

A New Orbital Analyst Tool for Associating Un-cataloged Analyst Debris with Historical Launches, Breakups, and Anomalous Events

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Abstract

A suite of astrodynamic software tools has been developed for associating un-catalogued analyst satellite debris with historical launches, breakups, and anomalous events. A semi-analytical orbit integrator is at the heart of the tool suite. To associate currently tracked analyst debris with events occurring over 30 to 45 years ago requires an orbit integrator that predicts orbit plane ascending node values accurately, with errors no more than a few degrees over thousands of degrees of node precession. Such an integrator, SGPE (Special General Perturbations Ephemeris), has been developed and proven to be extremely accurate. Orbit perturbations that are included in the tool are daily atmospheric density corrections, atmospheric rotation variations, high-order lunisolar gravitational effects, earth tidal effects, solar radiation pressure perturbations, and high-order geopotential zonal contributions. All these perturbations are computed with analytical equations evaluated on a half-day increment to achieve a very fast semi-analytical integrator. Also part of the tool suite is a non-linear least squares software package used to determine average ballistic coefficient values obtained from using years of mean element semi-major axis values. Once the B value has been fitted the SGPE integrator is used to integrate backwards from epoch to 1958, generating an ephemeris of mean element sets every 5 days. These elements are then compared with mean elements from every launch and breakup since the 1960s. Probabilities of possible associations are computed from tables based on over 550 validation cases using historical mean elements from catalogued debris. Associations of hundreds of analyst satellite debris with historical events have already been identified, and cataloging of the analyst satellites is currently underway. Examples are shown of breakup associations, and of associating currently tracked debris with previously lost satellites.

1. Introduction

Numerous attempts have been made to associate currently tracked un-cataloged analyst debris with historical launches and breakups. Prior to 1980 there have been numerous breakups, as shown in Fig. 1. The backward integration must span at least 25 to 45 years to associate with very early events. To associate debris with a particular launch or breakup the orbit plane orientation needs to be accurately correlated. This means that the orbit inclination and right ascension of the ascending node need to be accurately computed. The problem is to define the right ascension of the ascending node to an accuracy of only a few degrees after the very long integration span. Table 1 shows the amount of precession of the node over a 40 year span as a function of the orbit inclination. From 12,000 to over 65,000 degrees of precession must be accurately computed for the great majority of the events. In previous association attempts the integration has been done using SGP4 or SP, the special perturbation integrator. Using the analytical mean element SGP4 propagator means fast long term integrations are possible, but the predictions of the node are very inaccurate because of a lack of in-depth perturbation computations. Fig. 2 shows examples of four different long term SGP4 propagations for orbits of different inclinations and altitudes. Errors greater than 1000 degrees can occur using this method. Using a special perturbation integrator increases the accuracy of the integration but drastically increases the computer time for the integration. However, as will be shown later, using special perturbations still does not accurately account for all the necessary perturbations to achieve the desired accuracy of the node predictions.

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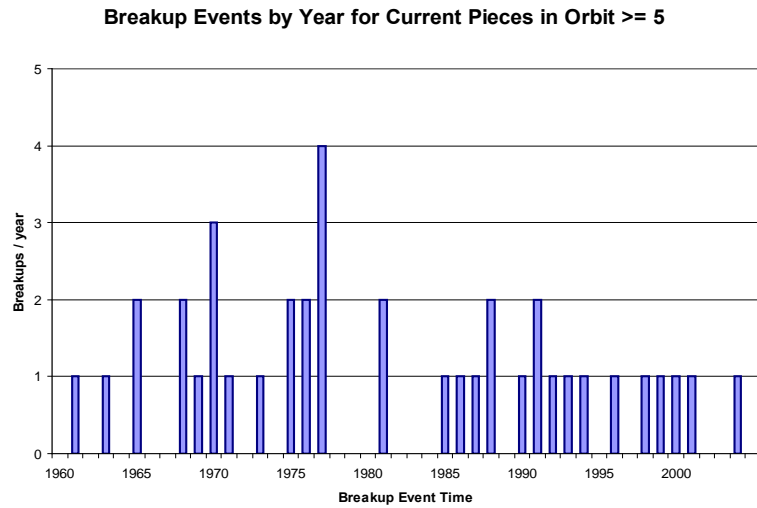


Fig. 1. Breakup events by year for breakups with more than 5 pieces currently in orbit.

