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Navigation Considerations for Low-Thrust Planetary Missions

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Motivated by the upcoming launch of the solar electric propulsion mission New Millennium Deep Space 1, an analysis of the operational navigation strategies affecting ground based, deep space orbit determination of low thrust missions is addressed. Simulations are performed to assess strategies for radiometric based calibration of the SEP engine. Covariance analysis studies are conducted for the cruise phase of the Deep Space 1 trajectory with particular attention to the transition from powered flight to ballistic coast. Orbit determination strategies for coping with a spacecraft whose dynamics are corrupted by relatively large, long term stochastic perturbations are assessed. Sensitivities of navigation accuracy to variations in data scheduling, observation data noise, and stochastic parameter mismodeling are compared.

INTRODUCTION

Over the past several years, deep space exploration missions which use low-thrust propulsion to accomplish mission goals have been gaining renewed interest. Proposals are being made for ion propulsion systems (IPS) (both solar and nuclear powered) and solar sailing configurations that all generate relatively low-level, long-term accelerations on the spacecraft. These low thrust technologies can provide a specific impulse an order of magnitude larger than that achievable with conventional chemical propulsion systems, and thus enable a whole new class of planetary exploration using smaller launch vehicles. Several of these innovative designs have been proposed to NASA's Discovery program and to NASA's New Millennium technology program. The upcoming launch of the solar electric propulsion (SEP) mission New Millennium Deep Space 1 provides the most immediate need for an analysis of the operational issues related to the navigation of low-thrust missions

The presence of a solar electric propulsion system can reduce or eliminate the need for planetary flybys which eases launch window constraints due to the alignment of solar system bodies. Also, low-thrust mission design can be more robust because a spacecraft does not have to be controlled to a single prescribed trajectory (as with standard ballistic missions). If a spacecraft deviates significantly from its nominal trajectory, a completely new trajectory can be reoptimized to satisfy the current mission objectives.

Although low-thrust technology provides enhanced flexibility and robustness in mission design, the inherent difficulties in characterizing the errors of the reconstructed thrust present significant challenges to the orbit determination analyst. SEP uses a low but continuous thrust which lasts for days or weeks. On the other hand, chemical propulsion provides thrusts at a higher level for much shorter periods of time (a few seconds to tens of minutes). When the thrust is deactivated, the spacecraft essentially follows a ballistic trajectory where the dynamics affecting the equations of motion are precisely known. This highly accurate dynamic modeling makes precision orbit determination possible. For typical low-thrust missions, a 1% random error in the execution of the commanded thrust profile induces stochastic perturbations which are three orders of magnitude larger than the typical stochastic disturbances (usually attributed to outgassing and solar pressure mismodeling) considered in conventional ballistic navigation. The presence of this high level of dynamic stochastic disturbances necessitates a more sophisticated analysis of the achievable orbit determination accuracies of low thrust missions.

This paper uses the proposed trajectory for the New Millennium Deep Space 1 (DS1) mission for data simulations and covariance analysis. New Millennium DS1 is a technology validation mission which is scheduled for a July 1998 launch. During its two year prime mission DS1 will encounter the near Earth asteroid 3352 McAuliffe, the planet Mars, and comet 76P/West-Kohoutek-Ikemura (WKI). Among the primary technologies demonstrated on DS1 are the xenon ion engine and an autonomous optical navigation system (AUTONAV). The AUTONAV system [1] processes optical observations of selected beacon asteroids against a background of known stars. Via onboard image processing, the asteroid position in the camera frame is precisely located. By incorporating the knowledge of the camera pointing, the ephermeris of the constellation of beacon asteroids, and the locations of the background stars the spacecraft position can be determined. After the onboard orbit determination system updates the trajectory, the guidance portion of the AUTONAV system computes any requisite updates to the ion propulsion system commanded thrust profile. Prior to targeted flybys, AUTONAV incorporates images of the target body into its orbit determination process. This produces highly accurate position information relative to the target body. All of the image processing, orbit determination, and guidance will be performed by the onboard computer without requiring any ground telecommunications. Although AUTONAV is designed to be the prime navigation system for the DS1 mission, conventional ground based radiometric navigation tracking data will be processed concurrently for the purpose of system validation and fault recovery. The analysis presented in this paper concentrates on the use of these conventional ground based navigation technologies applied to a deep space low-thrust mission.

