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A Long Duration Orbital Simulator (LDOS) Utilized in Technical Planning Activities

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ABSTRACT

This paper describes the development and use of a long duration orbital simulator (LDOS). The LDOS capabilities, options, accuracy, and central processing unit (CPU) usage are discussed. The applications described using the LDOS are associated with (a) Manned Space Flight Network (MSFN) support of long duration missions, (b) environmental effects on high earth orbits and (c) long term orbit decay. The appendix depicts the salient portions of the math model.

INTRODUCTION

The area of technical planning activity has become increasingly concerned with earth orbit missions which are "long duration" in nature. Highly accurate simulators exist which can be used to study a specific mission whose orbital elements, purpose, and time of occurrence are known. Such simulators, however, become impractical when used by a technical planning group looking ahead toward a vaguely defined mission. This situation is true from two standpoints. First, the accuracy offered by such simulators, while welcomed, is not required. Secondly, and perhaps more important, the cost of using such simulators to vary the wide range of possible input conditions is prohibitive. Because of this flexible and low cost requirement, the LDOS was developed under NASA MSFC Contract NAS8-14000.

The objectives of the LDOS, therefore, are to provide a versatile orbital simulator which can generate data applicable over a broad realm of planning situations, is economical to run, so as to allow for frequent use, and yields data of sufficient accuracy for mission planning. These objectives are met because:

1. A typical LDOS running time on an IBM System/360 Model 75 shows that 3000 orbits (over 6 months) can be generated in a CPU time of approximately 5 minutes, or approximately 10 orbits a second.

2. Comparison with radar tracking data for one of the early, long life-time NASA satellites showed agreement to within 1 kilometer in the semimajor axis and 0.01 radians in inclination angle after 6 months.

The LDOS was developed with the following capabilities:

1. Accept input in several coordinate systems

2. Allow for the variation in input of those parameters which affect the orbit

3. Allow for the skipping and modifying of orbits

4. Use of a closed-form integration technique where possible

5. Allow for a varying step size between output points.

LDOS OVERVIEW

Basic LDOS Program Organization

The basic program organization of the LDOS is described here for two major reasons:

1. Depicting the CPU running time advantage

2. Showing how its organization is different from the conventional methods

Figure 1 shows the LDOS at the modular level. A control subroutine identifies which of the input modules are to be used. This subroutine is designed so that many different modules may be called without altering the basic structure. The three modules presently in the program, which were found adequate for any application, are: new plumbline (Apollo 13), Ephemeral (Apollo 5), and orbital elements. These coordinate systems are well known and will not be further discussed.

Each of these modules calls the KEPAC module, which evaluates the trajectory. This subroutine uses closed-form integration methods to determine the components of state in a centrobaric field and determines the effects of solar radiation pressure and the atmosphere by perturbations on these closed-form solutions. The effects of drag are held constant over the orbit and updated at the end of each orbit. This procedure was found to be quite adequate for mission planning, and allowed (a) a considerable savings in CPU running time and (b) the use of a much simpler quadrature formula (Simpson's). The classical approach to this particular problem is to use a more complex quadrature formula (fourth-order Runge-Kutta is the most common) to evaluate drag effects. However, these more complex quadrature formulas are not required for LDOS planning applications because (a) the change in the orbital elements due to drag is continuous and smooth, which is well adapted to Simpson's method, (b) most complex quadrature formulas require excessive CPU time, and (c) the rate of change in the orbital elements is not rapid enough to require a method more sophisticated than Simpson's.

One feature of the KEPAC module is that eccentric anomaly (or true anomaly) becomes the independent variable rather than time. This feature, coupled with the closed-form expressions for the components of state, allows for orbit skipping (incorporating a quadratic update) which also saves considerable CPU time.

