A NEW TRAJECTORY PROPAGATION PROGRAM FOR DSPSE MISSION SUPPORT

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Abstract

A satellite trajectory propagation program called *N-Body* is being developed to support the upcoming Deep Space Program Science Experiment (DSPSE) mission which is also known as Clementine. The program will support the activities of the mission operations center which is run by The Naval Research Laboratory. The computer program is designed to predict a spacecraft orbit through the various phases in which the DSPSE vehicle will fly. These orbital phases include low Earth, translunar, lunar, and deep space. The design of *N-Body*, its functionality, and the mathematical methods it employs are presented.

1 Introduction

The Deep Space Science Experiment Program (DSPSE), also known as Clementine, is a joint Naval Research Laboratory, NASA, and Lawrence Livermore National Laboratory project. DSPSE is funded by the Ballistic Missile Defense Office. The primary goal of the DSPSE mission is to demonstrate the operation of lightweight sensors in the space environment. DSPSE has two secondary science missions in which it will fly in lunar orbit for two months and then depart the Moon for a four month cruise to rendezvous with the near Earth asteroid 1620 Geographos. In accomplishing its secondary missions, DSPSE will have made the first complete, multi-spectral mapping of the lunar surface, and executed the first close encounter with a minor planet (approximately 100 km).

Mission operations for DSPSE will be conducted by the Naval Research Laboratory's Naval Center for Space Technology at the DSPSE Mission Operations Center in Alexandria, Virginia. During the various mission phases, the DSPSE space vehicle will fly in low Earth orbit, highly elliptical orbit with apogee at lunar distances, lunar orbit, and deep space heliocentric orbit. The DSPSE Mission Operations Center, or DMOC is tasked to coordinate the science activities with the business of navigation, orbit determination, command and telemetry, and satellite housekeeping functions. The operations center must therefore possess a trajectory integration program which can accurately integrate the satellite orbit in each of these unique space environments.

It was decided that a program developed by members of the operations team itself could

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be tailored expressly to meet the DSPSE requirements in areas such as engineering and science support, contingency planning, maneuver design and verification and tracking station scheduling. Furthermore, by using a product developed in-house, software modifications can be effected more rapidly and less expensively than if an off-the-shelf product were used. The trajectory propagation program, called *N-Body*, is being developed by the author to satisfy this requirement.

Strictly speaking, the program does not address the classical n-body problem in which the motion of n objects under mutual gravitation is solved for. Rather, the program attempts to solve a restricted problem in which the trajectories of $n-1$ objects are known and the motion of the massless n^{th} object is solved for. In this case the $n-1$ objects are the Sun, Moon and planets and the n^{th} object is the space vehicle.

N-Body employs a Cowell method in which the total accelerations of all external forces acting on the vehicle are directly integrated. Cowell's method was selected due to its ease of implementation. For instance, Encke's method in which only the perturbations about the two body problem are integrated can be more efficient since the integration step size can usually be increased over that of the Cowell method without any associated loss of accuracy. However the Encke method requires that the perturbed orbit be rectified with a new reference orbit whenever the difference between the perturbed orbit and the reference orbit grows too large. The rectification process logic is dependent on the particular numerical integration procedure being used. For instance a multistep integration method would require restarting. By using Cowell's method, the executive algorithms can be independent of the integration procedures. The result is a major simplification in software design and maintenance.

Since the trajectories must be propagated in low Earth orbit, trans-lunar, lunar and deep space, the force model must contain the effects of the Earth and Moon gravitational potential fields, Earth atmospheric drag, solar radiation pressure and planetary gravitation. Several gravitational potential models for the Earth and Moon are incorporated into the program. Alternatively, a user defined potential may be specified. Atmospheric drag requires a model of the density variations of the Earth atmosphere and a physical model of the satellite. A model of the vehicle shape and its specular characteristics are needed to compute the acceleration due to solar radiation pressure impinging on the spacecraft. A planetary ephemeris is used to obtain the third body gravitational forces. The planetary ephemeris is also used to predict eclipsing events which affect the solar radiation pressure computation. Finally, the force model allows for impulsive spacecraft maneuvers.

