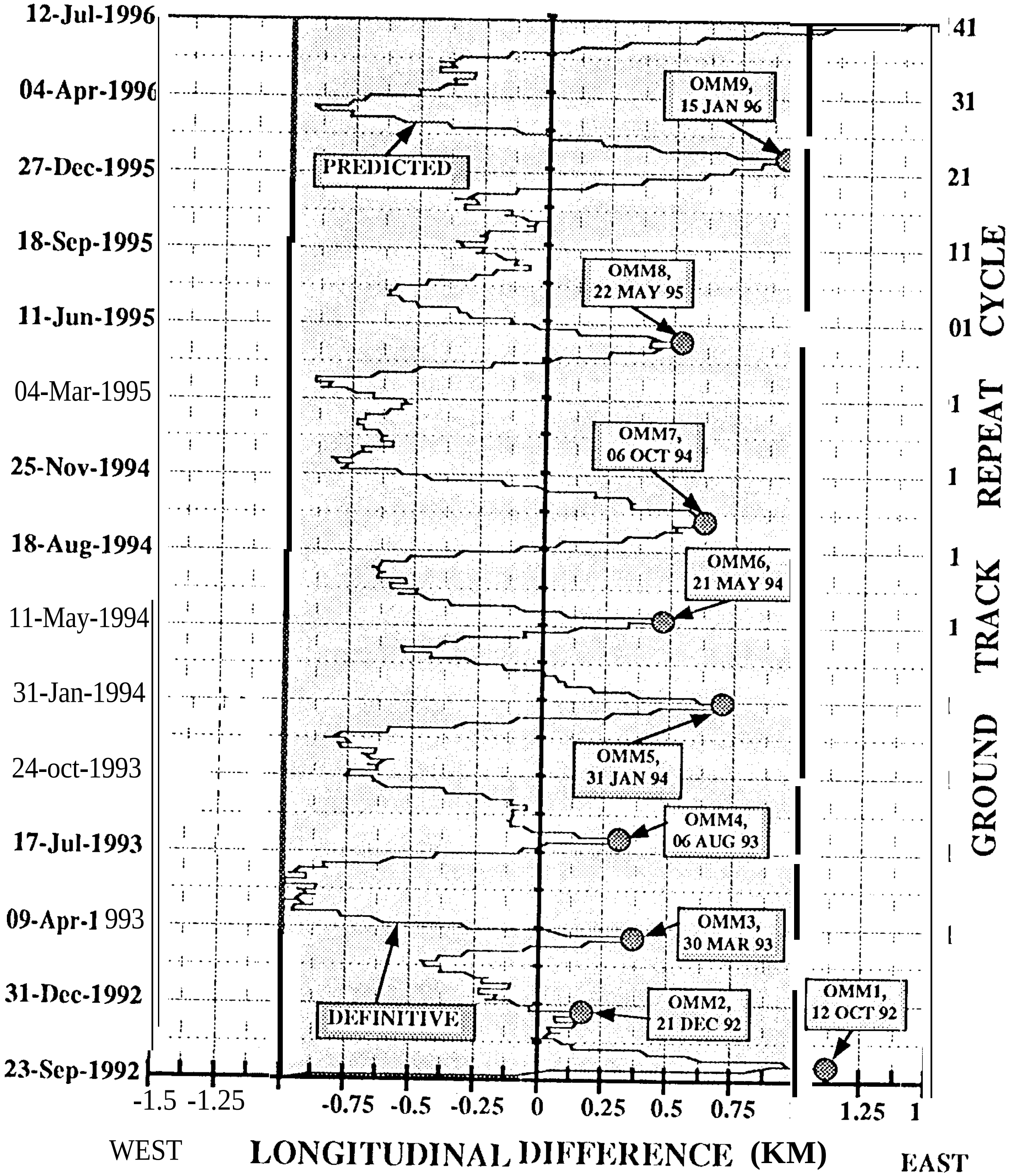

Fig.5 - Quadratic Fit to GPS Navigation Solutions

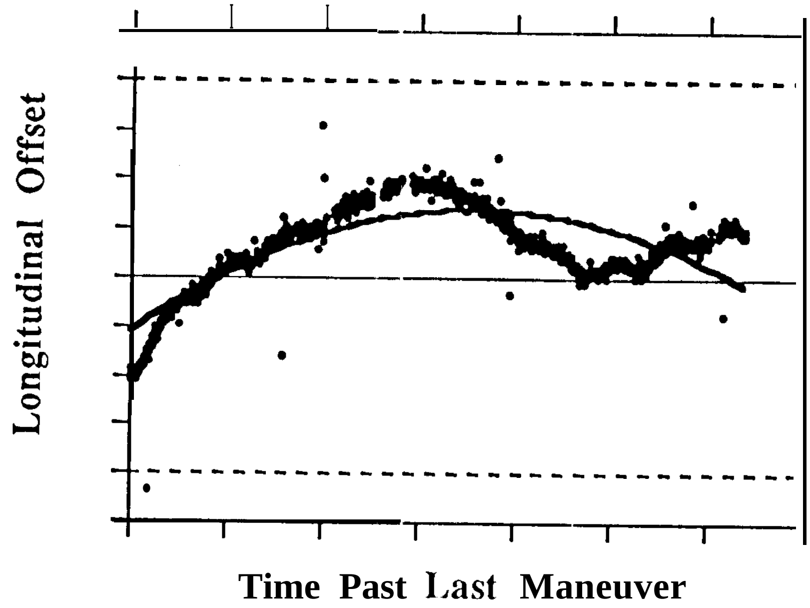

Fig.6 - West Boundary Maneuvers