A second feature is that large steps in the independent variable may be taken with little loss in accuracy. This is also made possible by the use of closed-form integration procedures and the way in which the perturbation effects, due to drag and solar radiation pressure, are incorporated into the components of state. However, keeping an accurate account of time (from lift-off, from perigee, etc.) with the eccentric anomaly as the independent

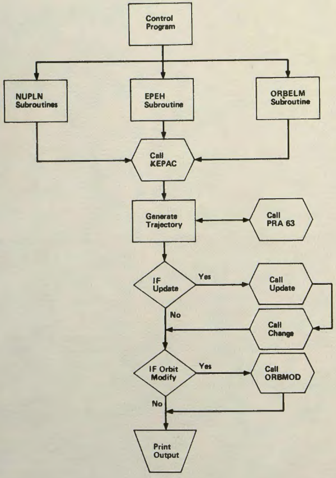


Figure 1. Program Flow Overview of LDOS

variable, while conceptually simple, is far from a simple logical problem. The procedure for accomplishing this is discussed later.

A third feature of this module is that the perturbation due to oblateness, long periodic and short periodic, may be updated at the end of each orbital period and factored into the closed-form solutions for the components of state with very less loss in accuracy. This again reduces the CPU time by a marked degree. For example, under conventional methods, if updating occurred 24 times an orbit (instead of 1 time an orbit as discussed in this paper), CPU time would increase approximately 30 percent.

The UPDATE module allows for the skipping of orbits. Up to 100 different updates can occur in any one simulation, each of a given increment on true anomaly. This orbit skipping procedure gives very accurate results when compared with continuous simulation and saves considerable CPU time over extended missions. The UPDATE module extrapolates results over the orbital span to be skipped by fitting a second-order curve, using finite differences, through the last three points prior to extrapolation. Results will be discussed later in the paper.

The ORBMOD module allows for modification of orbits by using velocity increments. This procedure, while well known and very simple, was found to give very useful results and, again, save CPU time in mission planning. This module will accommodate up to 20 orbital modifications in any one simulation. From these adjusted components of state, new orbital elements are generated, and a new orbital configuration is processed. Results using this module will also be discussed.

For a detailed description of the math model, input format for the different modules, and output format, the reader should consult Reference 1.

Because of the closed-form solution methods used, the simple but accurate method of orbit skipping, the procedure for evaluating atmospheric effects and solar radiation pressure, and the procedure for modifying the orbits, a simulator resulted which produces quite accurate results for LDOS missions at a fraction of the cost when compared to any other known current simulator.

Results Using Update and ORBMOD Modules

Typical results using the UPDATE and ORBMOD modules, respectively, are shown in Tables 1 and 2. Table 1 shows the loss of accuracy, using certain patterns of skipping in the change of the semimajor axis of a vehicle at 150 NM altitude and a 50° inclination. This table shows that the method of extrapolation using finite difference yields very accurate results in that about a 1-meter error is introduced in the worst case.

Table 2 shows the effect on the semimajor axis and numerical eccentricity by providing excessive velocity increments on a vehicle in high earth orbit (perigee $\approx$ 235 NM). In addition, Table 2 shows the expected result-higher impulse velocities (applied at perigee) increase the semimajor axis and numerical eccentricity.

Table 1. Change in Semimajor Axis (In Meters)

End of Day	Continuous Simulation	Simulate 2 Orbits, Skip 10 Orbits	Simulate 6 Orbits, Skip 6 Orbits
1	-136.56	-136.49	-136.54
2	-144.63	-144.46	-144.60
3	-153.36	-153.09	-153.33
4	-167.64	-167.04	-167.60
5	-190.50	-189.28	-190.47
6	-216.96	-215.19	-216.91

Time Flow Program Logic

As pointed out in the basic LDOS program organization section, the chosen independent variable is the eccentric (or true) anomaly, which provides the capability of taking large steps in the independent variable and provides for orbit skipping with very little loss of accuracy. However, as also noted, this feature produced a complex logic problem for accurately evaluating time in the LDOS. Figure 2 depicts the method whereby time is handled; it provides the reader with a "checked out" solution to this complex logic problem.

