FINAL REPORT

VOYAGER TERMINAL GUIDANCE STUDY

July 1968

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# VOYAGER TERMINAL GUIDANCE STUDY

July 1968

Contract NAS8-21168

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# FOREWORD

This report presents the results of work performed by Lockheed's Huntsville Research & Engineering Center while under contract to the Guidance Theory Branch/Astrodynamics and Guidance Theory Division of the George C. Marshall Space Flight Center, Huntsville, Alabama.

This document represents the final report for Contract NAS8-21168, "Voyager Terminal Guidance Study." This work was performed by personnel of Lockheed/Huntsville's Dynamics & Guidance Department.

## SUMMARY

This report presents the results of an analysis to determine the performance capability and orbit insertion accuracy of the Voyager spacecraft, utilizing constant inertial attitude guidance, to accomplish the deboost maneuver into a terminal orbit about Mars.

The performance capability of the nominal vehicle was obtained by generating nominal trajectories which accomplished transfers from the nominal class of hyperbolas specified by hyperbola specific energy into the nominal terminal orbit. The fraction of the initial vehicle mass which was delivered into the terminal orbit (performance) was found to be a function of the rotation angle between hyperbola and ellipse periapsis, as well as the difference in periapsis altitude. The optimum performance occurs for a periapsis to periapsis transfer in which hyperbola periapsis altitude is the same as that of the desired ellipse.

The deboost error analysis was accomplished by choosing representative nominal trajectories and applying correlated ignition position and velocity errors (navigation errors) and independent, normally distributed execution errors. This was done by means of a specially developed Monte Carlo deboost system error analysis program which samples the postulated error spaces for each Monte Carlo pass, and generates the statistics of the terminal orbit errors.

Both performance and error sensitivity information are presented as functions of terminal geometry, vehicle characteristics and terminal orbit size.

## NOMENCLATURE

| | |
|---|---|
| $\alpha$ | the angle between the vehicle velocity vector at deboost ignition and the nominal constant inertial thrust direction (deg) |
| $C_3$ | hyperbola specific energy $(km/sec)^2$ |
| $H_{ph}$ | altitude of hyperbola periapsis (km) |
| $h_p$ | altitude of ellipse periapsis (km) |
| $h_a$ | altitude of ellipse apoapsis (km) |
| $i$ | inclination of the terminal orbit (deg) |
| $I_{sp}$ | specific impulse (sec) |
| $M_F/M_o$ | mass ratio; mass in orbit divided by mass at ignition (dimensionless) |
| $\omega$ | argument of ellipse periapsis measured from the ascending node (deg) |
| $\Omega$ | argument of the line of nodes of the terminal orbit measured from an inertial reference (deg) |
| $P$ | period of the terminal orbit (sec) |
| $\phi_R$ | rotation of apsidal line from hyperbola to ellipse (positive in the direction of motion) (deg) |
| $r_p$ | radius of periapsis of the terminal orbit (km) |
| $r_{ph}$ | radius of periapsis of the hyperbola (km) |
| $T$ | thrust (lb) |
| $T/W$ | thrust to weight ratio (at ignition) (dimensionless) |
| $\theta_E$ | insertion anomaly on the ellipse (deg) |
| $\overline{V}_{h_i}$ | vehicle velocity vector at deboost ignition (m/sec) |

# Section 1
# INTRODUCTION

The object of the Voyager Terminal Guidance Study was to determine the terminal accuracy and performance capabilities of candidate guidance techniques to perform a spectrum of representative Voyager missions.

In order to accomplish the study objective, the effort was divided into three major tasks in the Work Statement as follows:

Task A: Determination of Perturbed Deboost Ignition Conditions

Task B: Simulation of Guidance System Error Sources

Task C: Analysis of Candidate Guidance Techniques

Implicit in Task C is the determination of the terminal accuracy and performance capability of each candidate guidance scheme. Early in the study, the decision was made to concentrate on constant inertial attitude (CIA) as the primary guidance scheme, and to do an extensive parametric performance and accuracy analysis for this guidance scheme.

Tasks A and B are preliminary steps to accomplish an analysis of orbit insertion accuracy. Task A corresponds to establishing the nature of the errors existing at deboost ignition, while Task B requires the construction of a model of the errors which occur during deboost burn (execution errors).

Essential to all of the above stated tasks is the generation of guidance "nominal" trajectories, i.e., sets of deboost ignition times, thrust angles, and cutoff criteria — which if accomplished — would achieve exact insertion into the desired orbit. These nominals yield the desired performance information for a range of expected terminal geometries and establish the basis for an error analysis of the system. In order to incorporate perturbed ignition conditions into the error simulation, ignition condition errors must be applied to "non-perturbed" ignition points. Also, for each execution error which occurs during deboost, a "nominal," or errorless value of that execution parameter must be defined. Thus, these "nominals" must be generated, not only to establish system performance, but to accomplish the insertion error analysis.

The approach taken in the study was to first generate CIA guidance nominals and established nominal vehicle performance (in terms of mass delivered into Mars orbit) as a function of terminal geometry. This was accomplished and reported in Voyager Terminal Guidance Study Phase I; Nominal Trajectory and Performance Analysis (Reference 1) and Addendum (Reference 2).

The second phase consisted of utilizing the nominals generated and combining them with error models developed to accomplish Tasks A and B.

Task A specifies that statistically significant variations in the target arrival hyperbola should be propagated to the deboost ignition point to define perturbed ignition conditions. In practice these variations in the approach hyperbola will actually consist of uncertainties in tracking, i.e., uncertainties in the knowledge of the actual hyperbola. Reference 3 presents a method for generating error covariance matrices as functions of the period of time before ignition at which the last tracking information was obtained. Uncertainties included in these matrices are the vehicle's position and velocity, the gravitational constant of the attracting body, the central body's position and velocity, and the location of the tracking station. Thus, to obtain perturbed ignition conditions, sets of normally distributed random variables properly applied to the diagonalized covariance matrix will yield correlated position and velocity errors.

Task B requires that a set of execution error sources and magnitude be defined. Reference 4 defines two levels of execution errors which were used throughout this study. All execution errors were assumed to be normally distributed and uncorrelated.

The ignition error results accomplished in Task A were combined with the execution error data for Task B in a Monte Carlo simulation developed to perform the terminal accuracy analysis specified in Task C. The results of this effort are presented in the Voyager Terminal Guidance Study Phase II report (Reference 5).

Phase I and Phase II of the Voyager Study are summarized in this report.

## Section 2
## SUMMARY OF DEBOOST MANEUVER PERFORMANCE ANALYSIS

This section presents a summary of the results of the initial phase of the Voyager Terminal Guidance Study. This study was conducted to evaluate nominal ignition requirements and vehicle performance for the deboost maneuver into a terminal orbit about Mars. As mentioned in Section 1, detailed accounts of the methodology, mission constraints, implementation requirements, and vehicle performance that were analyzed in the Phase I study, can be found in References 1 and 2. Before presenting these results, a brief review of parameters and conditions defining the deboost maneuver and vehicle performance will be given.

Arrival trajectories are classed by specific energy, while particular approach hyperbolas in the class were treated as variables and are identified by periapsis altitude ($H_{ph}$). Apsidal rotations ($\phi_R$), and the associated performance required to obtain that rotation is shown for both a selected set of nominal vehicle/mission conditions and parametric variations to these conditions. Figure 2.1 demonstrates the sign convention used in defining $\phi_R$. Performance capability to accomplish the deboost maneuver is measured in terms of the percent of initial mass which is delivered into the Mars orbit.

The nominal parameters describing the vehicle, terminal orbit and class of approach hyperbolas are defined as:

| | | |
|---|---|---|
| • Mass at ignition | ($M_o$) | 24,000 lb |
| • Specific impulse | ($I_{sp}$) | 304 sec |
| • Thrust | (T) | 10,175 lb |
| • Hyperbola specific energy | ($C_3$) | 10.8 $(km/sec)^2$ |
| • Terminal orbit | ($h_p \times h_a$) | 1,000 km x 20,000 km |

The following variations from the nominal conditions were evaluated independently:

- $I_{sp}$ — 320 sec and 290 sec
- $C_3$ — 6.7 $(km/sec)^2$ and 16.0 $(km/sec)^2$
- T/W — .25 and .75
- $h_p \times h_a$ — 1,100 x 10,000 km, 1,200 x 20,000 km, 1,000 x 25,000 km.

Figure 2.2 summarizes the performance possibilities for the nominal vehicle and hyperbola class as defined above. As may be seen, performance increases as hyperbola periapsis altitude approaches the periapsis altitude of the desired terminal orbit. When hyperbola periapsis is larger than the ellipse periapsis, the transfer can no longer be made at zero periapsis rotation angle. In the case of the 1,000 x 20,000 km ellipse when hyperbola periapsis altitude is above 1300 km, no rotations can be accomplished in the range $\pm 20^{\circ}$.

The effects on performance of variations in vehicle $I_{sp}$, in hyperbola specific energy ($C_3$), and in terminal orbit, all for $H_{ph}$ = 800 km are shown in Figures 2.3, 2.4 and 2.5, respectively.

The variations in $C_3$ produced the greatest difference in vehicle performance from nominal case. $I_{sp}$ and $h_p \times h_a$ variations, as noted above, also produced significant changes in performance from the nominal case. The T/W variations produced no effective change in performance, but did change ignition time and burn time appreciably from the values which resulted under nominal conditions.

Ellipse semi-major axis is parameterized and presented in Figures 2.6 through 2.8 as a function of mass ratio.

The larger the semi-major axis of the ellipse, the more efficient is the deboost transfer. Performance is reduced when an appreciable portion of the thrust is required to change the vehicle path in order to achieve tangency with the ellipse. This is the case when the hyperbola periapsis is closer to the planet than the ellipse periapsis ($r_p - r_{ph} > 0$). Given an ellipse, (fixed $r_p$ and semi-major axis a) and an ellipse-hyperbola orientation angle ($\phi_R$) performance will improve as $r_{ph}$ increases until the transfer can no longer be made with the chosen rotation angle.

