

Orbit Optimization For The Geospace Electrodynamics Connections (GEC) Mission

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Part of NASA's Solar Terrestrial Probe line of missions, the Geospace Electrodynamics Connections (GEC) mission will deploy a formation of three spacecraft to perform in-situ atmospheric research in the low Ionosphere-Thermosphere region. These spacecraft will fly together in a "string-of-pearls" formation with variable spacings ranging from 10 seconds to one-quarter of an orbit at perigee. Over the course of its two-year mission, the three spacecraft will perform ten, 1-week dipping campaigns whereby they maneuver to lower their perigee to near 134 km. Using available launch vehicle performance data, an optimal parking orbit of 222 x 1525 km was found to maximize the dry mass available while providing enough propellant to perform the ten deep-dipping campaigns over its two-year mission. The results were used to create multi-variable contour plots containing the orbit perigee, the orbit apogee, spacecraft dry mass, propellant mass, and T500 (a science data collection figure of merit that tabulates the cumulative time spent below 500 km). These plots illustrate how the mission can trade off science return relative to the cost in dry mass and propellant. Other optimal solutions such as minimum propellant or maximum T500 were found to either limit the science data collection or to be dry mass limiting, respectively. Sensitivity analyses were performed to find new optimal (maximum dry mass) solutions if the number of campaigns changed, if the coefficient of drag (C_D) were different, and if the propellant specific impulse were increased. A surprising result showed that the dry mass and T500 were both increased if the number of campaigns decreased. Changes in C_D provided the expected results – raising C_D lowered both the dry mass and T500 while lowering C_D raised both the dry mass and T500. Increases in the propellant specific impulse had the expected outcome of raising the dry mass and lowering the propellant load but there was no change in the T500 figure of merit. The orbit optimization was performed parametrically using a Matlab™ script and validated using FreeFlyer™, a commercial orbit analysis tool.

Nomenclature

ΔV	=	Velocity Change
Bc	=	Ballistic Coefficient
C_D	=	Coefficient of Drag
g	=	Gravitational Constant
Ha	=	Apogee Height
Hp	=	Perigee Height
Isp	=	Specific Impulse
T150	=	Cumulative time in orbit spent below 150 km
T500	=	Cumulative time in orbit spent below 500 km

I. Introduction

The Geospace Electrodynamics Connections (GEC) mission is part of NASA's Solar Terrestrial Probe (STP) program managed out of the Goddard Space Flight Center (GSFC). The STP program is part of NASA's Sun-Earth Connections (SEC) theme whose primary goal is to study the Sun and its effects on the Solar System. Among the SEC's major scientific pursuits are determining the origins of Solar variability and how the planets in the solar system react to this variability. The STP line of missions includes TIMED - Thermosphere Ionosphere Mesosphere

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Energetics and Dynamics (launched on December 7, 2001), STEREO - Solar Terrestrial Relations Observatory, Solar-B, MMS - the Magnetospheric Multi-Scale experiment, GEC, and MagCon - the Magnetospheric Constellation mission.

The GEC mission utilizes multiple spacecraft platforms to perform in-situ atmospheric science research in the Earth's Ionosphere-Thermosphere (IT) region. GEC endeavors to answer how the IT region responds to magnetospheric forcing and how this region is dynamically coupled with the Earth's magnetosphere¹⁻³. The GEC spacecraft will examine the IT region using a suite of instruments including electric field detectors (in the form of six orthogonal booms), a magnetometer, an ion mass spectrometer, a neutral wind meter, a neutral mass spectrometer, and energetic particle detectors. The most recent mission to the IT region was the Atmospheric Explorer-C (AE-C) in 1973. AE-C flew an eccentric orbit using maneuvers to lower its perigee to near 130 km at various times during its mission for a total duration at low altitude of roughly one day⁴. GEC will advance the science performed by AE-C by performing multiple low-perigee, atmospheric dipping campaigns for extended durations using multiple platforms. This will allow GEC to further the science performed by AE-C by capturing different temporal and spatial phenomena through the use of multiple spacecraft with variable baselines. As GEC is a "cost-capped" mission within the STP line, the number of spacecraft has yet to be determined. Early concept studies assumed four spacecraft but a three-spacecraft constellation will be used for this analysis.

The three GEC spacecraft will be launched simultaneously on a Delta-II 7920 into an elliptical parking orbit (early designs assumed a perigee of 185 km and an apogee of 2000 km) with an inclination of 83 degrees to provide for optimum viewing of the north and south auroral regions. The spacecraft will be separated into a "string of pearls" formation along the orbit with inter-satellite separations ranging from 10 seconds to a quarter orbit over the course of the 2-year mission. In order to study the deepest IT regions, the constellation will execute maneuvers to lower perigee to nearly 130 km. These deep-dipping campaigns are estimated to last for one week. As can be expected, the low-perigee nature of this mission will incur significant drag penalties on the spacecraft. Therefore, a substantial propulsion system will be needed to lower perigee to start a dipping campaign, raise perigee to end a dipping campaign, and re-boost apogee. Propellant will also be required to control the spacing between the spacecraft but the amount necessary for the constellation control will be greatly dwarfed by the amount required for campaign maneuvers and drag make-up. For this reason, it has been decided to de-couple analysis of the dipping campaign requirements from the analysis of the constellation control. Furthermore, the analysis will be performed on a single spacecraft as each spacecraft will be performing the same maneuvers for the campaigns.

The GEC Program has gone through two previous industry studies through the use of the NASA/Goddard Space Flight Center's Rapid Spacecraft Development Office (RSDO). The two RSDO studies examined the orbit selection strategy from the viewpoint of maximizing the number of dipping campaigns given their best estimate (with margin) of the spacecraft dry mass components. A small amount of optimization was performed relative to the parking orbit perigee height but no attempt was made to characterize the optimal apogee. This study will determine the optimal orbit, in terms of perigee and apogee, for the GEC mission with the goal of maximizing the spacecraft dry mass (i.e. maximizing the science payload) while executing ten, one-week dipping campaigns. This optimal solution will be contrasted against the optimal solution of minimizing the propellant required to perform the ten dipping campaigns. Further analysis will examine the sensitivity of the optimal solution to various mission parameters, in particular the propellant specific impulse, spacecraft coefficient of drag, and the number of campaigns. In comparing solutions, the time spent below a certain altitude will be used as a figure of merit. For this study, the time spent below 500 km will be used as the threshold as a majority of the science will be performed below this altitude.