Inclination	RA Precession (deg/day)	Degs after 40 Years
50°	-4.60	-67200°
70°	-2.30	-33600°
82°	-0.82	-12000°
90°	0.01	+-
98°	0.95	13880°

Table 1. Precession of the right ascension (RA) of the orbital node as a function of orbit inclination, with the number of degrees of precession after 40 years listed.

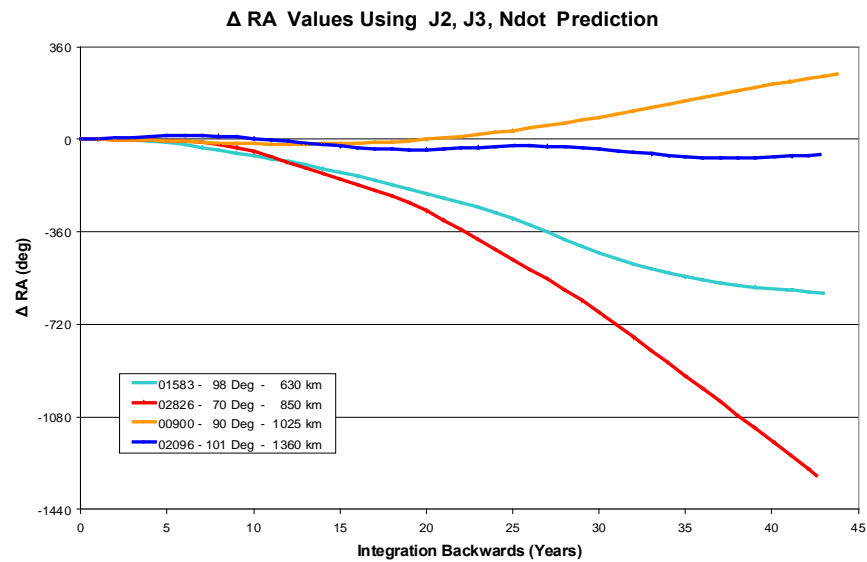


Fig. 2. Error in right ascension of ascending node (Δ RA) for selected satellites using SGP4 orbit perturbations.

2. Semi-analytical Orbit Integrator SGPE

To achieve an accuracy of only a few degrees in the precession of the orbit node requires in-depth perturbation computations to cover 40 years or more of time. A one step propagator like SGP4 would require exhaustive analytical equations accounting for all the variations of the orbit plane as a function of time. Using special perturbation techniques with detailed small step integration would require many hours of computer time. The solution for this problem is to use a semi-analytical integration technique with a step size on the order of a day that accurately accounts for all the perturbations. The semi-analytical integrator SGPE was originally developed in the 1970s to predict long term orbit perturbations of high eccentricity satellites. Only first-order perturbations of geopotential resonance and lunisolar gravitational effects were considered for the initial model. It was determined that a half-day step size would be sufficient to achieve accurate integrations. Recently SGPE was extended to accurately predict node precession by adding high order drag computations, increased lunisolar gravitation effects, earth tides changes, solar radiation pressure, and increased geopotential zonal perturbations.

From previous drag analysis (Bowman [1]) it was determined that using any current empirical atmospheric model in any integrator would result in inaccurate modeling of the atmosphere, resulting in inaccurate atmospheric drag computations. However, adding a daily density correction to the model would greatly reduce the errors in the drag computations. Fig. 3 shows the results of the ballistic coefficient solutions based on using the Jacchia 70 [2] model with and without a daily density correction based on observed density variations. Special perturbation orbit fits were used for the B solutions. Bowman [1] describes the method of using calibration satellites at different altitudes to compute daily temperature (i.e. density) corrections. This method was used to compute daily density corrections as a function of altitude from 1962 through 2009. The satellite 00060, used for the B computations in Fig. 3, is a blunt double-cone shape which behaves like a sphere for drag computations. Thus, the errors in the B values are a direct reflection of the errors in the atmospheric density. Fig. 3 shows that the density errors are significantly reduced from over 17% one standard deviation to just under 6% by applying an observed daily temperature (density) correction to the model during the B orbit fits.