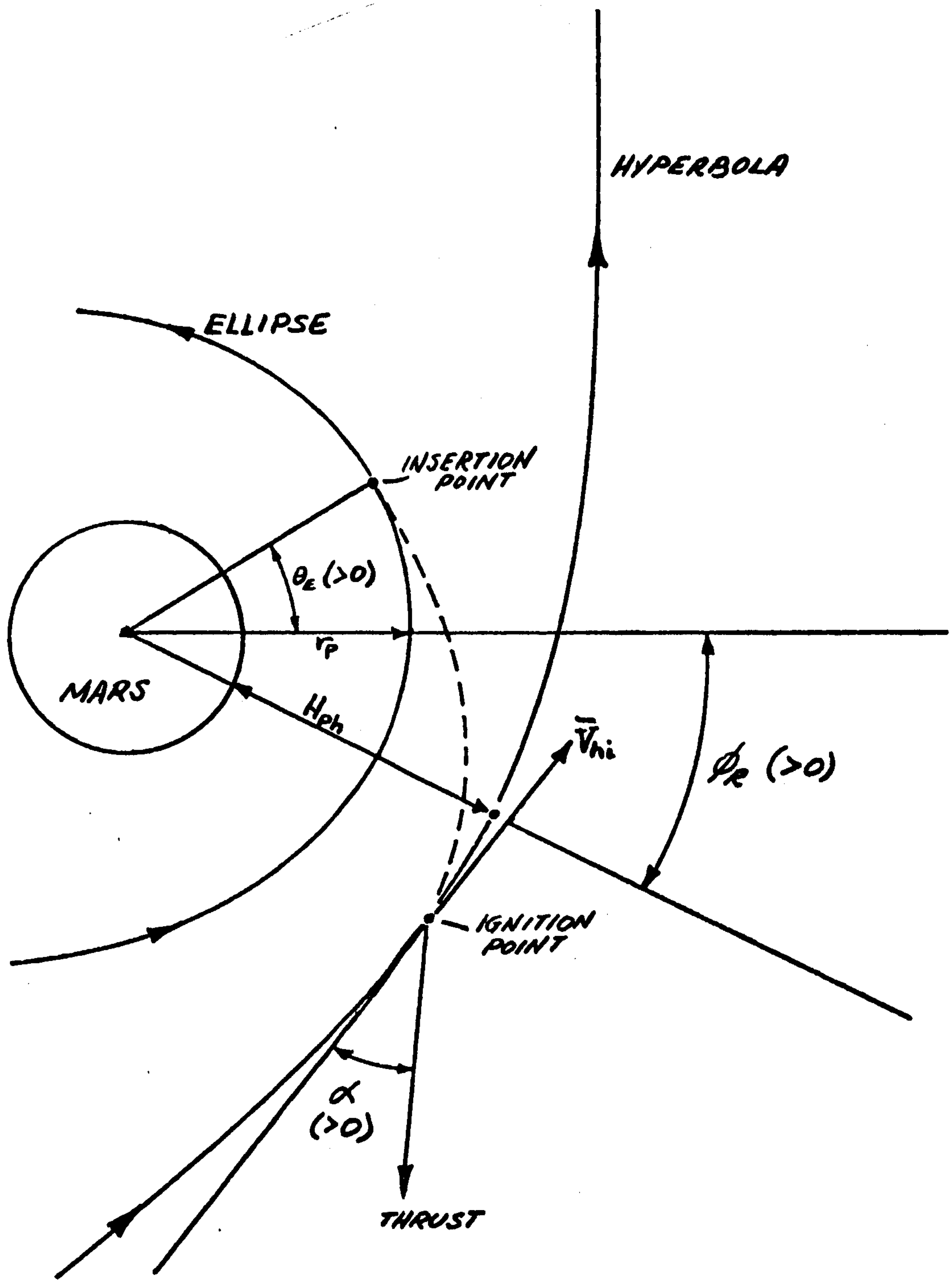


Figure 2-1 - Definition of Various Transfer Geometry/Parameters

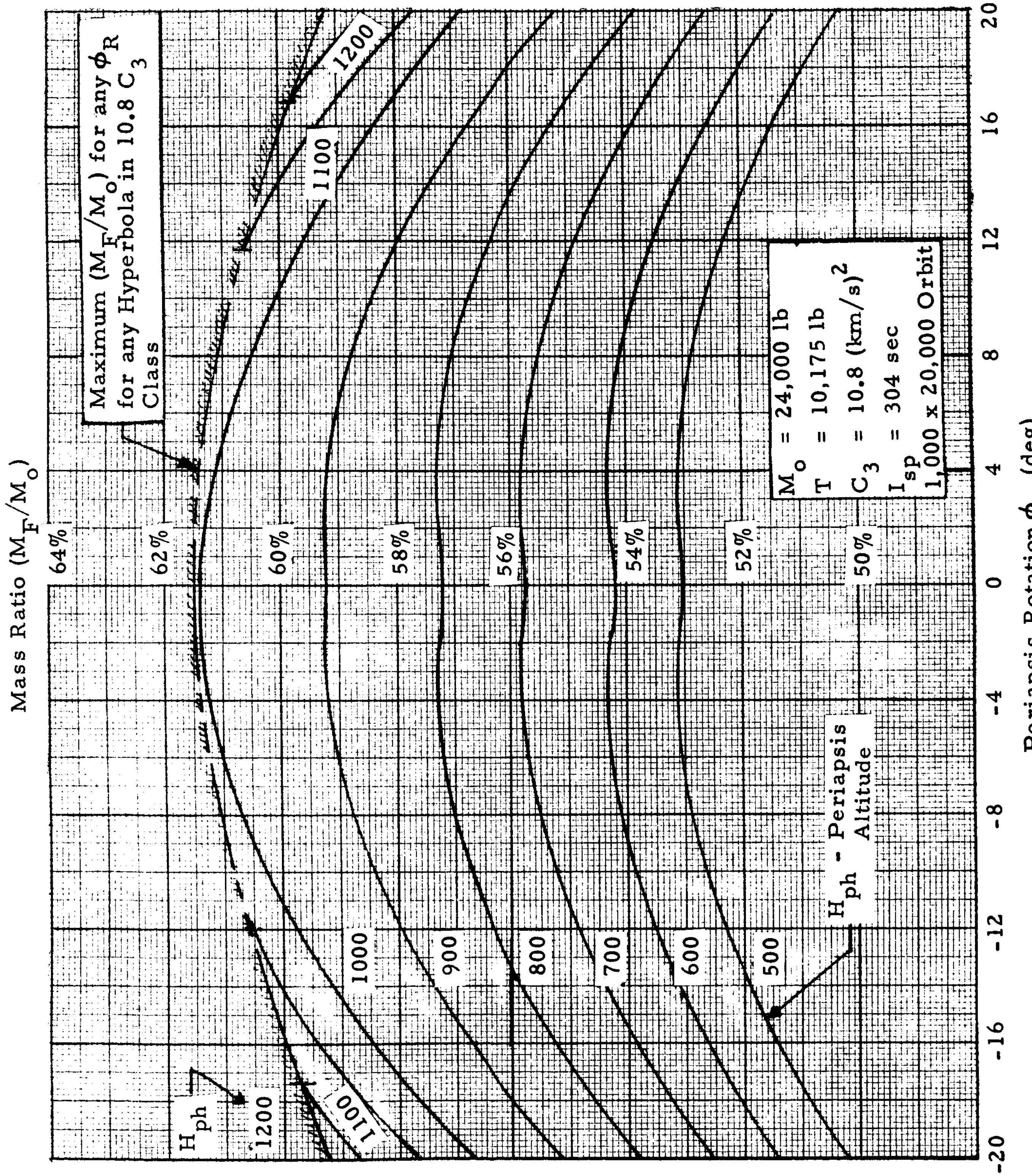


Figure 2-2 - Performance Capability/Nominal Conditions

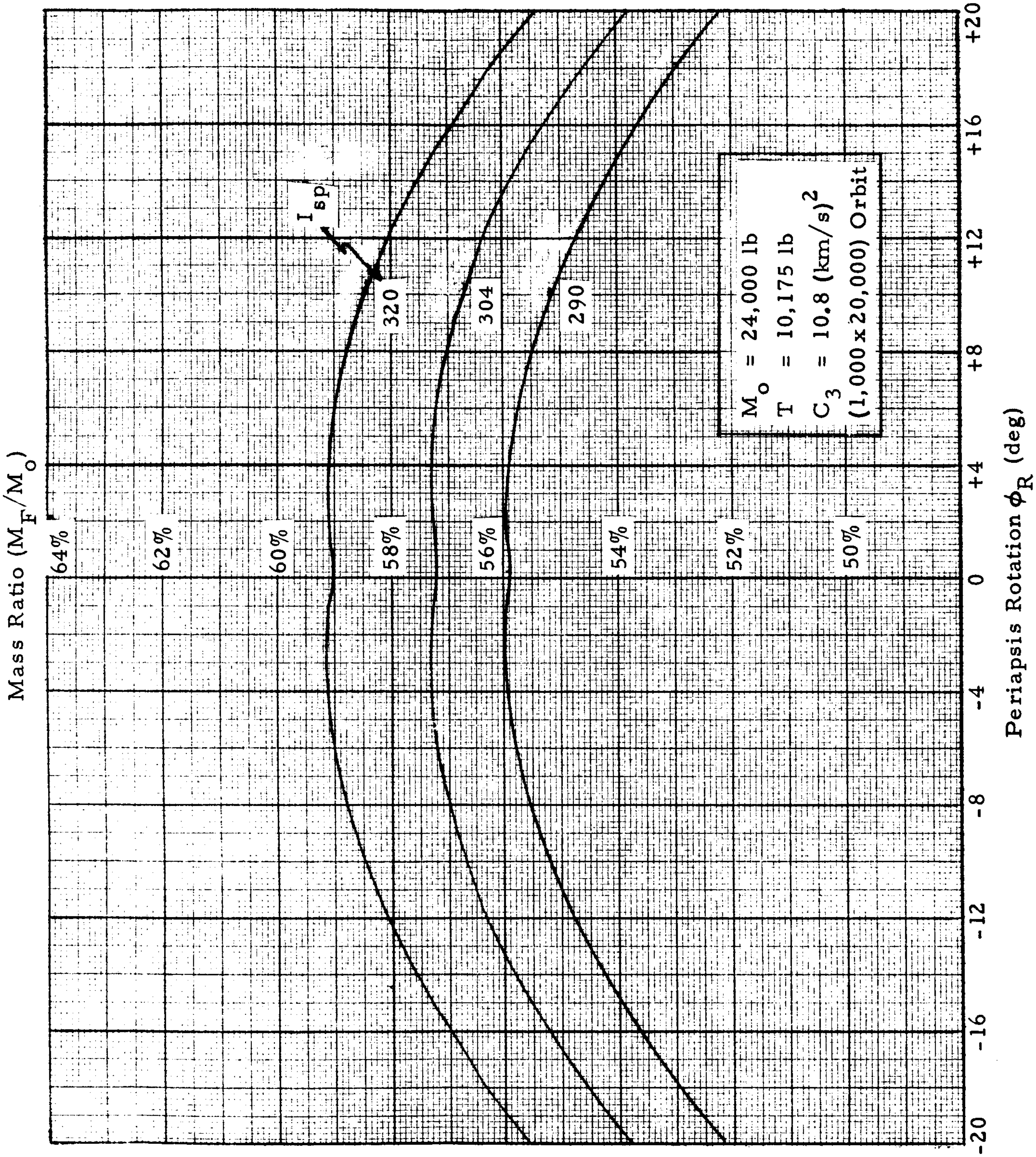


Figure 2-3 - Specific Impulse Variation ($H_{ph}$ = 800 km)

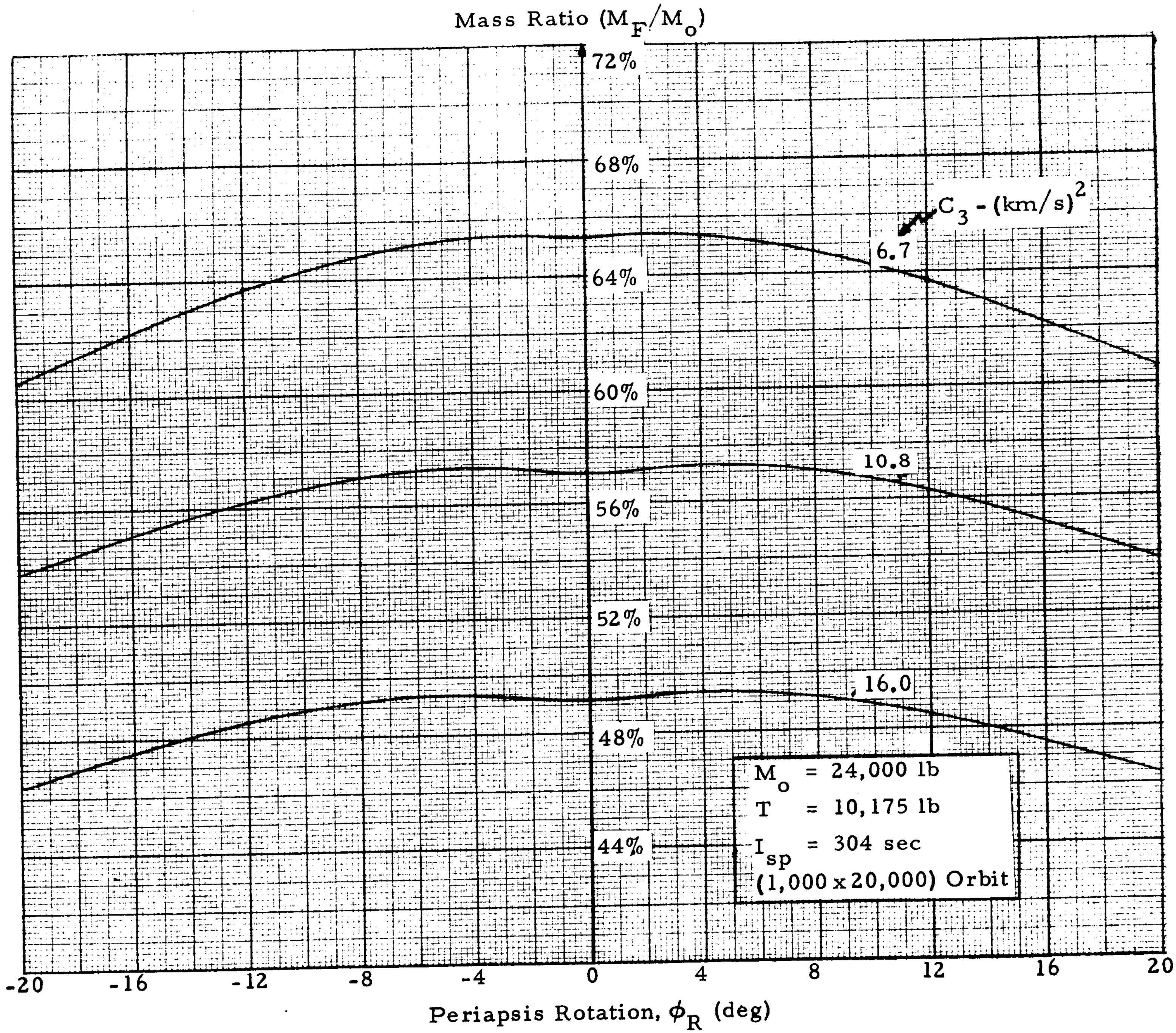


Figure 2-4 - Approach Energy ($C_3$) Variation ($H_{ph}$ = 800 km)