II. Methodology

A. Assumptions

For the purpose of this study, it was assumed that all three spacecraft were launched simultaneously on a Delta-II 7920-9.5 (i.e. using the 9.5-foot fairing). Performance data was received via the NASA/Kennedy Space Center's Launch Services Contract to support the second RSDO study performed in 2003. The launch vehicle performance data consisted of the delivered mass to orbit with perigee ranging from 185 to 300 km and apogee ranging from 1000 to 4000 km. All data assumed an inclination of 83 degrees. The single spacecraft mass was assumed by taking the total mass delivered by the launch vehicle, subtracting off an adapter mass (assumed to be 200 kg) and then dividing by three - the number of spacecraft delivered to orbit (Eq. 1). A contour plot of the single spacecraft mass as a function of the perigee and apogee heights is shown in Figure 1.

$$M_{\text{Spacecraft}} = (M_{\text{To Orbit}} - 200 \text{ kg}) / 3 \quad (1)$$

The assumed launch date of September 1, 2009 places the GEC mission near the peak of the next solar cycle (Figure 2). Atmospheric density calculations were performed using the Jacchia-Roberts density model^{5, 6} with the +2 σ Schatten F10.7 solar flux prediction from November 2002. In order to simplify the analysis, a constant solar flux that corresponds to the maximum of 205 was used for all cases. This will lead to a conservative solution based on the difference in total drag experienced over the 2-year mission.

The remaining spacecraft parameters that affect the orbital decay are the spacecraft frontal area and the coefficient of drag (C_D). The preliminary design for the GEC spacecraft is for the body to be a long cylinder (or possibly a hexagon or octagon) with a small cross-section (1.2 m² from early concept studies). Unfortunately, the current plan has the electric fields experiment utilizing six, 10-m orthogonal booms. While the booms are less than 30 cm in diameter, the boom length drives the total cross-sectional area for the booms to be over 1.3 m². Therefore, a total cross-sectional area of 2.5 m² was assumed for this study. This number is consistent with the RSDO industry studies. A C_D of 2.5 was selected as the baseline for this study. This C_D value may be highly variable, as GEC will be passing through such high-density regions where there is little observational data. For this reason, a sensitivity study with respect to the C_D was performed. Orbit maneuvers were computed assuming a conservative bi-propellant propulsion system with a specific impulse (Isp) of 285 s. This study also examined how the optimal solution varies as the Isp is increased.

B. Analysis Method

The spacecraft orbit decay was characterized by building look-up tables using high-fidelity simulations. The commercial software product FreeFlyer™ (developed by *a.i. solutions, Inc.* of Lanham, MD) was configured to calculate the single-orbit apogee decay given a perigee height, apogee height, and spacecraft mass. Look-up tables were created at constant perigee height for perigees spanning 185 to 300 km. A look-up table was also created for the dipping altitude of 134 km. This value was used as the dipping altitude because it corresponds to the altitude at which the atmospheric density is 6.7 kg/km³ – the target value for the science. As the mission is actually flown, the dipping altitude will be adjusted (depending on the solar flux level, solar local time, etc.) to achieve the target density. A contour plot of the single-orbit apogee decay at the target density as a function of the spacecraft mass is shown in Figure 3. The apogee decay look-up tables were loaded into Matlab™ (developed by Mathworks, Inc.) for later use.

Matlab™ was used to perform the parametric lifetime simulations. A script was developed which simulated, on an orbit-to-orbit basis, the evolution of the apogee decay, the rotations of the lines of apsides and nodes, the maneuver calculations, and the mass depletion during the parking orbit and dipping campaign phases of the 2-year mission. The script begins by calculating the spacecraft mass given the parking orbit perigee and the initial apogee values. Then, all parameters are advanced by a single orbit. Time is advanced based on the orbit period. The figure of merit T500 is calculated based on Kepler's equation⁷. Two-dimensional interpolation is used to determine the apogee decay based on the current apogee and mass. Finally, the right ascension of the ascending node (Ω) and the argument of perigee (ω) are adjusted based on the Earth's J2 perturbations⁷. These updates are continuously updated until a dipping campaign is required – 60 days for 10 campaigns. At this point, maneuvers are calculated to both raise apogee back to its initial parking orbit value and then lower perigee to the dipping height of 134 km. Then, all parameters are adjusted as during the parking orbit period except that the apogee decay is governed by a different look-up table. The dipping campaign continues for 7 days at which point maneuvers are computed to raise both perigee and apogee back to their parking orbit values whereupon the parking orbit cycle begins again. This sequence is repeated until all 10 campaigns are executed during the two-year mission duration. At the end of the script, the final mass is considered to be the “dry mass” of the spacecraft (after all propellant is used). This “dry mass” consists of both the spacecraft bus components (e.g. attitude control hardware, communications hardware, propulsion hardware, etc.) and instrument mass. In reality, some margin will have to be maintained but it is not accounted for in this study. The propellant mass, total ΔV , and cumulative T500 are also computed at the end of this script. Figure 4 shows the apsis history for a sample with parking orbit parameters fixed with a perigee at 200 km and apogee set at 2000 km.

III. Results

The process described above is for a single case - a single initial orbit and initial spacecraft mass. In order to optimize the problem, the simulation was repeated over the entire range of launch vehicle performance with perigee ranging from 185 to 300 km and apogee ranging from 1000 to 4000 km. A coarse scan with perigee increments of 5 km and apogee increments of 50 km was performed and data were tabulated at the completion of each individual 2-year simulation.

The composite simulation was executed in Matlab™ and allowed to run until completion (roughly 2 days of processor time). At the end, matrices of the important parameters (dry mass, propellant mass, & T500) were examined to see if there were any optimal solutions. Examining the contour plot for the spacecraft dry mass (Figure 5) we see a definite maximum. Closer inspection shows that a maximum dry mass of 610.7 kg is achieved with an orbit of 222 x 1525 km. At this optimal solution (maximum dry mass found from the ranges of launch vehicle performance that were studied), the propellant load is 407.6 kg and the T500 figure of merit is 215.5 days (roughly 30% of the 2-year mission). In Figure 6, we see that the minimum propellant case occurs at the highest apogee evaluated, 4000 km. The orbit perigee for this solution of 216 km is slightly lower than that for the maximum dry mass solution. Unfortunately there is a 77 kg (12%) penalty in available dry mass and a very large 111-day (51%) penalty in time spent below 500 km. Figure 7 shows the expected conclusion that the maximum T500 occurs at the lowest perigee and apogee values examined - 185 km and 1000 km, respectively. At this point, T500 is over 380 days - over half the mission is spent below 500 km. This benefit comes at a cost as the available spacecraft dry mass has dropped to 483 kg and the propellant load has increased to over 610 kg. This solution is clearly not attractive because of the low dry mass and large propellant requirements.

