A Pre-Processed Orbital Parameters Approach for Improving Cubesat Orbit Propagator and Attitude Determination

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Nanyang Technological University has built and successfully launched the VELOX-PII and VELOX-I cubesats. Both of the satellites carry a low-power consumption in-house-developed attitude determination and control subsystem (ADCS). The ADCS uses an 8-bit microcontroller. However, in-orbit propagation data indicates that the ADCS suffers high orbit propagation error due to the finite word length of the processor. High orbit propagation error could lead to a higher attitude determination error that affects the satellite’s pointing accuracy. This paper presents a pre-processed orbit parameter method to minimize the orbit propagation and attitude determination errors. The two-line-element (TLE) data is propagated to the nearest seven significant digits of a selected time epoch. The time epoch selection is required to fulfill the single precision floating point conversion constraint. The orbit propagator performance has been evaluated for modified orbital parameters in both Geocentric Celestial and true equator mean equinox reference frames, direct input of NORAD TLE, and SGP4 methods using the VELOX-I and FORMOSAT-3/Cosmic Global Positioning System data. The results show a reduction in propagation error on the order of one to two magnitudes. In addition, the method proposed has the lowest attitude determination and target pointing error when compared with different TLE input methods.

Key Words: VELOX-I, VELOX-PII, VELOX-II, Orbit Propagator, Attitude Determination

1. Introduction

Small satellites based on cubesat standards have the advantages of low development cost and time. However, the size constraint limits overall performance such as limited power budget due to smaller solar panels and less computational capability by each subsystem due to the use of a low-power microcontroller.

During VELOX-PII and VELOX-I1–3) (Fig. 1) nanosatellite missions, a high orbit propagation error of approximately 200 km was discovered. Although the VELOX-I carries a complementary metal-oxide semiconductor camera4) as the primary payload, the remote-sensing mission was always conducted in sun-pointing mode. On the other hand, the primary mission of VELOX-PII is to establish the flight heritage of several subsystems developed in-house, such as the attitude determination and control subsystem (ADCS) and power subsystem. Thus, the loss of attitude determination accuracy is less crucial in both VELOX-PII and VELOX-I missions.

However, accurate satellite target tracking or nadir pointing with better orbit propagator accuracy is highly desired for future nanosatellite programs. For example, the VELOX-II requires a target pointing error of 5 deg or less to operate the inter-satellite communication experiment. The VELOX-II attitude control accuracy is expected to be an error of 2 deg (under one sigma), inclusive of gyroscope noise and calibration error. Given that the VELOX-II’s fine sun sensor has an error of 0.8 deg and the magnetometer has an error of 0.125 µT Gaussian noises, the attitude determination (excluding attitude control error) and target line-of-sight vector acquisition errors are required to be less than 3 deg. High or-
bit propagation error in ADCS not only results in a higher attitude determination error, but also contributes to additional error in attitude control and pointing accuracy performance.

All VELOX pico and nanosatellites carry the same ADCS, which has an 8-bit microcontroller unit (MCU) with a maximum clock speed of 100 MHz. The 8-bit MCU has the advantage that it requires less power (1 W nominal operating power) than the commercial-off-the-shelf (COTS) cubesat ADCS (e.g., 4.3 W by Pumpkin IMI series). However, the MCU has a floating point input limitation (i.e., a maximum of seven significant digits). The limitation results in approximately 200 km to 400 km orbit propagation error due to truncation in the time epoch information when the two-line element (TLE) is uploaded onto the satellite. Furthermore, the IEEE standard 754-based single precision floating point to binary conversion results in an additional time information error in the orbit propagator. Consequently, an additional time offset of 5 to 30 s has been observed during the in-orbit experiment. All of these constraints affect the performance of determining attitude accuracy.

A study for improving the cubesat orbit propagation accuracy for formation flying has previously been conducted. The method focuses on introducing an appropriate force model, but it does not consider the finite word length of the MCU. On the other hand, the use of a Kalman filter with geomagnetic field measurement to improve the orbit propagation accuracy has also been investigated. Although this method avoids the input limitation of the MCU, additional computational complexity and memory are required.

A majority of the attitude accuracy determination improvement studies focus on improving the attitude determination algorithm, such as using the Unscented Kalman filter (UKF) and modified Rodrigues parameter-based nonlinear observer. In addition, a better attitude accuracy sensor, such as a star tracker has been considered to improve attitude determination accuracy. While the nonlinear filtering method, such as UKF, has the advantage of better accuracy performance, the limited MCU memory space and overall ADCS processing time have to be taken into consideration. Furthermore, an additional instrument such as a star tracker would result in additional mass and power being required for satellite operation, which leads to a higher mission cost.