Table 2. Modification of Orbit as a Result of Impulsive Velocity Increase

Impulsive Velocity Increment (Meters/Sec.)	Semimajor Axis (Meters)	Numerical Eccentricity
(Nominal)	6563484 (Nominal)	0.005407 (Nominal)
16	6598812	0.009505
32	6618242	0.013623
64	6674151	0.021886
128	6789561	0.038512
256	7035833	0.072167
512	7600388	0.141086

APPLICATIONS

Three applications of the LDOS which are associated with the planning phases of missions are:

1. MSFN support of a 30-day mission

2. Environmental effects on a high earth orbit

3. Long term orbital decay

MSFN Support of Long Duration Missions

For a vehicle in a $\approx$ 240 NM circular orbit (Skylab) with an inclination of 50°, the following data was needed concerning contact time above a 3° elevation angle:

1. Total contact time (data retrieval potential)

2. Total contact time when each contact is $\geq$6 minutes in duration (to plan dumps of tape recorders which require 6 minutes, etc.)

3. Degradation of the above parameters after 30 days

4. Large gaps between periods of contacts

The MSFN configuration for this particular analysis consisted of:

MILA	Carnarvon (CRO)	Goldstone (GDS)
Bermuda (BDA)	Hawaii (HAW)	Ascension (ACN)
Canary Islands (CYI)	Guam (GWM)	Honeysuckle (HSK)
Madrid (MAD)	Texas (TEX)	Newfoundland (NFL)*

*Proposed site

This particular analysis involved a one-month simulation stepping 0.25 degrees in true anomaly. The LDOS results then served as input parameters to a program which computed acquisition and loss data for the MSFN station locations and configuration. Coverage (or lack thereof) statistics were then computed for various configurations. Figures 3 and 4 reflect the resulting total and daily coverage to be expected. These figures show that a total of 253.29 hours of coverage above a 3° inclination angle could be expected over the course of the mission, and that 84.2 percent of such contact (213.33 hours) would involve contacts of at least 6 minutes in duration. These totals are a reflection of the 8.41 and 7.18 hours of daily coverage, and daily coverage $\geq$6 minutes output by the simulation. The high percent greater than or equal to 6 minutes is due, to some extent, to the overlapping condition that exists between the coverage circles of various stations (i.e., TEX, GDS, MILA; CYI, MAD).

Orbital characteristics, and their variations, which were computed by the LDOS for use in this coverage study are outlined in Table 3.

Table 3. Orbital Characteristics

Parameters	Start (Day = 0)	End (Day = 30)
Altitude (Meters)	447598	440316
Eccentricity	0.0000129246	0.0000129136
Period (Seconds)	5596.8740	5596.7239
Apogee (Meters)	6813387	6813263
Perigee (Meters)	6813210	6813087

The analysis indicated a loss of 0.18 hours a day in total coverage and a loss of 0.14 hours a day in a minimum contact of 6 minutes duration from the start to the end of the simulation, or a loss of approximately 0.02 percent in both statistics. From these results, it was determined that the "worst case" statistics for technical planning activity would be used.

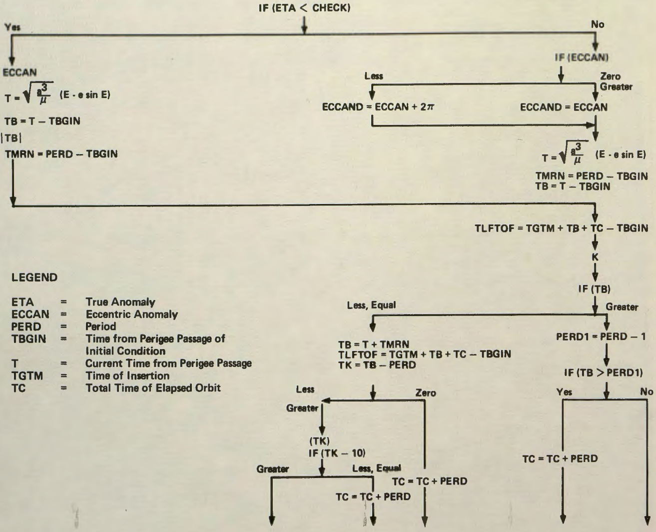
The analysis of large gaps in coverage revealed 52 periods of no contact 80 minutes duration or longer. A summary of these gaps is presented in Table 4.