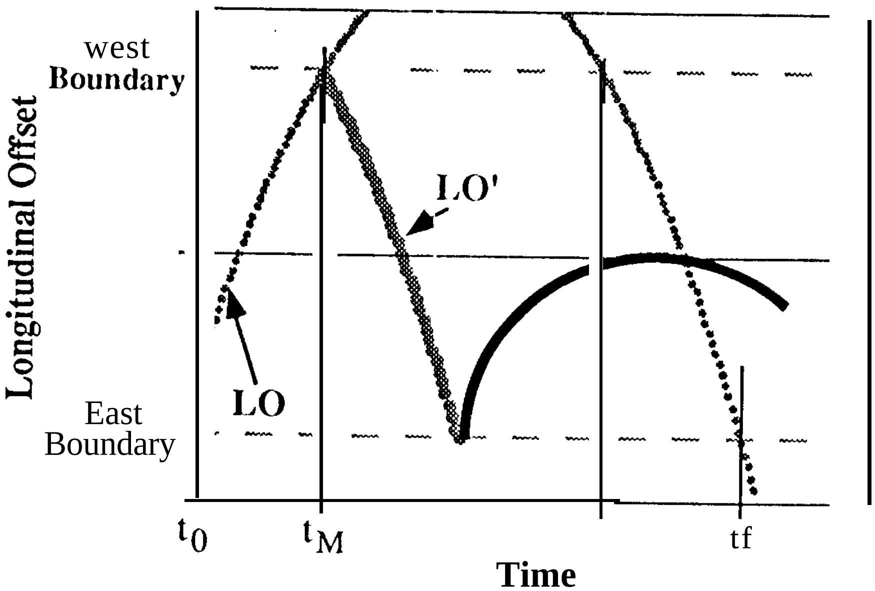

Fig.7 - East Boundary Maneuvers

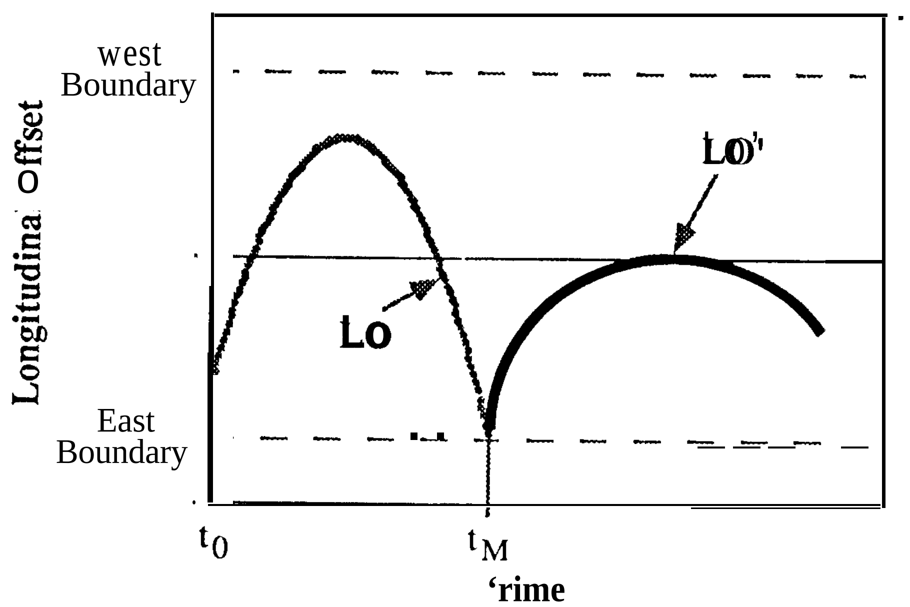

Fig. 8 - **TOPEX/Poseidon Example Fit**