By taking the computational requirements and MCU memory space into account, we propose a pre-process orbit parameters method to improve both the orbit propagation and attitude determination accuracy. Unlike the previous works whereby the focus is on either orbit propagation or attitude determination algorithm, the method proposed here is applied before the orbit parameter is uploaded onto the satellite’s ADCS MCU. It considers the finite word length limitation of on 8-bit MCU such as the input word length and single precision floating point to binary conversion error in a time epoch. The North American Aerospace Defense Command (NORAD) orbit parameters are propagated to appropriate time epochs to maximize the ADCS orbit propagation accuracy.

It is noted that the NORAD orbit parameters are expressed in true equator mean equinox (TEME) coordinate frame, while the majority of attitude determination methods assumed the Geocentric Celestial Reference Frame (GCRF). In this paper, the orbit propagation accuracy performance between TEME and GCRF are compared. Moreover, the performance of orbit propagation using the method proposed is compared using both VELOX-I and FORMOSAT-3/Cosmic GPS data, with respect to a simplified perturbation model (SGP4) and the orbit propagation data without an improved algorithm.

Furthermore, the impact of the method proposed on attitude determination performance is compared with respect to direct TLE input and mean anomaly only alternation method in terms of geomagnetic field modeling and attitude determination accuracy. In this study, the orbital parameters for both the method proposed and mean anomaly only alternation methods are expressed in terms of the TEME reference frame. Both nutation and precession effect during coordinate transformation from the earth-centered-earth-fixed (ECEF) reference frame to GCRF, and the equation of equinox effect during coordinate transformation between ECEF and TEME are neglected. This is because the results in the latter section show that the proposed method is able to fulfill the accuracy performance required without the need of transforming the orbital parameters from TEME to GCRF.

2. VELOX Attitude Determination and Control Subsystem

The data flow block diagram of the VELOX-I ADCS is shown in Fig. 2. The power subsystem supplies 5 V power to the ADCS. All ADCS mission commands, and orbital and calibration parameters are uploaded through the onboard data handling subsystem (OBDH) and stored in electrically erasable programmable read-only memory (EEPROM). In addition, the ADCS has a GPS receiver module to process GPS data.

The attitude control system (ACS) designed in-house is independent of the attitude determination system (ADS) during the sun-tracking mode. The sun-tracking algorithm only uses the sun sensor reading without the need of satellite attitude
information. This allows sun tracking to be efficiently performed during the initial launch phase because first ground contact often occurs 6 to 9 hr after orbit insertion, and updated orbital parameters cannot be uploaded to the satellite before first ground contact.

For VELOX-II, the inter-satellite experiment requires the satellite to track and point to either a low-Earth-orbit (LEO) or geostationary (GEO) satellite. The pointing error is required to be less than 5 deg. The ACS error is expected to be approximately 2 deg. Thus, the total error due to attitude determination, orbit propagation and target location acquisition is required to be less than 3 deg.

3. In-orbit Propagation Process

Figure 3 presents the uploading process for VELOX-I orbit parameters. As shown in Fig. 4, the latest orbit parameters and time epoch are directly obtained from the SpaceTrack website. Subsequently, the orbit parameters and corresponding time epoch information are uploaded to the satellite during ground contact. The VELOX satellite uses a two-step uplink procedure for in-orbit parameter uplink. First, the parameter is uploaded through ground command and stored in ADCS memory. An additional command is required to command the ADCS to permanently store the new parameter set in EEPROM. The two-step uplink command is to ensure that correct orbital parameters are uploaded into the satellite before it is permanently stored in satellite memory. Both real-time position and velocity are updated based on the clock information. The updated real-time position and velocity information are used to determine attitude with a TRIAD algorithm when both sun vector and magnetometer readings are available.

The orbit propagator uses the linearized J2 perturbation Kepler model, which requires six orbital parameters and time epoch information. The six orbital parameters are the mean motion, eccentricity, right ascension of ascending node (RAAN), argument of perigee, inclination and mean anomaly. The orbital parameters are then propagated and converted into position and velocity vectors. The position and velocity vectors are expressed in GCRF (or an Earth-centered inertial (ECI) reference frame).