N-Body is written in FORTRAN and is being developed on VAX workstations running under the VMS operating system. The computer code is largely complete and consists of about 10,000 lines. The program is presently undergoing verification testing by the DSPSE mission operations team. It is anticipated that the program will be fully operational in time to support DSPSE flight operations starting with the planned launch in January, 1994.

2 Design and Operation

N-Body is comprised of a program configuration procedure, an executive procedure, and an extensive mathematical subroutine library. A series of key words and variables is used to control the execution flow, define the input and output coordinate frames and times, configure the force model, select and configure the integrator, set the duration of the propagation and select the destination and frequency of the output data. The executive procedure transforms the input coordinates and epoch time to a standard internal format the coordinates of which are mean equator and equinox of J2000.0 and TAI (International Atomic Time) seconds. Using the mathematical subroutines, the executive procedure carries out the orbit propagation according to the key word configuration. While the propagation is in progress the executive procedure monitors its status to determine when output records should be produced. Output records can be requested at regular time intervals, after each integration step, at the first and last times in the run or they can be triggered by orbital events such as apoapsis and periapsis passages. The executive procedure also can be configured to detect the passage of the space vehicle into the umbra and penumbra of a planet, and the occultation of the vehicle by a planet as viewed from the Earth.

The program configuration procedure reads an input file containing key words and variables, as well as the initial condition for the trajectory that is to be propagated. Each key word and variable is assigned a default value so that the only item that is required to be in the input file is the initial condition. The initial condition may be specified in terms of Cartesian position and velocity coordinates or classical Keplerian orbital elements. In the case of Keplerian elements, the in-plane position of the satellite may be specified with either true anomaly, mean anomaly or time since periapsis passage. Any of the physical constants used by the program, including the Earth and Moon gravity models, may be changed by placing the appropriate key words and parameters into the input file. In this way the user may re-configure the orbit propagation model without changing any computer code.

The program output data may be directed to the screen or to a disk file. The output data begins with a listing of the input file used for the current run. Next a summary of the program configuration is listed with respect to the input and output coordinate frames, time definitions, force model parameters, integrator selection and print options. Each output record contains the current time in Julian and calendar dates, Cartesian position and velocity, Keplerian orbital elements and Earth referenced position of the space vehicle trajectory being integrated.

An ASCII ephemeris file may be created consisting of a header followed by time, position and velocity data records. The header and data in the ephemeris file conform to the Standard Ephemeris Record Format (SERF) which will be used by the DSPSE mission operations and science teams. The SERF header contains a description of the parameters, constants and models used to create the ephemeris. The input vector coordinate system, origin of coordinates, coordinate type, and position, velocity, and time units are listed followed by the actual input vector. Generation information is also provided which includes the time the run was made, the propagation start and stop times, the interval between ephemeris records, and the number of points in the ephemeris. The force model used for the run is summarized by listing on / off indicators for Earth harmonics, atmospheric drag, solar radiation pressure, third body perturbations, lunar harmonics and maneuvering. The output vector coordinates, time, etc. are described in the same manner as the input vector. Finally, The radius and gravitational parameter of the Earth Moon and Sun are listed along with the first four zonals for the Earth and Moon.

If it has been requested, the executive program will monitor the instantaneous state of the space vehicle in order to detect the occurrences of shadows and occultations. Shadows occur when a planet passes between the satellite and the Sun. Occultations are defined to occur when a planet passes between the satellite and an observer located at the center of the Earth. Accurate knowledge of shadow times is required for operating vehicle power and thermal subsystems. Knowledge of occultation times is important for communications scheduling. Exact start and stop times of these events are computed by iterative methods that are described below.