Table 4. Maximum Gap Statistics

Gap Length (Min)	Frequency of Occurrence	Last Station to See Prior to Start	First Station to Acquire at End
177	7	Texas	Hawaii
134	13	Hawaii	Ascension
90	8	Hawaii	Hawaii
90	2	Ascension	Ascension
85	4	Goldstone	Hawaii
85	2	Texas	Goldstone
83	2	Texas	Hawaii
82	13	MILA	Goldstone
82	1	Bermuda	Goldstone

From these results, particularly noting when larger gaps occurred, planning activity regarding work schedules was formulated.

Data generated during the simulation, reflecting the perturbations on the orbital elements, was maintained in a history file (a) to serve as a check on LDOS accuracy and (b) to give further insight into the effects of environmental factors and a variety of perturbations on vehicles in earth orbit.



LEGEND

ETA	=	True Anomaly
ECCAN	=	Eccentric Anomaly
PERD	=	Period
TBGIN	=	Time from Perigee Passage of Initial Condition
T	=	Current Time from Perigee Passage
TGTM	=	Time of Insertion
TC	=	Total Time of Elapsed Orbit

Figure 2. Time Logic Flow

Environmental Effects

The purpose of this analysis is to show the effect on pertinent orbital elements by varying parameters depicting the physics of the satellite and the environment. Even though excellent results have been derived in which mission planning data is available (Reference 2), the LDOS allows a wide variety of output data over a broad realm of planning situations in a fast, efficient, and inexpensive manner.

Changes in perigee, apogee, and numerical eccentricity are depicted in Figures 5 through 9. These changes were due to variations in mass, coefficient of drag, effective area, geomagnetic activity, and solar flux, respectively, for one of the early long-life NASA satellites. The initial values used for perigee, apogee and numerical eccentricity are 7,040.793 kilometers; 8,614.2311 kilometers; and 0.1005069, respectively. Because of the high altitude perigee (662.628 kilometers above an equatorial radius), small mass, small effective area, and drag coefficient, the effects of the atmosphere over 19 days are minimal. This is shown in Figures 5 through 7. Figure 5 also depicts the change in the semimajor axis. This change in semimajor axis may also be calculated from the perigee and apogee changes using $\delta a = 1/2 (\delta A + \delta P)$. For this reason the change in the semimajor axis is not depicted on the remaining graphs. Figures 5 through 7 show that the change in apogee is greater than the change in perigee, and the change in numerical eccentricity is small.

Changes in these same orbital elements as a function of change in geomagnetic activity and solar flux over the same time period are shown in Figures 8 and 9. These environmental changes also reflect minor modification in the orbit where again the major increment is in apogee, with the minor increment the perigee. The change in numerical eccentricity is again small. In summary,

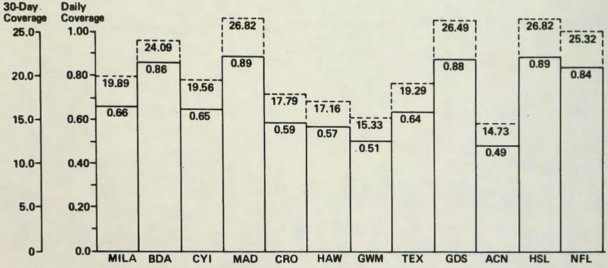


Figure 3. Total and Daily Contact

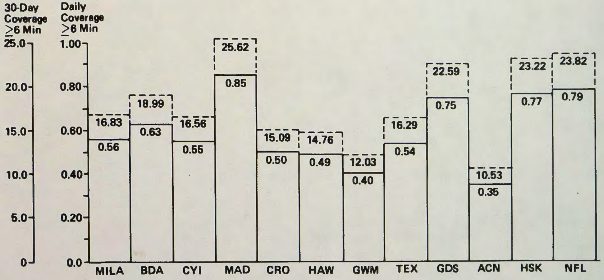


Figure 4. Total and Daily Contact $\geq$ 6 Minutes Duration

orbital changes over short time periods of less than one month are small, and the sensitivity of the orbit to the parameters outlined above is low, because of the high altitude and small size of the satellite.