The segment of the DS1 mission analyzed in this study spans from launch to the McAuliffe encounter. This segment encompasses a post launch calibration of the IPS, a long cruise phase with the thrusters operational, a transition from powered flight to ballistic coast, and a targeted flyby of a small body. From a trajectory design point of view, a mission to a near Earth asteroid is not representative of an ideal application of SEP technology (as discussed by Sauer, et al. [2,3]). However, the results of simulated ground based radiometric IPS calibration and the accuracy analysis for powered flight during cruise should be applicable to most classes of SEP missions. The analysis of the transition from powered flight to coast is relevant for fast high energy flyby missions which typically employ coast arcs prior to targeted gravity assist swingbys.

In contrast, missions which use low-thrust to accomplish a rendezvous and orbit about a planetary object must thrust continuously up to the point where the relative position and velocity are reduced enough so that an orbit injection can be accomplished. As in the navigation for a flyby mission, the cruise orbit determination errors could be allowed to grow quite large between times of updates, perhaps to tens of thousands of kilometers in position and several kilometers per second in velocity. Depending on the target body, thrusting and navigation guidance update cycles would be shortened and their frequency increased toward the end of the cruise, at a distance of several million kilometers, in order to deliver the spacecraft to the body with less uncertainty than that typically allowed during the earlier parts of a low-thrust thrusting period.

At about one hundred days prior to the rendezvous, a typical mission might perform weekly updates while using both radio metric tracking and optical navigation pictures of the target body. Exact distances for first optical detection depend on the particular mission and spacecraft design; however, it is possible to generalize the terminal navigation for many of the interplanetary rendezvous cases since their final trajectory ends up tangent to the target body with nominally zero velocity. Hence, starting with weekly optical navigation pictures, the relative positional navigation errors should quickly be reduced to a few hundred kilometers. Between pictures, the cross-track errors in the target plane (normal to the relative velocity vector) will quickly grow due to degradation of position knowledge from the stochastic errors in the low-thrust profile. In order to deliver the spacecraft to a target several radii from the body with a few kilometers error, the frequency of optical navigation pictures will have to be increased. With daily pictures, it should be possible to navigate the spacecraft to within a few days of closest approach with an error of about ten kilometers. Determining the range to the body to an accuracy any better than the a priori knowledge of the body ephemeris will remain difficult up until the very last hours of the rendezvous when the optical parallax will begin to produce relative motion in the focal plane for a sequence of pictures.

For a given mission, a judicious placement of low-thrust guidance update cycles and optical navigation pictures might produce relative errors of a few tens of kilometers in position and a few hundreds of meters per second in velocity. For rendezvous with an asteroid or comet, these errors are likely too large for direct injection into orbit about the body, so the low-thrust approach scenario would transition to a more conventional deep space navigation approach using impulsive chemical maneuvering capabilities during the last few hours of approach. For these small bodies it is important that the relative velocity not be allowed to become too small, say less than a couple of meters per second, when within a few radii of the body to avoid collision.

LOW THRUST NAVIGATION OVERVIEW

The goal of the orbit determination process is to ascertain whether the spacecraft is currently following the prescribed trajectory. If the spacecraft has been determined to deviate from the nominal trajectory, the guidance element determines the required adjustments to the thrust profile which nulls the errors in the trajectory. The guidance can be carried out via a complete (yet computationally intensive) trajectory reoptimization which amounts to solving a calculus of variations problem. This approach will be adopted for DS1 in the event that anomalies are encountered in the AUTONAV system which will require that the backup ground navigation and guidance system be used for mission operations. If deviations from the nominal are sufficiently small, a more computationally efficient linear perturbation approach to performing guidance could be adopted. This approach was examined in [4].