3 Implementation of Cowell's Method

Cowell's method is a straightforward solution technique used in the special perturbations solution technique of orbital motion. The formulation was first set down by Gauss but was used by Cowell at the beginning of this century for integrating planetary ephemerides in rectangular coordinates. The term Cowell's method is used now to describe any step by step numerical integration in which the equations of motion are integrated in rectangular coordinates. The technique has become more popular and useful with the advent of modern digital computers. The method consists of writing the perturbed equations of motion as

$$\ddot{\mathbf{r}} = \mathbf{a} \tag{1}$$

where $\mathbf{r}$ is the space vehicle position vector and $\mathbf{a}$ is the sum of all the accelerations acting on the vehicle. Equation (1) may be rewritten as a system of two first order differential equations,

$$\dot{\mathbf{r}} = \mathbf{v}, \quad \dot{\mathbf{v}} = \mathbf{a} \tag{2}$$

When further reduced to its rectangular components, Equation (2) represents six first order differential equations which may be numerically integrated by a variety of techniques.

For the force model contained in *N-Body*, the acceleration vector may be written explicitly in terms of its various constituents as

$$\mathbf{a} = \mathbf{a}_c + \mathbf{a}_h + \mathbf{a}_d + \mathbf{a}_s + \mathbf{a}_t \tag{3}$$

Where $\mathbf{a}_c$ is the central body acceleration, $\mathbf{a}_h$ is the acceleration from the harmonic model of the figure of the central body, $\mathbf{a}_d$ is the atmospheric drag acceleration, $\mathbf{a}_s$ is the acceleration from solar radiation pressure and $\mathbf{a}_t$ are the third body gravitational accelerations. No vehicle thrust acceleration appears in Equation (3) because maneuvers are accounted for in *N-Body* as instantaneous changes in velocity and not as vehicle accelerations.

The user may select one of three different numerical integration procedures: *RK4*, a fourth order Runge Kutta with constant step size, *RK45* a fourth order Runge Kutta with variable step size, or *SP*, the modified Adams Pece formulation of Shampine and Gordon[11]. *RK4* is the simplest method and requires only the step size to be specified. *RK45* dynamically controls its step size by monitoring the difference between the current solution from a fourth order formulation with that of a fifth order formulation. *RK45* requires an initial step size and an error tolerance parameter. *SP* is the most sophisticated numerical integration procedure provided with *N-body*. It uses a multistep Adams method with automatic step size control. *SP* is very efficient and has the advantage that very large step sizes may be taken without an associated loss in accuracy. *SP* requires both absolute and relative error tolerance parameters. Because of the large integration steps which are taken, the periapsis and apoapsis print modes are not permitted when using the *SP* integrator.

4 Force Model

To compute the acceleration of the space vehicle and integrate the equations of motion, all of the external forces acting on the vehicle must be modeled. The major forces acting on a space vehicle in orbit are gravitational attraction from the Sun and planets, atmospheric drag, and solar radiation pressure due to absorbtion and re-emission of the Sun's photons by the spacecraft surfaces. Each of these forces may be selected individually through the program configuration procedure. The Cartesian components of each force are computed and transformed to the mean equinox and equator of J2000.0 coordinate frame in which the equations of motion are cast. The accelerations are computed from the forces and the vehicle mass and summed to yield the total effective acceleration.

The origin of the coordinates of integration may be selected to be the Sun or any of the planets. When the central body is the Earth or the Moon, the force model allows the use of a harmonic expansion of the gravitational potential in order to account for perturbations due to the figure of the Earth or Moon. Two Earth potential models are provided: The 1984 World Geodetic Survey (WGS84)[1] truncated to order and degree 12, and the Goddard Earth Model (GEM9)[2] truncated to order and degree 7. For orbits about the Moon, The Bills and Ferrari lunar potential model[3] of order and degree 16 is included. To provide greater flexibility the user may specify the coefficients of any normalized Earth or Moon potential model of up to order and degree 21. The coefficients are defined through use of an input file in which the non-zero terms in the potential model are placed.

Gravitational forces exerted on the vehicle by bodies other than the central body are referred to as third-body forces. Third-body forces may be selected independently for the Sun and planets. The third-body gravitational forces are computed assuming point mass attraction. Thus the third body forces are functions of the relative positions of the central body, the perturbing body and the space vehicle as well as the gravitational parameters of the central and third bodies. The relative planetary positions are computed using the JPL Planetary Ephemeris System DE200[4]. By default, the gravitational parameters for each of the bodies in the solar system are set equal to those used in DE200. However, if a potential model is selected, the gravitational parameter associated with that model replaces the default value. Any of the gravitational parameters may be changed from its default value or from the value in the potential model via the input file.

