Enhancement of Collision Probability Accuracy Using Improved Orbit Prediction Method

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Korea Aerospace Research Institute (KARI) is currently monitoring objects that come close to the KOMPSAT-2 (KOrea Multi-Purpose SATellite-2) using collision analysis software called Automated Conjunction Analysis System (ACAS) which is developed by KARI. For initial screening, Two-Line Element (TLE) provided by US Strategic Command (USSTRATCOM) is used as the orbit information of space objects. In addition, we have been receiving information of secondary object from Conjunction Summary Message (CSM) distributed by JSpOC. For TLE consistency analysis of the objects, we analyzed TLE statistics with TLEs generated for 15 days before Time of Closest Approach. The TLE accuracy is important to analyze a proximity risk of secondary object which approaches to satellite. The TLE uncertainties is originated from semi-analytic SGP4 orbit model for TLE orbit propagation, radar tracking and processing and would be dependent on solar flux, object size and orbit shape. To mitigate these uncertainties, we improved the accuracy of predicted KOMPSAT-2 position using TLE orbit fitting. Also, in order to enhance collision probability analysis, covariance matrices are generated from the radial, in-track, and cross-track components between TLE propagation and orbit propagated by improved orbit prediction method. The performance of the ACAS which applies to more accurate covariance of each object was improved than with the constant covariance applied to the previous ACAS. In this paper, we found more reliable covariance matrix by comparing covariance matrix determined from TLE propagation with the result that applies to the improved orbit prediction method.

Nomenclature

KARI	=	Korea Aerospace Research Institute
ACAS	=	Automated Conjunction Analysis System
TLE	=	Two-Line Elements
JSpOC	=	The United States Joint Space Operations Center
USSTRATCOM	=	US Strategic Command

I. Introduction

In KARI the KOMPSAT-2, the multi-purpose low earth orbit (LEO) satellite, currently is being operated on the mission orbit located at 685 km. Due to increase of space objects, the risk of collisions between the operational satellites and space objects on space grows and the operators continuously are aware of a space situation for satellites and space objects. In order to analyze the risk of collisions between two objects, the probability of collision must be determined after accurately performing orbit prediction of two objects. We have operated the orbit of satellites through our own production that use the GPS navigation solution from an on-board GPS receiver and the other space objects are dependent to TLE Catalog which tracks, gathers and distributes orbit information in

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Joint Space Operations Center under the U.S. Strategic Command. For a preliminary conjunction analysis, the orbit prediction of KOMPSAT-2 and space objects is carried out by using the TLEs. After screening a result of orbit prediction performed through Automated Conjunction Analysis System (ACAS) and the conjunction information received from CSM, KARI establishes the avoidance maneuver as a collision action plan of KOMPSAT-2. However, there is no information about the orbit accuracy in TLE and there are uncertainties in SGP4 orbit model using propagating a TLE orbit as well as TLE orbit determination and prediction accuracy. Therefore, to make a high accuracy analysis continually for potential collision, it can be resolved by using the data reduced uncertainties and finally it can be used to prevent the unnecessary avoidance maneuvers.

In this paper, the TLE uncertainty of KOMPSAT-2 was analyzed using the TLE pair-wise differencing method in Ref. 1 and the accuracy of the predicted position the object was enhanced from the improved orbit prediction method in Ref. 3 that used the TLE orbit fitting to reduce the uncertainty. In addition, the covariance matrix is computed through orbit propagation of the TLE and the predicted orbit from high precision orbit propagator and is computed from the improved orbit. The results were compared with the covariance matrices computed from the truth orbit and presented as table.

II. TLE Accuracy Analysis with truth orbit

To analyze the TLE accuracy, the TLE pair-wise differencing method in Ref. 1 was applied to KOMPSAT-2. Every TLE of KOMPSAT-2 distributed for 15 days is propagated to epoch of the reference TLE as the latest TLE using the SGP4 propagator. After propagating, the reference TLE is transferred to previous epoch and the propagation process to the reference TLE epoch is repeated and performed until 2nd TLE becomes the reference TLE. The state vector of the reference TLE and propagated state vector at equal epoch are determined and to compute the relative state vector as the residuals, differences of the state vectors (position and velocity), is determined through RIC (Radial, In-track, and Cross-track) as space object based coordinate system. Because the TLE data is provided at unequal time internals, the bin for the classification of delta epoch is selected properly at constant intervals and classified residuals for the statistics then mean, variance, and standard deviation of residuals belong to each bin is computed. As a result, the variations of the RIC component can be analyzed.

