Improved orbit predictions using two-line elements

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Abstract

The density of orbital space debris constitutes an increasing environmental challenge. There are two ways to alleviate the problem: debris mitigation and debris removal. This paper addresses collision avoidance, a key aspect of debris mitigation. We describe a method that contributes to achieving a requisite increase in orbit prediction accuracy for objects in the publicly available two-line element (TLE) catalog. Batch least-squares differential correction is applied to the TLEs. Using a high-precision numerical propagator, we fit an orbit to state vectors derived from successive TLEs. We then propagate the fitted orbit further forward in time. These predictions are validated against to precision ephemeris data derived from the International Laser Ranging Service (ILRS) for several satellites, including objects in the congested sun-synchronous orbital region. The method leads to a predicted range error that increases at a typical rate of 100 meters per day, approximately a 10-fold improvement over individual TLE’s propagated with their associated analytic propagator (SGP4). Corresponding improvements for debris trajectories could potentially provide conjunction analysis sufficiently accurate for an operationally viable collision avoidance system based on TLEs only.

We discuss additional optimization and the computational requirements for applying all-on-all conjunction analysis to the whole TLE catalog, present and near future. Finally, we outline a scheme for debris-debris collision avoidance that may become practicable given these developments.

Key words:
Space debris, Conjunction analysis, Orbit prediction

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1 Introduction

Collisions in orbit pose a threat to spacecraft, to astronauts and to the global commons of near-earth space. Several collisions have already occurred between spacecraft and debris, while the Iridium 33/Cosmos 2251 collision of January 2009 represented the first documented satellite-satellite collision. Unfortunately, the cumulative number of collisions thus far is consistent with the prescient predictions of a runaway chain reaction (Kessler and Cour-Palais, 1978).

Some traditional methods of debris mitigation include minimising the number of components created in separation events, minimising the probability of accidental explosion by depleting fuel tanks and manoeuvring to re-entry or graveyard orbits at the end of life. These constitute important measures. However, models have shown that even the extreme (and unrealistic) situation of “no new launches” is inadequate to curb runaway growth (Liou and Johnson, 2008). Two additional ways to alleviate the problem are debris removal and collision avoidance.

For collision avoidance one needs orbit predictions to be sufficiently accurate so as not to imply an impractical number of collision avoidance maneuvers. This paper describes a method to increase the orbit prediction accuracy based on publicly available TLEs. In addition, a means of doing debris-debris collision avoidance is desired since collision avoidance only on the subset of conjunctions involving a maneuverable spacecraft probably does not suffice to curb debris growth: this is also addressed in the paper by proposing a scheme for debris-debris collision avoidance.

Many satellite owner operations have inadequate (if any) processes for conjunction assessment and collision avoidance. They would have to screen their asset(s) against all other space objects. The only source of knowledge at their disposal for the majority of other objects is the publicly available two-line element (TLE) sets. But predictions based on TLEs using the associated analytic propagator (SGP4) are not sufficiently accurate to warrant maneuvering to avoid potential collisions: resulting in an unacceptably large number of potential collisions per space object, each of which has very low probability. The problem is similar for debris-debris conjunctions except then both objects, not just one, are subject to these imprecisions.

To address this TLE/SGP4 accuracy problem, we investigated several methods to improve the propagation errors for non-maneuvering orbital objects whilst using only TLEs as input data. The following research target was set: “increase the predictive accuracy for orbital objects, using only historical TLE data, such that it enables operational conjunction assessment for collision
avoidance”.

In the following sections, we describe one approach to this target, assess its accuracy, and discuss the requirements for extension to the entire space object catalog. Finally, we propose a new method of debris-debris collision avoidance enabled by long-term high-accuracy conjunction assessment.

2 Method: TLE orbit fitting and propagation

Fieger (1987) performed least-squares fits to (very) long sequences of TLEs using a semi-analytic (not a numerical) propagator, obtaining dramatic improvements in (very) long-term predictions of certain orbital elements for artificial satellites.

ESA have developed techniques to use TLEs from the publicly available catalog to initially screen their sun-synchronous orbit (SSO) spacecraft ERS2 and Envisat for conjunctions (Flohrer et al., 2009). Telemetry from their operational spacecraft provide precision orbital ephemerides (POEs) for those spacecraft. For screening against all other potentially conjuncting objects, only TLEs are used for initial screening. Flohrer et al. (2008) describe a method to estimate error covariances of TLEs in order to quantify collision probability assessments. Their method provided inspiration for the present work. Here we extend and adapt their approach: based solely on the object’s historical TLEs, we improve the accuracy of the object’s predicted position, as opposed to quantifying the accuracy of the object’s SGP4 propagation errors.