During both the VELOX-PII and VELOX-I in-orbit ADCS experiments, several issues have been observed. First, if the full time epoch of YDDD.dddddddd is input into the MCU (first element of year, Y is omitted), it will be truncated into YDDD.ddd. The loss of 5 decimals in the time epoch results in a propagation error of 30 to 60 s, which is an error of approximately 200 to 400 km. The propagation error of 200 to 400 km has minimal impact on both VELOX-I and VELOX-PII missions as the VELOX-I mission mainly operates in sun tracking mode and the main objective of VELOX-PII is to establish the flight heritage of several subsystems built in-house. However, a better orbit propagator and more accurate attitude determination are required for future VELOX programs such as VELOX-II, where target tracking is required for the purpose of inter-satellite communication.

While the NORAD TLE is expressed in terms of the TEME reference frame, the sun model is computed in GCRF and the geomagnetic field model is expressed in the ECEF reference frame. Thus, there is a need to compute all of the vectors in the same reference frame. Because the transformation from GCRF to ECEF includes additional nutation and precession transformation, which requires a high computational cost, it is required to study the impact on geomagnetic field modeling accuracy if both nutation and precession are neglected during transformation. On the other hand, the TEME does not require the precession information to transform from TEME to ECEF. Therefore, this motivates the study of satellite attitude determination accuracy perform-
ance for input orbit parameters expressed between TEME and GCRF.

4. Improved Propagation Algorithm using Pre-Process Method

As discussed in Section 3, the finite word length not only affects the accuracy of the time epoch information, but also eccentricity and mean motion. However, it can be shown that the error due to finite word length effect on eccentricity and mean motion is relatively small compared to the error in time epoch. The approximate error in mean motion can be shown as

$$\Delta\mu = \left(\frac{86400}{2\pi}\right)^{1/3} \left(\frac{1}{\sqrt{n} + \Delta n} - \frac{1}{\sqrt{n}}\right)$$

(1)

From Eq. (1), the approximated $\Delta\mu$ is always within meter error range for LEO with the assumption that $\Delta n \ll n$, which is always true. To determine the error in eccentricity, the relationship between the satellite distance and eccentricity is used:

$$r = \frac{a(1 - e^2)}{1 + e \cos \theta}$$

(2)

Using first-order Taylor series approximation, the error can be determined as

$$\Delta r = -\frac{2e a \Delta e}{1 + e \cos \theta} - \frac{a(1 - e^2) \Delta e \cos \theta}{(1 + e \cos \theta)^2}$$

(3)

By assuming $a$ is 7,000 km, the propagation error based on Eq. (3) due to truncation of 7th digital in eccentricity is presented in Fig. 3. The error is presented with respect to a given range of eccentricity and every location of the satellite, when in orbit. Figure 4 shows that the error is less than 5 m, which is insignificant when compared to the error due to time epoch information loss (between 200 km to 400 km). Therefore, the error in eccentricity and mean motion can be neglected.

The flow diagram of the proposed pre-processed orbital parameters algorithm is presented in Fig. 5. First, the updated NORAD TLE obtained from SpaceTrack is either time-forwardly or time-backwardly propagated from $i_0(M)$ at (Y)YDDD.dddddddd to $i_0(M)$ at (Y)YDDD.xxx000000, which can be written as

$$i_0(M) = i_0(N) + \Delta t_0$$

$$= \text{INT}(i_0(M)) + \tau_0(M)$$

(4)

where $\text{INT}(\cdot)$ denotes the integer portion of $i_0(M)$. Due to the floating point to binary conversion, which is based on the IEEE floating point format standard (see Fig. 6), there is an additional propagation error of 5 to 30 s. Therefore, the time fraction, $\tau_0(M)$ (or 0.xxx) of $i_0(M)$ is required to fulfill the following condition to minimize the binary conversion loss:

$$\tau_0(M) = \frac{k_1}{2^1} + \frac{k_2}{2^2} + \frac{k_3}{2^3}$$

(5)

where $k_j$ is an arbitrary positive integer, with the constraint of $0 \leq k_j < 2^{j-1}$ and $\tau_0(M) < 1.0$.

