REAL-TIME ESTIMATION OF SGP4 ORBITAL ELEMENTS FROM GPS NAVIGATION DATA

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ABSTRACT – The availability of GPS receivers for spaceborne applications has opened new prospects for satellite orbit determination over the past decade. In combination with a Kalman filter, accurate estimates of the spacecraft position and velocity can be obtained in real-time onboard the space vehicle. Traditionally, numerical methods of varying complexity are applied for propagating the state vector between measurements and updates of the state vector are referred to the epoch of the latest measurement. This approach is well-suited for applications requiring instantaneous state estimates (e.g. sensor pointing or geocoding of images), but cannot easily be applied for onboard mission planning purposes (resource allocation, eclipse prediction, station contact forecast, etc.) that require orbit predictions. A different concept has therefore been developed, which offers increased onboard autonomy and is particularly promising for small satellites with moderate accuracy requirements. Here, an analytical orbit model is used to describe the spacecraft trajectory and mean epoch elements are estimated instead of the instantaneous, osculating state vector. Making use of the SGP4 orbit model that is compatible with NORAD twoline elements, an epoch state Kalman filter has been implemented and tested with GPS flight data of GPS/MET and SAC-A. It is demonstrated that the proposed method provides accuracies compatible with that of the analytical model and is robust enough to handle large data gaps in case of limited onboard resources for GPS operations. By adjusting the ballistic coefficient along with the mean epoch elements, a considerable improvement of mid-term orbit predictions is achieved over methods that are restricted to the estimation of the state vector alone.

1 - INTRODUCTION
Real-time onboard orbit determination is commonly considered as a first and vital step towards increased autonomy in the area of spacecraft navigation. Today, onboard orbit determination is of growing interest even for small-satellite missions due to the availability of low-cost GPS receivers [1,7]. Irrespective of the particular sensor type employed, some sort of filtering and dynamical modeling must be applied to the measurements to obtain reliable estimates of the spacecraft state vector. Even for GPS receivers providing both position and velocity as part of the navigation solution, a 1-2 orders of magnitude increase in accuracy may be achieved by applying a dynamical orbit determination algorithm [10].

A Kalman filter based orbit determination process generates estimates of the instantaneous spacecraft state vector (or, equivalently the osculating orbital elements) at discrete time steps that usually coincide with the arrival of new measurements. Information on the spacecraft location resulting from the onboard orbit determination can directly be used in a variety of real-time
applications like geo-coding of images, sensor and antenna orientation, as well as 3-axis attitude control. On the other hand, the derived orbital parameters would be difficult to use for onboard orbit forecasts over several orbits ahead of time. This is due to the fact that reliable orbit predictions from osculating orbital elements or state vectors require complex numerical propagation models which often exceed the available onboard computer resources. Therefore, onboard scheduling functions that require the prediction of spacecraft-related events like shadows, ground station contacts and image target hits, remain difficult to implement, despite the apparent availability of onboard orbit information.

The disadvantage of numerical orbit prediction may be overcome by the use of analytical orbit models, that can be evaluated at arbitrary times and do not require a step-wise integration of the trajectory. This allows off-line predictions over mid- and long-term time scales (multiple revolutions to multiple days) at the expense of a decreased short-term (<1 rev) accuracy. Analytical models cannot, however, be used with osculating orbit information but require a dedicated set of mean orbital elements. Furthermore, no explicit conversion from osculating to mean elements does exist. Any erroneous use of mean elements in a numerical model or osculating elements in an analytical model would result in semi-major axis offsets of 1-10 km with associated along track errors of up to 100 km per orbit. To cope with this problem, the onboard orbit determination algorithm must be reformulated for the direct estimation of mean orbital parameters that are compatible with the applied analytical orbit model [9].

In the sequel an epoch state Kalman filter for use with the SGP4 orbit [6] is described and tested with GPS flight data from various low Earth-orbiting spacecraft. The choice of SGP4 is based on its widespread application for near-circular, low altitude satellites and its high communality with existing ground equipment and commercial off-the-shelf software (COTS) products. Aside from this, the proposed concept is general enough to be likewise applied with other analytical orbit theories.

2 - ESTIMATION OF SGP4 MEAN ELEMENTS FROM GPS OBSERVATIONS

2.1 – The SGP4 Model

In maintaining their catalogue of known space objects, the North American Aerospace Defense Command (NORAD) determines orbital parameters of some 8000 satellites and space debris from radar tracking data. Traditionally, analytical orbit models of moderate complexity are used for this purpose, since high accuracy is less important than computational speed.