For validation of the results by TLE data, the independent orbit data with higher accuracy was used. If there is a mutual consistency when compared position variation which determined using only the TLE data with position variation determined using the higher accurate truth data, the reliability of the TLE data can be ensured. These characters of variation will make it easy to find covariance properties that represent the uncertainty.

In this paper, orbit propagation was performed using SGP4 orbit propagator of STK with respect to all TLEs distributed from 12 September 2011 to 26 September 2011 and the state vectors of RIC direction in each reference epoch was determined. Figure 1 shows residuals variations of the state vector of KOMPSAT-2 in RIC direction using only TLE and with truth orbit. The mean and standard deviation of each bin as time intervals of one day reasonable for initial study was computed then centered on the mean of the bin and extended in both directions by the standard deviation. A dotted line fitted to a second-order polynomial by the least squares method was added. For the validation, the variation between the true state vector of the satellite and the state vector determined from TLE data was investigated. Because the residuals is computed in each epoch for analysis of residuals variation of the state vectors, the epoch of the true state vector of the satellite should be matched to each epoch of the TLE data (Ref. 2). The true position was synchronized by propagating from the closest state vector of the previous TLE to the TLE epoch using HPOP (High Precision Orbit Propagator). The specification of the force models are as follows:

- 70 x 70 JGM-3 gravity field
- Lunar-solar gravity field
- Solar radiation pressure
- Jacchia71 atmospheric drag model

The residuals between the state vector determined by TLE data and the truth orbit are computed and the mean, variation, and standard deviation of each bin are computed. As the results of two state vectors, the residual of the state vectors is distributed by curve fit that presented characteristics of used the SGP4 dynamic model. The error in the in-track direction is larger than the others and the maximum error is about -17.38 km (only TLE) and -16.63 km (with truth orbit), relatively. KOMPSAT-2 belonging to low earth orbit seems to be dominant the perturbation by atmosphere in the in-track components. In addition, in case of TLE propagation, the deviation of each bin in RIC components with respect to truth orbit is small. Because TLE data is mean orbit, the variation can be guess to be less than it of truth orbit which the osculating orbit.

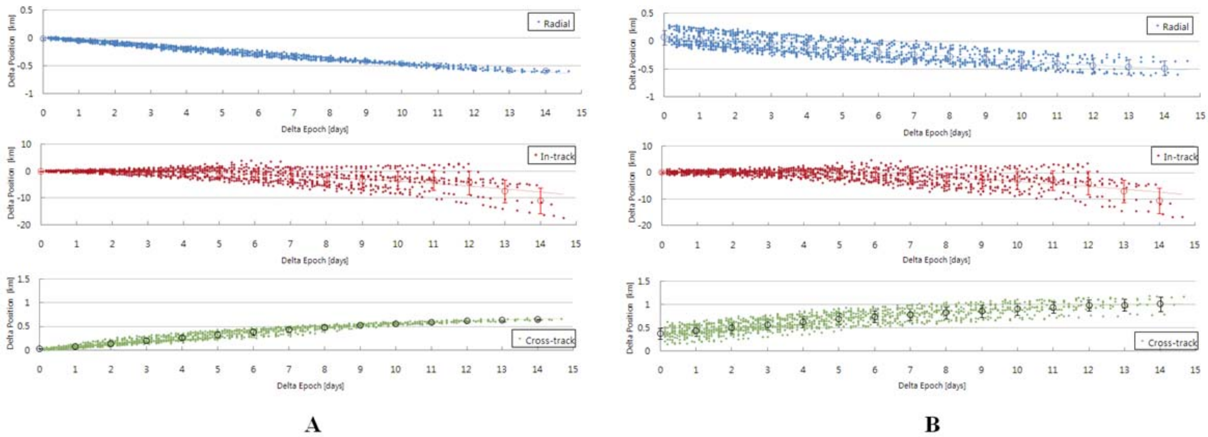

Figure 1 Residual Variation of KOMPSAT-2 (A: only TLEs B: with truth orbit)

III. Analysis of Orbit Information

A. TLE orbit fitting and propagation

The improved orbit prediction method that contributes to the orbit prediction accuracy for objects in the publicly available TLE catalog in Ref. 3 was applied to KOMPSAT-2. The TLE data was used as pseudo-measurements and the state vectors generated from a successive pseudo-measurement were fitted on the orbit using high precision special perturbation propagator and Bayesian least square estimation technique. The fitted orbit is propagated forward using the same high precision propagator. The high precision propagator incorporates the force model mentioned in section II.