A typical ground-based navigation cycle involves acquiring radiometric tracking data from the Deep Space Network at regular time intervals (typically two eight hour passes per week). From the downlink spacecraft telemetry the thrust profile is reconstructed and incorporated into the dynamic models used by the orbit determination system. After the orbit determination is completed, the optimal flight path is recomputed and the associated update of the SEP thrust profile for the subsequent cycle is uploaded to the spacecraft. Large orbit determination uncertainties can be tolerated until the approach of a coast phase boundary or the final target. At these critical junctures, the navigation and guidance update cycles will need to be executed more often. Thus, a non-autonomous low-thrust navigation system may incur greater operational costs due to the increased communications requirements to support these complex scenarios. These cost considerations motivated the development of the onboard autonomous navigation system which is to be demonstrated on the DS1 mission. The ground navigation system for DS1 will provide independent validation of the orbit information output by the onboard system. The covariance computations discussed in this paper can be used to develop success criteria for the trajectory comparisons between the ground and onboard navigation systems.

LOW-THRUST CALIBRATION

In order to calibrate the IPS thrusters, the spacecraft will be commanded to orient the thrust vector along the spacecraft-to-Earth line of sight to provide a strong doppler signature in the radiometric tracking data. Strategies for calibrating the SEP engine are analyzed by data simulations where a truth model of the thrust profile is constructed and the orbit determination system attempts to recover the thrust model by reducing the simulated data. Doppler and range observations are simulated using a spacecraft trajectory whose dynamics are derived from the "truth" thrust profile. Noise is applied to the simulated observations via a pseudorandom number generator. Then an incorrect thrust model is hypothesized and used to initialize the iterative orbit determination system. Each iteration of the orbit determination process adjusts parameters of the thrust model until the residuals between the modeled observations and the simulated "truth" observations is minimized. This iterative process is depicted in figure 1. After the orbit determination process has converged the computed thrust profile is compared to the "truth" profile. If the profiles match then the simulated calibration is judged to be successful.

The IPS calibration was scheduled to be performed 5 days after launch. The IPS was scheduled to be turned on for a short period of time and then turned off. The thrust profile consisted of a sequence of one hour periods of constant thrust interspersed with 30 minute transition periods modeled as linear ramps. We are only concerned with matching the constant thrust segments with their corresponding components in the "truth" profile. The calibration simulation contained three constant segments with thrust levels of 25, 50, and 70 mN. The errors in the initial modeled thrust profile ranged from 10% to 20% of the true values. Doppler and range measurements were scheduled continuously throughout the thrusting phase in addition to 24 hour periods prior to and following the thrust period. Pass through observation residuals exhibit a distinct signature as shown in figure 2. After iterating three times through the orbit determination system, the residuals are reduced to the levels of the data noise and the recovered thrust profile differs from the true model by less than 0.2 %.

The above describes an idealized scenario where the thrust model is perfectly parameterized and orbit determination filter state includes includes the entire set of parameters which were mismodeled. Another simulation was generated with a thrust orientation offset 5 degrees from the spacecraft-Earth line of sight. If the orientation is not adjusted in the orbit determination filter, the residuals after several iterations display a low level yet systematic signature as shown in figure 3. One may expect that the errors in the orientation would alias into errors in the recovered thrust profile. Despite the gross mismodeling of the thrust orientation, the relative errors in the recovered thrust model ranged from 0.6% to 2.0% with a maximum absolute error of 0.5 mN. If the orbit determination filter adjusted the orientation, the signature on the postfit residuals was removed and the maximum relative thrust error was reduced to 1.2%. However, the errors in the recovered orientation angles were quite large and ranged from 0.5 to 5 degrees. The formal sigmas on the most poorly determined angles were approximately 10 degrees which implies that the short thrusting data arc provides poor information content for recovering thrust orientation errors. Another simulation scenario applied a large timing error of 5 seconds to the thrust profile. The postfit residuals in figure 3 exhibit a signature during the thrusting period which indicates mismodeling of the thrust. However, the recovered values of the thrust magnitudes were still quite good with a maximum absolute thrust error of 0.2 mN (0.3% relative error). In operations, the timing errors can be reduced by compressing the Doppler data to a subsecond count time. The timing of IPS events can be recovered (to the time resolution of the radiometric data) by correlating changes in the Doppler signature with state changes of the IPS. These preliminary results indicate the feasibility of calibrating the thrust magnitude of the IPS by using short arcs of Doppler and range data. However, thurst orientation errors cannot be reliably recovered in a short calibration run.