Our method is essentially to use TLE data as “pseudo-observations” and to fit an orbit to these pseudo-observations using a high-precision special perturbations propagator and traditional batch least-squares differential correction. The fitted orbit is then propagated into the future using the same high-precision orbit propagator. The prediction accuracy is assessed by comparison with precision orbital ephemeris (POE) data from the International Laser Ranging Service (Pearlman et al., 2002).

For each object we wish to analyze we choose a time window with two sections: a fitting period and a subsequent prediction period. The fitting period is initially set to ten days, typical of the period over which U.S. Space Surveillance Network observations are fitted when generating TLEs for LEO objects (Danielson et al., 2000). Section 4 details a more principled approach to determining the fitting period. The prediction period is 30 days.

For a given object, all TLEs with epochs within its entire window are obtained. However, only TLEs with epochs within the fitting period are used for
the fitting process; those in the prediction period are only used for validation and test.

Essentially we take an initial guess at a starting state vector and propagating it using a high precision propagate through the fit period. We then adjust the starting state of that trajectory to do a least squares fit of the trajectory to a series of TLEs at (or close to) their epochs.

In detail, each TLE in the fitting period is interpolated using SGP4 until the epoch of the subsequent TLE. We then generate a series of order 100 state vectors (“pseudo-observations”) equally spaced in time within the fitting period. We initialize the differential correction with a state vector derived from the first TLE in the fitting period. This is our initial guess at the state vector. This state is propagated using a high precision propagator until the end of the fit period. We measure the RMS error in the radial, in track and cross track (RIC) components of relative positions between the fitted orbit and the pseudo-observations. We then apply batch least-squares differential correction to minimize the RMS error by adjusting the parameters of the initial state. The trajectory obtained from the converged differential corrector is our fit. Finally this trajectory is further propagated through the end of the prediction period.

The high precision propagator incorporates a $60 \times 60$ EGM2008 gravity field (Pavlis et al., 2008) including solid Earth tides, point masses for the solar and lunar gravity fields and the MSISE 1990 atmospheric drag model (Hedin, 1991). Solar radiation pressure is modelled using a simple biconic approximation with Earth and Moon as eclipsing bodies. During prediction we assume constant values for solar F10.7 and Ap equal to their averages over the fitting period.

3 Results: Prediction Accuracy

We applied our method to four non-maneouvring spacecraft for which POE data were readily available (Ries, 2009): Stella, Starlette, Ajisai and Etalon-2. The summary of basic orbital and physical properties for these spacecraft are shown in Table 1. They were picked in part for their range of orbital altitudes.

Figure 1 shows examples for two of these spacecraft – the highest and lowest altitude – comparing our method’s predictions to those of a single TLE propagated with SGP4. The TLEs for these two objects exhibit quite different behaviours. Nevertheless, in both cases the TLEs propagated with SGP4 (black) depart rapidly from the true position of the satellite whereas our numerically fitted and numerically propagated orbits (red) maintain higher
Fig. 1. Typical position errors with respect to POE (“truth”) using our method (red), and using SGP4 (black) for Stella (upper panels) and Etalon-2 (lower panels). The prediction period is $0 < t \leq 30$ days. The fitting period is $-10 \leq t \leq 0$ days for Stella and $-90 \leq t \leq 0$ days for Etalon-2. All fitting uses only publicly available TLEs. Truth data are only used for plotting. Selection of the length of the fitting period is discussed in Section 4. Also shown are the updated TLEs (no prediction) in blue.

Accuracy, particularly over the long term.

Precision orbit ephemerides (“truth data”) define the $x$-axis for plotting, but truth data are not used in the fitting process. Nor are they used for tuning hyperparameters (e.g. length of fitting interval, see Section 4). Improvements
Table 1

<table>
<thead>
<tr>
<th>Satellite</th>
<th>Perigee (km)</th>
<th>Eccentricity</th>
<th>Period (min)</th>
<th>Inclination (deg)</th>
<th>Mass (kg)</th>
<th>Diameter (m)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Stella</td>
<td>800</td>
<td>0.0206</td>
<td>98.6</td>
<td>101</td>
<td>48</td>
<td>0.24</td>
</tr>
<tr>
<td>Starlette</td>
<td>812</td>
<td>0.0206</td>
<td>104</td>
<td>49.8</td>
<td>47</td>
<td>0.24</td>
</tr>
<tr>
<td>Ajisai</td>
<td>1490</td>
<td>0.0010</td>
<td>116</td>
<td>50.0</td>
<td>685</td>
<td>2.15</td>
</tr>
<tr>
<td>Etalon-2</td>
<td>19120</td>
<td>0.0007</td>
<td>675</td>
<td>65.5</td>
<td>1415</td>
<td>1.29</td>
</tr>
</tbody>
</table>

in prediction accuracy are typically seen in all three axis (RIC), especially more than a day into the future. For predictions of less than one day the errors appear to be dominated by TLE bias (with respect to truth) which our method cannot remove.