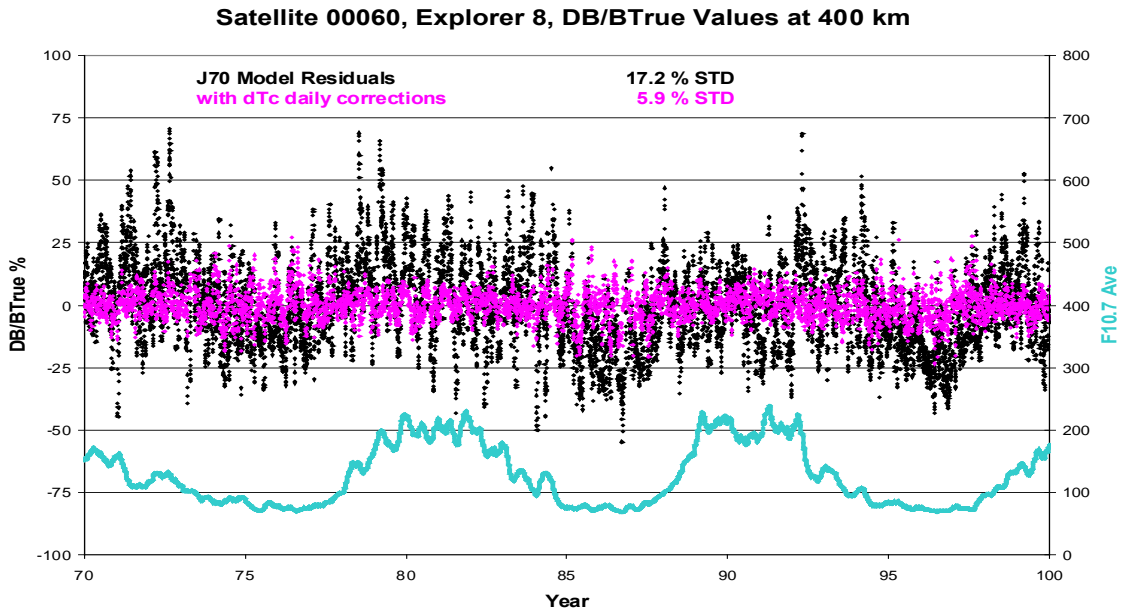


Fig. 3. Ballistic coefficient B error (%) from orbit differential corrections as a function of year (1970 to 2000) using the Jacchia 70 model with and without daily temperature corrections (dTc).

King-Hele [3] analytical drag computations at the orbit perigee location were selected for use in SGPE, coupled with the CIRA 72 [4] atmospheric density model using the daily temperature corrections. The atmospheric drag on the orbit is recomputed every SGPE half-day integration step. The CIRA 72 model was selected because at every integration step the density and atmospheric molecular constituent amounts are computed from an altitude integration of the diffusion equations. This is necessary because the drag coefficient is a function of molecular number densities (Bowman [5]) which varies with altitude and solar conditions, as displayed in Fig. 4. Therefore, the drag coefficient value needs to be reevaluated at every SGPE integration step for the drag computations. Also, the atmospheric rotation rate needs to be used to account for the drag effect on the orbit plane orientation. King-Hele [6] determined that the zonal winds are variable as a function of altitude, so these effects were also added into the SGPE drag computations. For low eccentricity orbits ($e < 0.05$) it was determined that the day-night diurnal density variations were too large to use the King-Hele equations evaluated just at the perigee point of the orbit. Therefore, additional development in SGPE was undertaken to account for the diurnal effects for low eccentricity orbits.