Atmospheric drag may be included in the force model when propagating a space vehicle in Earth orbit. The model used to compute atmospheric drag is:

$$\mathbf{a}_d = -\frac{1}{2} C_d A \frac{\rho}{m} \frac{\mathbf{v}^3}{v} \tag{4}$$

In this model C_d is the coefficient of drag, A is the effective cross-sectional area of the vehicle, ρ is the atmospheric density, m is the mass of the vehicle, and $\mathbf{v}$ is the velocity of the vehicle in a coordinate frame which is rotating with the Earth. The use of the Earth relative velocity of the space vehicle is equivalent to assuming the atmosphere is rotating with the Earth.

The atmospheric density may be computed using either of two models selectable through the input file. The first is a simple exponential function of altitude and may be useful for applications which do not require high accuracy. The second density model is the Jacchia 1971[5] in which atmospheric density is computed as a function of space vehicle position, day of year, solar flux and the geomagnetic index.

Solar radiation pressure is accounted for by assuming a constant value for solar flux, and scaling it by the space vehicle area to mass ratio. The vector force acts in the direction from the Sun to the vehicle. Solar radiation is not computed when the vehicle is determined to be in the shadow of the central body. This simple model will probably not be of sufficient accuracy for propagating the trajectory of a deep space, heliocentric orbit. An upgrade of this module is planned wherein the solar flux will be modeled as a function of heliocentric distance and an index of reflectivity will be applied to the solar radiation pressure scale factor.

5 Coordinates

Coordinate frame definitions are of primary importance in any trajectory propagation system. The fundamental frame chosen for *N-Body* is the mean equator and equinox of J2000.0. J2000.0 refers to the standard epoch at 0 hours, on January 1^{st}, in the year 2000. The x, y plane is defined by the Earth's mean equatorial plane of J2000.0. The x direction in the x, y plane is defined by the intersection of the ecliptic plane with the mean Earth equator of J2000.0. The terminology mean equator of J2000.0 refers to the actual orientation of the Earth equatorial plane at J2000.0 corrected for nutation. The origin of the reference frame may be chosen to be any planet in the solar system or the Moon.

The mean equator and equinox of J2000.0 coordinate system is a convenient one in which to carry out the trajectory integration. However, depending on the application, the state of the vehicle may have to be referred to another reference frame. For instance, the position and velocity of the space vehicle relative to the true of date stellar background may be of interest. In addition, true of date geographic coordinates are required to evaluate the gravitational force from a harmonic potential model. *N-body* allows space vehicle coordinates to be input and output in 1) mean equator and equinox of J2000.0, 2) true equator and equinox of date, 3) true ecliptic and equinox of date, 4) mean equator and equinox of B1950.0, 5) geographic (Earth fixed) and 6) selenographic (Moon fixed) coordinate frames. The Earth and Moon referenced velocity coordinates may be defined either as inertial or with respect to the rotating body. The selenographic coordinates may be computed with or without including the physical libration of the Moon.

In transforming from mean equator and equinox of J2000.0 to true equator and equinox of date, rotations of precession and nutation are applied in order. The precession rotation takes place about the polar axis. The angle being defined in reference 6 page S15. The nutation rotation matrix is computed according to the 1980 IAU Theory of Nutation and Reduction[7].

An additional rotation about the pole through the Greenwich hour angle yields the geographic (Earth fixed) inertial coordinate frame. When the velocity coordinates are corrected for the rotation of Earth the geographic rotating frame results.

Selenographic (Moon fixed) inertial coordinates are computed via a transformation directly from J2000.0. Two such transformations are included. In the first transformation, the optical libration of the Moon is included[8]. In the second, the physical as well as optical librations are included. The analytic theory for the libration of the Moon is that of Eckhardt (1981, 1982) and the computer code was provided by the U.S. Naval Observatory. The reason for maintaining both models is that most lunar potential models are reduced without including the physical libration. Thus when evaluating lunar gravitation one should likewise not include the physical libration. However when it is desired to obtain the true position of the space vehicle over the lunar surface to high accuracy, the physical libration should be included. Selenographic rotating coordinates are computed by assuming a mean rotational rate for the Moon and applying the appropriate correction to the velocity coordinates.