Table 1: Summary of Optimal Solutions for 10 Campaigns

Mission Parameter	Maximum Dry Mass	Minimum Propellant	Maximum T500
Hp (km)	222	216	185
Ha (km)	1525	4000	1000
Dry mass (kg)	610.7	532.9	483.2
Propellant Mass (kg)	408.8	255.8	610.4
T500 (days)	215.5	104.4	380.2

All three data sets have been combined into a complex contour plot in Figure 8. This figure can be used in making system-level trades relative to the effects of changing the parking orbit (perigee and apogee) and how that cascades into changes in the dry mass and propellant load. Then, the return on the investment is seen in the form of T500. It is apparent from Figure 6 and Figure 7 that decreasing the parking orbit perigee changes the propellant required and T500 minimally. Lowering the parking orbit apogee can cause greater changes in T500. For instance, in order to gain an extra month (4 weeks) of science, the parking orbit apogee can be reduced from the optimal (maximum dry mass) location of 1525 km to 1330 km, while keeping perigee at 222 km. The effects of this change are to lower the available dry mass by 2 kg while raising the propellant required (due to a smaller initial mass and therefore a lower ballistic coefficient) by 27 kg. This plot will be very helpful in determining how to maximize science return (i.e. T500) while allowing for sufficient dry mass and propellant storage capability.

IV. Model Validation

The analytical simulations were validated using a script developed in FreeFlyer™ that employed full perturbations. The script used the same initial conditions (mass and parking orbit) and followed the same control strategy for executing the ten dipping campaigns. The orbit was advanced using a 9th-order Runge-Kutta propagator along with a 4x4 geopotential model along with Sun & Moon perturbations. The results from this simulation compare very favorably with the analytic simulation. In the evaluation of the optimum, maximum dry mass case (with an orbit of 222 x 1525), the FreeFlyer™ dry mass following the 10 campaigns was 621.3 kg - 10.6 kg (1.7%) higher than the analytic Matlab™ solution. This corresponds to an equivalent 10.6 kg decrease in propellant mass for the FreeFlyer™ solution, as both simulations started with the same initial mass. Graphically, the apsis history for both perigee and apogee is shown in Figure 9 where the small symbols representing the Matlab™ solution

superimposed on top of the large symbols representing the FreeFlyer™ solution. A small scan around the FreeFlyer™ solution showed the maximum to occur at a slightly smaller perigee of 219 km with a small 0.2 kg increase in the dry mass to 621.5 kg. The small difference between the two methods is likely from the different density seen by the FreeFlyer™ profile as it is responding to the “true” solar flux estimate (with proper adjustments made for the local solar time) while the Matlab™ profile is keeping the flux constant through every single perigee pass. While the curves in Figure 9 match very well, there is a big deviation between the 4th and 5th dipping campaign where the FreeFlyer™ simulation experiences significantly less apogee decay (almost 40 km) than the Matlab™ simulation. In fact, careful examination of the apogee decay during the parking orbit show the Matlab™ solution to be more conservative in all but one case. Hence, the Matlab™ simulation uses more ΔV (i.e. propellant) each cycle and as a result maintains a slightly lower ballistic coefficient for the remainder of the simulation.

V. Solution Sensitivity

Once an optimal solution was obtained for the maximum dry mass, a sensitivity study was performed to determine how the optimal solution changed if different parameters changed. Sensitivity studies were performed by varying the number of campaigns, the Coefficient of Drag, and the propellant specific impulse.

A. Number of Campaigns

For this study, the number of 1-week dipping campaigns was varied from 8 to 12 in order to gauge this effect on the optimal (maximum dry mass) solution. As was done above, an entire simulation was performed with the perigee ranging from 185 to 300 km and the apogee ranging from 1000 to 4000 km and contour plots were generated to determine where the maximum dry mass occurred. The important mission parameters at the optimal solution were gathered and presented in Table 2 and in Figure 10.

The first observation is that as the number of campaigns rises from 8 to 12, the parking orbit perigee decreases while the parking orbit apogee rises. Also, as the number of campaigns grew, the dry mass decreased while the propellant mass increased as expected. The most surprising result is that as the number of campaigns grew the figure of merit T500 decreased. This results from an examination of the two optimal orbits for the 8- and 12-campaign solutions. The 8-campaign parking orbit of 230 x 1350 km spends 3 minutes more time below 500 km per orbit than the 12-campaign parking orbit of 215 x 1710. It is then easy to see why T500 is greater in the 8-campaign case when that 3-minute differential is multiplied by over 10,000 over the 2-year mission duration.

A new figure of merit called T150 (the cumulative time spent below 150 km) has been introduced to gauge the consequences of science data capture during the dipping phase of the mission. As expected, T150 rises as the number of campaigns rises, though the rise is small. In fact, examining the dipping orbits for the extreme cases shows that the 8-campaign case (at 134 x 1350 km) spends nearly 1 minute per orbit more below 150 km than the 12-campaign case (at 134 x 1710 km) though this is offset by the greater number of dipping orbits for the 12-campaign case (almost 400 more).

Table 2: Sensitivity of Mission Parameters to Number of 1-week Dipping Campaigns for Maximum Dry Mass Solution

Mission Parameter	Number Of Campaigns				
	8	9	10	11	12
Hp (km)	230.0	227.0	222.0	218.0	215.0
Ha (km)	1350.0	1410.0	1525.0	1625.0	1710.0
Mdry (kg)	675.3	641.9	610.7	581.8	554.4
Mprop (kg)	366.2	391.3	408.8	426.4	444.1
T500 (days)	235.4	227.4	215.5	206.5	199.6
T150 (days)	4.0	4.4	4.6	4.8	5.1
# Dipping Orbits	832	930	1021	1112	1203

This sensitivity study will be very useful when discussing with the GEC science team the trade-offs in the number of campaigns.

B. Coefficient of Drag

As mentioned above, very few missions have performed in-situ research in the deep IT region and knowledge of the gas-surface interactions when the spacecraft pass through this high-density region will be a great unknown. Therefore, the spacecraft C_D was varied by $\pm 10\%$ from its nominal value of 2.5 to examine the effect on the optimal, 10-campaign solution.