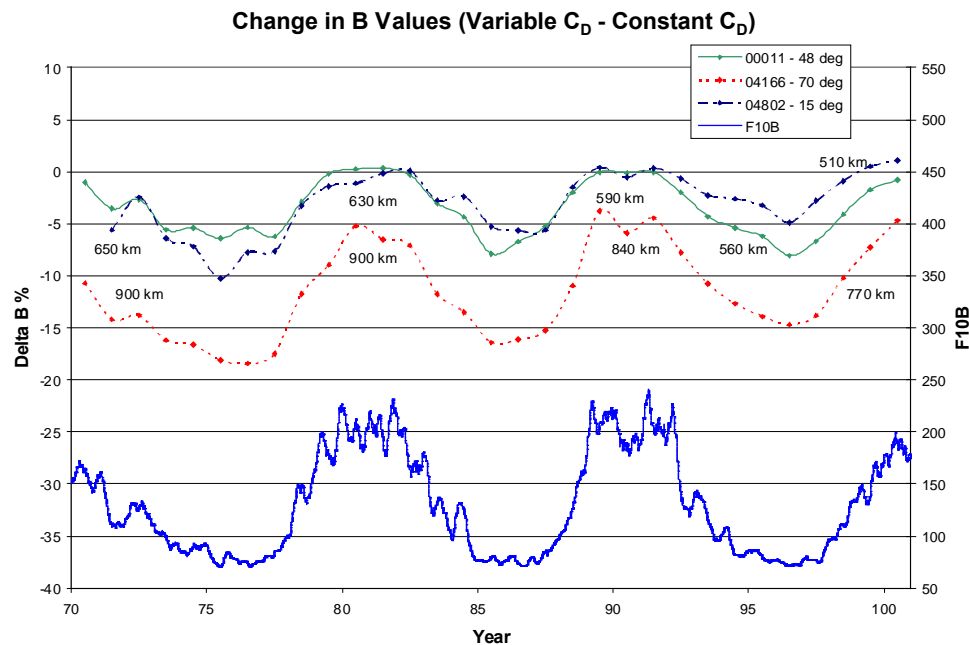


Fig. 4. Change in ballistic coefficient B values as a function of year from changes in the drag coefficient C_D due to changes in atmospheric constituent densities at high altitudes

Fig. 5 shows the results of the backward SGPE integration from 2010 to 1968 for satellite 02826, a calibration sphere, with and without applying a daily temperature (density) correction. All the other orbit perturbations, discussed below, were included in the 42 year integrations. The SGPE integration with the daily corrections compares almost precisely with the historical operational mean elements, while the SGPE integration without the daily corrections is in error in orbital period by about 0.25 minutes after the 42 year integration. Even though this period difference appears small it is enough to cause a large error in the precession of the node as displayed in Fig. 6. The error in the SGPE integration with the daily corrections is just less than 6 degrees after 42 years, while the integration without the daily corrections is in error by almost 80 degrees after the 42 year backward integration. Thus, using the daily density corrections is absolutely vital to obtaining accurate node precession predictions over long periods of time.

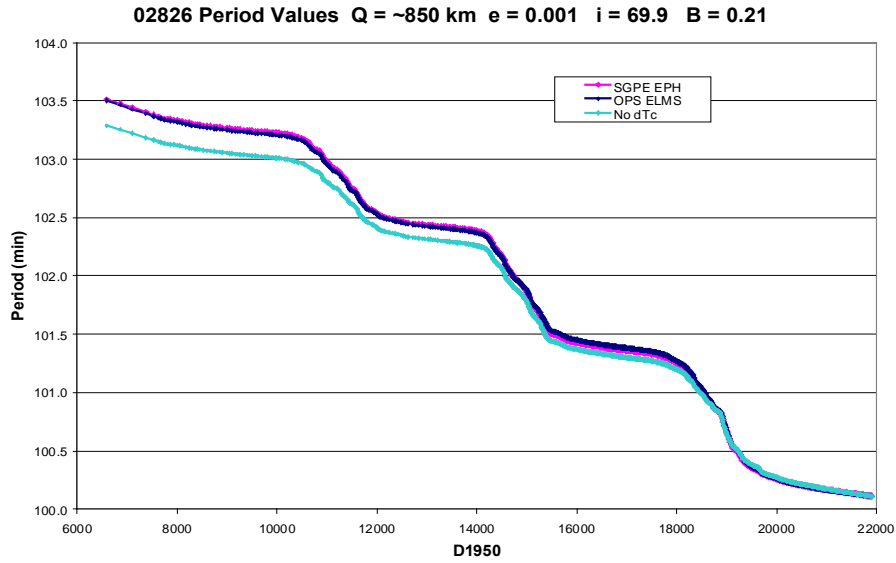


Fig. 5. Variation in orbital period of sphere 02826 as a function of backward integration in days since 1950, with and without daily temperature corrections, plotted with operational mean elements. Q is the perigee height for this near circular orbit.