For each of the four satellites, we performed 50 runs of our method using different starting dates distributed uniformly in 2004. Summary statistics of these runs using “box-and-whisker” plots appear in Figures 2 and 3. The TLE+SGP4 errors grow more rapidly in every case. The typical prediction errors for our method are 3km at 30 days out, corresponding to ~ 100 m/day prediction error growth. The worst case is Stella where the median error grows < 150 m/day and the best is Etalon at < 30 m/day. The maximum error for our method over all four satellites is < 300 m/day. These compare to typical growth of 1.5 km/day for TLE’s propagated with SGP4 for these objects. The ratio of median prediction error using SGP4 error vs. our method error at 30 days range from 3 for Etalon-2 to 50 for Starlette, and averages 15.

Thus for these satellites, our method exhibits approximately one order of magnitude improvement in prediction accuracy over TLEs propagated with SGP4. Since the improvement is in all three directions (RIC) the resultant decrease in position covariance ellipsoid volume is likely to reduce false positive conjunctions by at least an order of magnitude.

Since the TLEs for these objects have an instantaneous range bias from truth of 0.8±0.3 km and error growth of approximately 1.5 km/day, only after approximately one day can one detect the benefits of our method. Our fitting method has a similar initial range bias to TLEs but error growth of only 100 m/day. These latter errors are similar to those published in the open literature the high accuracy special perturbations catalog(s) maintained by the US Space Command (Neal et al., 1997; Coffey et al., 1998; Boers et al., 2000). Table 2 compares the accuracy of various prediction methods. All of these methods use data based on observations from the U.S. Space Surveillance Network (SSN). The space fence, one element of the SSN, has raw directions cosines with an accuracy of approximately 10 meters for objects in LEO(Hayden,
Fig. 2. “Box and whisker” plots of range errors for predictions using our fit method (red), and using SGP4 (black) for four satellites. Each panel summarizes 50 runs corresponding to 50 random starting dates in 2004. The $x$-axis is prediction time in days. The $y$-axis is prediction error in km. The lower and upper whiskers extending from each box bound the minimum and maximum prediction errors, respectively. The lower and upper box edges bound the 25th and 75th percentiles, respectively. The glyphs inside the boxes mark the median prediction error. All errors are plotted with respect to POEs obtained from the ILRS. Truth data are only used for plotting. All fitting uses only publicly available TLEs.

1962; Gilbreath, 1997; Schumacher et al., 2001). ILRS sensor range accuracy, in comparison, is approximately 10cm for objects in LEO (Pearlman et al., 2002).

It is important to note that improved predictions require both more accurate initial states and more accurate propagators. An initial attempt was made to simply take the most recent TLE, convert it to a state vector and
Fig. 3. Additional summaries of range errors for predictions using our method (red), and using SGP4 (black) for four satellites. The LHS shows our method’s errors for predictions \( \leq 30 \) days. The RHS compares our error (red) to TLEs propagated with SGP4 (black) for the first 10 days (5 in the case of Starlette since SGP4 was diverging rapidly). Each box and whisker plot summarizes 50 runs corresponding to 50 random starting dates in 2004. The \( x \)-axis is prediction time in days. The \( y \)-axis is prediction error in km. All errors are plotted with respect to “truth” (i.e. POE obtained from the ILRS). Truth data are only used for plotting. All fitting uses only publicly available TLEs.

propagate using a high precision propagator. This did not yield statistically significant improvements over the TLE propagated with SGP4. The key to
Table 2
Prediction accuracy of different methods for LEO satellites between 600km and 800km altitude.