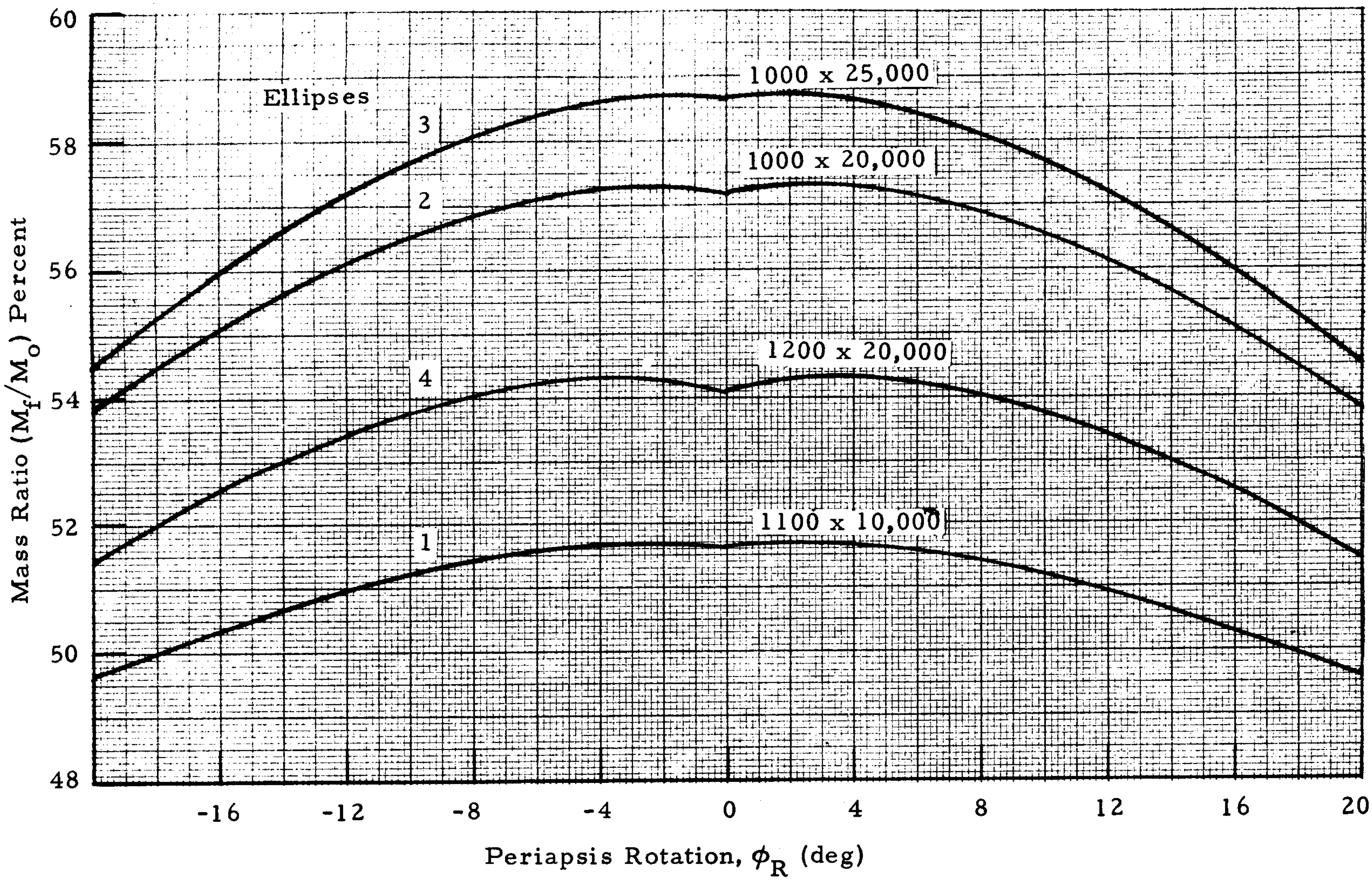


Figure 2-5 - Mass Ratio vs Rotation Angle ($H_{ph}$ = 800)

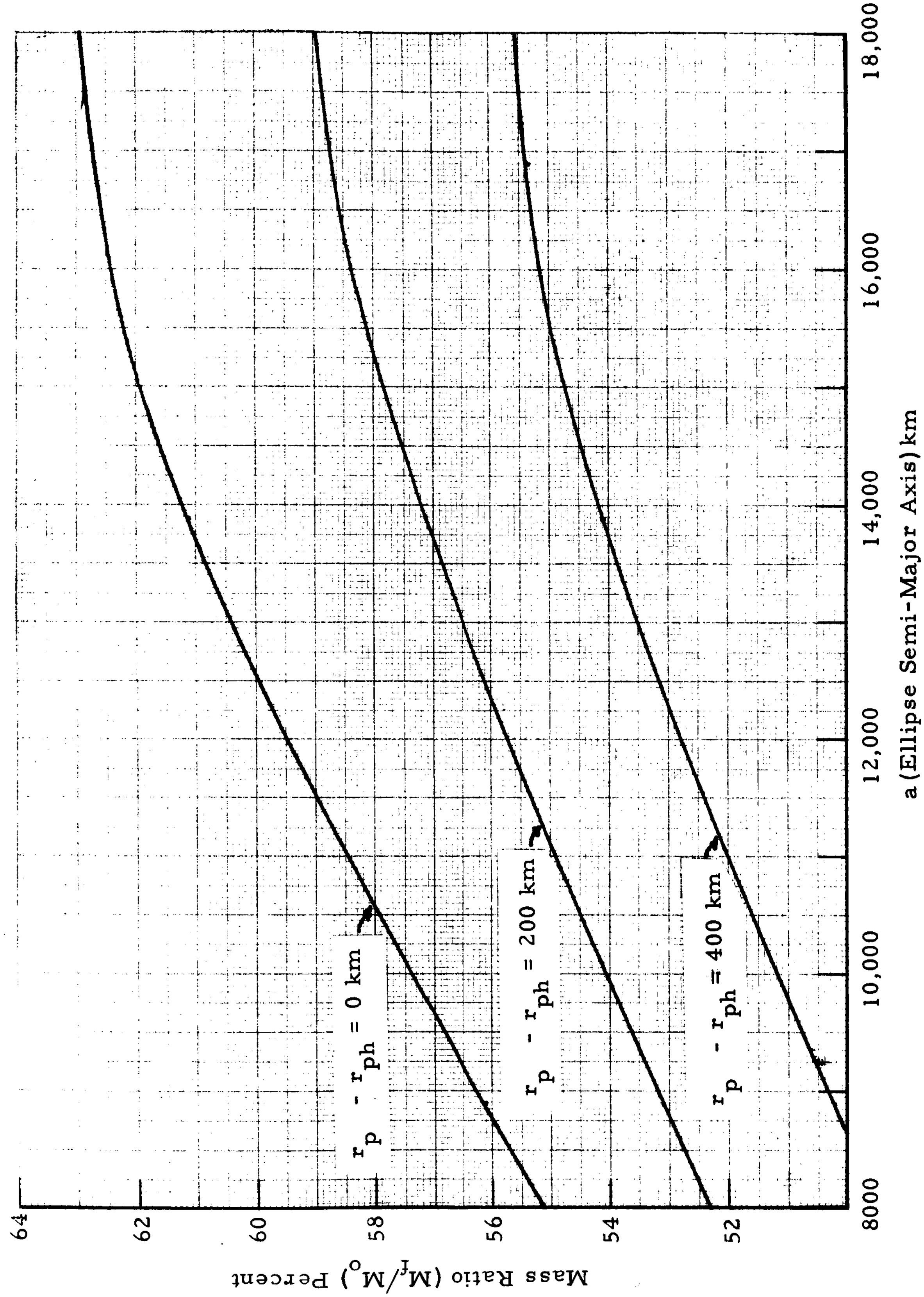


Figure 2-6- Mass Ratio vs Delta Periapsis ($\phi_R$ = 0 degrees)

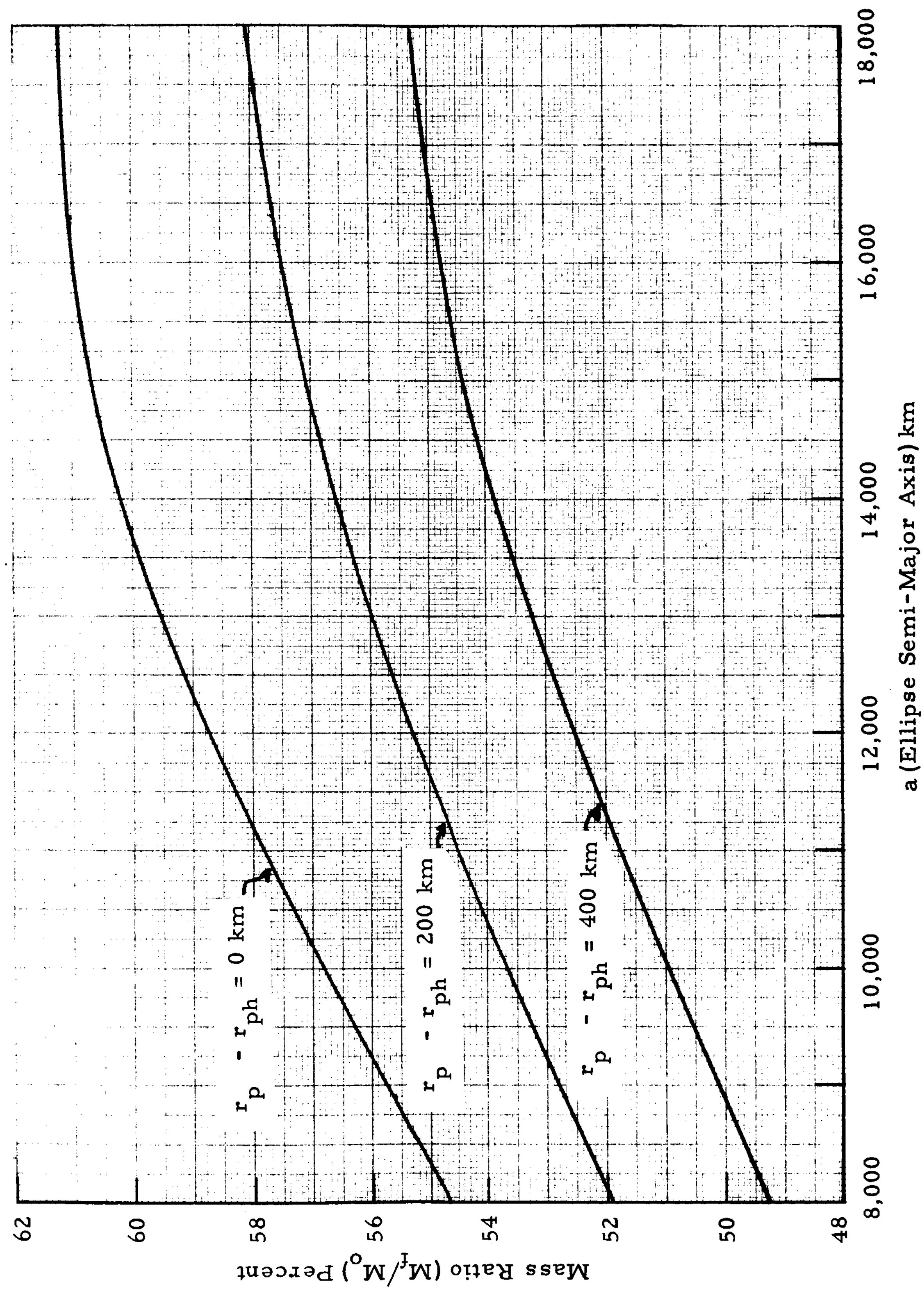


Figure 2-7 - Mass Ratio vs Delta Periapsis ($\phi_R = \pm 10$ degrees)

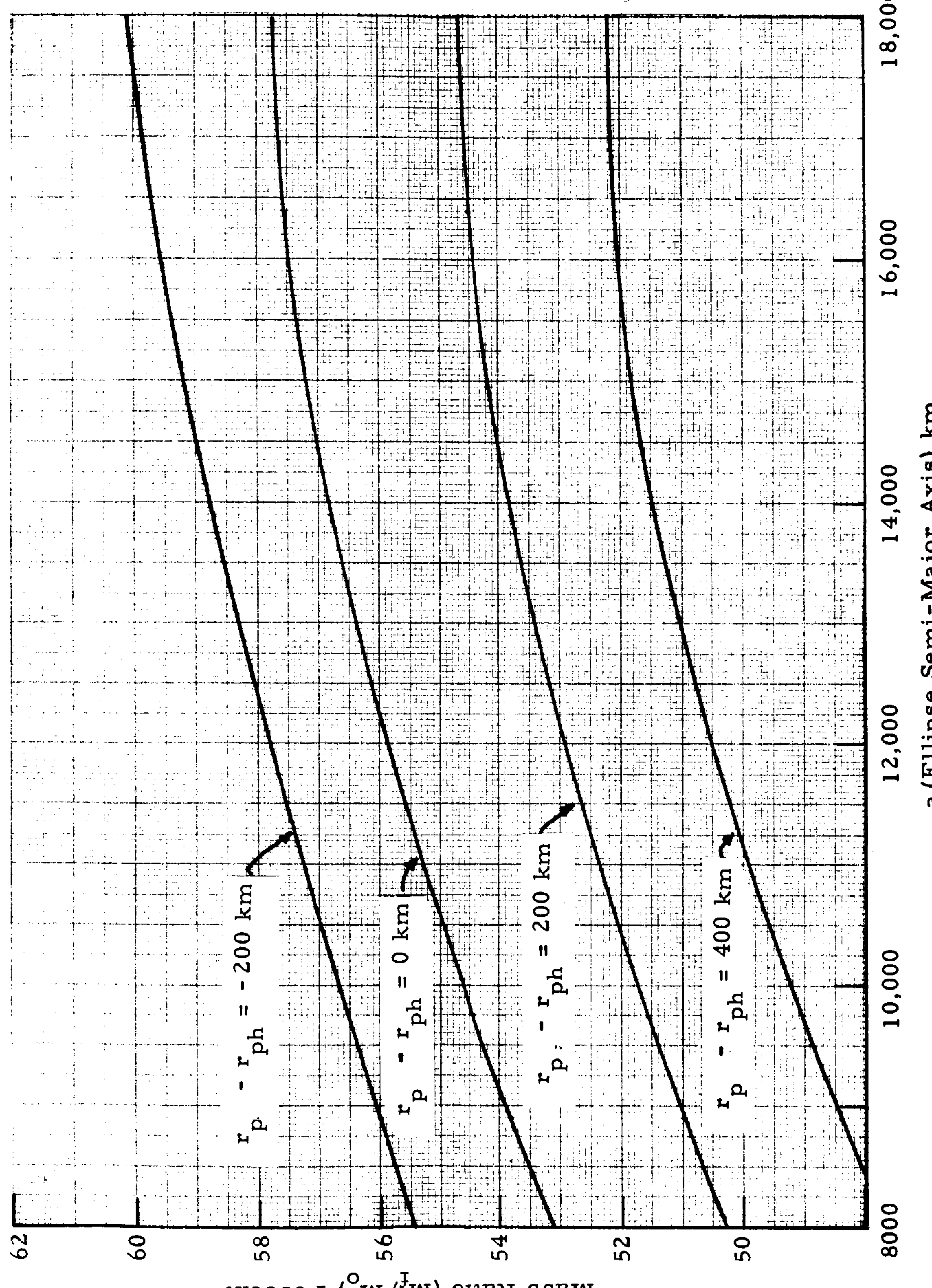


Figure 2-8 - Mass Ratio vs Delta Periapsis ($\phi_R = \pm 20$ degrees)

## Section 3
## SUMMARY OF DEBOOST MANEUVER ORBIT INSERTION ACCURACY

This section summarizes the results of Phase II of the voyager terminal guidance study and the orbit insertion error analysis. As with the variations in performance presented in Phase I, the variations in orbital element errors may be represented as functions of terminal geometry, as well as functions of vehicle and conic parameters.