CRUISE COVARIANCE ANALYSIS

A covariance analysis was conducted for the portion of the DS1 trajectory spanning from 30 days after launch to the flyby of McAuliffe. The IPS was operational during the entire time until 20 days before the McAuliffe flyby where the spacecraft enters a terminal coast phase. The baseline tracking data schedule consisted of two eight hour passes of coherent x-band doppler and range each week. One of the two weekly passes was from the Goldstone, California complex while the second eight hour pass was acquired from Spain and Australia on alternate weeks. The data noise for the doppler was 0.1 mm/s normalized to a 60 second compression time; the noise on the range data was 10 meters. Random errors in the reconstructed IPS thrust profiles were modeled as Gauss-Markov process noise accelerations. The selection of five day time constant for the stochastic process noise was based upon a previous analysis described by McDanell [5] which yielded a robust filter for a low thrust comet mission. The batch update interval was 6 hours and the steady state process noise uncertainty was chosen to be 10^{-9} km/s^2 which corresponds to a thrust error of approximately 1%. The adjusted parameters in the orbit determination filter consisted of the initial position and velocity, solar pressure scale factors, and the piecewise constant stochastic accelerations. Consider parameters are not adjusted but their uncertainties are allowed to inflate the data noise covariance via the sensitivity of the estimate error with respect to parameter variations. The consider parameters and their assumed uncertainties are tabulated in figure 4.

The position error covariance results were mapped over a 38 day period where the spacecraft transitioned from powered flight to ballistic coast on the 20th day of the 38 day mapping period. The covariance in the thrusting region indicates the steady state performance during powered flight. Of interest in the coast region are the time required to reach the new, reduced steady state uncertainty and the magnitude of the steady state error. The position error for the baseline strategy is depicted in figure 5. The "scatter" in the 3D position uncertainty during powered flight is the result of undersampling the mapped covariance. If the mapped position error is plotted at a higher time resolution (figure 6) the dependence of the magnitude of position error on the tracking data schedule becomes apparent. The error is at a relative minimum immediately after a pass of data has been processsed. In the intervals between passes, the covariance propagation is driven by a large stochastic component which results in a steep rise in the position uncertainty. Figure 6 shows "stray" values of the uncertainty between relative maximum and minimum errors. These points correspond to the filtered covariance mapped to the middle of a tracking pass. Under the baseline filter scenario, missed tracking passes result in serious degradation of the position knowledge. The baseline scenario was altered so that only doppler data were processed. This corresponds to a spacecraft whose transponder lacks a ranging capability. The doppler only position uncertainty was magnified by up to a factor of four in the thrusting region but more importantly, the steady state error in the coast region increased by a factor of six. From the standpoint of navigation accuracy the removal of a ranging capability is inadvisable because of severely compromised orbit knowledge during the critical coast phase.