The first of the NORAD orbit models (SGP - Simplified General Perturbations) was developed in 1966 and used to provide orbital elements in the form of twoline elements. They contain the Keplerian elements eccentricity $e$, inclination $i$, right ascension $\Omega$, argument of perigee $\omega$ and mean anomaly $M_0$ at epoch time $t_0$. Traditionally, the anomalistic mean motion $n_0$ is used as sixth orbital element instead of the semi-major axis $a$. The SGP4 theory was developed in 1970 and used continuously since then for the production of twoline element sets for low-Earth satellites by NORAD. It is based on the analytical theory of Brouwer and accounts for the Earth gravitational field through zonal terms $J_2$, $J_3$, $J_4$, and the atmospheric drag through a power density function assuming a non-rotating spherical atmosphere. Short periodic perturbations, however, are only modeled to first order ($J_2$). A detailed description of the SGP4 model as applied for the generation of NORAD 2-line elements is given in [6].

The SGP4 model, which is denoted by the symbol $S$ in the sequel, relates the spacecraft state vector

$$y(t) = \begin{pmatrix} r(t) \\ v(t) \end{pmatrix} = S(\sigma_0, B, t)$$

(1)
at time $t$ to a set of (SGP4 specific) mean elements $\vec{\alpha}_0 = (\vec{a}_0, \vec{e}_0, \vec{\Omega}_0, \vec{\omega}_0, \vec{M}_0)$ at epoch $t_0$ and a ballistic coefficient $B$ describing the effective area-to-mass ratio. Here, we have chosen a Keplerian set of orbital elements, which is identical to the NORAD elements set, except for the semi-major axis $\vec{a}_0$. The latter parameter is identical to the “original mean motion” $\alpha^*_0$ of the SGP4 theory, that results from the “mean mean motion” $n_0$ by removing the secular $J_2$ perturbations from the associated Keplerian semi-major axis.

For the subsequent discussion SGP4 is considered as a 6-dimensional, continuous and differentiable function of time, depending on seven dynamic parameters. Because of the well-known singularity of Keplerian orbital elements for orbits that are either circular or equatorial, a different parameterization of the SGP4 model is, however, required for the adjustment of orbital parameters from observations. We therefore introduce the concept of a “mean SGP4 state vector”, which is free from singularities, and – in the ideal sense – free from orbital perturbations. Making use of the well known mapping between osculating Keplerian elements $a = (a, e, i, \Omega, \omega, M)$ and the six dimensional (osculating) state vector $\mathbf{y} = K(\mathbf{a})$ (cf. [3]) one may define the mean state vector at epoch $t_0$ by the expression

$$\mathbf{y}_0 = K(\vec{\alpha}_0) \quad (2)$$

where, again, $\vec{\alpha}_0$ denotes the SGP4 mean elements at the same epoch. Making use of the inverse function $K^{-1}[3]$, eqns. (1) and (2) may be combined into the resulting expression

$$\mathbf{y}(t) = S(K^{-1}(\mathbf{y}_0), B, t) = s(x, t) \quad , \quad (3)$$

which relates the osculating state vector for a given time $t$ to the combined parameter vector $x = (\mathbf{y}_0, B)$ via the composite function $s$. Compared to the original formulation, $s$ is non-singular even for circular or equatorial orbits and the partial derivatives of $s$ with respect to the orbital parameters are well defined throughout the phase space of interest.

### 2.2 - Osculating to Mean State Vector Conversion

For epoch $t_0$ and given ballistic coefficient $B$ (e.g. $B = 0$), eqn. (3) may be inverted using a fixed point iteration of the form

$$\mathbf{y}_0^{(0)} = \mathbf{y}_0 \quad , \quad \mathbf{y}_0^{(i+1)} = \mathbf{y}_0^{(i)} + (\mathbf{y}_0 - s((\mathbf{y}_0^{(i)}, B), t_0)) \quad , \quad (4)$$

to find the mean state $\mathbf{y}_0$ at this epoch from the corresponding osculating state vector $\mathbf{y}_0 = \mathbf{y}(t_0)$.