<table>
<thead>
<tr>
<th>Method</th>
<th>Accuracy (m/day)</th>
<th>References</th>
</tr>
</thead>
<tbody>
<tr>
<td>TLEs + SGP4</td>
<td>100-3,000</td>
<td>Boyce (2004); Chan and Navarro (2001); Flohrer et al. (2009); Kelso (2007); Muldoon and Elkaim (2008); Snow and Kaya (1999); Wang et al. (2009)</td>
</tr>
<tr>
<td>TLEs + new scheme</td>
<td>50-200</td>
<td>present paper</td>
</tr>
<tr>
<td>High Accuracy catalog + SP</td>
<td>50-200</td>
<td>Neal et al. (1997); Coffey et al. (1998); Boers et al. (2000)</td>
</tr>
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</table>

The new method is finding a better starting state, one that cannot be derived from any single TLE. Evidence for this result (that a high precision numerical propagator alone is insufficient to improve the accuracy of TLE-based predictions) can be seen in Figure 4: as the fitting interval decreases, the prediction accuracy diverges. In the limit, with the “fitting interval” reduced to a single TLE, a “high precision” propagator does not produce high precision results.

4 Fitting Debris objects

The majority of tracked space objects (i.e debris) do not have precision orbital ephemerides – they are not tracked by the ILRS and they transmit nothing. Many are, however, tracked by the United States Space Surveillance Network (SSN). Unfortunately, the only publicly available source of data derived from these SSN observations are, at present, the TLEs accessible at space-track.org (and archived e.g. at celestrak.com).

While there is currently no publicly available source of high-accuracy trajectories for debris objects, we believe the method described above could be applied to refine the predictions of debris objects’ future positions based only upon comparatively inaccurate TLEs, facilitating conjunction assessment and collision avoidance. However, there are issues which complicate the application of our procedure to debris.

One set of issues arise when TLEs are the only source of data - when we have no “truth” in order to perform validation and test. This is solved by the use of TLEs from beyond the fitting period as a proxy for truth data. While such “future TLE” data are not as accurate as precision ephemerides from laser ranging, they are accurate enough for verification and validation, and for
Fig. 4. When optimizing the length of the fitting period, TLEs from beyond the fitting period can be used as “proxy” data and substitute for precision ephemerides (“truth”). Left: prediction error with respect to future TLE proxy vs. fitting period. Right: prediction error with respect to truth vs. fitting period. Each box and whisker plot summarises 50 runs corresponding to 50 random starting dates in 1994.

most objects they are all that is publicly available. Fieger (1987) used TLE data alone to determine the accuracy of his predictions. We used such data for selecting the length of the fitting interval for different satellites.

Our initial manual experiments suggested that different fitting periods were
optimal for different orbit categories. These results are consistent with Alfriend et al. (2002); Danielson et al. (2000). We made a more principled investigation, utilizing cross-validation, to optimize the length of the fitting interval on a per-object basis using TLEs alone for both fitting and cross-validation. Figure 4 shows how a fitting period of 10 days was determined to be optimal for satellite Ajisai. The important point shown in the figure is that a 10 day fitting period is optimal both when validating predictions with truth data and when validating predictions with future TLEs. And the 10 day fitting period is optimal whether predicting 5, 10 or 30 days into the future. Thus it appears TLEs can be used as a proxy for truth data when optimizing the length of the fitting period on a per object basis.

Another set of issues is that for debris objects one must also solve for area to mass ratio (drag) and validate this as well using only future TLEs. We are currently performing these experiments.

5 Outlook

5.1 Limitations

The error in a sequence of TLEs consists of two components: bias and variance. We use least-squares to minimize the deleterious effects of TLE variance, essentially by smoothing. But smoothing cannot remove bias and this is a limitation. Nevertheless, by smoothing out the variance in a principled way we have shown it is possible to make significantly better predictions, even without addressing bias.

Another limitation is that of drag, particularly whether the technique will work for lower altitude objects where drag becomes more important and for debris objects with unknown masses and effectively non-constant areas. We believe we can solve for at least some drag-related parameters (area-to-mass ratio), but this has not yet been tested.

5.2 Methodological Improvements

Further accuracy improvements could come from a principled analysis of the number of fitting points and their weighting as a function of location within the fitting period, the removal of outlier data (i.e. filtering) and other techniques from statistical orbit determination, treating TLEs as observations. Also, recent work Legendre et al. (2008); Muldoon and Elkaim (2008) suggests
that there may be simple global transformations of TLE data that can improve TLE/SGP4 accuracy, essentially by reducing systematic biases. Such systematic corrections might also be further refined on the basis of orbital parameters.

In addition, the scheme should be tested on actual known conjunctions in order to analyse its efficiency for conjunction assessment. These issues will be the subject of a future paper.