The robustness of the baseline filter design can be assessed by analyzing the resulting covariances under assumptions of incorrect stochastic modeling and incorrect data noise. The analysis of a non-optimal filter requires the precomputation of the Kalman gains for the baseline filter. The non-optimal covariance is generated by driving the baseline filter (defined by its gain profile) with mismodeled stochastic processes or incorrect observation data noise. We first consider incorrect stochastic modeling of the thrust reconstruction error. When the steady state sigma for the stochastic acceleration was doubled, the position error within the thrust region increased from 40 to 60% over the uncertainties in the baseline case (figure 8). When the steady state sigma was increased by a factor of five, the position error in the thrust region ranged from a low of 159 km to a high of 978 km which amounts to a relative error increase of up to 400%. Because the non-optimal stochastic acceleration model is active only during the thrusting phase, the steady state sigmas within the coast phase are affected minimally by the non-optimal filter. If the random error in the thrust profile is significantly less than the assumed level of the baseline filter (1%), then the baseline statistics would be pessimistic. Figure 9 shows the reduction in the position error due to improvements in modeling the thrust profile. A new set of non-optimal covariances was generated by systematically changing the time constant for the stochastic thurst model. When the time constant is increased, the acceleration error approaches the behavior of a non-stochastic bias. However, the filter was much less sensitive to the mismodeled time constant compared to errors in characterizing the steady state stochastic acceleration. If the baseline filter is executed with degraded noise assumptions on the range data, the mapped position error changes very slightly compared to the optimal baseline filter. Thus, the output position errors are relatively insensitive to errors in characterizing the data noise for range. However, when non-optimally weighted doppler data are processed the degradation in the position information is much more severe (figure 10). This increased sensitivity is probably due to the fact that the doppler is more effective than range for observing the time-varying stochastic thrust. If the non-optimally weighted doppler results in greater errors in the estimation of the stochastic thrust, then the position errors will be increased due to the high sensitivity of the position error to thrust errors. The sensitivity to mismodeled doppler data noise is important because unfavorable Earth-spacecarft-sun geometries can result in greater fluctuations in the solar plasma along the signal path which increases the doppler system noise to beyond the baseline assumptions of 0.1 mm/s.

ASTEROID FLYBY COVARIANCE ANALYSIS

Several days before the McAuliffe flyby, optical images of the target asteroid are incorporated into the navigation solution. The center finding error for the optical navigation frames was assumed to be 1/2 the apparent diameter of the asteroid. During this final approach coast phase, the filter estimates several trajectory correction maneuvers using an apriori uncertainty of 1% of the ΔV's. In addition the McAuliffe ephemeris is adjusted.

CONCLUSION

Simulations for the IPS calibration show that neither errors in the thrust orientation nor thruster timing errors which are large relative to the performance expected on New Millennium DS1 will seriously hamper the recovery of the thrust magnitude. This suggests that reliable ground based calibration of the output of the IPS is feasible. However, short duration (on the order of several hours) engine calibration strategies were ineffective in correcting thrust orientation errors. Covariance analyses for the cruise phase showed that range data were necessary for reducing the orbit uncertainty, but the position uncertainty is relatively insensitive to variations in the quality of the range data. On the other hand, position errors were much more sensitive to mismodeled doppler data noise during the thrusting phase of a mission. The covariance analysis also quantitatively showed the improvement in orbit uncertainty that results from both a reduction and a correct characterization of the errors in the reconstructed thrust profile.

The approach used in this paper to generate the navigation covariance results did not account for any guidance laws. The guidance during the powered flight mission phases may be carried out by using either the same techniques used to search the nominal trajectory or by using various linearized perturbation techniques. In order to completely analyze the navigation errors, a scenario for re-targeting and calculating control gains for the linearized guidance scheme needs to be specified and included. Such a scenario was studied previously for a comet Halley rendezvous mission [8], and would be a good starting point for refining the results presented here.

ACKNOWLEDGMENTS

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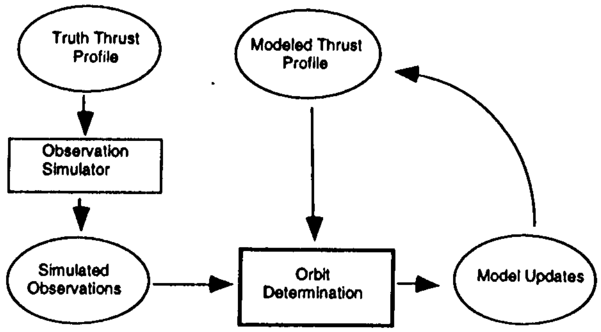

