ORIGINAL PAPER

ORBITAL DECAY PREDICTION AND SPACE DEBRIS IMPACT ON NANO-SATELLITES

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Abstract. *The equation of two body motion was derived with the assumption of mass with spherically symmetric distribution. This equation includes perturbions which represent the deviation from the ideal Kepler's orbit as a result of external forces effect. These forces can be classified into two types: gravitational forces and non-gravitational forces. In this work gravitational forces includes Earth's oblateness effect and non-gravitational include Atmospheric drag and Solar radiation pressure.*

The prediction of orbital lifetime of KufaSat using orbital parameters and engineering specifications as inputs to the Debris Assessment Software (DAS) was done, it has been verified that the orbital lifetime will not be longer than 25 years after completion of mission which is compatible with recommendation of Inter-Agency Space Debris Coordination Committee (IADC).The probability of KufaSat collisions with orbital debris and satellites which operating in the same orbit during orbital lifetime was determined. Apogee/Perigee Altitude History was used to graph apogee and perigee altitudes over KufaSat lifetime .The change in velocity required for maneuvers needed to achieve atmospheric reentry within 25 years was calculated.

Keywords: *Orbital Decay, Orbital Lifetime, DAS, KufaSat, Space Debris*

1. INTRODUCTION

The risk of collisions a spacecraft with orbital debris and satellites which operating in the same orbit has become extremely important. This risk is increased in case of low earth orbit. To avoid or minimize these risks the Inter-Agency Space Debris Coordination Committee (IADC) recommended that the appropriate lifetime limit is 25-year. The Scientific and Technical Subcommittee (STSC) of the United Nations Committee on the Peaceful Uses of Outer Space states that space mission planners, designers, manufacturers and operators should "Limit the long-term presence of spacecraft and launch vehicle orbital stages in the low-Earth orbit (LEO) region after the end of their mission [1] .Prediction the orbit lifetime of a satellite is an important component and one of the most important aspects of a satellite mission. The prediction of orbit lifetime depends on the initial orbital parameters, satellite characteristics, the atmospheric density, the drag force, geomagnetic activity, and the solar radiation pressure. Orbital lifetime analysis of a satellite is important to demonstrate compatibility with standards, support spacecraft mission design, predict a future orbit demise, and post-decay debris mitigation. KufaSat is the first satellite of the University of Kufa, Iraq. It is developed within the framework of program called KufaSat Project, whose goal is to provide hands-on experience to aerospace students in cooperation with another three Iraqi universities. The calculated Keplerian elements for kufaSat are listed in Table 1 [2].

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Table 1. KufaSat orbital elements

Field	Value	Unit
Semi-Major Axis (SMA)	6978	km
Eccentricity (ECC)	0	----
Inclination (INC)	97	deg
Argument of Perigee (AOP)	150	deg
Right Ascension of Ascending Node (RAAN)	0	deg
True Anomaly (TA)	10	deg

2. BACKGROUND

Using Newton's law of gravitation and Kepler's laws of orbit motion, the equation of two body motion can be derived with the assumption of mass with spherically symmetrical distribution. This equation can be given in the relative form as:

$$\frac{d^2\vec{r}}{dt^2} = -\frac{\mu}{r^3}\vec{r} \tag{1}$$

where r is the position vector of the satellite measured from the center of the primary body, and μ is the gravitational constant .

Because of the presence of gravitational and non-gravitational perturbations, equation (1) will be representing approximation of the actual motion. The general form for the relative motion of two bodies with perturbations can be expressed as:

$$\frac{d^2\vec{r}}{dt^2} = -\frac{\mu}{r^3}\vec{r} + a_p \tag{2}$$

where a_p is the sum of all the perturbing accelerations. [3]

3. PERTURBING ACCELERATIONS

A perturbation is a deviation from the ideal Kepler's orbit as a result of external forces effect which can be classified into two types: gravitational forces such gravitational force of the moon and sun, the non-spherical shape and non-uniform mass distribution of the earth and non-gravitational forces such as solar radiation pressure, atmospheric drag, and geomagnetic field.

3.1. NON-GRAVITATIONAL PERTURBATIONS

a. Atmospheric drag

The primary factor that affects the 'orbital lifetime' of a satellite is atmospheric drag, which itself depends on atmospheric density and the form factor of the object flying into that atmosphere. Drag forces effect on a satellite's motion and change the orbit shape as a result of the presence of neutral gases molecules in the Earth's upper atmosphere.

Atmospheric drag acts in opposite direction of the velocity and reduces the energy from the orbit which causes the orbit to decay until the satellite reenters the atmosphere. The equation for the acceleration due to drag is [3]:

$$a_d = -\frac{1}{2}\rho\left(\frac{C_d A}{m}\right)v_r^2 \tag{3}$$

where ρ is the atmospheric density at that altitude, C_d is the drag coefficient ≈ 2.2, A is the effective cross-section area of the satellite normal to its direction of travel (m^2), m is the total mass of spacecraft (kg), and v_r is the satellite's velocity relative to the atmosphere . $\left(\frac{C_d A}{m}\right)$ represent ballistic coefficient.