Table 3: Sensitivity of Mission Parameters to Coefficient of Drag for Maximum Dry Mass Solution

Mission Parameter	CD		
	-10% 2.25	Nominal 2.5	+10% 2.75
Hp (km)	220.0	222.0	225.0
Ha (km)	1400.0	1525.0	1650.0
Mdry (kg)	637.3	610.7	585.7
Mprop (kg)	398.2	408.8	418.4
T150 (days)	4.8	4.6	4.4
T500 (days)	230.4	215.5	202.3

The results from the C_D sensitivity study could also be applied to a change in another mission parameter – the spacecraft frontal area. Since the ballistic coefficient is inversely proportional to both C_D and frontal area (Eq. 2), a change in C_D (while keeping area fixed) is equivalent to the same magnitude change in area (while keeping C_D fixed). In previous studies of GEC, there was some uncertainty in some of the design parameters for the electric field booms. The possibility existed of obtaining smaller diameter booms or shortening the booms, both of which would lower the spacecraft frontal area.

$$B_c = \frac{\text{Mass}}{C_D * \text{Area}} \quad (2)$$

C. Specific Impulse

For spacecraft that require orbit maneuvers, the propellant specific impulse (Isp) is an important parameter. The specific impulse is defined as the thrust produced per unit of propellant flow rate⁷. A rearrangement of the rocket equation shows the relationship between the propellant mass and the Isp, given a constant ΔV Eq. (3).

$$M_{\text{Prop}} = M_{\text{Initial}} \left[1 - \exp\left(\frac{-\Delta V}{g I_{\text{sp}}}\right) \right] \quad (3)$$

So, for a constant ΔV , an increase in the Isp will lower the propellant required for a maneuver. For GEC, this can have a great benefit for as less propellant is used per maneuver, the spacecraft mass will be greater allowing it to pass through its high-density perigee passes with less apogee decay.

All of the previous studies have been performed using a conservative bi-propellant Isp of 285 s. An examination of future thruster design showed that Isp's of up to 310 s could be obtained with advances in combustion chamber materials. Table 4 shows the sensitivity of the key mission parameters to the propellant Isp for the 10-campaign simulation.

Table 4: Sensitivity of Mission Parameters to Propellant Specific Impulse for Maximum Dry Mass Solution

Mission Parameter	Isp (s)			
	285	295	305	310
Hp (km)	222	223	222	222
Ha (km)	1525	1425	1430	1446
Mdry (kg)	610.7	623.6	636.0	641.9
Mprop (kg)	408.8	408.2	395.4	390.8
T150 (days)	4.6	4.8	4.8	4.8
T500 (days)	215.5	227.0	227.1	227.9

It appears that changes in the propellant Isp only affect the dry mass and propellant mass. There is essentially no change in the optimal orbit for the maximum dry mass solution. The difference in the optimal apogee for the 285 s Isp relative to the other cases was puzzling until it was observed that the results are fairly flat. Examining the 285 s Isp results at a similar 222 x 1425 km orbit we see that the dry mass is 610.5 – only 0.2 kg different from the maximum. There is also no change observed in the key figures of merit T150 and T500. This makes sense as these values are mostly affected by a change in apogee and all four cases presented are at nearly the same apogee.

VI. Conclusions

An optimal orbit was determined for the GEC mission given the available launch vehicle performance and the requirement to perform 10, 1-week dipping campaigns over the course of a 2-year mission. The optimal orbit of 222 x 1525 km was found to maximize the available dry mass necessary for the science payload and the spacecraft subsystems. A minimum propellant solution was found to be too limiting for science collection while a maximum T500 solution was too limiting in available dry mass. Sensitivity studies were performed to determine how the optimal solution (maximum dry mass) varied with respect to the number of campaigns, C_D , and propellant Isp. A surprising result showed that reducing the number of campaigns both raised the available dry mass and increased the T500 figure of merit. Changes in C_D produced the expected results – a smaller C_D raised the dry mass and T500 and vice versa. It was also determined that the C_D sensitivity study was equivalent to projecting changes in the spacecraft frontal area. This area change could be the result of advances in technology (thinner booms) or a potential science de-scope (reductions in the length of the booms). Finally, changes in the Isp seemed only to affect the dry mass and the propellant mass. The T150 and T500 figures of merit were not affected.

There is more work to be done for this problem. Spacecraft contact time (to a ground station) could also be included as a figure of merit. The solutions found above tended to have lower apogee values – which mean shorter ground contact opportunities. Future work on this problem will be to examine alternate dipping scenarios. These scenarios could include allowing apogee to decay constantly until a limit was reached. This would hopefully take advantage of the higher apogee cases where spacecraft contact times to a ground station are greater. Another option includes examining the concept of staging. In this case, the spacecraft are delivered to an orbit with the lowest apogee (1000 km) and spacecraft propellant is used to raise apogee to the parking orbit value. Early results show some increases in dry mass. It is very apparent that there is plenty of additional analysis to be done for the GEC mission.

VII. References

¹Grebowsky, J., et al., "Understanding Plasma Interactions with the Atmosphere: The Geospace Electrodynamic Connections (GEC) Mission", NASA/TM—2001-209980, July 2001.

²Sojka, J. J., et al., "GEC: A Mission to the Threshold of Space - Updated Overview of the Report of the NASA Science and Technology Definition Team for the Geospace Electrodynamic Connections Mission", May 2004.

³Grebowsky, J., and Buchanon, R., "Ongoing Evolution of the Geospace Electrodynamic Connections Mission", 53rd International Astronautical Congress, IAA-02-IAA.11.1.05, Houston, TX, October 2002.

⁴Barbieri, R., and Cooley, J. L., "Atmosphere Explorer (AE-C) Adaptive Mission Design and Trajectory Performance", AAS 75-052, 1975.

⁵Jacchia, L. G., "Revised Static Models of the Thermosphere and Exosphere with Empirical Temperature Profiles", Smithsonian Astrophysical Observatory Special Report No. 332, Cambridge, Massachusetts, May 1971.

⁶Roberts, E. R., Jr., "An Analytic Model for Upper Atmosphere Densities Based Upon Jacchia's 1970 Models", *Celestial Mechanics*, 4 (3/4), December 1971, pp. 368-377.

⁷Brown, C. D., "Spacecraft Mission Design – Second Edition", AIAA Education Series, AIAA, Reston, VA, 1998.