Nine nominal trajectories were chosen from those generated in Phase I, representing possible transfers of the nominal vehicle from a 10.8 $(km/sec)^2$ specific energy hyperbola into the 1,000 x 20,000 km ellipse. These nominals were chosen with the objectives of obtaining a representative spread of geometric possibilities.

Navigation and execution errors were incorporated into each of these nominals, by means of the DSEA Monte Carlo program described in Reference 5, which repetitively integrates the deboost trajectory, sampling navigation and execution errors for each pass. For a sufficiently large sample size, statistically significant means and standard deviations in the orbital element errors are generated.

It was found that the geometric parameters alpha ($\alpha$) and theta ($\theta_E$) (Figure 2-1) had the largest effect on the sensitivity of orbit element errors to navigation and/or execution errors. Alpha is defined as the angle between the thrust vector and the negative of the velocity vector at deboost ignition, and theta is defined as the anomaly on the ellipse at which insertion is completed. The values of these parameters for the nominal vehicle and conics in terms of the more familiar terminal parameters, periapsis rotation ($\phi_R$) and hyperbola periapsis altitude ($H_{ph}$) are shown in Figures 3-1 and 3-2, respectively.

The sensitivity of argument of periapsis, and orbit period are functions of alpha primarily, while radius of periapsis, argument of the line of nodes, and orbit inclination are mild functions of the insertion anomaly.

Figures 3-3 through 3-7 show the standard deviations of the orbit elements as functions of alpha, or insertion anomaly, and the periapsis altitude of the incoming hyperbola. All vehicle parameters are nominal as defined in Section 2, and the terminal orbit size is the nominal 1,000 x 20,000 km.

Variations in execution error and navigation error were then run. The DSEA program was also applied to deboost trajectories representing variations in vehicle $I_{sp}$ and thrust; hyperbola $C_3$, and terminal orbit.

The results of these runs are summarized in Figures 3-8 through 3-12 in which the deviations in each orbital element are treated separately. Of particular interest is Figure 3-10, which presents the effects on orbit period of parametric variations in the vehicle, in the hyperbola class, and in the navigation and execution error levels. Preliminary accuracy requirements on period seem to be quite stringent (Reference 6) in view of the accuracy that appears attainable with an open loop guidance system.

A simulation of the errors that may be expected with a closed loop guidance system was also run. References 5 and 7 furnish a description of this model which, in effect, reduces the magnitude of the execution errors to that of the guidance platform sensors. Figures 3-13 through 3-17 present the standard deviations for the five orbital elements for several levels of navigation error as obtained from the DSEA Monte Carlo Program. As may be expected, a large improvement in accuracy is obtained when tracking can be continued very close to the ignition point.

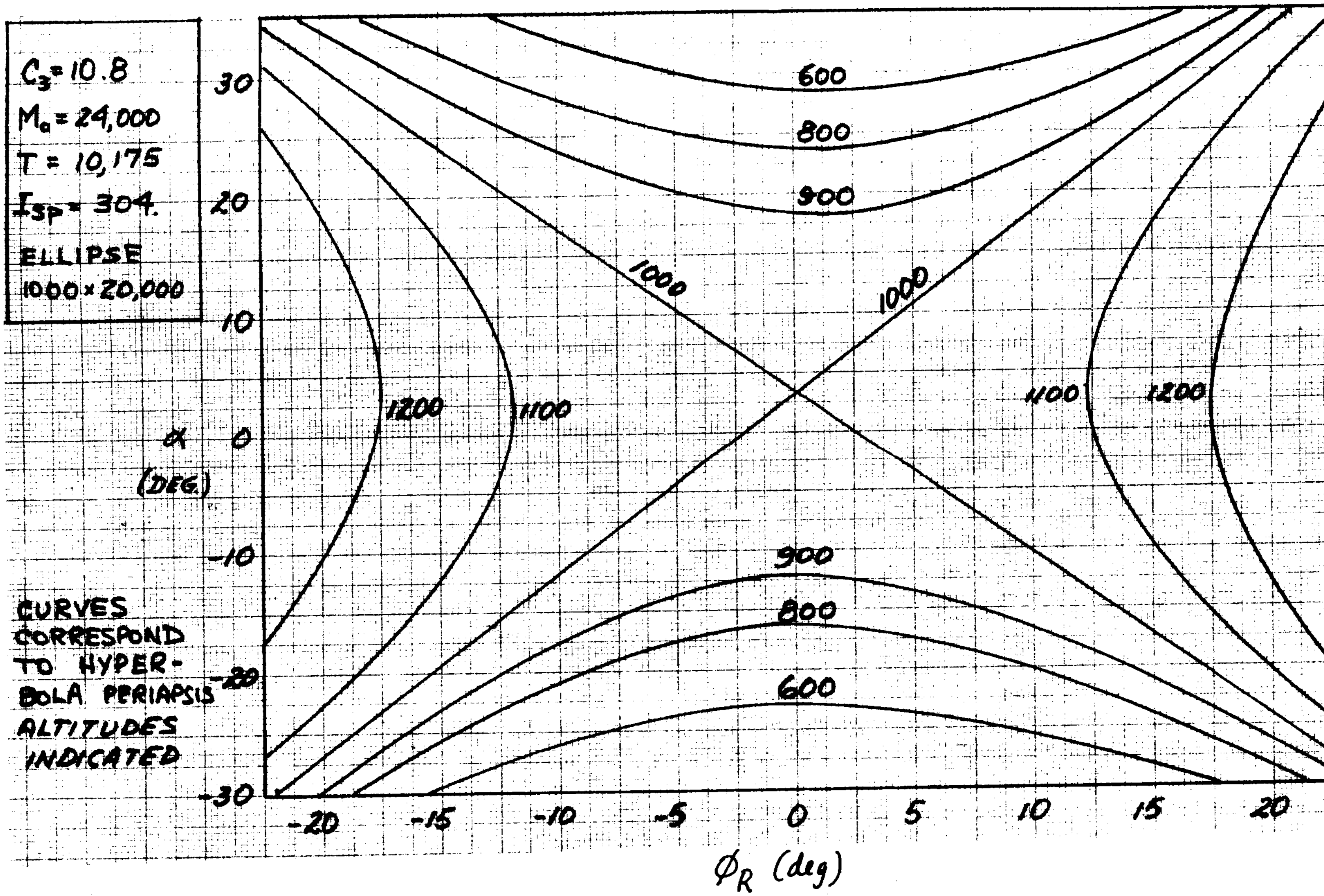


Figure 3-1 - Alpha vs $H_{ph}$ and $\phi_R$

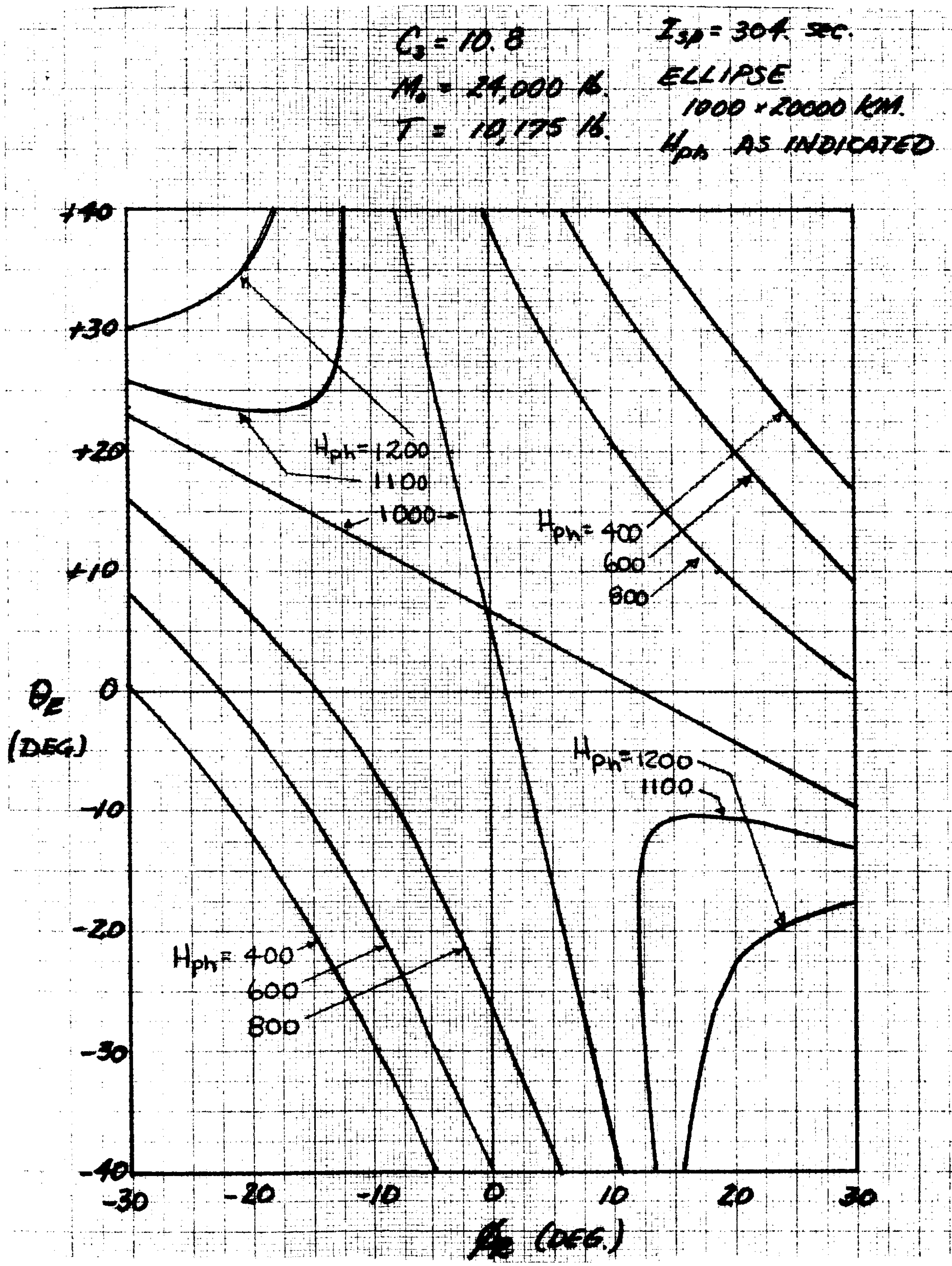


Figure 3-2 - Insertion Anomaly vs Rotation Angle and Hyperbola Periapsis Altitude

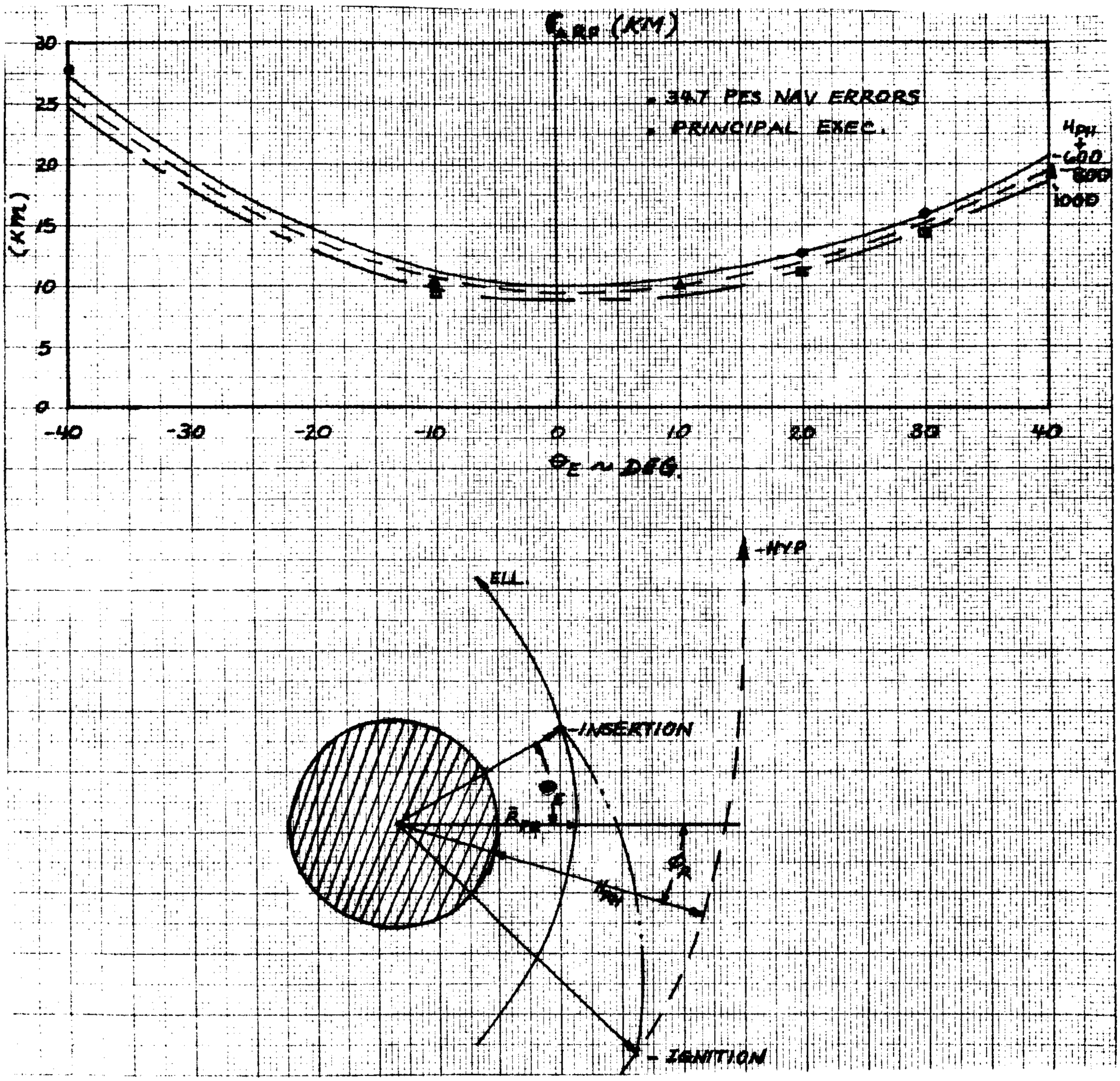


Figure 3-3 - Nominal Vehicle; Radius of Periapsis Standard Deviation vs Insertion Anomaly