A similar analysis (with no environmental perturbations) was carried out for 6 months and the results compared to radar tracking data for one of the NASA satellites. This comparison showed agreement to within 1 kilometer in semimajor axis and 0.01 radians in inclination angle.

Long Term Orbital Decay

The third example using the LDOS involves an evaluation of the effects of orbital dacay. The study also proved most useful in demonstrating the running time advantage of the LDOS. A typical running time on Model 75 shows that 2500 orbits (6 months) can be generated in a CPU time of 4 minutes and 29 seconds. This time is approximately 0.1-second per orbit. Figures 10 and 11 show the change in apogee, perigee, semimajor axis and numerical eccentricity over the 6-month period. These figures depict the

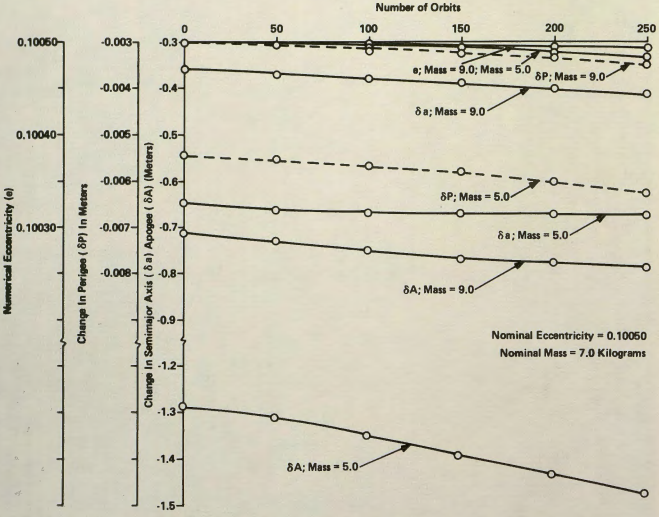


Figure 5. Change In Orbital Elements as a Function of Mass Change M = 9.0, M = 5.0

expected result – as the orbit becomes more circular, it loses energy and finally begins to fall at an increasing rate into the atmosphere.

CONCLUSIONS

The development and use of the LDOS resulted in the following conclusions:

1. A typical LDOS running time on a Model 75 shows that 3000 orbits (over 6 months) can be generated in a CPU time of approximately 5 minutes, or about 10 orbits per second.

2. Comparison with radar tracking data from one of the early, long-lifetime NASA satellites showed agreement to within 1 kilometer in the semimajor axis and 0.01 radians in inclination angle after a 6-month period.

3. For another early NASA satellite the decay in the orbit of over 19.14 days is small, and the sensitivity of the drag orbit to changes in mass, coefficient of drag, effective area, geomagnetic activity and solar flux is small because of (a) the high altitude (perigee of 662 kilometers) and (b) the small size of the satellite.

4. Using the LDOS in conjunction with an acquisition/loss program, it was found that a total of 253.29 hours of coverage above a 3° elevation angle would be provided by the MSFN over the course of the Skylab mission.

5. The MSFN Skylab acquisition/loss analysis also showed that there was a loss of less than 1 percent in total coverage and in contacts of at least 6 minutes duration over the 30-day simulation.