The new modified time epoch, $t_0(M)$ in Eq. (4) allows the last five decimals of the time epoch to be omitted. Furthermore, the constraint of $\tau_0(M)$ in Eq. (5) is able to minimize the error when the time epoch floating point is converted into binary. Then, the following orbit parameters, RAAN, argument of perigee, mean anomaly and inclination in TEME reference frame are propagated using the SGP4 propagator to time, $t_0(M)$:

$$\{\Omega(i_0(M)), \omega(i_0(M)), \theta(i_0(M)), i(i_0(N)), \Delta t_0\}$$

(6)

where $\Omega(i_0(M))$ and $i(i_0(M))$ are $\Omega_0$ and $i_0$, respectively, in Appendix A of Hoots et al.,18 which can be obtained after the short-period periodic update. In addition, $\omega(i_0(M))$ and $\theta(i_0(M))$ can be obtained after the secular update for earth zonal gravity in Appendix A of Hoots et al.18

It is noted that $\Omega(i_0(M))$, $\omega(i_0(M))$, $\theta(i_0(M))$ and $i(i_0(M))$ are computed using SGP4 in Eq. (6) and are expressed in TEME. To obtain the orbit parameters expressed in GCRF, position and velocity vectors at $i_0(M)$ are transformed from TEME to the position and velocity vectors in GCRF (details of position and velocity vectors computation in TEME is shown Hoots et al.18):

$$r_{GCRF} = [P][N]R_z(E_{\text{Equinox}}) r_{\text{T}E}$$

where $[P]$ is the rotation matrix due to the precession angle, $[N]$ is the rotation matrix due to the nutation angle, and
\( E_{\text{Equinox}} \) is the equation of equinox parameters (see Vallado\(^{14}\)) with \( R_0(E_{\text{Equinox}}) \) denoting the rotation matrix of equinox parameters about the \( z \)-axis.

The four orbital parameters in GCRF can then be computed using the \( \theta_{\text{GCRF}} \) and \( \theta_{\text{GCRF}} \) from Eq. (7). The details of position/velocity to orbital elements transformation, and coordinate transformation between TEME to GCRF are detailed in Bate et al.\(^{19}\) and Vallado,\(^{20}\) respectively. It is noted that the conversion of orbital parameters from TEME to GCRF in Eq. (7) is optional. A detailed comparison on orbit propagator and attitude determination accuracy based on the input of orbital parameters in TEME and GCRF will be studied in Sections 5 and 6.

As eccentricity and mean motion of the satellite are oscillating over time, we consider that eccentricity and mean motion remain constant during the short time period of propagation. The assumption is acceptable because both semi-major axis and eccentricity computed by NORAD are the average over a given time period. The assumption is acceptable because both semi-major axis and eccentricity of the satellite moving over time, we consider that eccentricity and mean motion remain constant during the short time period of propagation. The assumption is acceptable because both semi-major axis and eccentricity computed by NORAD are the average over a given time period. Thus, we have

\[
\begin{align*}
&n(t_0^{(M)}) = n(t_0^{(N)}) \\
&e(t_0^{(M)}) = e(t_0^{(N)})
\end{align*}
\]  

(8)

The newly computed time epoch in Eq. (4) and six orbital parameters computed using Eqs. (6) to (8) are uploaded to the satellite through the procedure shown in Fig. 5. Finally, the orbit propagator computes the position and velocity vector in TEME/GCRF based on the input parameter set. Both position and velocity vectors can then be used for attitude determination, nadir and target pointing purposes.

### 5. Experimental Results

Experiments have been conducted to compare the propagation accuracy of the following algorithms:

(a) Pre-processed algorithm in GCRF (GCRF + J2),
(b) Pre-processed algorithm in TEME (TEME + J2),
(c) Only mean anomaly alternation (TEME-MA),
(d) SGP4-based propagation, and

(e) Direct NORAD TLE input.

All GCRF+J2, TEME + J2, TEME-MA and direct NORAD TLE input methods are conducted using VELOX ADCS hardware. The orbital parameters for both cases (a) and (b) utilize the pre-process method in Section 4, with the orbital parameters in case (b) expressed in the TEME reference frame. For the TEME-MA method, the mean anomaly of TLE is propagated based on the time epoch given by the pre-process method in Section 4. However, RAAN, argument of perigee and inclination remain the same. On the other hand, the SGP4-based propagation is performed using Matlab software to serve as the performance benchmark.

The experimental configuration and TLE selected are shown in Table 1 to 3. All of the algorithms are compared with respect to the satellite position obtained through the GPS data in the ECEF reference frame.