Figure 1 Overview of the data simulation process

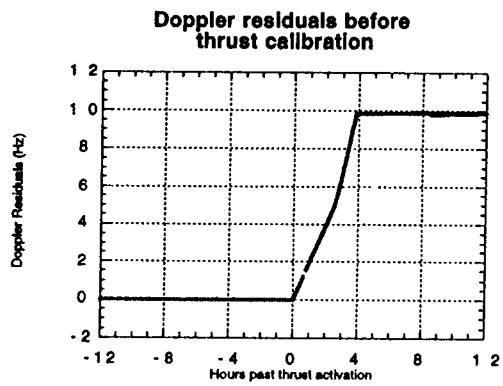

Figure 2 Residual Doppler signature due to uncalibrated IPS

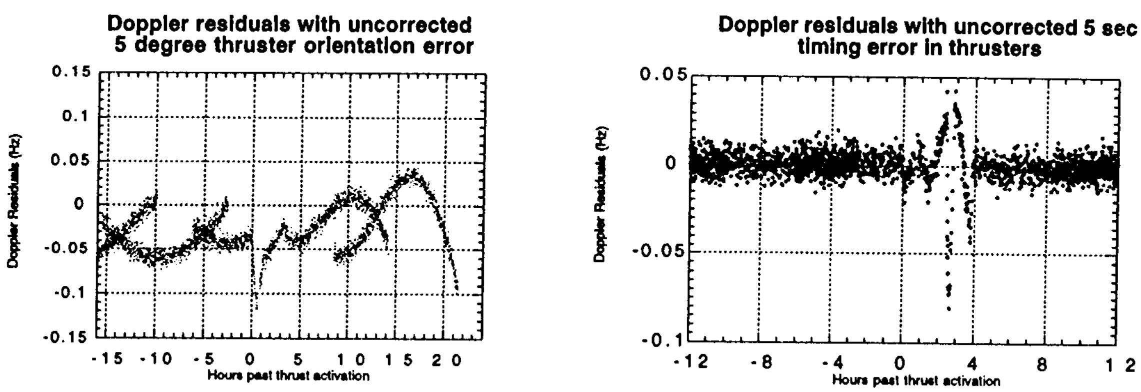

Figure 3 Postfit residuals with unadjusted orientation and timing errors

Estimated Parameters	Apriori Uncertainty
Position and velocity	1.5×10^7 km, 20 m/s
Solar pressure	10% of nominal value
Stochastic acceleration	10^{-9} km/s^2
	5 day time constant, 6 hr batch size, steady state sigma 10^{-9} km/s^2

Impulsive chemical maneuvers (active only during approach phase)	1% execution error
McAuliffe ephemeris (active only during approach phase)	Correlated covariance provided by JPL's Solar System Dynamics group
Consider Parameters	
Station locations	Correlated covariance determined from VLBI processing
UT1 and polar motion	.2 msec, 15 nrad.
Troposphere (wet and dry)	4 cm, 10 cm
Ionosphere	$.5 \times 10^{17}$ elec/m^2