For the true ecliptic of date reference frame, precession and nutation are applied to the J2000.0 x axis to yield the true equinox of date. Then a rotation about the true equinox through the true obliquity of date is made to establish the fundamental plane as the ecliptic plane.

Transformation from J2000.0 to B1950.0 coordinates is computed according to the rotation matrix provided in reference 7. The rotation accounts for a change in the value of nodal precession as well as a correction to the origin of the B1950.0 coordinate frame.

6 Time

The epoch times of the initial condition and propagated state vectors may be specified in several different timescales. Input and output state vector times may be independently chosen from four different timescales: International Atomic Time, Coordinated Universal Time, Terrestrial Time, or Barycentric Dynamical Time. Universal time and Sidereal time are also used internally by the program to compute Earth referenced coordinate transformations. The program must be capable of translating from one timescale to another because the various theories of motion incorporated by the program to compute coordinate frame orientations, planetary ephemerides, etc. use different reference timescales for their independent variables. For instance, most mission planning is done with respect to the UTC timescale. The independent variable in the JPL planetary ephemeris is TDB. In the theory of Earth nutation it is TT. In addition, for some applications such as comparing propagated trajectories to data from other programs or astronautical tables, it is convenient to select different timescale for the input and output state vector times.

The independent variable over which the equations of motion are integrated is, of course, time. The internal timescale used in the integration is International Atomic Time (TAI). The fundamental unit of TAI is the Systéme International (SI) second. The SI second is defined to be a fixed number transitions of a particular energy state of the cesium-133 atom. The TAI timescale provides a stable, very nearly uniform measure of time. It is used as the basis for prediction of the other timescales.

Universal Time (UT) is the timescale which is most commonly used in every day life. It is closely related to the motion of the Sun, which involves the rotational and orbital motion of the Earth, and may be derived directly from observations of the transit times of stars and radio sources. UT is used internally by the program in order to compute Sidereal time. Sidereal time is the measure of the hour angle of the intersection of the true equator of date with the ecliptic of date. Sidereal time must be known in order to relate positions in an Earth-fixed reference frame to the J2000.0 reference frame. When the observationally derived UT is corrected for longitudinal errors caused by polar motion, it is designated UT1. UT1 is related to sidereal time by means of a numerical formula. UT1 and Sidereal Time are not available as input and output timescales, rather they are used internally for the purpose of transforming from geographic coordinate frames to the J2000.0 frame.

Coordinated Universal Time (UTC) is the timescale which is distributed by all broadcast time services world wide. UTC differs from TAI by an integer number of seconds, and the fundamental unit of UTC is the SI second. UTC is kept to within 0.90 seconds of UT1 by adding (or subtracting) leap seconds. As of July 1993, UTC is 28 seconds behind TAI. *N-Body* keeps track of the difference between UTC and TAI through use of a data base which must be updated each time an additional leap second occurs.

Terrestrial Time (TT) is a theoretical, idealized timescale which is the independent parameter in the apparent geocentric ephemerides. In the past this timescale has been called Ephemeris Time (ET) and more recently Terrestrial Dynamical Time (TDT). The fundamental unit of TT is the SI second. The value of TT is TAI + 32.184 seconds.

Barycentric Dynamical Time (TDB) is another theoretical timescale which is the time argument in the equations of motion of the solar system with respect to its center of mass. TDB is determined from TT depending on the motion bodies in the solar system and the particular theory being used. The variation of TT and TDB is periodic and involves no secular terms. An approximate formula is given in reference 7:

$$\mathrm{TDB} = \mathrm{TT} + 0.001658^s \sin g + 0.000014^s \sin 2g \tag{5}$$

where

$$g = 357.53^\circ + 0.9856003(\mathrm{JD} - 2451545.0)$$

and JD is the Julian date.