Figure 7 shows the root mean square error (RMSE) of each propagation method with respect to the positioning data obtained from VELOX-I GPS. The SGP4 propagator has the lowest error as it considers the satellite’s drag effect. But the result shows that the position RMSE is at least 100 km in the positioning data.

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**Table 1. VELOX-I: TLE data and experiment setup.**

| Satellite No. | Start time               | Sampling rate (s) | Orbit parameter epoch | Mean motion (rev/day) | Eccentricity |
|--------------|-------------------------|-------------------|-----------------------|----------------------|--------------|
| 40057        | 11 Jul 2014/02:48:14 UTC | 10                | 14191.81334084        | 14.73956287          | 0.0010367   |
| 29051        | 23 Apr 2014/00:00:44 UTC | 60                | 14112.81429524        | 14.37965553          | 0.0056991   |
| 29052        | 01 Sep 2014/02:33:44 UTC | 60                | 14243.82154369        | 14.281717185         | 0.0039314   |
| 29053        | 02 Sep 2014/02:33:44 UTC | 60                | 14243.82154369        | 14.281717185         | 0.0039314   |
| 29054        | 03 Sep 2014/02:33:44 UTC | 60                | 14243.82154369        | 14.281717185         | 0.0039314   |
| 29055        | 04 Sep 2014/02:33:44 UTC | 60                | 14243.82154369        | 14.281717185         | 0.0039314   |

**Table 2. Formosat 3E (Cosmic-004): TLE data and experiment setup.**

| Satellite No. | Start time               | Sampling rate (s) | Orbit parameter epoch | Mean motion (rev/day) | Eccentricity |
|--------------|-------------------------|-------------------|-----------------------|----------------------|--------------|
| 29057        | 23 Apr 2014/00:00:44 UTC | 60                | 14112.81429524        | 14.37965553          | 0.0056991   |
| 29058        | 01 Sep 2014/02:33:44 UTC | 60                | 14243.82154369        | 14.281717185         | 0.0039314   |
| 29059        | 02 Sep 2014/02:33:44 UTC | 60                | 14243.82154369        | 14.281717185         | 0.0039314   |
| 29060        | 03 Sep 2014/02:33:44 UTC | 60                | 14243.82154369        | 14.281717185         | 0.0039314   |
| 29061        | 04 Sep 2014/02:33:44 UTC | 60                | 14243.82154369        | 14.281717185         | 0.0039314   |

**Table 3. Formosat 3F (Cosmic-002): TLE data and experiment setup.**

| Satellite No. | Start time               | Sampling rate (s) | Orbit parameter epoch | Mean motion (rev/day) | Eccentricity |
|--------------|-------------------------|-------------------|-----------------------|----------------------|--------------|
| 29057        | 23 Apr 2014/00:00:44 UTC | 60                | 14112.81429524        | 14.37965553          | 0.0056991   |
| 29058        | 01 Sep 2014/02:33:44 UTC | 60                | 14243.82154369        | 14.281717185         | 0.0039314   |
| 29059        | 02 Sep 2014/02:33:44 UTC | 60                | 14243.82154369        | 14.281717185         | 0.0039314   |
| 29060        | 03 Sep 2014/02:33:44 UTC | 60                | 14243.82154369        | 14.281717185         | 0.0039314   |
| 29061        | 04 Sep 2014/02:33:44 UTC | 60                | 14243.82154369        | 14.281717185         | 0.0039314   |
magnitude if no modification is done on TLE before it is uploaded into VELOX ADCS. Using the proposed pre-processing method, it is noted that the accuracy is improved significantly. From Fig. 7, it shows that there is a slight accuracy performance difference when the orbital parameters are input in terms of either TEME or GCRF. The result shows that the input orbital parameters in GCRF have a lower RMSE than the orbital parameters in TEME. This could be attributed to the fact that the J2 linearized Kepler model is optimized for orbital parameters in GCRF. On the hand, it is also possible to lower the propagator error by one order of magnitude through only propagating the mean anomaly based on the time epoch obtained in Section 4 without modifying other orbital parameters. But overall, pre-process methods TEME + J2 and GCRF+J2 have a lower propagator error than TEME-MA.