The accuracy of prediction is evaluated by comparing the truth orbit data. All TLEs distributed within 10 days (from 16 September 2011 to 26 September 2011) as fit period were interpolated using the SGP4 propagator until the epoch of the subsequent TLE. A series of order 100 state vectors generated equally spaced in time within the fitting period was extracted as the pseudo-measurement and the orbit was fitted by applying Bayesian least square method. The fitted orbit was propagated for 30 days forward using high precision orbit propagator.

The position error compared with the truth orbit of KOMPSAT-2 is shown in Figure 2. The result of applying to the improved orbit prediction method (red) and the result propagated from epoch of the latest TLE (black) using the SGP4 propagation model are compared with truth orbit. The interpolated result with respect to each TLE (blue; the updated data) is illustrated and this result can be made a reference in case of the space debris which the truth orbit does not exist (Ref. 3).

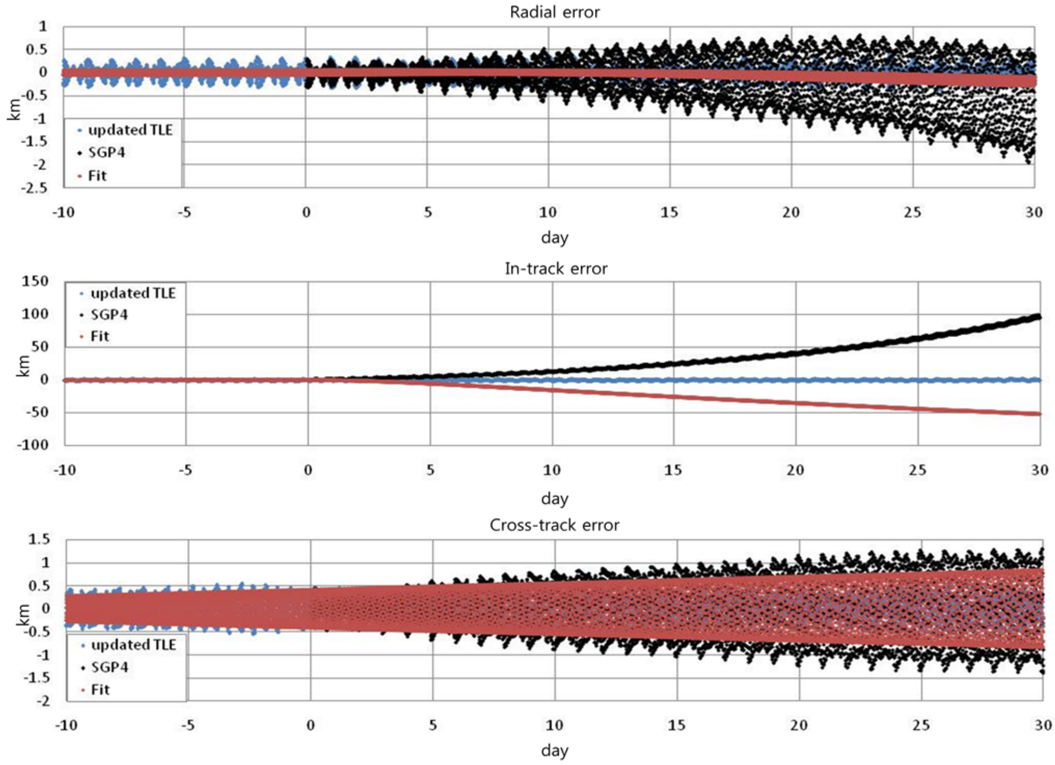

Figure 2 Position errors with respect to truth orbit using improved orbit prediction method and using SGP4

B. Covariance Matrix

The covariance matrix indicating the uncertainty of the position and velocity is estimated. In detailed analysis of the collision, the covariance matrices were computed through the propagated result for 2 days, 1 day and 8 hours respectively corresponds to 2 days before a collision expectation as the avoidance maneuvers planning phase and 1 day before the avoidance maneuver phase. The RMS errors in Table 1 show the prediction errors of each prediction period with respect to truth orbits. Because there is no existing accurate future data, the latest TLE is propagated using SGP4 orbit model. The covariance matrix is computed by the RIC residual errors determined at equal epoch with the result propagated from epoch of each state vector to epoch of best state vector through the method in section II. In this paper, the covariance matrices with respect to KOMPSAT-2 were computed by the TLE orbit propagation using SGP4 orbit model and improved orbit prediction, respectively. As for the covariance matrix computations, the result of the TLE propagation is not good data as a reference. Therefore, the covariance matrix is computed by the RIC error at equal epoch with the GPS navigation solution of KOMPSAT-2 using the high precision orbit propagator. The parameters are generated for 60-second intervals in two propagators and this process was repeated with the truth orbit.