While (4) provides a useful point-to-point conversion from osculating to mean state vectors, the result is only approximate due to inevitable modeling deficiencies in the SGP4 theory. Considering the neglect of higher order perturbations as well as sectorial and tesseral gravity field components, eqn. (3) should properly be written as

$$\mathbf{y}(t) = s((\mathbf{y}_0, B), t) + \epsilon(t) \quad , \quad (5)$$

where $\epsilon$ denotes the time dependent model errors. As a rule of thumb, the neglected terms in the SGP4 orbit model give rise to position errors of 2 km. Assuming that these errors have a zero mean value over multiple revolutions, an iterative least-squares fit may be used for a rigorous determination of $\mathbf{y}_0$ from a given $\mathbf{y}_0$. To this end, a reference trajectory $\mathbf{y}_i = \mathbf{y}(t_i, \mathbf{y}_0)$ at discrete time steps $t_i$ needs to be computed from the osculating state at epoch $t_0$ using a reliable force model and numerical integration. Then, starting from an initial guess (provided e.g. by (4)), a differential correction scheme is applied to find the epoch mean state vector and ballistic coefficient that minimizes the root-mean-square difference between the reference trajectory and the one computed by the SGP4 orbit model [Herman 1997].
2.3 – Kalman Filtering of the Mean Epoch State Using GPS Measurements

As suggested in [7] and [10] one may directly adjust SGP4 elements (or, equivalently, SGP4 mean state vectors) to given GPS data in a least-squares fit. Due to the processing of all measurements in a single batch as well as the use of multiple iterations, the least-squares approach to the solution of the GPS-to-SGP4 orbit determination problem is robust enough to handle erroneous data points or bad a priori parameters. Furthermore, the process can be implemented in a self-starting manner, since an a priori state vector can always be derived from the GPS navigation measurements. This makes it particularly attractive for automated, ground-based orbit determination of satellites carrying GPS receivers [8].

For the envisaged onboard application a least-squares formulation is hardly useful, however, since it implies a need for storing measurements over extended periods of time. A classical Kalman filter, estimating the instantaneous state vector, is likewise undesirable due to the non-trivial mapping of osculating to mean orbit information. As a solution to this problem an extended epoch state filter has been developed [9], which processes all measurements sequentially (and only once) to update an a priori value of the mean state vector at epoch as well as its covariance. In contrast to the classical Kalman filter, the epoch state filter does not include a state update since propagation of the estimated state to the measurement epoch is not required. Instead, it consists of a measurement update, only, which comprises the computation of the Kalman gain, correction of the current parameter estimate and the computation of the a posteriori covariance.

Residuals of the \(i\)-th GPS position measurement

\[
\mathbf{z}_i = \begin{pmatrix} x_{WGS84} \\ y_{WGS84} \\ z_{WGS84} \end{pmatrix}
\]

are formed with respect to the measurement model function

\[
h_i(x) = \Theta(t_i) \cdot (1_{3 \times 3} \cdot 0_{3 \times 3}) \cdot s(x, t_i)
\]

which is evaluated with the latest estimate \(x_{i-1}\) of the parameter vector. Here \(\Theta(t) = R(t)(GHA(t))\) denotes the Earth rotation matrix that describes the transformation from the equator and equinox of date to the Earth-fixed Greenwich meridian system. Likewise, the partial derivatives \(H_{i-1} = \partial h_i / \partial x_{i-1}\) of the computed measurements at time \(t_i\) are computed with respect to the estimated parameter vector \(x_{i-1}\) as obtained from all previous measurements. Due to the complexity of the analytical formulation of the SGP4 theory, these partials are best obtained from a numerical difference quotient approximation. In practical tests, first order differences have been found to be sufficient when used with increments of \(10^{-5}\)m (position), \(10^{-8}\)m (velocity) and \(10^{-7}\) (ballistic coefficient) and double precision computer arithmetics.

Denoting by \(x_i\) and \(P_i\), respectively, the estimated parameter vector and its covariance after processing the \(i\)-th measurement vector \(z_i\), the resulting filter equations are given by

\[
K = P_{i-1}H_{i-1}^\top(W_{i-1}^{-1} + H_{i-1}P_{i-1}H_{i-1}^\top)^{-1},
\]

\[
x_i = x_{i-1} + K(z_i - h_i(x_{i-1}))
\]

\[
P_i = (1 - KH_{i-1})P_{i-1} + Q_i.
\]

Here, \(K\) is the Kalman gain, that determines by how much the new measurement affects the estimated parameters. \(W_{i-1}^{-1}\) is the covariance of the measurement vector, which in the absence of more rigorous information may be assumed to be a multiple \(\sigma^2 I\) of the identity matrix with \(\sigma\)
denoting the measurement standard deviation of roughly 100 m for the Standard Positioning Service (C/A code navigation with S/A on). If appropriate, process noise can be added in the covariance update as indicated in (10). In this case, the process noise matrix $Q$ should be adjusted to tune the filter memory or to avoid filter divergence in the case of long data arcs and high data rates.