Figure 8 compares the accuracy between the GCRF + J2 and TEME + J2 pre-processed methods, direct NORAD TLE input and SGP4-based propagation using the FORMOSAT-3F/Cosmic-004 GPS data. The result shows that at least 100 km propagation error is observed if the NORAD TLE is directly uploaded to the satellite without any modification. The propagation error of either GCRF + J2 or TEME + J2 is significantly reduced to within 2 to 20 km. Although there is no time offset issue in the pre-processed method, the J2 model only includes a second-order zonal harmonic effect, and no perturbation effect such as atmospheric drag. Therefore, additional propagation error may be observed because COSMIC satellites have a larger surface area and higher eccentric orbit. Similarly, TEME-MA has a lower propagator error than the direct NORAD TLE input method. But it has a higher propagator error than either GCRF + J2 or TEME + J2.

Figure 9 compares the propagation error with respect to FORMOSAT-3F/Cosmic-002 GPS data. The VELOX ADCS propagation error could be as high as 200 km if direct NORAD TLE is used. However, the propagation error of the method proposed is only within 5 to 20 km error, without taking the satellite’s drag into consideration. In addition, the result confirms that both GCRF + J2 and TEME + J2 have a lower propagation error when compared to direct NORAD TLE input. The result also shows that TEME-MA has a similar accuracy performance with a slightly higher maximum error when compared to GCRF + J2 and TEME + J2.

The average RMSE with standard deviation of propagator error of different input methods is compared in Table 4. The result shows that the direct NORAD input method has the highest average error. Result also shows that the TEME-MA has a higher average RMSE than TEME + J2 and GCRF+J2. This is due to the reason that the input orbital parameters of TEME-MA does not consider either the zonal nor secular effects on RAAN, argument of perigee and inclination over short period of time (within 3 hr or 0.125 days). But TEME + J2 and GCRF+J2 have taken both the zonal and secular effects on the modified TLE, thus achieving better propagation accuracy.

6. Impact on Attitude Determination Error

Next, the impact of various TLE input methods on attitude determination and target pointing accuracy is studied using the VELOX-II (satellite ID 15077F) available on 2 February 2016. First, the impact on geomagnetic field modeling by direct NORAD input, the proposed TEME + J2 and alternative TEME-MA methods are compared. The geomagnetic
field is computed based on 8th degree-6th order international geomagnetic field model. The computational cost of the model selected has been proven in VELOX-I flight data and VELOX-II flatsat, where one ADCS process cycle includes attitude determination and control requires less than 200 ms. In addition, the expected error simulated of the selected degree and order is 0.4 μT when benchmarked with a 13th degree and 13th order geomagnetic field model.

The comparison of sun modeling error is omitted as the difference between methods is less than 1 mdeg. This is due to the reason that the distance between the sun and satellite is much higher than the error distance by the orbit propagator (1.5 × 10⁸ km vs. 200 km).

Figure 10 compares the geomagnetic field magnitude between direct TLE, the proposed TEME + J2 and alternative TEME-MA methods. The result shows that TEME + J2 has the lowest overall magnitude difference, with an average of 0.16 μT (or approximately 0.1 μT error for each axis), followed by the TEME-MA method, with an average of 0.32 μT (or approximately 0.18 μT error for each axis). The TEME + J2 geomagnetic field average modeling error proposed is lower than the magnetometer reading error used in the VELOX nanosatellite (approximately 0.125 μT error in each axis), but the TEME-MA method has a higher geomagnetic field average modeling error than the magnetometer used in flight hardware. On the other hand, the geomagnetic field modeling error using the direct input method is at least four to five times higher than TEME + J2 and TEME-MA methods.

Figure 11 compares the attitude determination accuracy between the direct TLE input method, the proposed TEME + J2 method and the TEME-MA method. Here, it is assumed that sun sensor and magnetometer readings are available, and the quaternion estimation algorithm (or QUEST10) is used to determine the orientation of the satellite. The sun sensor ranges between 0.1 to 1.0 deg and the magnetometer errors ranges between 0.1 to 1.0 μT. In addition, 50 Monte Carlo simulations are considered, where sensor errors and satellite orientation are randomly generated during each Monte Carlo simulation.

The results in Fig. 11 show that the TEME + J2 method proposed has the lowest attitude determination error, followed by the TEME-MA method. Both Fig. 11(b) and Fig. 11(c) show that the errors are approximately 1 deg lower than Fig. 11 (a), the direct input method, at all times. This is due to the high geomagnetic field modeling error of direct TLE input, as shown in Fig. 10. Thus, the orbital parameters and time epoch alternation, such as that introduced in either the TEME + J2 method proposed or the TEME-MA method not only improves orbit propagator accuracy, but also improves attitude determination accuracy.