Atmospheric density model

The atmospheric density can be specified by a simple exponential law but due to spatial and temporal variations of space environment, a precise specification cannot be achieved. The atmospheric model is used to describe atmospheric density variations in time, season, altitude, latitude, solar activity, and geomagnetic [4].

The Jacchia series and the Mass Spectrometer and Incoherent Scatter (MSIS) series are the most commonly used of atmospheric density models. The Jacchia model is a dynamic model assumes a temperature profile. It used for orbit decay predictions with improved accuracy. The MSIS model has been derived based on data from instruments and contains a density profile extending all the way to the Earth's surface. It is considered the most accurate density mode [5].

b. Solar radiation pressure

The vertical profile of atmospheric density depends on the distributions of pressure and temperature, and in particular on the heating and cooling rates at the altitude of interest. One important factor that would strongly affect the density profile is solar activity, as an enhanced influx of radiation and particles would provide some additional heating to the atmosphere and therefore change its properties .The Sun radiation, radiation reflected from the illuminated Earth hemisphere, IR radiation re-emitted from Earth, and the IR radiation emitted from the spacecraft are the major radiation sources. Solar radiation pressure produces acceleration in a radial direction away from the sun.

The equation for solar radiation pressure can be written as follows:

$$a_{rsp} = \Gamma\left(\frac{T}{c}\right)\left(\frac{A_s}{m}\right) \tag{4}$$

where A_s is the satellite's average area projected normal to the direction of the sun in m^2, m is the satellite's mass in kg, T is the solar flux near the Earth, c is the speed of light, and Γ is the satellite's reflection coefficient [6].

3.2. GRAVITATIONAL PERTURBATIONS

The gravitational potential of the Earth is broadly symmetrical at large scale, but quite heterogeneous at smaller scales. This is because of the non-spherical shape of the planet, important variations in mass distribution (mountain peaks versus ocean dips) and density (rocks, water and air), as well as the displacement of masses within the system (in the Earth's core, in geophysical processes responsible for plate tectonics, or in atmospheric and oceanic currents, including tides, for instance). For a circular orbit, the gravitational force is essentially perpendicular to the velocity vector, while the force that brings the satellite back down is a drag that acts in the opposite direction of the velocity vector.

Earth's oblateness

Earth is not a symmetrical body and seems to be flat at the poles with bulge at the equator when compared to perfect sphere. The difference in force due to the earth's oblateness is referred to it as the J2 perturbation .This perturbation is the main gravitational force that acts on a satellite in LEO. The acceleration due oblatness of earth can be expressed in spherical coordinate system as [7] :

$$\overrightarrow{a_{J_2}} = -\frac{3J_2\mu r_e^4}{r^4}\left[\vec{e}_r(0.5 - 1.5\sin^2 i \sin^2 u) + \vec{e}_i \sin^2 i \sin u \cos u + \vec{e}_z \sin i \cos i \sin u\right] \quad (5)$$

when J_2 is geopotential coefficient representing Earths oblatness, μ is a gravitational earth constant, r_e is radius of earth , u is argument of latitude , $\vec{e}_r$ is a unit vector along the satellite orbit radius vector direction, $\vec{e}_i$ is a unit vector along the local horizontal direction, $\vec{e}_z$ is a unit vector along the orbit normal direction.

The other types of gravitational perturbations have a secondary effect in LEO compared to the other processes, and will perturb the orbit, but the impact on the lifetime of the satellite will be minimal [8]. The sum of all effective perturbing acceleration in low earth orbit can be expressed as:

$$a_p = a_d + a_{rsp} + a_{J_2} \quad (6)$$

By substitute equations (3), (4), and (5) in equation (6)

$$a_p = -\frac{1}{2}\rho\left(\frac{C_d A}{m}\right)v_r^2 + \Gamma\left(\frac{T}{c}\right)\left(\frac{A_s}{m}\right) - \frac{3J_2\mu r_e^4}{r^4}\left[\vec{e}_r(0.5 - 1.5\sin^2 i \sin^2 u) + \vec{e}_i \sin^2 i \sin u \cos u + ez \sin i \cos i \sin u\right] \quad (7)$$