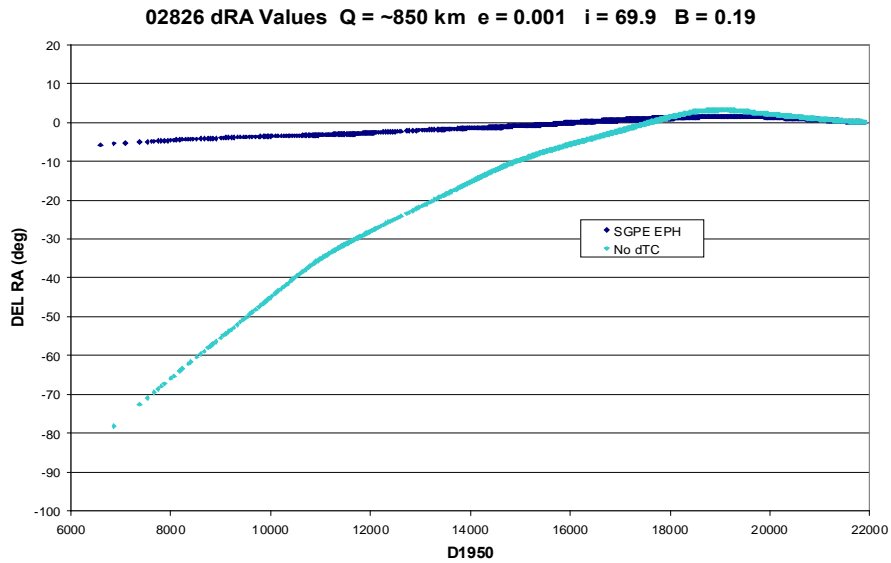


Fig. 6. Error in right ascension of node of sphere 02826 as a function of backward integration in days since 1950, with and without daily temperature corrections (dTc).

For lunisolar gravitational perturbations the original SGPE used first-order theory equations obtained from integration of the first term of the disturbing function. These analytical equations accounted for secular and long-periodic perturbations of the orbital elements. These perturbations were then extended in the updated SGPE from Kozai [7] who used the first 5 terms of the disturbing function to compute analytic equations of the orbital element variations as a function of the polar coordinates of the sun and moon. These equations are ideal for use in a semi-analytical integrator with a half-day step size, where the perturbational effects are continually updated each step with new sun and moon positions. Kozai [7] also included the perturbations due to the tidal deformation of the earth, expressed again as first-order analytical equations. Fig. 7 displays the SGPE integration of satellite 00900, a

calibration sphere at 1025 km altitude. The integration is with and without earth tides, and is compared to the historical operational mean elements. The precession of the node is highly dependant on the orbital inclination, and errors in the inclination result in large errors in the precession of the node. The SGPE integrations in Fig. 7 include all other perturbations discussed in this paper. At an inclination of precisely 90 degrees there should not be any node precession cause by geopotential zonal effects (see below). However, the solid earth tidal effects move the orbit plane away from the precise 90 degree inclination orbit, and thus cause the node to start precessing due to zonal perturbations. Fig. 8 shows the results of the nodal precession mainly due to these tides. Accounting for the tides in SGPE produces a 45 year ephemeris that almost precisely mirrors the historical observed variations in the inclination. However, very large errors occur in the integration by not including the earth tides effects. Additionally, general precession and nutation effects were included in SGPE , both of which are caused by the gravitational attraction of the sun and moon on the equatorial bulge of the earth (Cooley [8] and Newton[9]).

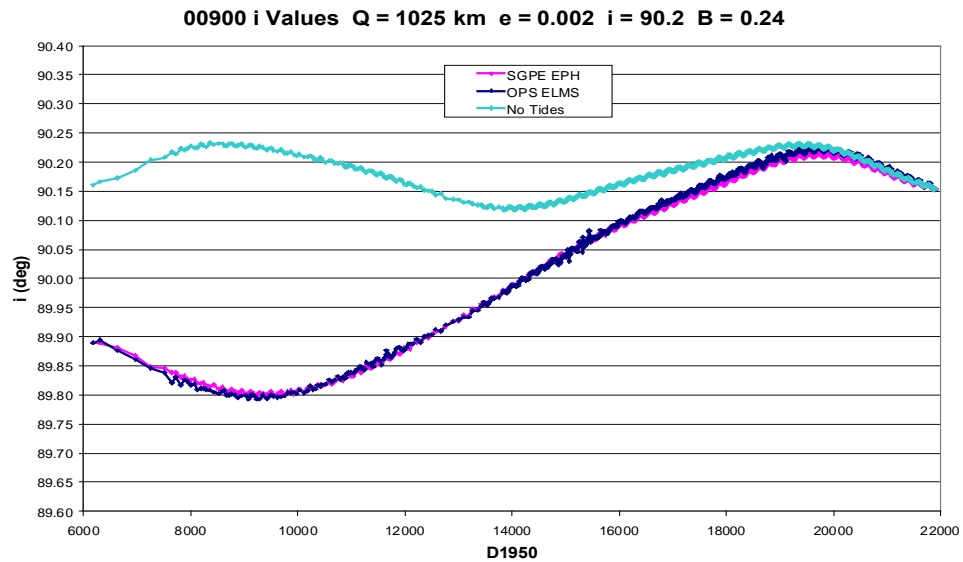


Fig. 7. Variation in inclination of sphere 00900 as a function of backward integration in days since 1950 with and without tidal effects, plotted with operational mean elements.