7 Searching for Orbital Events

During the course of an integration, the program executive can be instructed to watch for various orbital events which when detected are included part of the program output. Techniques are employed to search for periapsis and apoapsis passages, umbra and penumbra passages, and occultations of the satellite by the planets with respect to the center of the Earth.

When the program is operated in the periapsis or apoapsis print modes, it will attempt to detect the radius of closest or farthest approach to the central body. When it has determined such an event, the exact time of passage is approximated by an iterative bisection method which converges very rapidly to an accuracy which may be specified through the input file. The passages are detected by comparing from one integration step to the next, the dot product of the position and velocity vectors. When the dot product changes sign, an apsis has been passed. Whether it is periapsis or apoapsis is determined by the direction of the sign change.

It must be noted that for a near circular orbit the periapsis and apoapsis locations may not be well defined based on the osculating states. Therefore the program may report multiple apsis passages per revolution for orbits with very low eccentricity. Also since the Shampine integration technique can take very large step sizes, the periapsis and apoapsis print modes may only be selected when using the Runge Kutta integration techniques.

The times of passage of the space vehicle into and out of the shadows of the Moon and planets may also be determined. Subsequent integrated states are checked to detect the passages from full sun light, penumbra, and umbra. The computation is carried out using the values of the planetary radii defined in the JPL Developmental Ephemeris System, DE200. The planetary radii values, as with most other constants used in *N-Body*, may be changed via the input file. When a shadow transition occurs, the space vehicle trajectory is integrated in a secondary procedure from the previous state to the current state with a small constant step size in order to better approximate the exact shadow entrance or exit time. The accuracy of the approximation is thus equal to the step size of the secondary integration. The step size may be selected using the appropriate input file parameters. It is possible, depending on the step size of the primary trajectory integration process, that a shadow event of sufficiently small duration could be missed by this detection method. Thus if all such shadow events are to be found it may be advisable to configure the program executive to use a Runge Kutta integration method with an appropriately small, constant step size.

The beginning and ending times of occultations of the space vehicle by the Moon, Sun and planets may be determined using the same bisection process used in determining the periapsis and apoapsis passages. The occultation condition is computed for an observer located at the center of the Earth. Therefore the parallax effect for an observer located somewhere on the surface of the Earth is neglected. In this computation, the radii of the planets defined in DE200 are used but may be overridden through the appropriate input file key words and parameters.

Note that in the periapsis and apoapsis print modes the vehicle state is written to the ephemeris and program output files. When occultation or shadow detection is enabled only the times of occurrence are reported to the program output file. The ephemeris file is not updated when these events are found to occur.

8 Code Verification

Verification of the computer code is being accomplished in part by comparing its trajectory predictions to that of other propagation programs. In this process the orbital elements of a space object are propagated in time by each program and the predicted positions and velocities are directly compared. To facilitate this process care must be taken that all physical constants such as Earth and Moon gravitational potential models, planetary constants and the like are identical in each program. All such constants in *N-Body* may be changed through the input file without any code revision. Thus input files may be tailored to cause the force model of *N-Body* to emulate that of most other trajectory propagation programs. Obviously when doing such comparisons, care must be taken to ensure that the comparison states from each program are referred to similar coordinate reference frames and timescales.

Extensive verification tests of *N-Body* have been conducted by comparing its output to the TRIP trajectory propagation program developed by Kaufman at the Naval Research Laboratory[9]. One such comparison exercise is presented here in which the trajectory of a space vehicle in low Earth orbit (LEO) is computed. The reference orbit is that planned for the LEO phase of the DSPSE mission. The