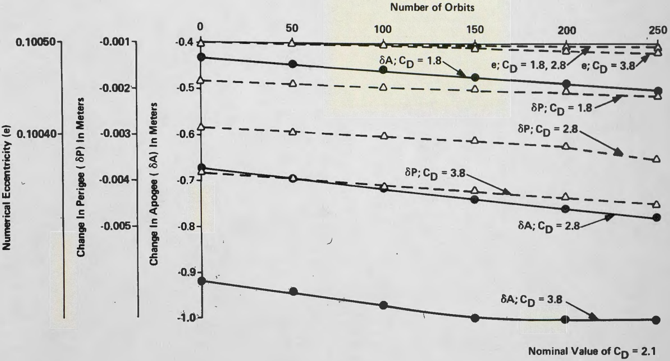


Figure 6. Change In Orbital Elements as a Function of Change In Coefficient of Drag

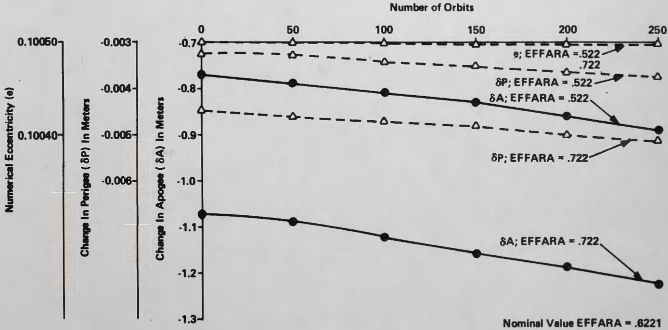


Figure 7. Change In Orbital Elements Due to Change In Effective Vehicle Cross Sectional Area (EFFARA)

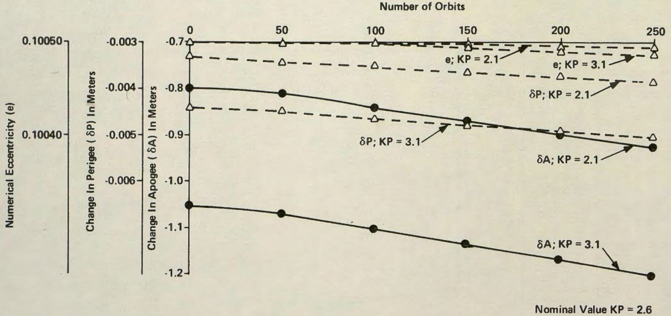


Figure 8. Change In Orbital Elements Due to Change In Planetary Index of Geomagnetic Activity (KP)

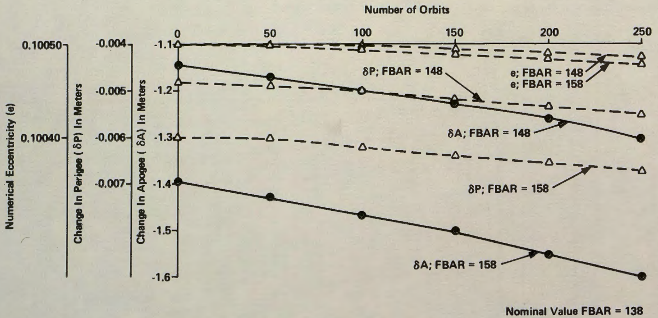


Figure 9. Change In Orbital Elements Due to Change In Solar Flux (FBAR)

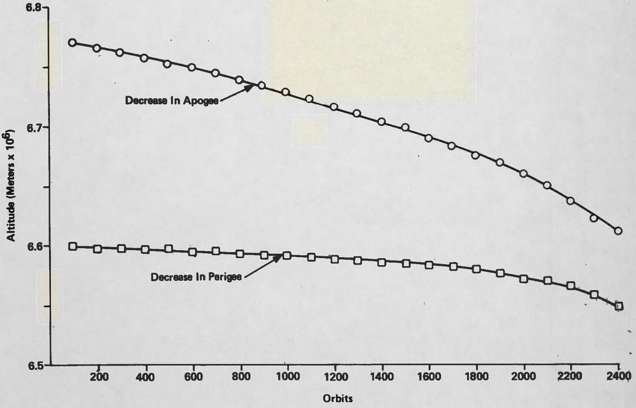


Figure 10. Change In Apogee and Perigee Over 6-Month Period (Oblate Earth)