While the above equations have been formulated for vector valued measurements, it is generally more efficient to process each coordinate axis ($x,y,z$) individually to avoid the inversion of a 3x3 matrix. In this case $W, z$ and $h$ are reduced to scalar quantities, while $H$ and $K$ become seven dimensional row and column vectors, respectively. However, to avoid a costly recomputation of the measurement function after each individual scalar update, $h$ and $H$ are computed only once per measurement and (9) is replaced by the linearized expression

$$x'_{i} = x_{i-1} + K(x_{WGS84,i} - h^X_i(x_{i-1}))$$

$$x''_{i} = x'_{i} + K'(y_{WGS84,i} - h^Y_i(x_{i-1}) - H^Y_i(x'_{i} - x_{i-1}))$$

$$x_{i} = x''_{i} + K''(z_{WGS84,i} - h^Z_i(x_{i-1}) - H^Z_i(x''_{i} - x'_{i})) .$$

For the rejection of bad measurement data the difference $z - h$ between observed and computed measurements may be used, but care must be taken that typical errors of the SGP4 model (ca. 1 km) by far exceed the native measurement accuracy (ca. 100 m). These errors are mainly due to the neglect of short periodic perturbation terms in the analytical SGP4 model and exhibit a pronounced correlation over fractions of an orbit. As a consequence, the filter is required to process data over at least one orbit to ensure a proper averaging of the model induced errors. While short arc solutions may exhibit oscillations of the estimated state at epoch as the tracking arc increases, a suitably stable solution is obtained after at most one day as shown by the subsequent applications. This is in contrast to high fidelity numerical orbit models, that require a considerably shorter initialization phase when used with GPS measurements of the given precision.

It may be noted that GPS velocity measurements have not been considered in the above formulation of the estimation problem. Even though most GPS sensors do actually provide velocity data based on Doppler frequency measurements or filtering of the position solution, these have generally been found to be of inferior quality in comparison with position data [Montenbruck & Gill 1996]. Typical velocity errors amount to 1 m/s , i.e. 0.015% of the orbital velocity, whereas the relative position accuracy is roughly one order of magnitude better. Velocity information is, however, helpful to obtain a priori values of the mean and osculating state vectors at epoch that are required to start the non-linear estimation process. Otherwise a coarse value of the velocity at epoch can be obtained by numerical differentiation of subsequent position measurements.

4 - APPLICATIONS

The presented algorithms have been implemented into the RTSGP4 (" software for ground-based testing and filter tuning. Tests conducted for both low-drag and high-drag spacecraft as well continuous and discontinuous measurement schedules are presented below. All tests are based on actual flight data to valuate the performance under realistic conditions.

4.1 - GPS/MET (MicroLab-1)

The GPS/MET experiment [4] onboard the MicroLab-1 satellite made use of a dual-frequency TurboStar GPS receiver to probe the atmosphere during occultations of the GPS satellites near the Earth’s rim. The spacecraft was launched in April 1995 using a Pegasus launch vehicle and injected into a circular orbit of 740 km altitude and 70° inclination.

For the subsequent analysis, GPS based navigation solutions covering a seven days data arc starting on 1 October 1996 were made available by the GPS/MET project. The data are sampled at 5 minute intervals and include both position and velocity in Earth-fixed WGS84 coordinates. Based on a
least-squares orbit determination using a reliable numerical trajectory model, the root-mean-square position error of the navigation solutions was determined as 40 m in each axis. Since the resulting osculating state vector cannot, however, be compared directly to the output of the RTSGP4 filter, an independent least-squares adjustment to the SGP4 model has been carried out. It covered the entire seven days interval and provided a reference state vector at epoch for assessing the state estimates generated by the filter. Besides the state at epoch, the ballistic coefficient was also adjusted in the least squares fit, even though atmospheric drag has only little impact at the given altitude. Residuals of the GPS position measurements with respect to the adjusted SGP4 trajectory are less than 1.5 km in each axis with an r.m.s. error of 350 m over the entire data arc. This is typically ten times higher than the native measurement noise and reflects the internal accuracy of the SGP4 orbit model.