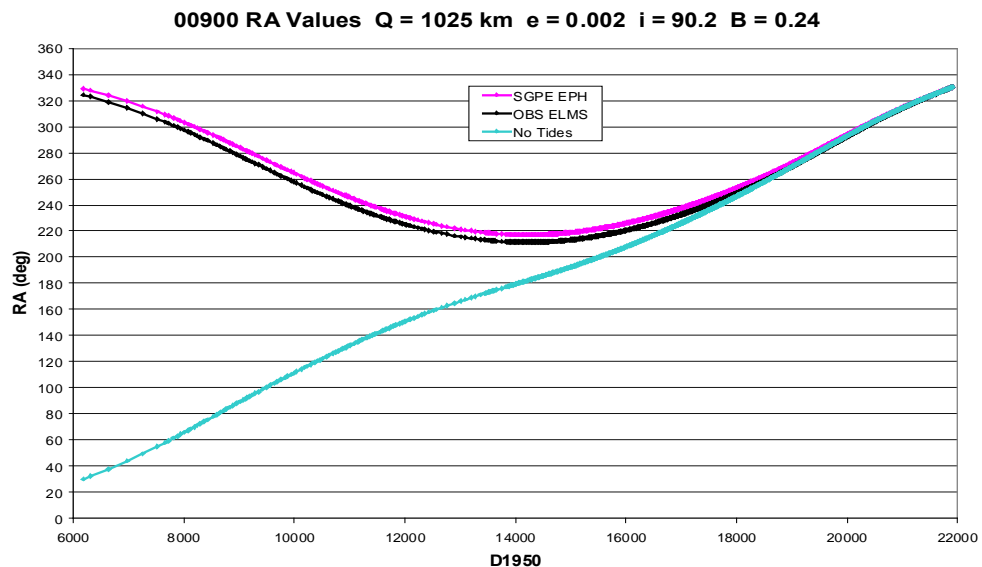


Fig. 8. Right ascension of node of 00900 as a function of backward integration in days since 1950, with and without tidal effects, plotted with operational mean elements.

The geopotential zonal perturbations produce the main effect of the precession of the orbital plan. The very large node changes listed in Table 1 are the result of this perturbational effect. These variations are represented by the even zonal coefficients of the earth's geopotential field. SGP4 uses only the first two major coefficients J_2 and J_3 for the secular and long-periodic variations in the orbital elements. However, for an accurate long-term orbit integration many more higher order terms are needed. Mueller [10] computed analytical equations for these perturbations with all inclusive terms through J_{10} , including perturbations on the node, inclination, eccentricity, and argument of perigee. These equations were included in SGPE. The equations were further extended to include J_{12} and J_{14} from Kozai [11] for additional accuracy. Finally the J_2 squared term from King-Hele [12] was included in SGPE for the node perturbation to capture the use of Kozai type mean elements used in the SGPE integration.

The last SGPE included perturbation of the node affecting low earth orbits is solar radiation pressure. Koskela [13] developed a complete theory for handling the effects of solar radiation pressure on satellite orbits. The effect of the satellite being eclipsed by the Earth was considered, as well as the case when the satellite is continuously exposed to sunlight for many orbits. The eccentric anomaly was used as the independent variable in the analytical approach, with perturbational equations being obtained in closed form. The key to obtaining an accurate solar radiation perturbation is determining an accurate radiation pressure coefficient. The non-linear least squares solution for B below discusses the solar radiation pressure coefficient determination.

3. Non-Linear Least Squares Solution for B Value

A major key to accurate long term predictions is an accurate atmospheric drag computation, which means using an accurate ballistic coefficient B value with accurate density values. A non-linear least squares model was developed for use with the SGPE integrator. This model uses multiple years of mean element semi-major axis values to solve for an average B value and an initial mean motion value. The great majority of satellite debris has B variations due to frontal area variations resulting from the non-spherical shapes of the pieces. Therefore, it is necessary to obtain the best average value possible for long term predictions. Obtaining an accurate B average value assumes that the rotation and precession of the rotation axis of the pieces will have periodicities shorter than the fit span used for the average B determination. More than 550 known catalogued pieces of debris were used in the validation process to determine integration accuracy as a function of the B fit span (in years), the magnitude of the B value, the altitude of the orbit, and the length of the SGPE integration. One sigma standard deviation error tables were computed from the validation cases, and these are used, as described below, to determine boundary values for assigning event association probabilities.