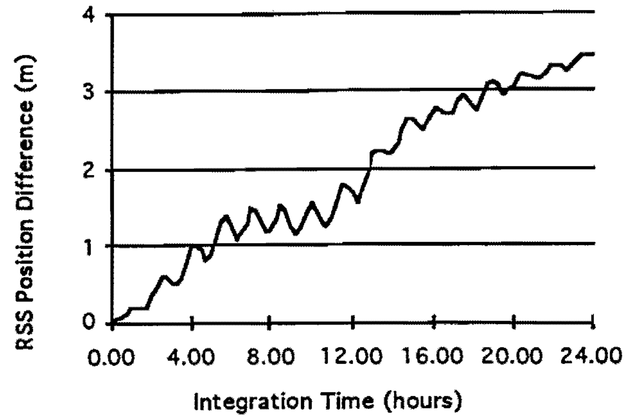

Figure 1: One Day RSS Position Comparison To TRIP Propagation

LEO injection state is propagated for one day by *N-Body* and TRIP and the resulting mean equator and equinox of J2000.0 positions are compared by computing the norm of the position vector differences. The orbit parameters at injection are that of a 140 by 160 nautical mile orbit inclined at approximately 66 degrees. The force model in both programs includes Moon and Sun third body gravitation and uses the Goddard Earth Model GEM9 truncated to order and degree 4.

The results of the comparison of the integrated space vehicle position between *N-Body* and TRIP are plotted in Figure 1. It is seen that after one day of integration the two programs predicted the space vehicle position to under 4 meters of each other. This small difference can likely be attributed to differences in evaluation of the geopotential model (TRIP uses an unnormalized form, while *N-Body* uses a normalized form of the geopotential), to differences in the implementation of the numerical integration procedures, and to the addition of leap seconds which have occurred in since the latest update of TRIP. This small difference is viewed to be acceptable and provides a validation of the *N-Body* computer code.

In another verification test a comparison was made using the JPL computed ephemeris of the asteroid Geographos[10]. The initial state of the ephemeris is propagated for six months at which time it is compared again to the JPL ephemeris. In this instance the force model included the third body perturbations of the major planets and solar radiation pressure. The computed RSS position difference was found to be approximately 24 kilometers after the end of the six month integration. This discrepancy can likely be attributed to variations in the solar radiation pressure model. Also, the JPL ephemeris is computed using a relativistic gravitational theory whereas *N-body* uses classical Newtonian gravitation and the relativistic masses from DE200.

9 Execution Performance

Program development and testing is being carried out on a VaxStation 4000 model 90 workstation. The low Earth orbit verification test case described above was carried out on this machine and the execution run time was calculated to serve as an indication of program execution performance. The force model used in the integration consisted of the GEM9 Earth model truncated to order and degree 4, as well as Moon and Sun third body forces. The fourth order Runge Kutta numerical integration method was selected. The force model was evaluated four times per integration step. The total elapsed time for the one day propagation was 22.8 seconds and a total of 19.1 CPU seconds were used.

10 Planned Enhancements

The *N-Body* program is still undergoing development and verification. Several enhancements are planned for the short term in order support the beginning of the DSPSE mission in January 1994.

The force model for solar radiation pressure will be upgraded to better model the variation of solar flux with heliocentric distance. The solar radiation pressure model will also be modified to include the spacecraft reflectivity coefficient and to check for solar occultation from planetary bodies other than the central body. The Earth atmospheric drag model will be modified to accept as input current or predicted values of solar flux and the geomagnetic index. The JPL developed Earth gravitational potential model JGM2 and the Goddard GEMT3 model will be added to the available selections. For lunar orbit propagation the Konopliv Lunar gravity model will be added.

The capability to automatically switch the central body of integration will also be added. The body switching logic will use the planetary gravitational parameters to compute the spheres of influence of each of the bodies in the solar system. Body switching will automatically take place when the space vehicle trajectory leaves the sphere of influence of one body and enters that of another.

11 Summary

Development and verification of the orbit propagation program *N-Body* is underway and is planned for completion in time to support the Deep Space Program Science Experiment mission planned to begin in January 1994. The program contains a high fidelity orbital force model that will support orbit propagation in all phases of the mission: Low Earth orbit, cislunar orbit, lunar orbit and deep space.

The program is designed for flexibility by allowing the selection of several different force model options as well as the use of user defined physical constants. Optional specifications for time and coordinate reference systems enable the program to be useful for a wide variety of space mission planning and support applications.

Code verification is progressing through comparison of computed ephemerides with those generated by other programs. Initial testing indicates very good agreement with other accepted models.

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