VIII. Related Web Sites

Solar Terrestrial Probes: <http://stp.gsfc.nasa.gov/>

Sun-Earth Connection: <http://sec.gsfc.nasa.gov/>

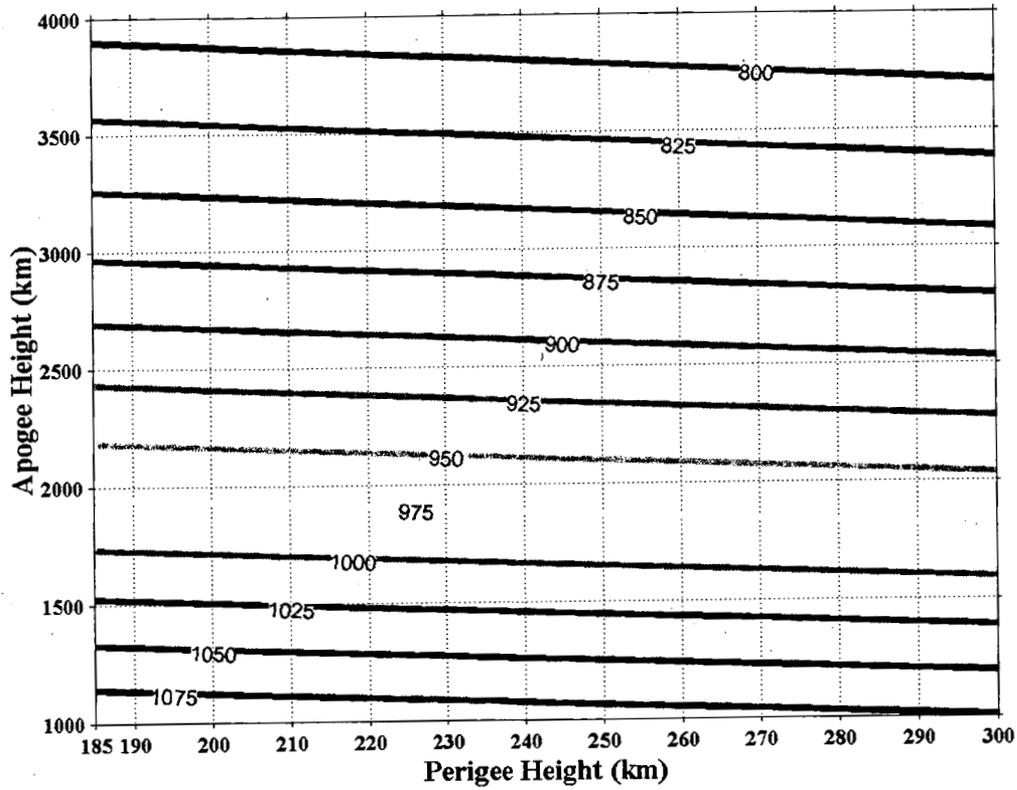


Figure 1: GEC Single Spacecraft Mass (kg) Delivered to Orbit by Delta-II 7920-9.5

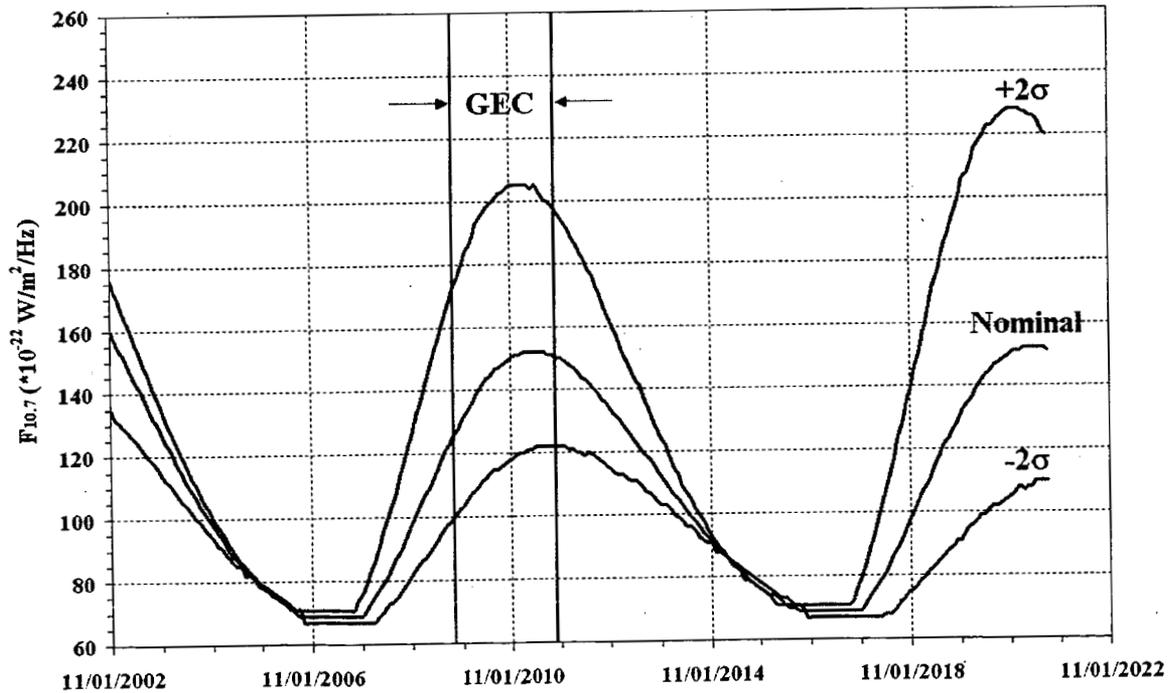


Figure 2: GEC Will Operate Near the Next Predicted Solar Flux Peak (11/2002 +2σ Schatten Solar Flux Prediction)