By substitute equation (7) in equation (2), the general form for the relative motion equation of two bodies under the influence of atmospheric drag and solar radiation pressure and earth's oblateness becomes:

$$\frac{d^2\vec{r}}{dt^2} = -\frac{\mu}{r^3}\vec{r} - \frac{1}{2}\rho\left(\frac{C_d A}{m}\right)v_r^2 + \Gamma\left(\frac{T}{c}\right)\left(\frac{A_s}{m}\right) - \frac{3J_2\mu r_e^4}{r^4}\left[\vec{e}_r(0.5 - 1.5\sin^2 i \sin^2 u) + \vec{e}_i \sin^2 i \sin u \cos u + \vec{e}_z \sin i \cos i \sin u\right] \quad (8)$$

4. PERTURBATIONS TECHNIQUES

Special perturbation, general perturbation, and semi-analytic are the three main methods to solve equation of motion with perturbations. Special perturbation method uses numerical integration of equation of motion which including all important perturbations accelerations. Theoretically it provides the most accurate solution. Cowell's method and Encke's method are examples of these approaches. General perturbation method uses a low order analytical solution of perturbed equation of motion instead of the original equation of motion and permits analytical integration over some limited interval .The SGP4 propagator used in Two Line Elements processing is an application of general perturbation. Semi-analytic method uses a combination of the two techniques, numerical (special perturbation) and analytic (general perturbation) [9].

5. LIFETIME PREDICTION TOOLS

The NASA Debris Assessment Software (DAS), AGI's STK software, and 1 Earth's QProp lifetime estimation tools are some of orbit propagation tools available for lifetime prediction of the CubeSat. [10]

The Debris Assessment Software (DAS) version 2.0 which is designed to assist NASA programs in performing orbital debris assessments (ODA), as described in NASA Technical Standard 8719.14, (Process for Limiting Orbital Debris) is used in this work.

DAS 2.0 uses the NASA propagator "PROP3D" which is designed to maintain integration accuracy over long propagation periods with reasonable computation speed. Atmospheric drag, Solar and Lunar gravity, Solar radiation pressure, and Earth's gravity field with Zonal harmonics (J2, J3, and J4) are the perturbations included in the DAS orbit propagator. The atmospheric model used in DAS was the Jacchia 1976 Standard Model. The coefficient of drag is assumed to be 2.2 and the coefficient of reflectivity (for the solar radiation pressure perturbation) is assumed to be 1.25. The major functions of DAS are divided into three sections: mission editor, requirement assessments, and associated science and engineering utilities.The mission information must be entered into the mission editor . Requirement assessments section of DAS includes routines to assess the mission's compliance with each NASA debris-limiting requirement.Science and engineering utilities provide a number of functions useful for mission planning and allow analyzing some aspects of orbit/mission design [11].

6. SIMULATION AND RESULTS

Science and engineering utilities are divided into six categories: on-orbit collision, analysis of postmission disposal maneuver, orbit evolution analysis, delta-V postmission maneuver analysis, orbit to orbit transfer, and other utilities [11].The first four categories are used in this work.

On-Orbit Collisions

These utilities assist in the assessment of compliance with Requirement, to (limit the probability of operating space systems becoming a source of debris through collisions with orbital debris or meteoroids). Probability of debris impacts versus a verity of factors can be graphed to provide a visual aid. Figures (1, 2, and 3) are the results of the three contour plotting tools which are available within this category. These figures explain the relations between Debris Impacts vs. Orbit Altitude, Debris Impacts vs. Debris Diameter, and Debris Impacts vs. Date respectively.

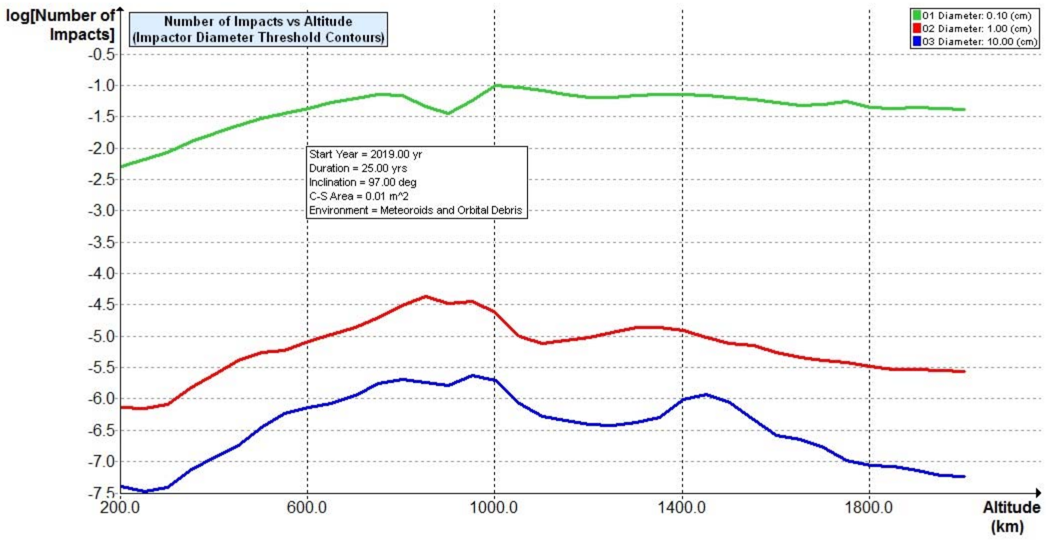

Figure 1. Debris Impacts vs. Orbit Altitude.