The non-linear least squares program is designed to process at least 3 iterations until the B solution reaches a consistent value. Numerical partials are computed for iterating on the initial mean motion and B values. Following the B solution the left over residuals in inclination, right ascension, and argument of perigee are then least squares fit to account for any remaining unmodeled perturbation effects. These new terms are then applied as additional perturbations in the SGPE backward integration to 1958.

Once the long-term average B value has been determined the SGPE integrator is used to generate 2-line mean elements sets every 1 to 5 days back to 1958. This mean element "ephemeris" is then compared with the mean elements from initial launch and breakup elements of the parent satellites and rocket bodies. Databases of mean elements were built using the first 30 days of elements from every launch, and elements from 20 days before and after every breakup that occurred since the 1960s. All these launch and breakup element sets (over 200,000 for launches and over 230,000 for breakups) are then compared to the element sets generated in the SGPE ephemeris file. The probability of association of a SGPE ephemeris element set with a launch or breakup element set is based upon the integration error obtained from tables of validation data described above. This association computer run (ELMCOMP) only takes a few seconds of computer time to complete. The longest computer time required for this entire process is for the SGPE integration, which takes approximately 1 second per integration year.

4. Analyst Debris Association Examples

Following an association, the launch or breakup parent element sets and the SGPE ephemeris element sets are plotted. Fig. 9 is an example of associations of several analyst debris with 2 separate breakups occurring in late 1992 and early 1993. To remove the very large nodal precession that occurred over the last two decades the element

set node values of the parent satellite 22285 are used as a reference. The node differences from the parent values of the analyst debris are then computed and this “relative” right ascension of the ascending node is plotted for each debris piece. Fig. 9 shows very high correlations (intersection of the relative nodes) of analyst debris 81535 and 89173 with the breakup of satellite 22285 in late 1992. Also, analyst debris 81215 and 89115 shows very high correlations with the breakup of 22566 occurring in early 1993. All of the elements of the analyst debris were generated from an SGPE integration going backward from the mid 2000s. This method of first computer association and then verification by plotting have been used to identify many hundreds of pieces associated with historical launches and breakups.

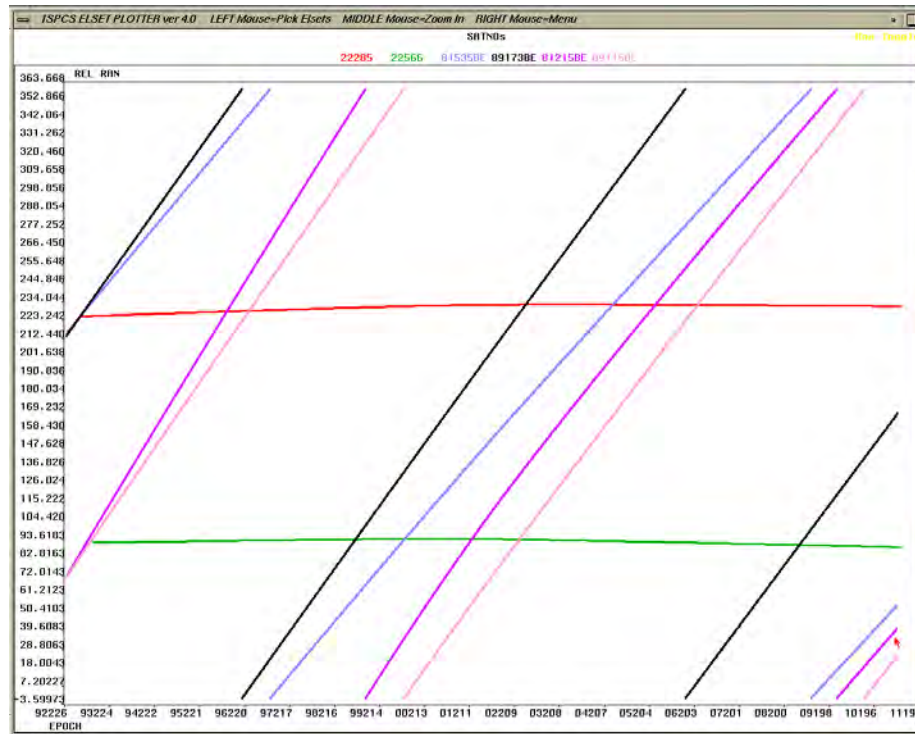


Fig. 9. Associations of analyst satellite nodes with 2 separate breakups occurring in 1992 and 1993. The relative right ascension of the ascending nodes are plotted as a function of date.