5.3 Computational Scaling

To assess the computational requirements for performing conjunction analysis on all space objects in the publicly available catalog, a simple conjunction analysis system was parallelized\(^1\) on the Pleiades supercomputer at NASA Ames Research Center. Using approximately 200 CPU cores, checking all objects against all objects for conjunctions 7 days into the future with takes about a minute. The conjunctions found this way involving a spacecraft (a small fraction of all conjunctions) precisely match those found by SOCRATES (Kelso and Alfano, 2005). The same system takes about 40 minutes to perform 7 days of all-on-all conjunction assessment of a “simulated S-band fence” catalog (obtained from the NASA Orbital Debris Office) containing simulated TLEs for approximately 2.5 million objects of size $\geq 2\text{cm}$. This demonstrates one critical aspect of performing all-on-all conjunction assessment on the present and expected near-future catalogs. Scaling the fitting method discussed above to the future catalog will be the subject of a later paper.

5.4 A New Scheme for Debris-Debris collision avoidance

Here we briefly outline a concept for debris-debris collision avoidance. It is speculative at this stage but could be very significant if it turns out to be feasible in practice. An along track $\Delta V$ of 1 cm/s provides of order 1km displacement per day. Since error growth for our method is 100m per day, such small manoeuvres might suffice for collision avoidance. For manoeuvres this small one could consider using radiation pressure to impart the necessary momentum. To verify the conjecture, we increased the flux of radiation by a factor of 10x the solar radiation constant for 10 minutes on a typical debris object (area to mass ratio 0.1 $\text{m}^2/\text{kg}$) in SSO. We then propagated that for a further 5 days. The displacement was $> 0.1 \text{km/day}$ along track compared to a

\(^1\) This work was performed by Chris Henze of the NASA Advanced Supercomputing Division at NASA Ames Research Center. Details will be forthcoming in a future publication.
reference object that was not so irradiated. Two cumulative such events, if one ensured appropriate geometry, would thus lead to the necessary along-track displacement.\footnote{This idea was first conceived in discussions with Dr. Rüdiger Jehn from ESA, July 2009.}

A factor of 10x the solar rational constant could conceivably be applied from the ground. For example, a 10kW class laser attached to a 1m class telescope with appropriate tracking, could, (considering diffraction limits only) provide the necessary radiation pressure for a debris object of area $<1\text{m}^2$ in SSO.

There are some strong requirements for the architecture of such a system, e.g. pointing accuracy, overcoming atmospheric distortion, laser deconfliction as well as political issues. One would need to perform a detailed engineering analysis of the feasibility of such a system, particularly the viability of maintaining small beam divergence and tracking. However, if feasible, laser-radiation-pressure-induced debris-debris collision avoidance might reduce or eliminate the need for an active debris removal program. We interpret the results of Liou and Johnson (2009) to suggest that performing \textit{continuous} debris-debris collision avoidance for a relatively small number of carefully selected objects could curb debris field growth. If removal of approximately 5-10 additional carefully selected objects per annum reverses debris growth, then continuous debris-debris collision avoidance for those objects would, to first order, have the same effect. Though the scheme can not prevent collisions with objects below the detection threshold, this can presumably be counteracted by applying the technique to more objects.

Additional debris-debris collision avoidance could reduce the net debris density, eventually to below a critical level: nulling debris creating collisions would once again allow atmospheric drag caused re-entry to dominate. Thus, in principle applying debris-debris collision avoidance on a small subset of the debris population for a finite period of time could reduce the debris density to below critical. Thus this method might provide not just a stop gap measure, but a permanent solution.

\section{Conclusions}

This fitting and propagation method, based solely on TLEs, could potentially provide initial conjunction analysis sufficiently accurate for an operationally viable collision avoidance system. If similar improvements are possible for debris objects, as we expect, then it could be used for all-on-all conjunction assessment for collisional avoidance based on the publicly available catalog.
However, to ensure that the scheme works in an operational setting, it would be necessary to have additional data: the so-called “uncorrelated objects” (a.k.a. “analysts set”) which account for approximately 30 percent of potential conjunctions in LEO (Newman, 2008). Further, we show that key elements of all-on-all conjunction assessment is possible with moderate computer infrastructure, even with the large increase in size of the catalog of tracked objects that is expected in the next few years.

Finally, given accurate predictions, we claim debris-debris collision avoidance may be possible by externally inducing small manoeuvres using radiation pressure from a 10 kW class power density ground-based laser. If feasible, this could negate the need for a large scale and costly debris removal program.

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