Figure 4 Covariance analysis assumptions

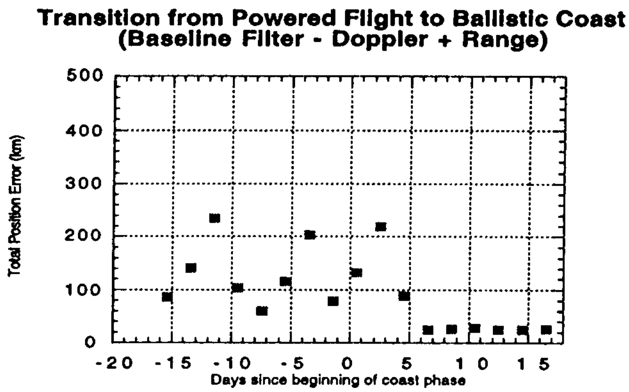

Figure 5 Mapped position error for baseline strategy

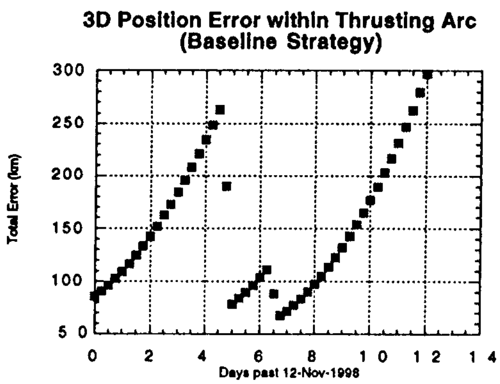

Figure 6 Position uncertainty during powered flight

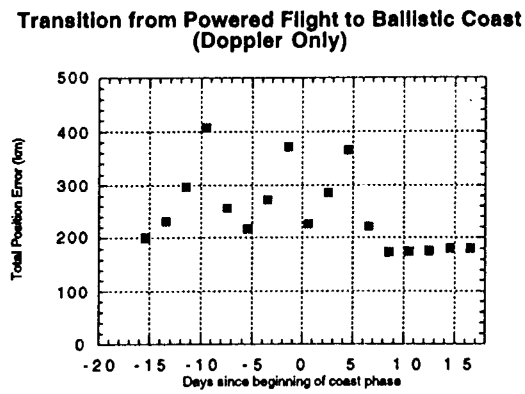

Figure 7 Position uncertainty for doppler only strategy

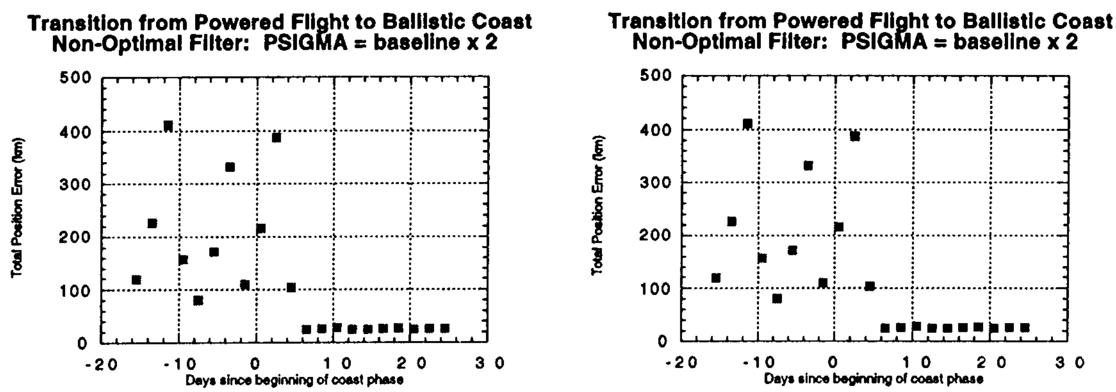

Figure 8 Covariance due to mismodeled process noise sigma

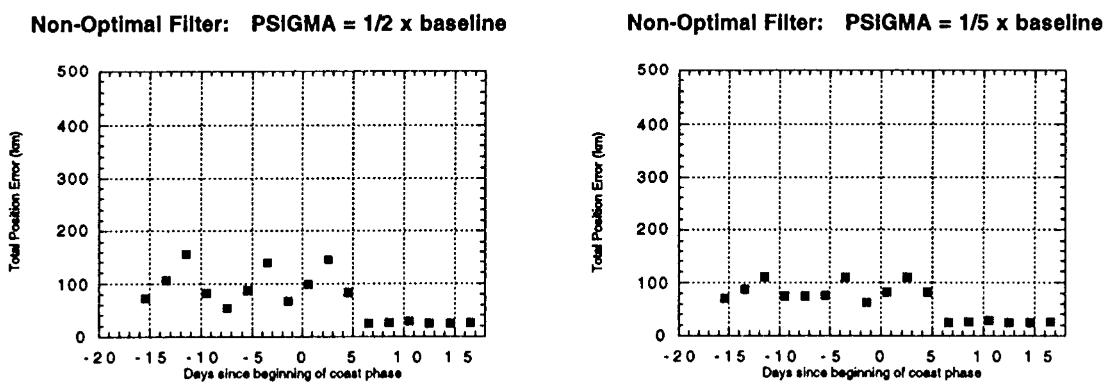

Figure 9 Effect of enhanced calibration of the thrust error