In addition to associations with historical events this association method can be used to recover lost satellites. Fig. 10 displays an example of identifying a newly tracked piece of debris with a previously lost satellite. Satellite 23293 is a moderately eccentric small piece of SL-12 debris with an apogee of $\sim 19,000$ km and a perigee height varying between 500 km and 900 km. The argument of perigee of the orbit has a very slow precession, and this fact coupled with the varying perigee height produced conditions such that the debris was not able to be tracked after mid 2005. The green curve in the figure shows the tracking from breakup to 2005. The non-linear least squares B solution was obtained with the elements prior to loss of tracking, and the SGPE integrator was used to generate an ephemeris of element sets from 2005 through 2010 (red curve). This element set ephemeris was then compared with known analyst debris and a high correlation was obtained with debris piece 83984 that was only tracked since the beginning of 2010. Fig. 10 displays the correlation in inclination, node, apogee and perigee heights, period, and eccentricity, and leaves no doubt that 83984 is indeed the lost 23293 piece. This example illustrates the use of SGPE not just for historical event associations but also for the recovery of lost satellites.

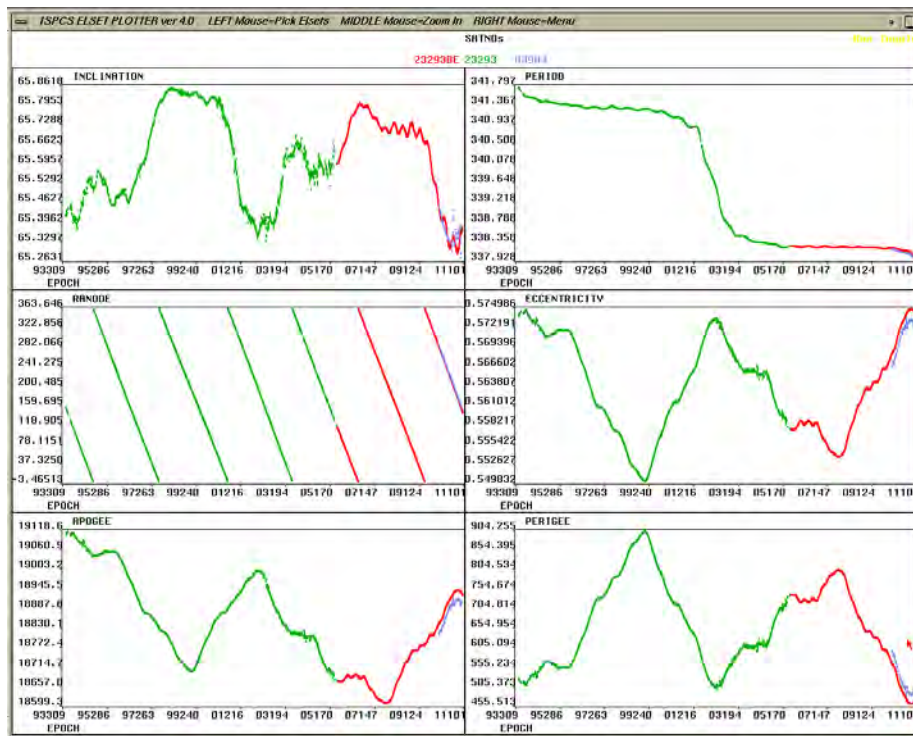


Fig. 10. Mean element plots of satellite 23293 (green) up until time of loss (2005), SGPE predicted elements (red), and analyst satellite 83904 elements (blue).

5. Conclusions

It has been demonstrated that currently tracked un-catalogued analyst satellites can be associated with historical launch, breakup, or other anomalous events that have occurred as far back as the 1960s. A fast semi-analytical integrator SGPE has been developed using analytical equations to account for the great majority of perturbations of the orbit plane. Additionally a fast accurate method of determining the average ballistic coefficient has been developed using a non-linear least squares program fitting standard 2-line semi-major axis mean elements. Associations of currently tracked satellite debris with historically lost satellites are now also possible. Cataloging the currently thousands of pieces of analyst debris with accurate international designators is now possible.

6. References

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