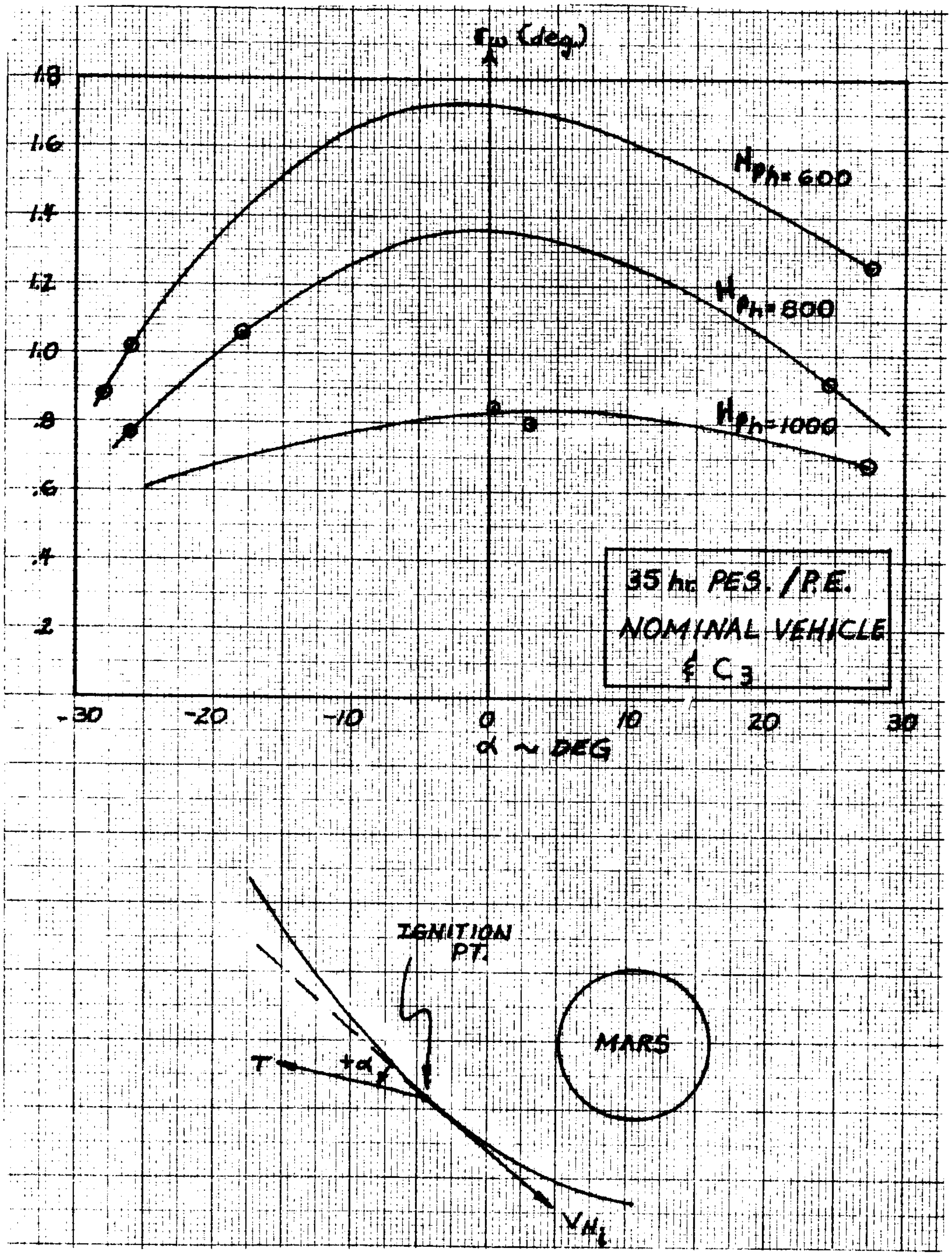


Figure 3-4 - Nominal Vehicle; Argument of Periapsis Standard Deviation vs Alpha

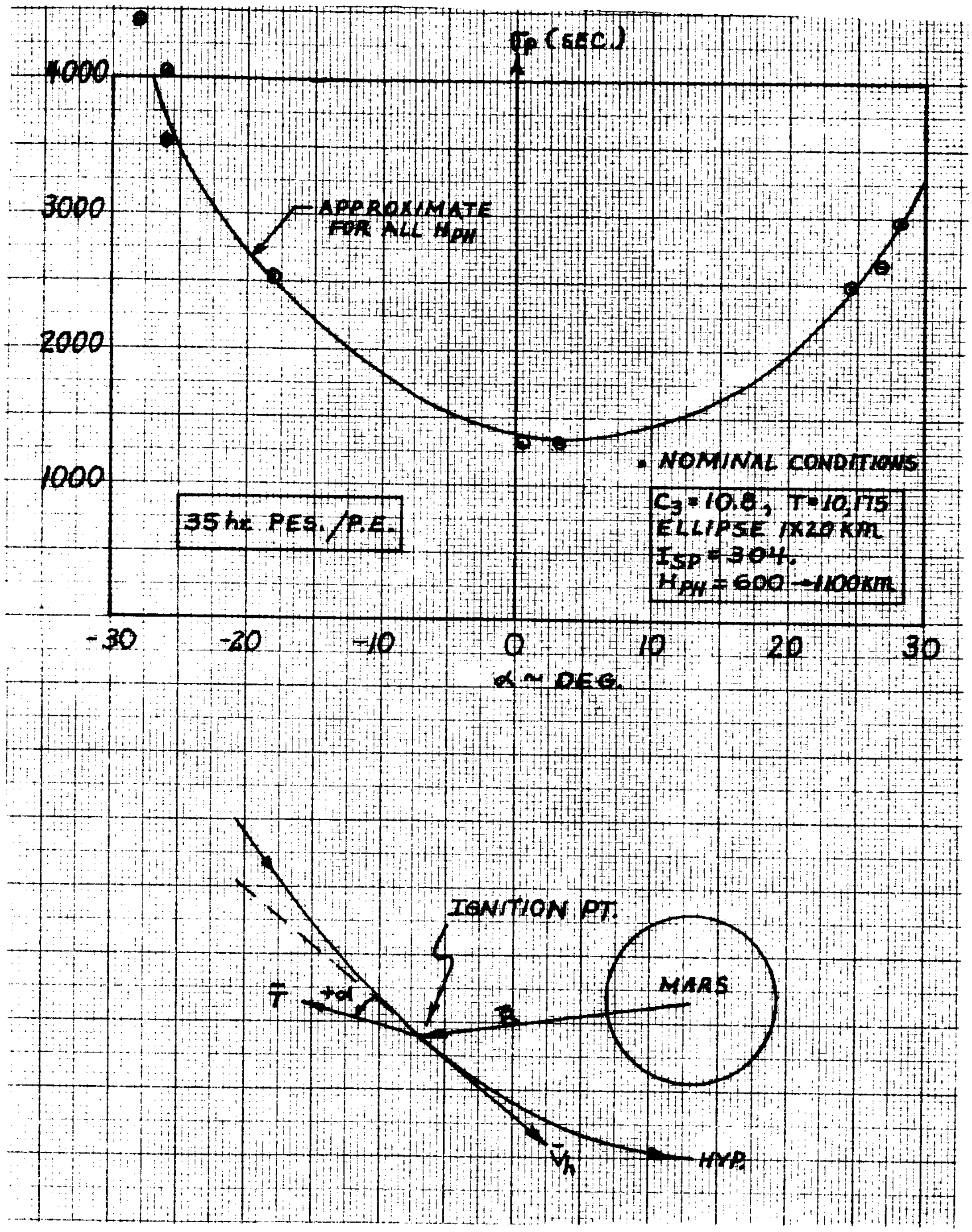


Figure 3-5 - Nominal Vehicle; Period Standard Deviation vs Alpha

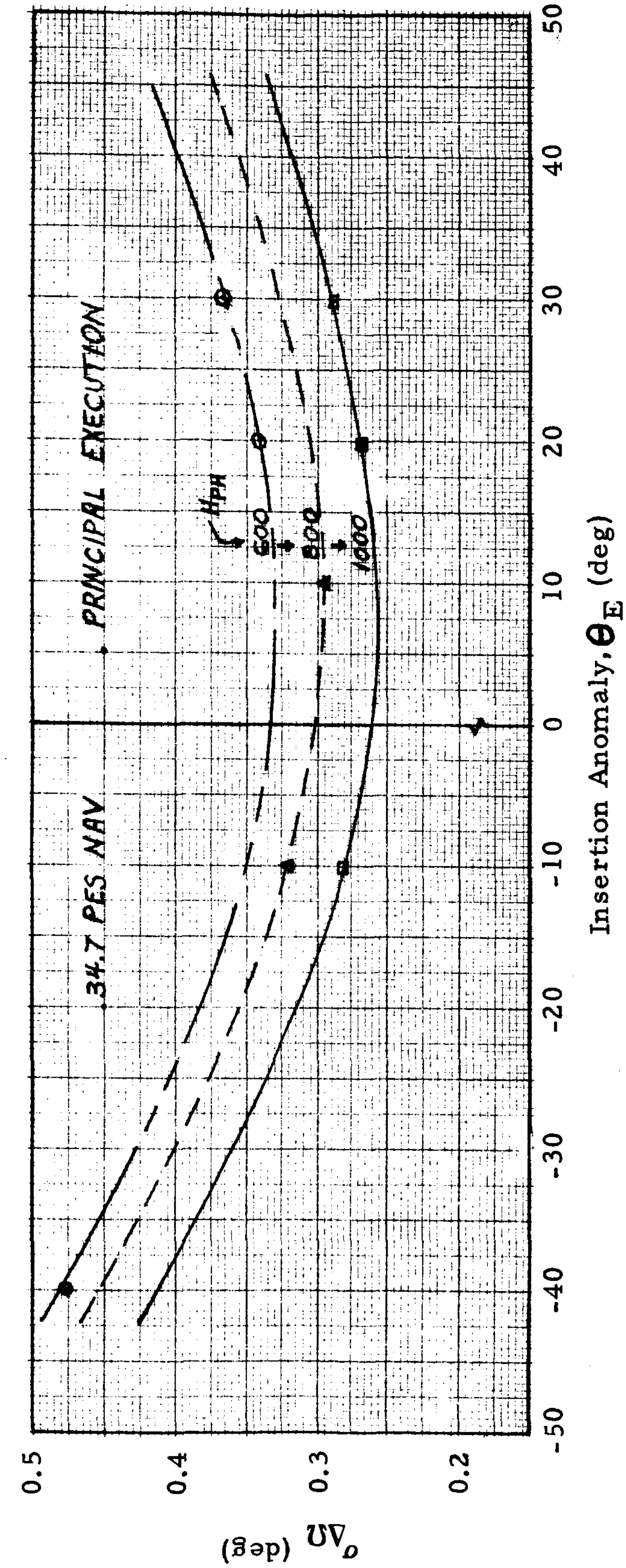


Figure 3-6 - Nominal Vehicle; Line of Nodes Standard Deviation vs Insertion Anomaly