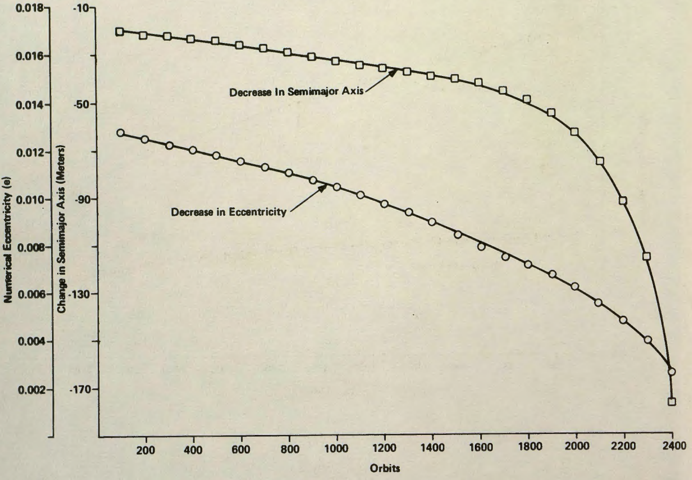


Figure 11. Change In Semimajor Axis and Numerical Eccentricity Over 6-Month Period

APPENDIX

Secular perturbations are defined as ever increasing or decreasing changes from set EPOCH values and are periodic about these EPOCH values. In the LDOS the variations in (Ω), (ω) and (M) are considered. The variations in (a), (e), and (i) are considered constant and averaged to $J_2 = 1.0823 \times 10^{-3}$. The expressions, which are used for calculating these changes obtained from first-order theory, are well known, and therefore are not repeated here.

The long periodic perturbation incorporated is reflected in the change in the inclination angle i_L, while the short periodic perturbations considered are those of the ascending node and argument of perigee. The form of these perturbations is also well known and, therefore, is not included.

Drag Equations

The effects of drag are taken into consideration for a, i, P, A, Ω, ω. The effects of other parameters are not included due to demonstrated minimal influence on the results. The form of the drag equations (with eccentric anomaly as the independent variable) is given by:

$$\frac{da}{dE} = -2b\rho a^2 \sqrt{\frac{D1^3}{D2}} \; (D4)^2$$

$$\frac{di}{dE} = -\frac{1}{2}\,\frac{(b\rho a\,\omega_e \sin i)}{N_o\sqrt{1-e^2}}\,(1-\cos 2\mu)\,(\sqrt{D2^5 * D1})\;D4$$

$$\frac{dP}{dE} = 2b\rho a^2(1-e^2)\sqrt{\frac{D1}{D2}}$$

$$*D4*\left\{(1+e)(\cos E + D5)*D3 - D1*D4\right\}$$

$$\frac{dA}{dE} = -2b\rho a^2(1+e^2)\sqrt{\frac{D1}{D2}}$$

$$*D4*\left\{D1*D4 + (1-e)(\cos E + D5)*D3\right\}$$

$$\frac{d\Omega}{dE} = -\frac{1}{2}\,\frac{(b\rho a\omega_e)}{N_o\sqrt{1-e^2}}\;*(\sin 2\omega\cos 2\eta + \cos 2\omega \sin 2\eta)$$

$$*(\sqrt{D2^5*D1})\;*D4$$

$$\frac{d\omega}{dE} = -\cos i\,\frac{d\Omega}{dE} - 2b\rho a\,\frac{(1-e^2)^{1/2}}{e} * \left[\frac{1+e\cos E}{1-e\cos E}\right]^{1/2}$$

$$*(\sin E*D4)*\left[1 - \frac{D*D2*(2-e^2-e\cos E}{2(1-e^2)}\right]$$

where

$$D1 = (1 + e\cos E)$$

$$D2 = (1 - e\cos E)$$

$$D4 = (1 - \frac{D*D2}{D1})$$

$$D = \frac{\omega e}{N_o}(\sqrt{1-e^2})\quad \cos i$$

$$D5 = -\frac{D*D2}{2(1-e^2)}$$

$$N_o = \sqrt{\frac{\mu e}{a^3}}$$

$$b = C_D A_E/2Mv$$

$$(1-\cos 2\mu) = (1 + \cos 2\omega\cos 2\eta - \sin 2\omega \sin 2\eta).$$

A detailed description of the derivative of these equations may be found in Reference 3.