Although the TEME + J2 and TEME-MA methods show similar performance in attitude determination accuracy performance, with the TEME + J2 method proposed having slightly better accuracy performance, but a significant difference in performance could be observed when either target tracking or nadir pointing reference vector was taken into consideration. Table 5 compares the reference line of sight vector error for different tracking scenarios. We consider Iriddium 8 as the LEO target satellite and OPTUS D1 as the

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Table 4. Average RMSE (with standard deviation) with different input methods.

| Method       | VELOX-I | Cosmic-004 | Cosmic-002 |
|--------------|---------|------------|------------|
| Direct       | 98.80 ± 0.25 km | 86.45 ± 10.30 km | 153.70 ± 9.85 km |
| TEME+J2      | 4.51 ± 2.94 km  | 11.02 ± 2.75 km  | 11.11 ± 2.72 km  |
| GCRF+J2      | 2.00 ± 0.29 km  | 8.39 ± 4.69 km   | 11.45 ± 5.00 km  |
| TEME-MA      | 11.64 ± 0.06 km | 27.44 ± 8.58 km  | 19.43 ± 7.00 km  |

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Fig. 10. Geomagnetic field modeling error comparison based on VELOX-II (SAT ID 15077F) orbit propagation data.

Fig. 11. Attitude determination accuracy comparison with respect to sun sensor and magnetometer error: (a) Direct TLE input, (b) proposed method in TEME + J2 and (c) TEME-MA. Based on VELOX-II (SAT ID 15077F) orbit propagation data.

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GEO target satellite. By considering the attitude determination accuracy based on the result in Fig. 11, with a fine sun sensor error of 0.8 deg and magnetometer error of 0.125 µT; Table 5 shows that the TEME + J2 method proposed is able to meet all of the pointing performance requirements under one sigma error condition (or worst case). However, the TEME-MA method fails to meet the pointing accuracy requirement for the LEO satellite tracking scenario. The TEME + J2 method proposed has a maximum pointing error of approximately 2.3 deg, while both TEME-MA and the direct input method have maximum pointing errors of approximately 5.1 deg and 27 deg, respectively, for target tracking of the LEO satellite. It is noted that the direct input method is able to achieve a low pointing error for GEO satellite tracking compared to either LEO satellite tracking or nadir pointing due to the reason that the distance between the GEO satellite and LEO satellite is extremely high (approximately 30,000 km), which results in the error due to the orbit propagator being negligible.

Table 5. Target and nadir pointing reference vector error comparison for an intersatellite communication mission.

|                | TEME + J2 | TEME-MA | Direct |
|----------------|-----------|---------|--------|
| LEO pointing error (deg) | 0.37 ± 0.69 | 1.81 ± 1.91 | 9.94 ± 15.09 |
| GEO pointing error (deg)  | 0.24 ± 0.03 | 0.30 ± 0.07 | 0.39 ± 0.21 |
| Nadir pointing error (deg)| 0.17 ± 0.01 | 0.69 ± 0.02 | 5.23 ± 0.05 |

7. Conclusion

The finite word length effect of a MCU has not been well addressed in the literature for cubasat design. This paper proposes a pre-processed NORAD TLE method to cope with the finite word length limitation. The NORAD orbital elements are first propagated before the four orbit parameters, namely RAAN, argument of perigee, mean anomaly and inclination, are computed and uploaded into the satellite. Experiments have been conducted to compare the propagation accuracy between the pre-processed method in TEME (or TEME + J2) and GCRF (or GCRF + J2), only mean anomaly alternation (TEME-MA) method SGP4 and direct NORAD TLE input. The SGP4 propagator using Matlab has the lowest error as it includes the satellite’s drag effect. The experimental results show that the pre-processed method improves propagation accuracy at least one to two magnitude as compared to the direct input NORAD TLE into the satellite.

In addition, the attitude determination accuracy performance based on the TEME + J2 method proposed has been studied and compared with the direct TLE input method and TEME-MA method based on VELOX-II orbital parameters. The result indicates that the method proposed has the lowest magnetic field modeling and overall attitude determination errors compared to the other two methods. In addition, the study shows that only the TEME + J2 method proposed is able to meet the mission requirements for all of the target and nadir pointing scenarios. Although the TEME-MA method has similar attitude determination accuracy as the TEME + J2 method proposed, it fails to meet the LEO satellite target tracking accuracy performance requirement.

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