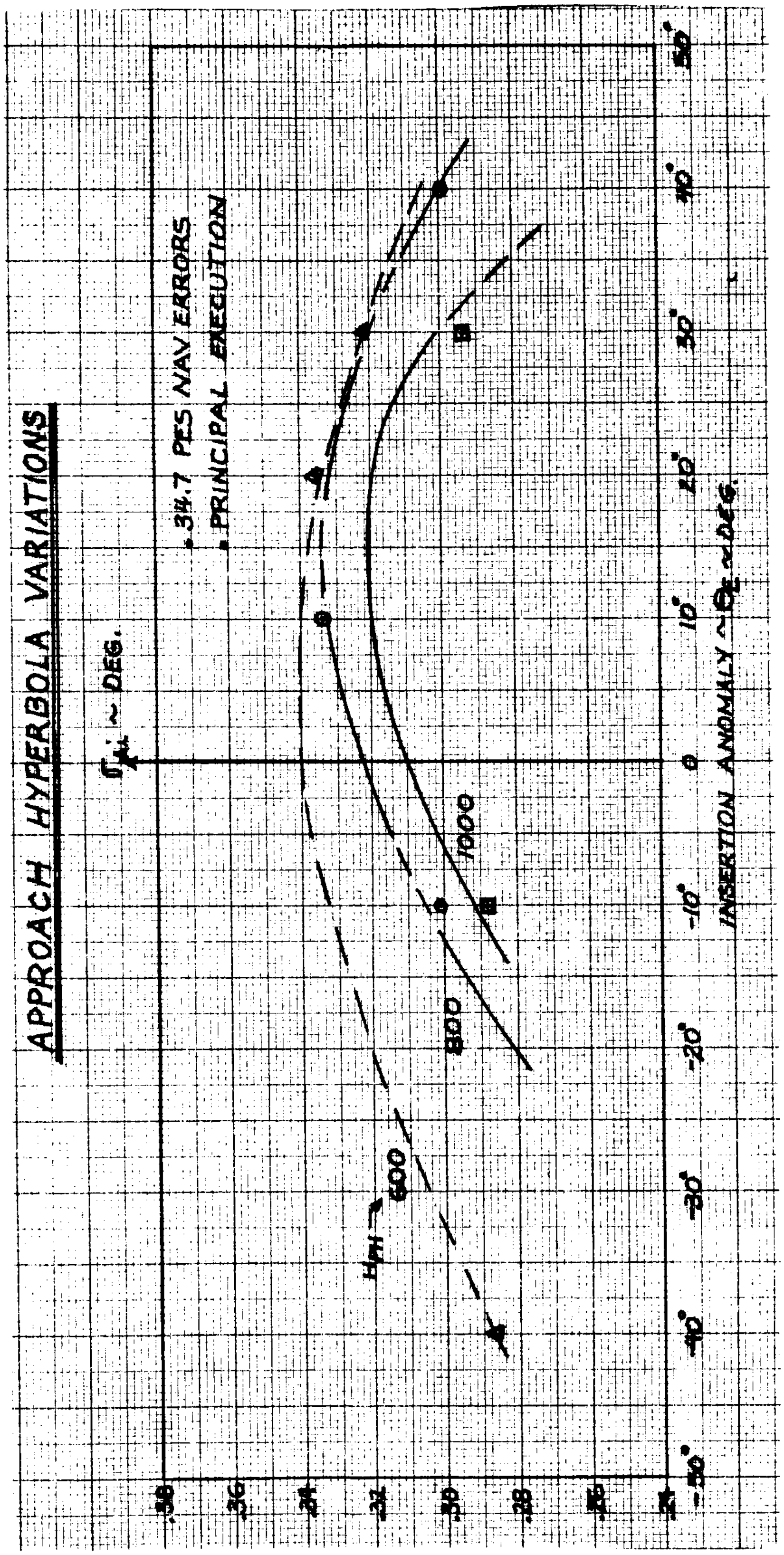


Figure 3-7 - Nominal Vehicle; Inclination Standard Deviation vs Insertion Anomaly

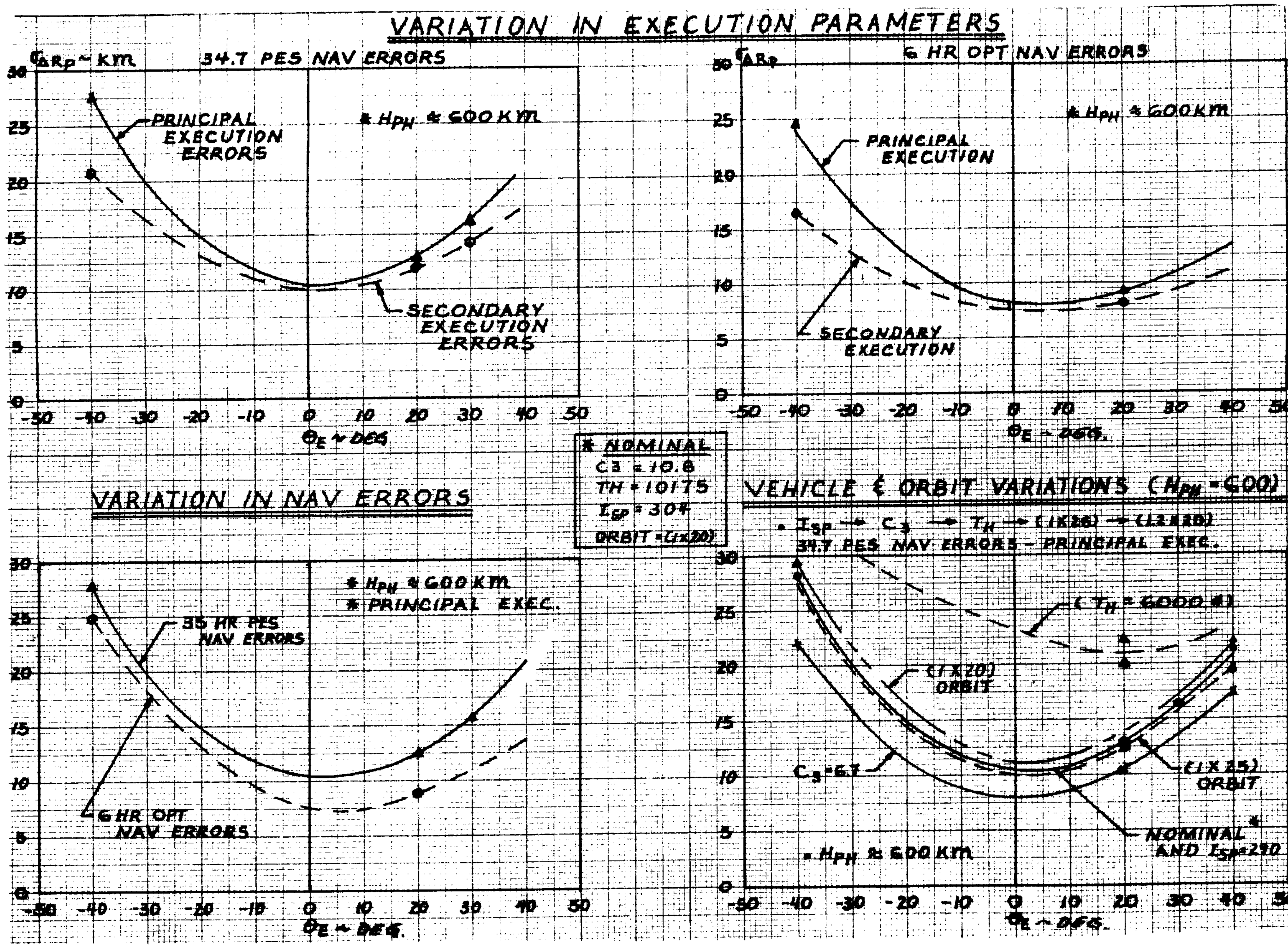


Figure 3-8 - Effects of System Parametric Variations on Radius of Periapsis Error

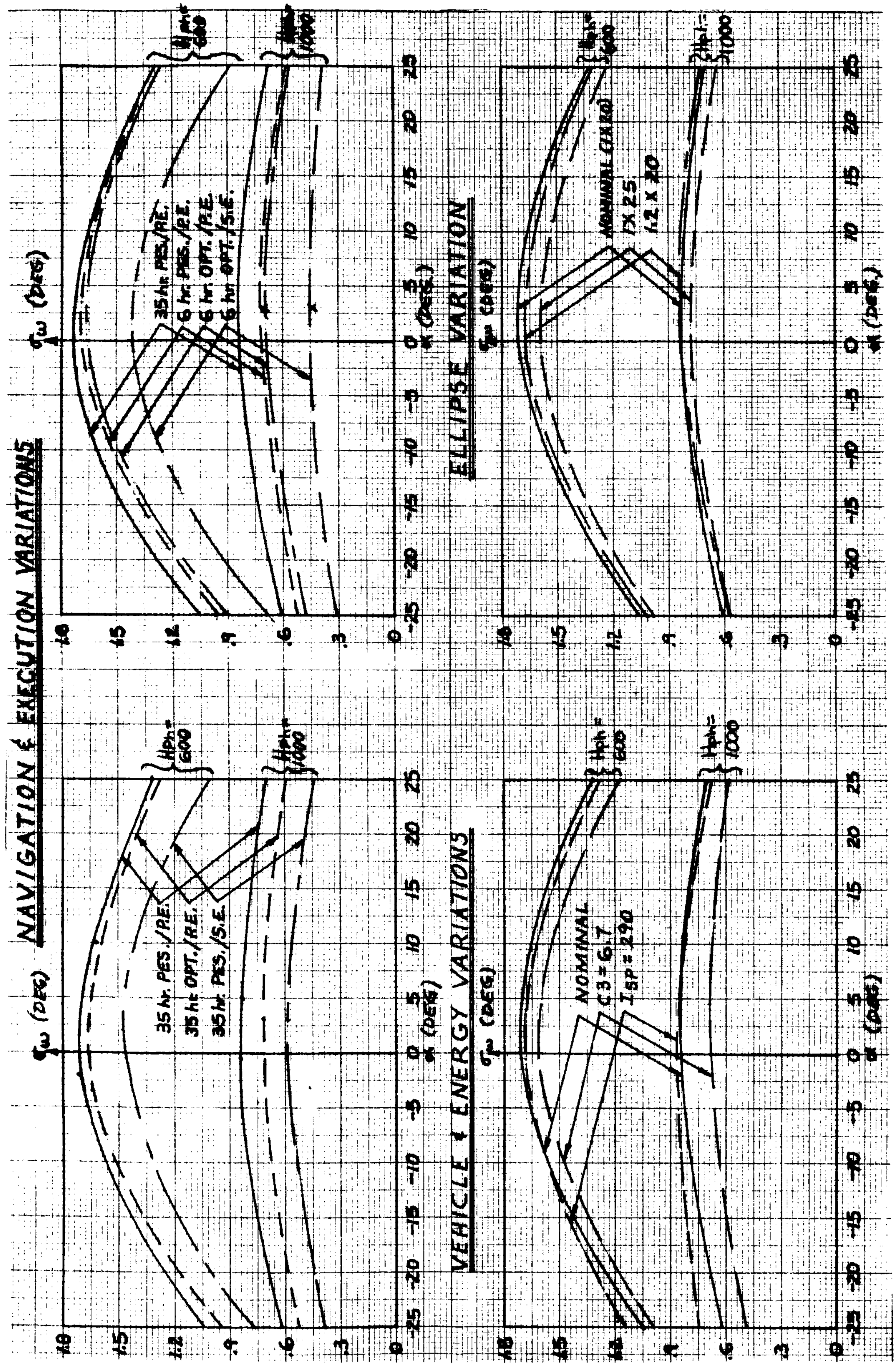


Figure 3-9 - Effects of System Parametric Variations on Argument of Periapsis Error