Mathematical Expressions Relating the Effect of Solar Radiation Pressure and Geomagnetic Activity on Atmospheric Density

Define S as a "heating parameter."

$$S = \bar{S}\, e^{g(t)}$$

where

$$\bar{S} = 25 + 0.8\,\bar{F}_{10.7} + 0.4\,(F_{10.7} - \bar{F}_{10.7}) + 10\,KP$$

$$g(t) = 0.025\cos\left[2\pi\left(\frac{t-38047}{365.25}\right)\right] - 0.06\cos\left[4\pi\left(\frac{t-38047}{365.25}\right)\right]$$

$e^{g(t)}$ = Correction for seasonal effects

t = Time in modified Julian Days (Julian Day - 2400000.5)

KP = 3-hour planetary index of geomagnetic activity

$F_{10.7}$ = Daily value of 10.7-centimeter solar flux

$\bar{F}_{10.7}$ = S modified value of 10.7-centimeter solar flux

Assuming only a mean value of solar flux (i.e., $F_{10.7} = \bar{F}_{10.7}$) then $\bar{S} = 25 + 0.8\,\bar{F}_{10.7} + 10$ KP.

The effect of S on the atmospheric density RHO (ρ) is:

$$\frac{\rho\,corr}{\rho_o} = \left(\frac{S}{200}\right)^{0.81}\left[3 + 2.5\left(\frac{\frac{ALT}{1000} - 360}{240}\right) - 0.5\left(\frac{\frac{ALT}{1000} - 360}{240}\right)^2\right]$$

For more extensive discussion, the reader is directed to Reference 4.

Atmospheric Effects

PRA 63 Atmosphere (Patrick Air Force Base Reference Atmosphere, 1963).

The PRA 63 atmosphere was used in the LDOS. For a more detailed description, the reader is directed to Reference 5.

Orbit Modification

The analysis assumes an impulsive velocity correction which adds or detracts from vehicle energy depending on the direction of application.

The expressions used to modify the orbit are as follows:

$$\cos\alpha = \frac{\dot{X}}{V};\ \cos\beta = \frac{\dot{Z}}{V};\ \cos\delta = \frac{\dot{Y}}{V}$$

$$\Delta\dot{X} = \Delta V(\cos\alpha);\ \Delta\dot{Z} = \Delta V(\cos\beta);\ \Delta\dot{Y} = \Delta V(\cos\delta)$$

Increment Equation

The change in the magnitude of orbital elements is generally nonlinear and increasingly negative. The equation used to estimate the change during an update step must consider these nonlinearities. The equation used takes into account the nominal step differences and the differences between these differences (the effects of skipping). Specifically, consider the last three points X_3, X_2, X_1 prior to an update of N steps. The equations used are:

$$\Delta_1 = X_1 - X_2$$

$$\Delta_2 = X_2 - X_3$$

$$\Delta\Delta_1 = \Delta_1 - \Delta_2$$

$$\text{Jump} = (N)\left\{X_1 + [(\tfrac{N}{2})\ \Delta_1] + \Delta\Delta_1\right\}$$

Step Size/Time

The true anomaly (η) is incremented by the step size delta (Δ).

$$\eta = \eta + \Delta\ .$$

Compute eccentric anomaly E. Depending on the sign of η, the logic to handle time diverges into one of the paths shown in Figure 2. The K is a block of logic inserted to avoid the case (only possible in a 360^{o} step), where an overstep in time is made (usually not more than 10 microseconds). The logic flow outlined ensures that: (a) time from lift-off is accurately updated, and (b) (sign) changes do not introduce errors in η and E.

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