Table 1 RMS of prediction with respect to truth orbits

	Improved orbit prediction			SGP4 propatation			Updated TLEs		
	R [m]	I [m]	C [m]	R [m]	I [m]	C [m]	R [m]	I [m]	C [m]
48-hour prediction	41.39	455.67	308.69	128.92	1096.66	227.05	133.31	658.78	216.27
24-hour prediction	41.59	589.32	304.00	130.20	760.78	226.19	130.90	678.33	223.33
8-hour prediction	41.35	643.87	300.52	128.61	556.98	223.78	128.94	546.29	224.44

IV. Improved Covariance matrix

Table 2 Comparison of Covariance matrices

		TLE w.r.t. numerically prop orbit			Improved orbit w.r.t. numerically prop orbit			TLE w.r.t. truth orbits			Impoved orbit w.r.t. truth orbits		
		R [m]	I [m]	C [m]	R [m]	I [m]	C [m]	R [m]	I [m]	C [m]	R [m]	I [m]	C [m]
48-hour prediction	R [m]	16.81	-1.16	1.94	1.61	0.01	14.75	16.46	2.35	-0.92	1.70	-0.39	11.43
	I [m]	-1.16	1089.40	28.54	0.01	20.65	-16.11	2.35	613.31	20.11	-0.39	133.99	-8.96
	C [m]	1.94	28.54	95.04	14.75	-16.11	173.48	-0.92	20.11	51.57	11.43	-8.96	95.32
24-hour prediction	R [m]	16.98	2.10	4.21	1.69	0.12	14.98	16.80	2.85	-0.88	1.72	-0.20	11.55
	I [m]	2.10	445.95	35.99	0.12	21.85	-16.56	2.85	449.45	23.19	-0.20	17.62	-11.00
	C [m]	4.21	35.99	90.16	14.98	-16.56	169.47	-0.88	23.19	51.19	11.55	-11.00	92.46
8-hour prediction	R [m]	27.49	1.41	23.70	1.76	0.15	14.86	27.24	1.73	8.35	1.69	0.08	11.22
	I [m]	1.41	354.42	93.11	0.15	11.99	-16.63	1.73	385.61	76.18	0.08	8.13	-9.55
	C [m]	23.70	93.11	113.48	14.86	-16.63	162.34	8.35	76.18	70.40	11.22	-9.55	87.97

To estimate the covariance matrix in preliminary analysis of the collision in the future, the covariance matrix is computed by the result propagated the latest TLE to forward epoch using the SGP4 propagator and the result propagated by orbit determination obtained from the GPS navigation solution using the high precision orbit propagator. In addition, the covariance matrix is computed by the result determined from an improved orbit prediction mentioned in Section III and the propagated result using the high precision orbit propagator and is compared with the previously computed covariance matrix.

As shown in Table 2, as a reference the orbit obtained by high precision orbit propagation, the covariance matrix computed by improved orbit predictions has more accurate results than the covariance matrix computed by the result of the TLE propagation. In the case of in-track direction that has a large deviation of the error in low earth orbit, due to the fact that the propagation time becomes shorter, the error becomes smaller; the variables of the covariance are reduced. However, the variance, the in-track component of the orbit propagated for 48 hours is less than the variance of the orbit propagated for 24 hours. Because the result of the improved orbit prediction within 1 day is worse than the prediction result of 1 day or more (Table 1). In comparison with the truth orbit, the covariance matrix obtained by the improved orbit prediction through fitting of the least square method has a more accurate value than the covariance matrix obtained through the result of the TLE propagation. This is because the propagation error is decreased by improved orbit prediction. Thus, in case of space debris that obtained the orbit information through only TLE, the more accurate covariance matrix can be obtained by the improved orbit prediction through fitting of the least squares method.

V. Conclusions

For the preliminary conjunction analysis, the residual variation of the TLE state vector is performed through only TLEs and verified the reliability with truth orbit. The TLE uncertainty was examined by the character of residuals of the state vector and in the case of LEO satellites affected by atmospheric in the TLE uncertainty analysis, the error in the in-track direction is larger than the others.

The error of propagation was reduced by improved orbit prediction method. According to the prediction period of the analysis, the covariance matrices were computed through the TLE propagation and with improved orbit. In addition, when compared with the covariance matrix computed using the truth orbit, the accuracy of the covariance matrix computed by improved orbit is higher than the covariance matrix computed by the TLE orbit propagation. In this study, the case of space debris that does not have the truth data, the covariance matrix can be computed by improved orbit prediction and more accurate initial analysis can be performed.

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