NORAD Propagators and Two Line Element Sets

The NORAD orbit propagators are popular, accurate and easy to use. They are based on the pioneering work of Dirk Brouwer, Lane and Cranford, and others. The Two Line Element (TLE) sets required by these propagators are widely distributed on the Internet and maintained by NORAD. Each orbit propagation algorithm is documented in the classic SpaceTrack Report No. 3, “Models for Propagation of NORAD Element Sets” by Felix R. Hoots and Ronald L. Roehrich. This document also contains Fortran source code and test cases for each type of propagator. A good source of TLE data and NORAD documents is Dr. T. S. Kelso’s web site (www.celestrak.com).

The following is a typical TLE for the NOAA 14 spacecraft:

```
NOAA 14
1 23455U 94089A   97320.90946019  .00000140  00000-0  10191-3 0  2621
2 23455  99.0090 272.6745 0008546 223.1686 136.8816 14.11711747148495
```

The *mean* orbital elements contained in this data are Earth-centered-inertial (ECI) coordinates with respect to the true equator of date and the mean equinox of date. They do not include the effect of nutation.

The following is a brief description of the data contained in each line of a TLE. Each item must appear in its column field in exactly the format specified.

Line 1

The first line is a twenty four character satellite name. Software that reads a database of TLEs will usually look for this name in order to find the correct data.

Line 2

```
Column          Description

01         line number of element data
03-07      satellite number
08         classification (u=unclassified)
10-11      international designator (last two digits of launch year)
12-14      international designator (launch number of the year)
15-17      international designator (piece of the launch)
19-20      epoch year (last two digits of year)
21-32      epoch (day of the year and fractional portion of the day)
34-43      first time derivative of the mean motion
45-52      second time derivative of mean motion (decimal point assumed)
54-61      bstar drag term (decimal point assumed)
63         ephemeris type
65-68      element number
69         checksum (modulo 10)
           (letters, blanks, periods, plus signs = 0; minus signs = 1)
```

Line 3

```
Column          Description

01        line number of element data
03-07     satellite number
09-16     orbital inclination (degrees)
18-25     right ascension of the ascending node (degrees)
27-33     orbital eccentricity (decimal point assumed)
35-42     argument of perigee (degrees)
44-51     mean anomaly (degrees)
53-63     mean motion (orbits per day)
64-68     revolution number at epoch (orbits)
69        checksum (modulo 10)
```

All other columns are blank or fixed.

Converting a state vector to a TLE

This section describes a numerical technique that can be used to estimate an SGP4 compatible TLE from an *osculating* state vector (position and velocity vectors). The computational steps of this algorithm are as follows:

(1) Compute the classical *osculating* orbital elements from the position $\mathbf{r}_{osc}$ and velocity $\mathbf{v}_{osc}$ vectors.

(2) Set the initial guess for the SGP4 compatible *mean* orbital elements (subscript *sgp*) to the osculating (subscript *osc*) values computed in step (1) as follows:

$$n_{sgp} = n_{osc} = \text{ mean motion}$$
$$e_{sgp} = e_{osc} = \text{ orbital eccentricity}$$
$$i_{sgp} = i_{osc} = \text{ orbital inclination}$$
$$\omega_{sgp} = \omega_{osc} = \text{ argument of perigee}$$
$$\Omega_{sgp} = \Omega_{osc} = \text{ right ascension of ascending node}$$
$$M_{sgp} = M_{osc} = \text{ mean anomaly}$$

(3) Convert the TLE to osculating position and velocity vectors using the SGP algorithm. Note that `TSINCE` in the SGP4 subroutine parameter list is set to zero.

(4) Solve the following <u>vector</u> system of nonlinear equations:

$$\mathbf{r}_{osc} - \mathbf{r}_{sgp} = 0$$

$$\mathbf{v}_{osc} - \mathbf{v}_{sgp} = 0$$

Note that $\mathbf{r}_{sgp}$ and $\mathbf{v}_{sgp}$ are updated using the SGP4 algorithm as described in step (3) during each iteration.

Essentially, this numerical method is trying to minimize the difference between the six components of the state vector input to the algorithm and the state vector computed by SGP4 during the iterative solution of the system of nonlinear equations. Since the two state vectors are not dramatically different, the algorithm converges in a reasonable time. However, it may be necessary to use a *constrained*, nonlinear equation solver in order to avoid negative values of orbital eccentricity during the solution.

Since we usually do not know the value of the first time derivative of the mean motion, second time derivative of mean motion and the *bstar* drag term, these SGP4 orbital elements are set to zero for all computations. This information could be computed by processing several state vectors centered around the time of interest and performing some type of *differential correction* process.

Warning: Do not use TLEs with other orbit propagators. They are only compatible with the SGP4, SDP4 and other algorithms used by NORAD.