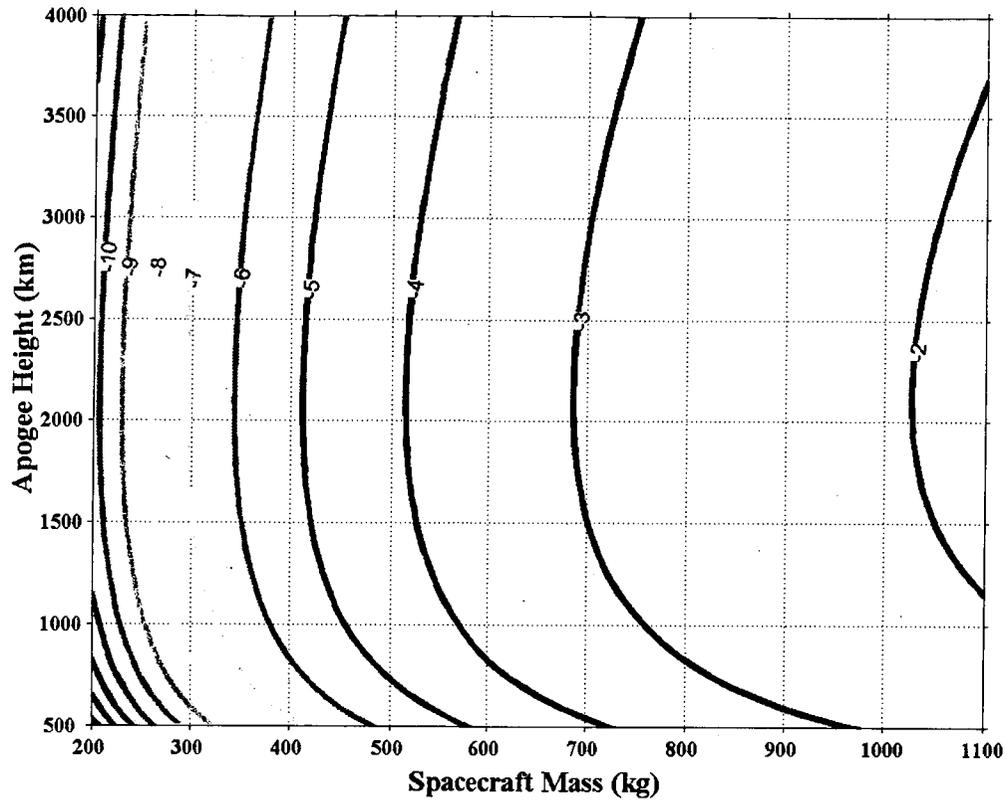


Figure 3: Single Orbit Apogee Decay (km) at Dipping Density of 6.7 kg/km^3

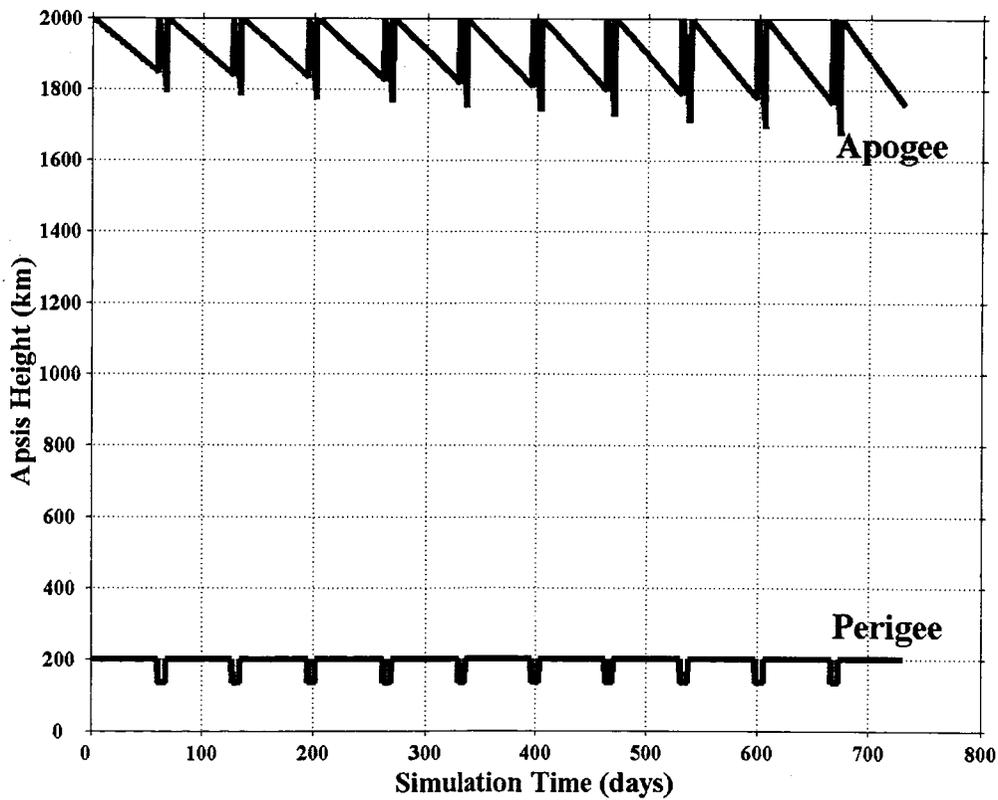


Figure 4: Sample Apsis History 200 km x 2000 km Orbit

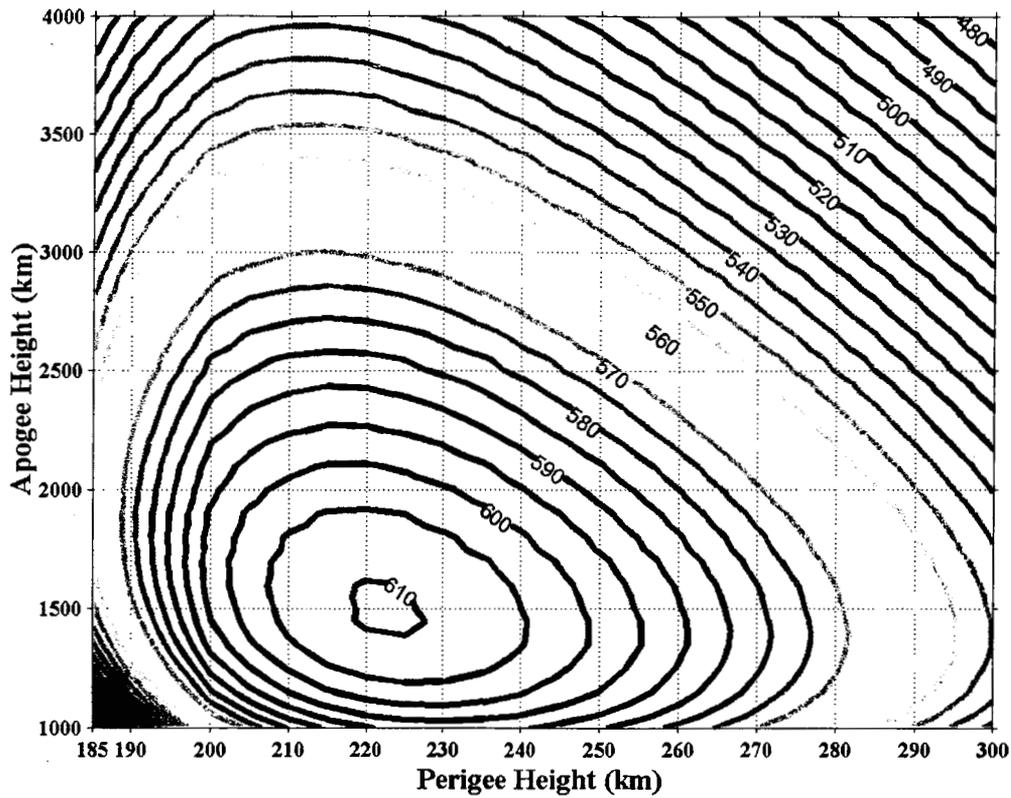


Figure 5: Contour Plot of GEC Dry Mass (kg) Remaining After 10 Dipping Campaigns