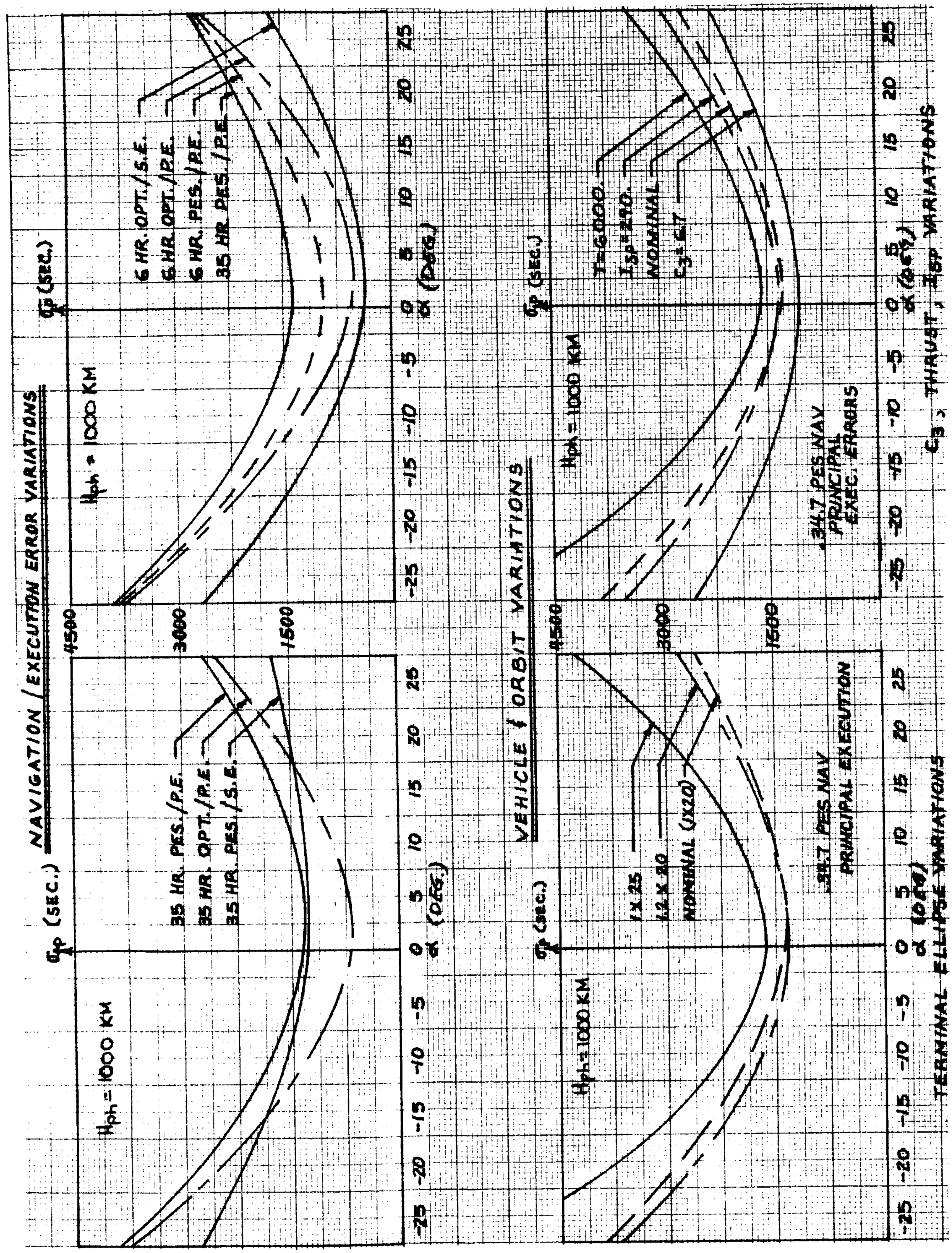


Figure 3-10 - Effects of System Parametric Variations on Period Error

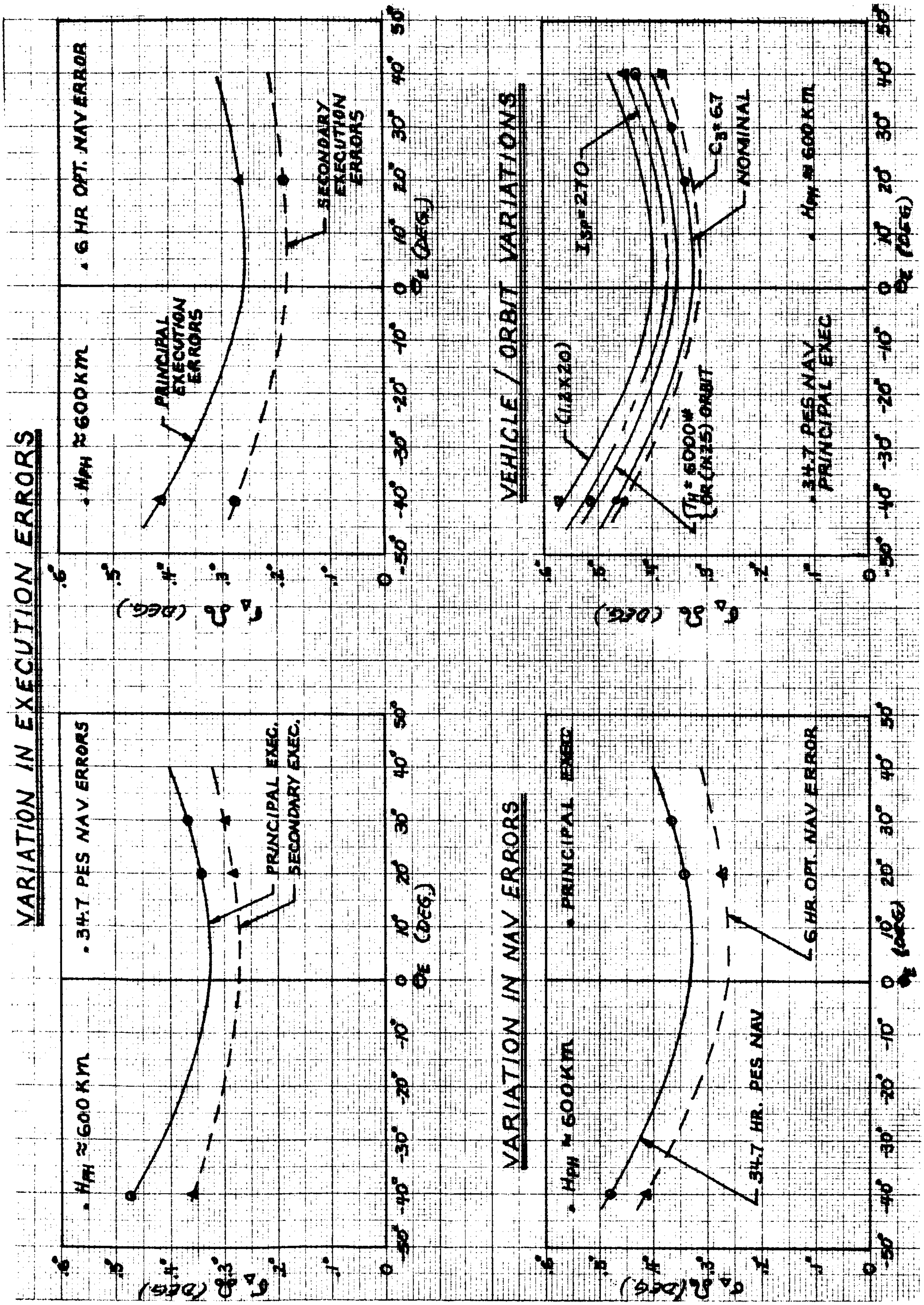


Figure 3-11 - Effects of System Parametric Variations on Line of Nodes Error

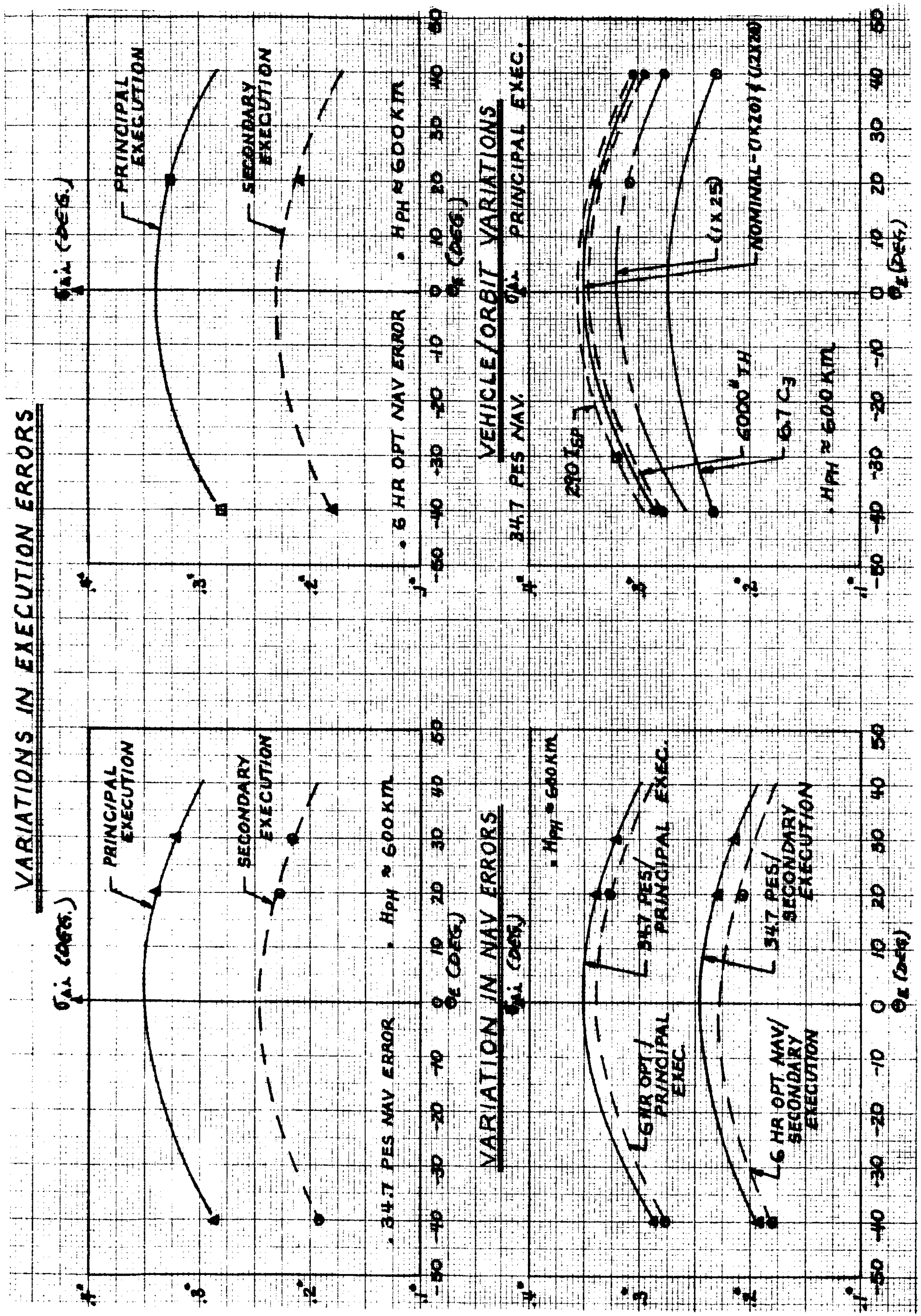


Figure 3-12 - Effects of System Parametric Variations on Inclination Errors

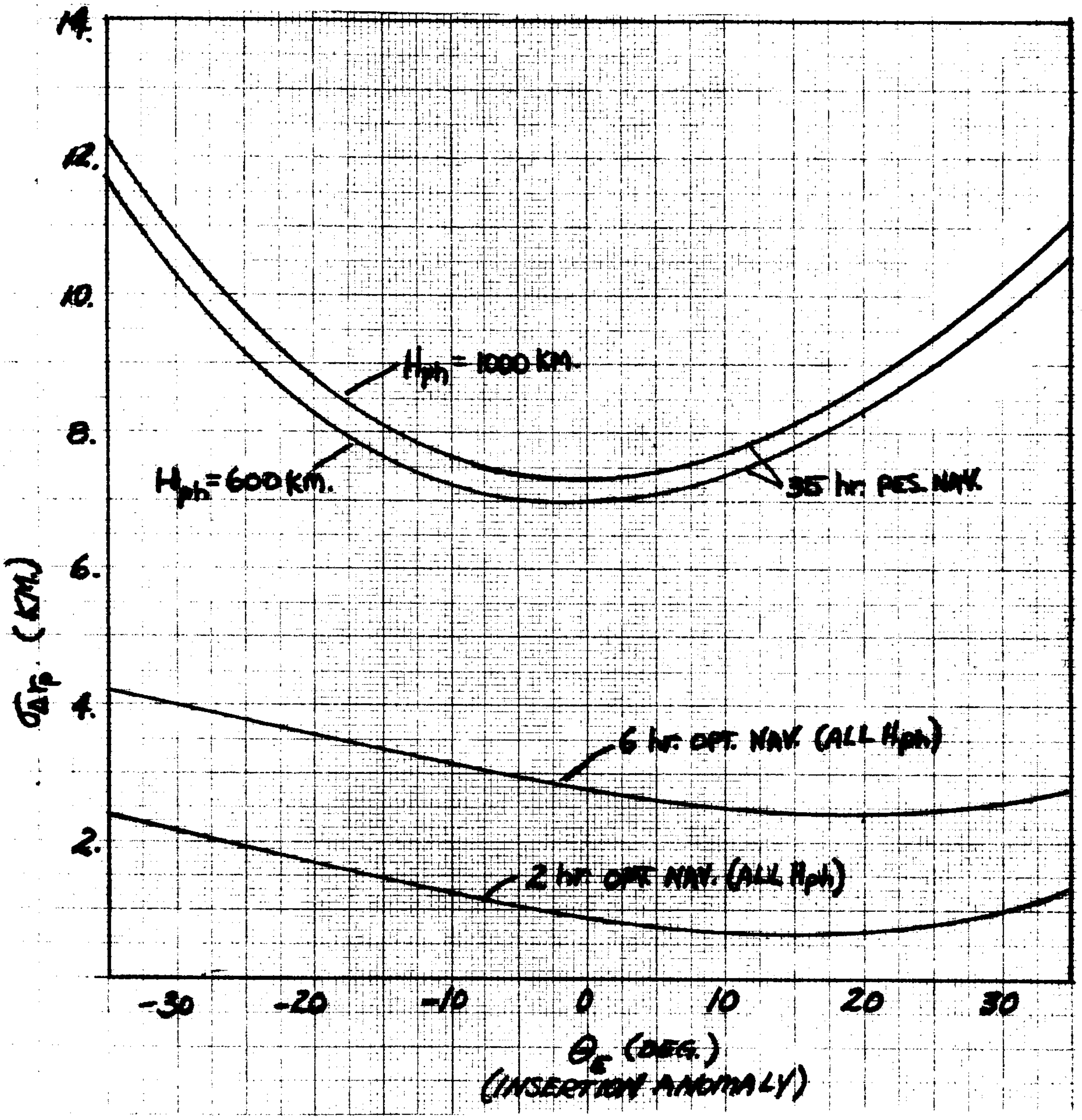


Figure 3-13 - Closed Loop Guidance; Radius of Periapsis Error — Standard Deviations