To account for this deficiency, the statistical accuracy of the measurements was set to 1 km upon processing the data in the RTSGP4 filter. Initial values of the state vector at epoch were obtained by converting the WGS84 position and velocity of the first data point to inertial values and removing the osculating perturbations as outlined above. In the absence of other assumptions, the ballistic coefficient was initially set to zero. A priori standard deviations of the estimation parameters were assumed to be 1 km (position), 1 m/s (velocity) and 0.001 (ballistic coefficient). No process noise was applied since various test runs indicated a stable filter behavior over the entire one week data arc.

Results of the GPS/MET data processing are summarized in Figs. 1 & 2, which show the variation of the estimated position and semi-major in relation to the time of the latest processed measurement. The filter takes about one day to converge to a stable solution for the position and velocity at epoch, whereas about three days are required for a reliable estimate of the ballistic coefficient. Between the second and seventh day, a linear shift in the estimated state vector is observed, which, at first sight, might be interpreted as filter divergence. As indicated by both the position and velocity standard deviation, the covariance is still large enough, however, to ensure that the filter remains adequately sensitive to new measurements. Furthermore, the linear shift of the state vector at epoch can likewise be observed, if process noise is added to the updated covariance matrix in each step. In fact, it isn’t caused by filter divergence, but reflects a dependence of the reference state at epoch on the total data arc length. This is demonstrated by supplementary least-squares adjustments of GPS/MET data to the SGP4 orbit model covering data arcs of two days, four days and six days, respectively. A comparison with the seven days data arcs exhibits a similar dependence of the adjusted state at epoch on the total data interval as observed for the filter solution. As indicated in Fig. 1, the least squares solution matches the corresponding filter output to a small fraction of the standard deviation. This confirms the proper function of the filter and illustrates its ability to cope with the inherent non-linearity of the problem.

While the observed variation of the position and velocity at epoch following the initialization phase lies within the overall accuracy of the applied SGP4 model, it is important to assess possible impacts of these errors on a long-term orbit prediction. Considering the state transition matrix of the two-body problem, it is well known that state errors at the epoch usually map into state errors of similar magnitude, provided that they do not result in a change of the semi-major axis. Restricting oneself to near-circular orbits, a semi-major axis offset \( \Delta a \) at epoch \( t_0 \) gives rise to a secularly increasing along-track error \( \Delta L = 3\pi \Delta a (t - t_0) \) at later times \( t \). It is therefore worthwhile to monitor the accuracy of the estimated semi-major axis in addition to the state at epoch itself. As may be recognized from Fig. 2, the semi-major axis converges rapidly towards its final value. A stable estimate is achieved within a day, after which it differs by less than 10 m from the least squares reference solution. This implies a maximum secular along track error of less than 1.5 km/d as early as one day after the start of the filter.
Fig. 1: GPS/MET: Error of the mean position vector at epoch as estimated by the RTSGP4 filter with respect to the result of a 7 days batch orbit determination. Diamonds indicate the results of least squares solutions covering 2, 4, and 6 days, respectively.

Fig. 2: GPS/MET: Error of the mean semi-major as obtained from the RTSGP4 filter with respect to the result of a 7 days batch orbit determination
4.2 – SAC-A

The Argentine SAC-A satellite was launched on 14 December 1998 onboard the US Space Shuttle and ejected into a 381 x 398 km orbit of 51.6° inclination. SAC-A served as an engineering test bench for new space science technology instruments and equipment that will be used in a more complex spacecraft for the Argentine space program. For the purpose of this analysis, selected data from the SAC-A GPS receiver were provided by the Comision Nacional de Actividades Espaciales (CONAE). The data set covers seven batches with a typical duration of 20-45 mins. Gaps between consecutive batches vary from 8 to 24 hours, during which the orbit is notably affected by the high atmospheric drag experienced by the satellite. Processing of the GPS navigation data with conventional onboard orbit determination algorithms based on Kalman filtering and low fidelity numerical models is therefore essentially outruled. The SGP4 formulation in contrast is well suited to handle this situation in view of its capability to model air drag and provide reasonably accurate mid-term forecasts.