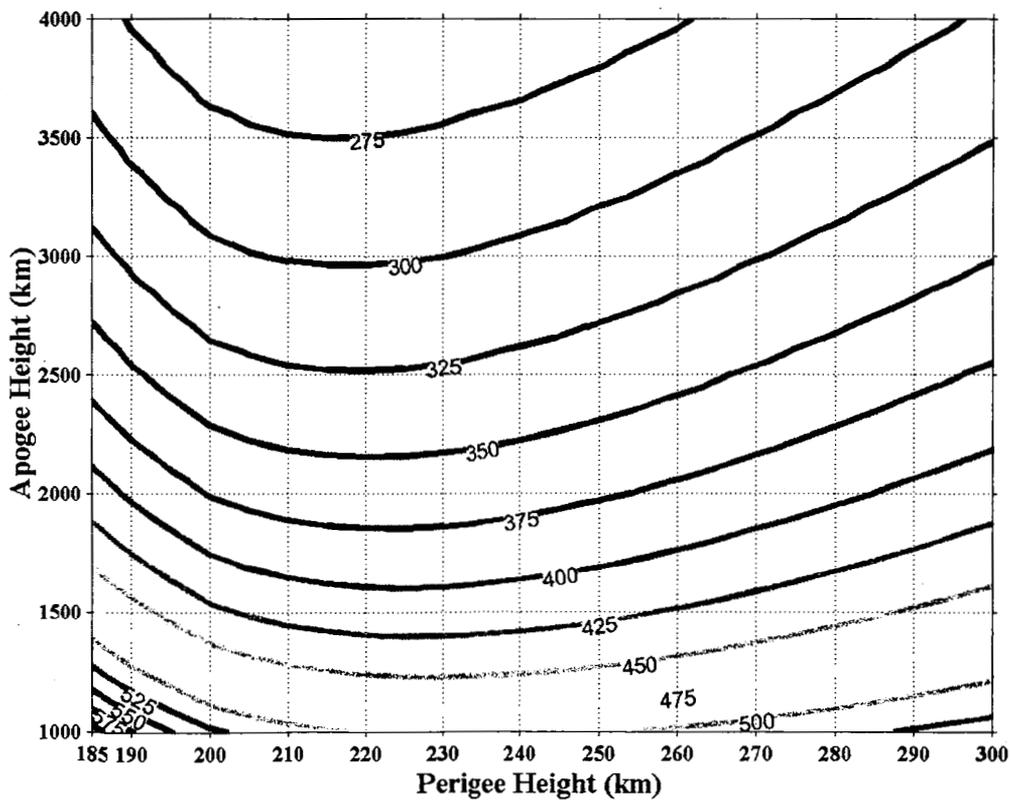


Figure 6: Contour Plot of GEC Propellant Mass (kg) Required for 10 Campaigns

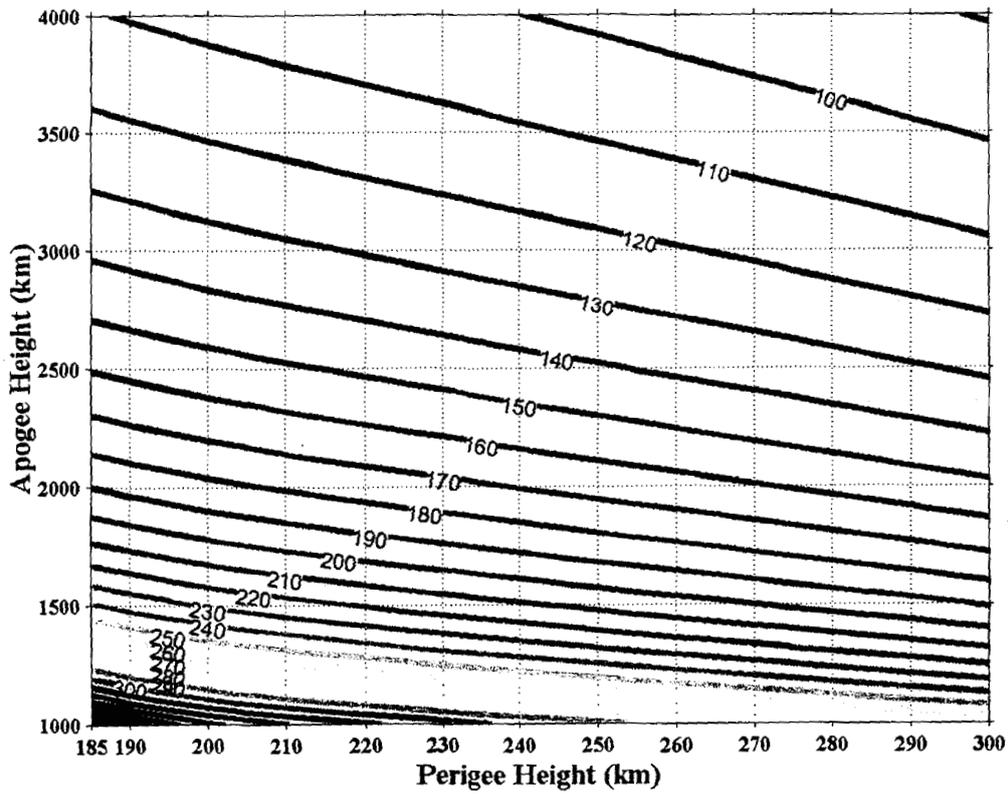


Figure 7: Contour Plot of Cumulative Time (days) Below 500 km Over 2-Year Mission

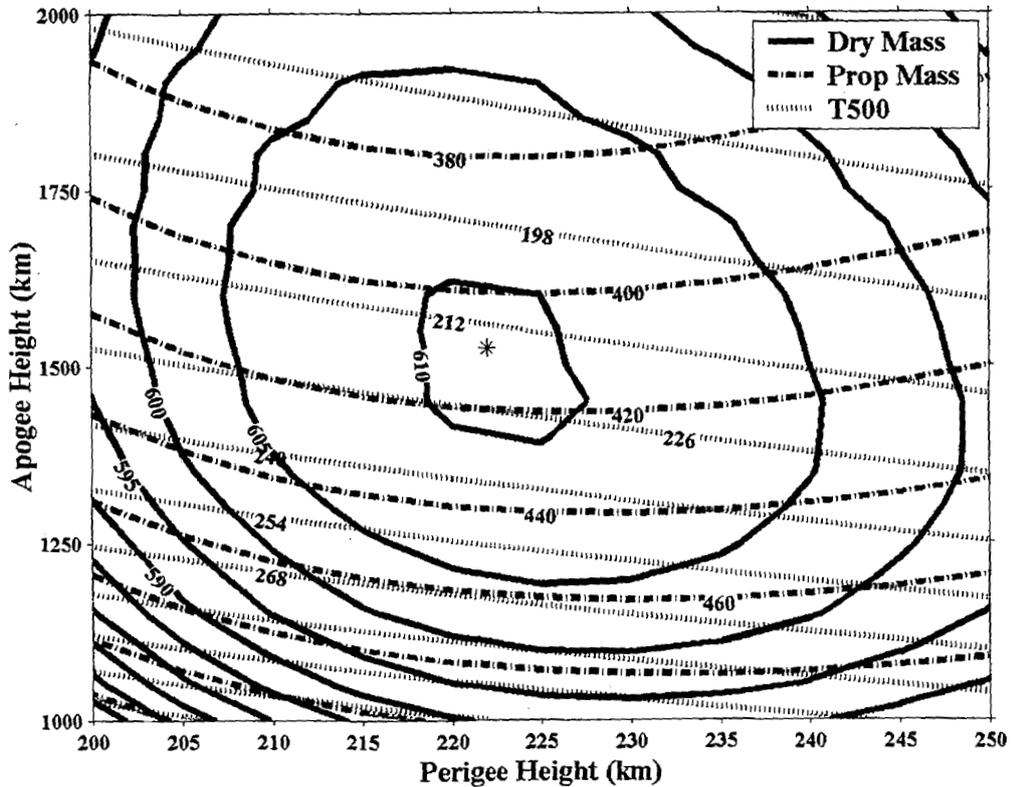


Figure 8: Multiple Contour Plots of Dry Mass (kg), Propellant Mass (kg), and T500 (days)