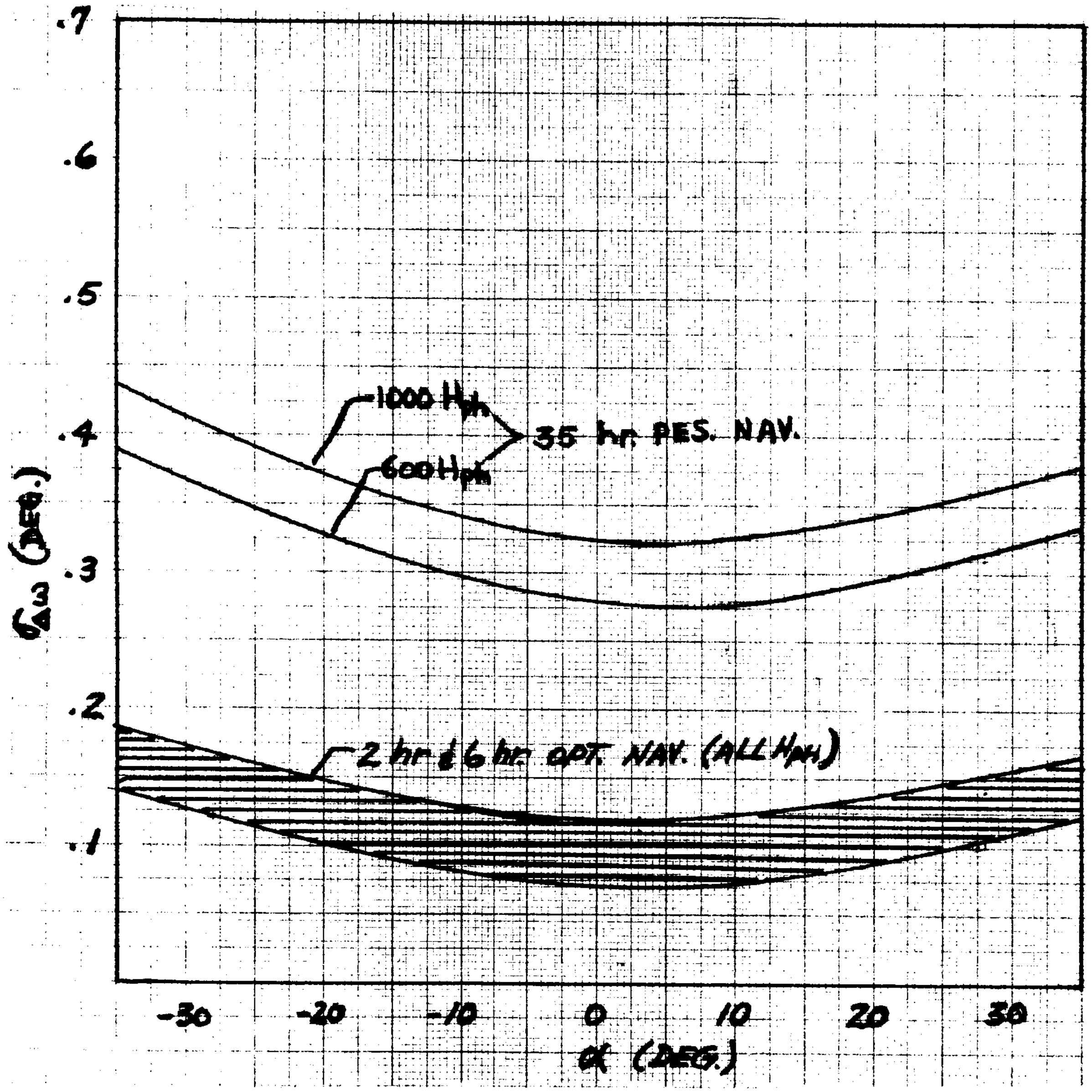


Figure 3-14 - Closed Loop Guidance; Argument of Periapsis Error — Standard Deviations

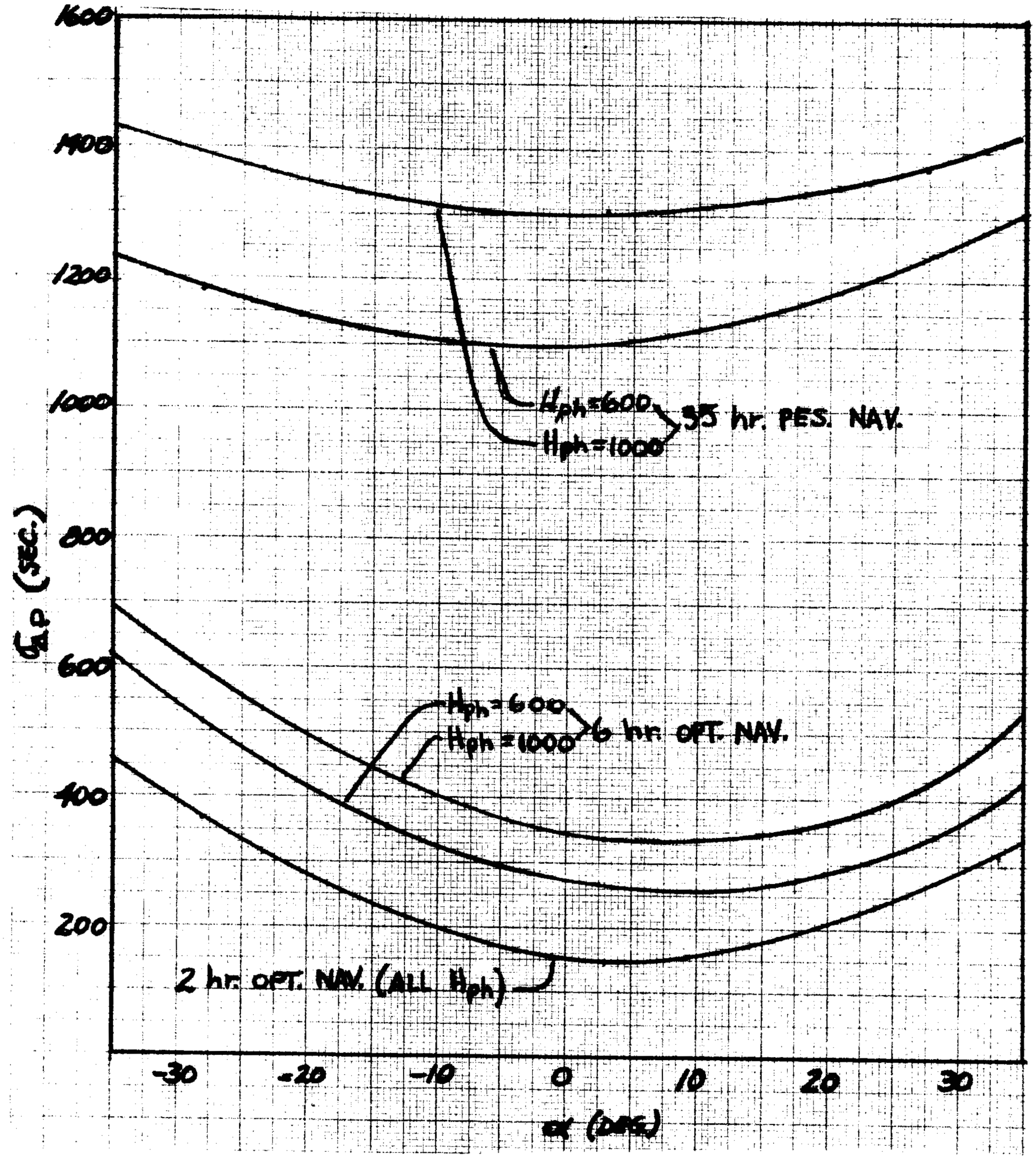


Figure 3-15 - Closed Loop Guidance; Period Error — Standard Deviations

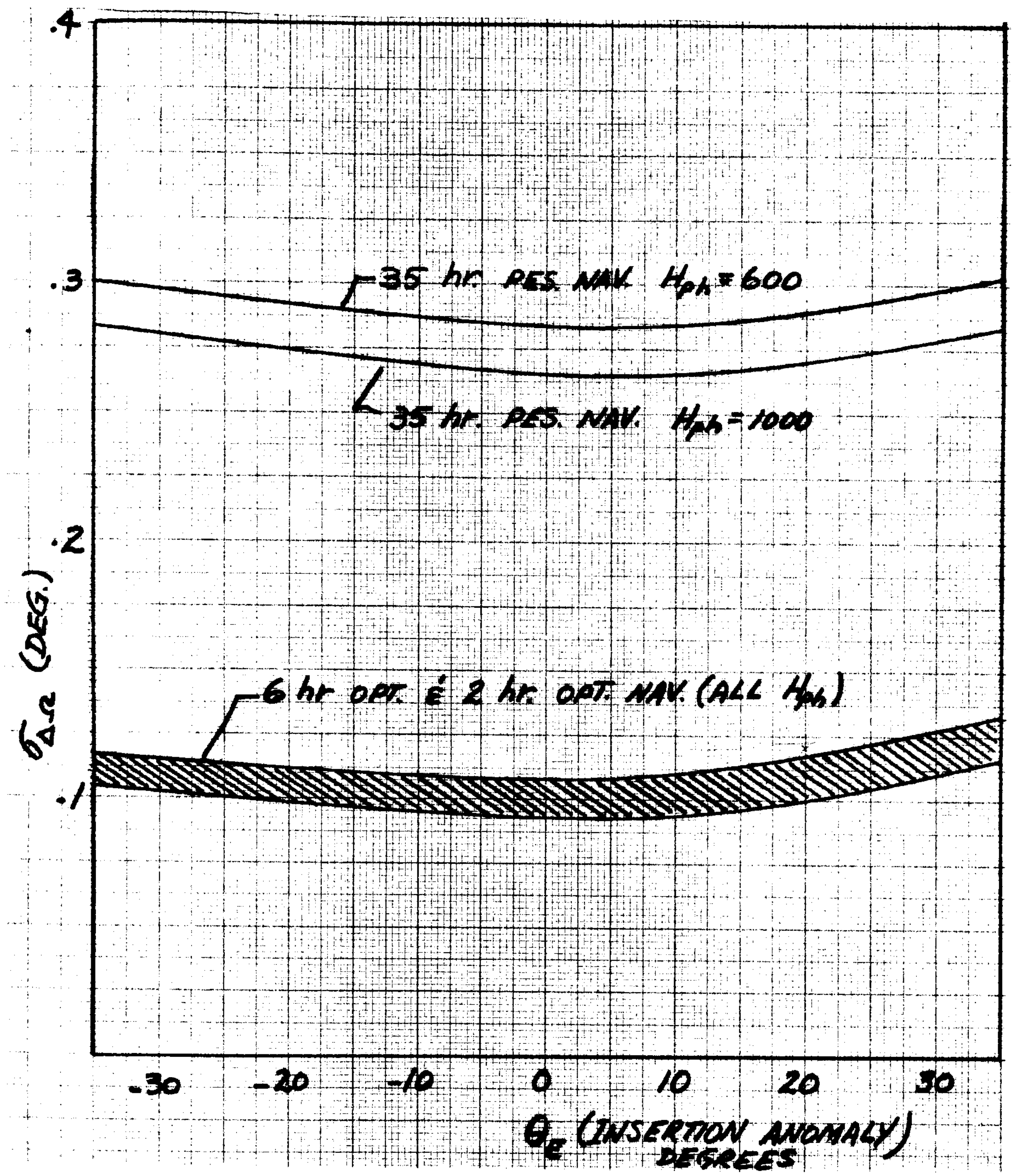


Figure 3-16 - Closed Loop Guidance; Line of Nodes Error — Standard Deviations

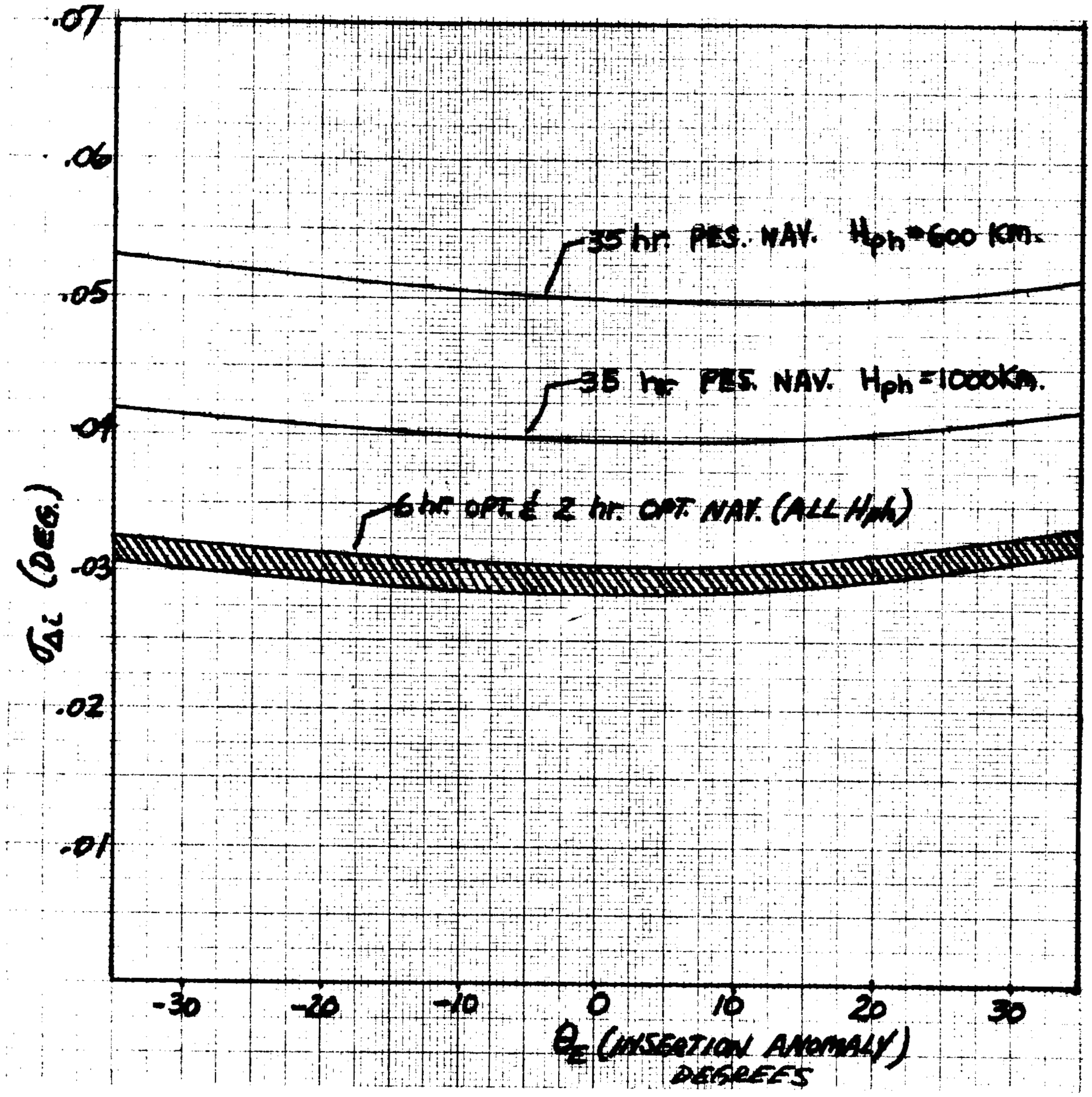


Figure 3-17 - Closed Loop Guidance; Inclination Error — Standard Deviations

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