Results obtained by RTSGP4 for the SAC-A test data set are illustrated in Fig. 3. As may be expected from the less favorable conditions, the estimated position values are statistically less accurate than in the GPS/MET case. Uncertainties in the estimated epoch position amount to roughly 0.5 km throughout the data arc, whereas the actual accuracy is about a factor of five better. In addition to the epoch state the filter provides an estimate of the ballistic coefficient, which allows the modeling of atmospheric drag in the orbit prediction. Within 2.3 days (i.e. after processing the third data batch), the ballistic coefficient converges to within 5% of its final value. At the same time a semi-major axis accuracy of better than 10 m is achieved, which induces an along-track position uncertainty of less than 1.5 km per day. As in the GPS/MET case, no process noise was applied in the covariance update, since the filter remained receptive to new measurements over the entire data arc.

Fig. 3: SAC-A: Error of the mean position vector at epoch as estimated by the RTSGP4 filter with respect to the result of a 4 days batch orbit determination.
5 - ONBOARD PROCESSOR IMPLEMENTATION

In order to assess the real-time characteristics of GPS-based twoline elements estimation onboard a satellite, the algorithms described above have been implemented on the development version of the onboard processor for the German BIRD mission [2]. The BIRD onboard processor is built by the Institute for Computer Architecture and Software Technology of the German National Research Center for Information Technology (GMD/First). It features an industrial Power PC 823 processor operated at a 48 MHz clock rate (without floating point support), 8MB of RAM memory as well as 8 serial and 1 parallel port for external communication. The real-time operating system BIRT developed by GMD/First separates the kernel run-time system and a hardware dependant layer, which allows an emulation on standard Linux workstations as well as an easy adaptation to different processors. BIRT is a preemptive multitasking operating system well suited for real-time and onboard applications. Processes are executed as separate threads, which are controlled by a central scheduler based on pre-assigned priorities and timers. In this way, short and high-priority activities (e.g. commanding, attitude control) can well be separated from computation intensive tasks with long duty cycles.

For execution under the BIRT operating system, a real-time version of the RTSGP4 software has been implemented in C++ making use of standard classes for vector-matrix operations and Kalman filtering [11]. A dedicated communication thread retrieves GPS navigation measurements from a global data array and passes them to the RTSGP4 thread for further processing at the respective measurement time. The interval between subsequent activations of the RTSGP4 thread may be chosen between 30 s and 5 mins, depending on a block-wise or continuous availability of GPS data. Using double precision arithmetic, a single evaluation of the SGP4 function takes about 22 msecs, while the computation of Keplerian elements from a given state vector requires 5 msec. Thus the partial derivatives of the current state with respect to the epoch state and the ballistic coefficients can be computed in a total of 200 msecs. The complete processing of a single GPS position vector takes 250 msecs on the PPC823 processor, when using three scalars as described in (11). In addition, the computation of an approximate mean epoch state vector from position and velocity, which is performed once for the initialization of the estimation process, can be performed in about 120 msecs (four iterations). Considering a maximum applicable data rate of 1 measurement per 30s, the twoline estimation requires less than one percent of the available computing time on the respective processor board.

The above figures may be compared to a traditional onboard navigation system using a numerical orbit model and Kalman filtering of the current state vector. In a representative double precision implementation using a 10x10 gravity field and a 4th order Runge-Kutta integrator, a total execution time of roughly 600 msecs per 30 s step has been measured on the same processor board. While the computation time is of the same order of magnitude for both approaches, it is evident that the numerical model provides a much higher instantaneous accuracy. The analytical methods, on the other hand, can easily cope with a reduced data rate of about one measurement per 5-10 minutes and bridge long data gaps. Thus it can even be implemented on considerably less powerful processors, whithout sacrificing the accuracy of the adjusted twoline elements.

6 - SUMMARY AND CONCLUSION

A concept for real-time estimation of SGP4 mean elements based on GPS navigation measurements has been presented. Compared to classical Kalman filters using numerical orbit models, the new approach can cope with small measurement rates and data gaps up to several days in size. Its built-in capability to adjust a free drag parameter, as well as the analytical formulation of the orbit model, facilitates mid-term forecasts and allows the implementation of onboard algorithms for the prediction of station contacts or eclipse times. The performance of the proposed algorithm has been studied using actual GPS flight data of two spacecraft in low-Earth orbit. It could be demonstrated that the filter produces adequate results both in high-drag and low-drag environments. While the use
of continuous measurements adds to the stability and accuracy of the adjusted parameters, the process is robust enough to work with a data coverage of even less than one orbit per day. This makes it particularly useful for micro-satellites with limited onboard resources and tight constraints on the permissible time of GPS receiver operations.

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