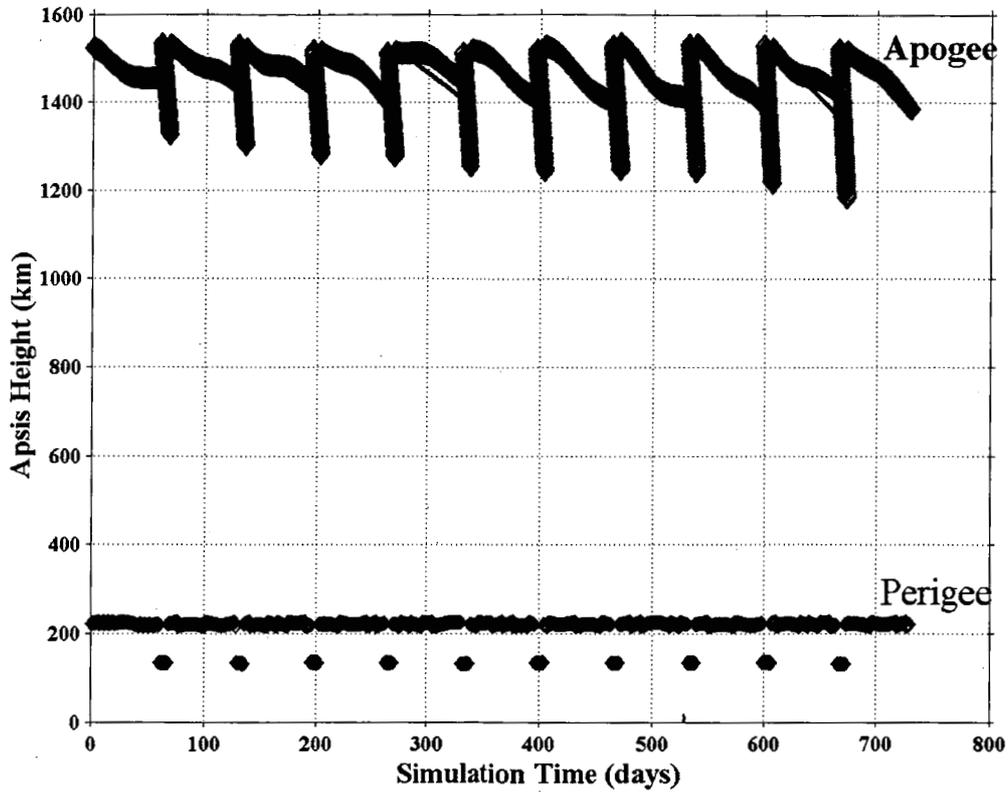


Figure 9: Comparison of Truth Model (large symbols) to Analytic Estimation (small symbols) for Maximum Dry Mass Solution (222 x 1525 km)

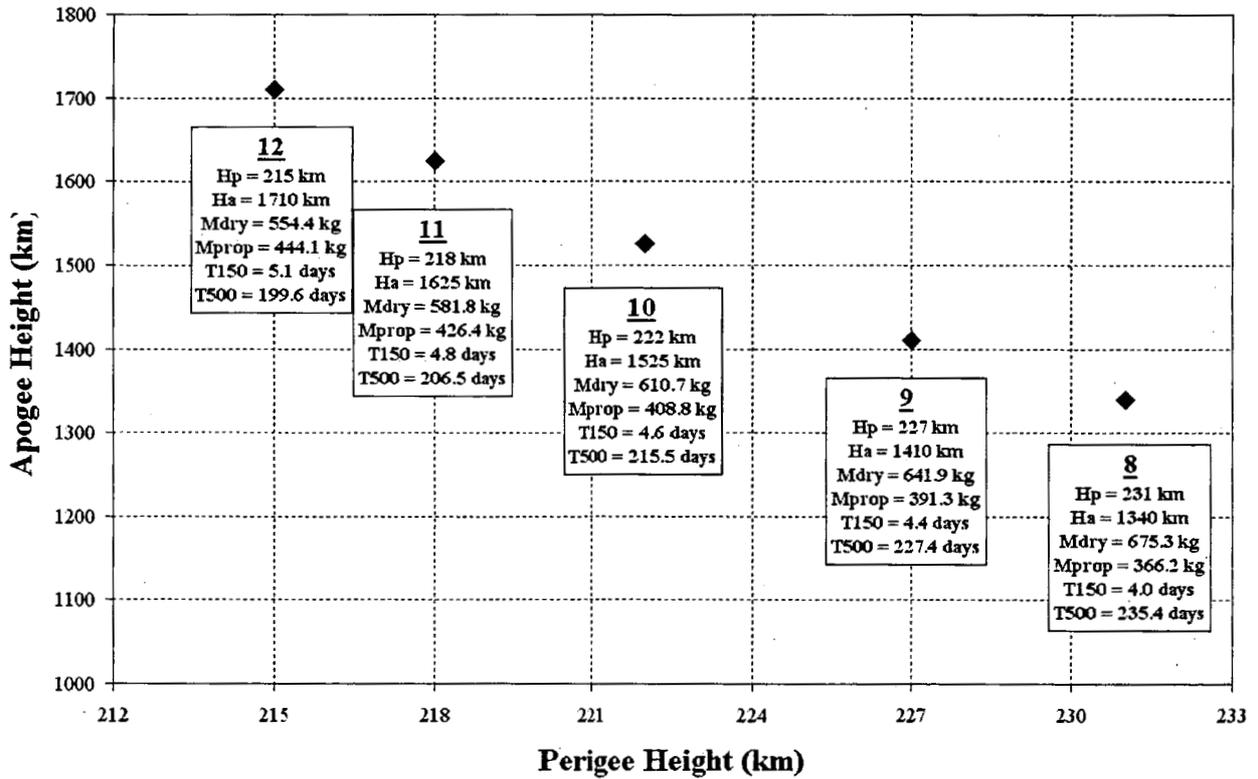


Figure 10: Sensitivity of GEC Optimal Parking Orbit Parameters to Number of Campaigns