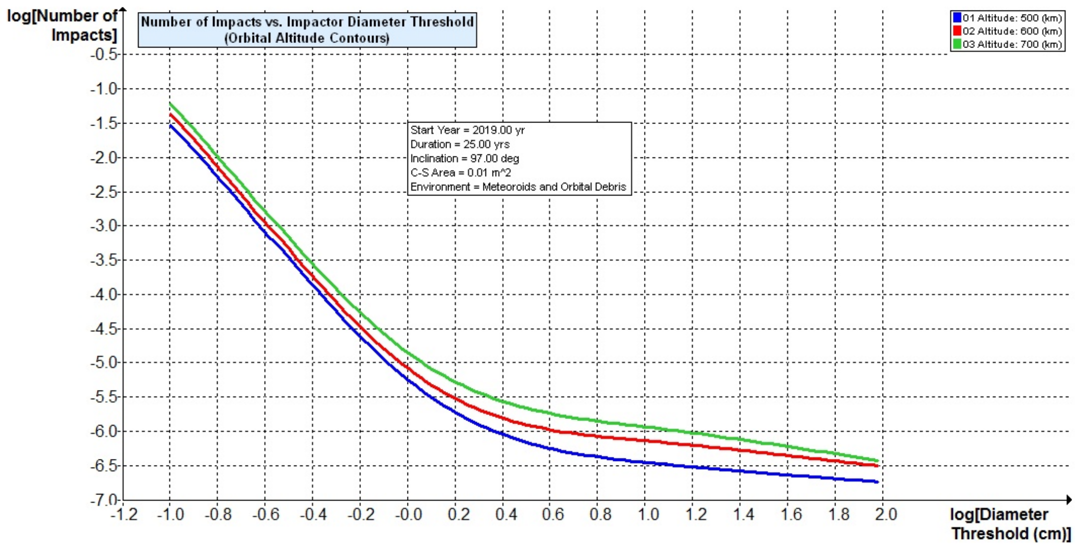

Figure 2. Debris Impacts vs. Debris Diameter.

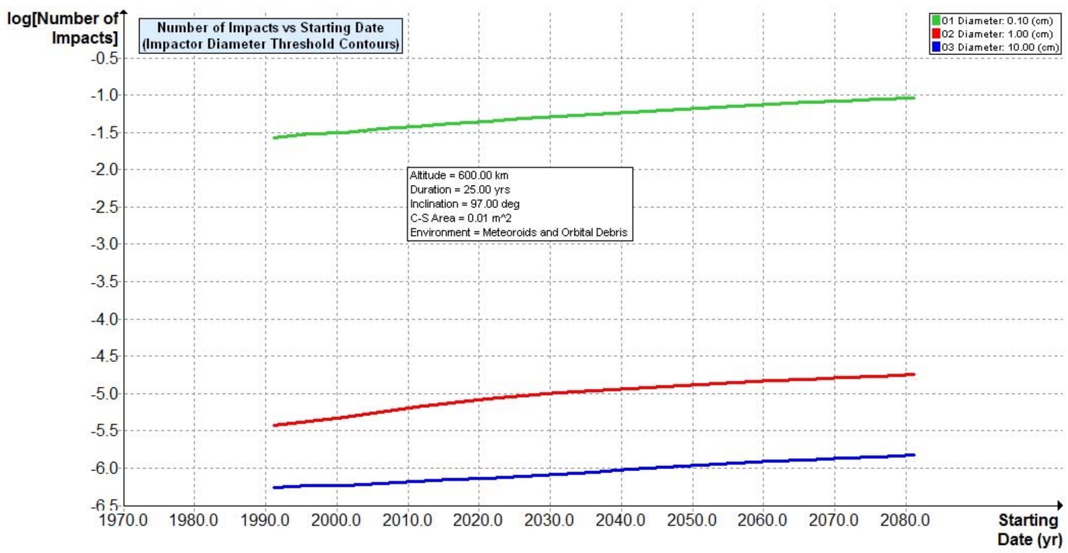

Figure 3. Debris Impacts vs. Date.

a) Analysis of Postmission Disposal Maneuvers

These utilities assist in the assessment of compliance with Requirement the (postmission disposal of space structures). Disposal by Atmospheric Reentry utility plots contours of Delta-V corresponding to the Delta-V required moving from LEO to a decay orbit with specified lifetime. This may aid in determining the cost of deorbit maneuvers. Figure (4) explain disposal by atmospheric reentry.

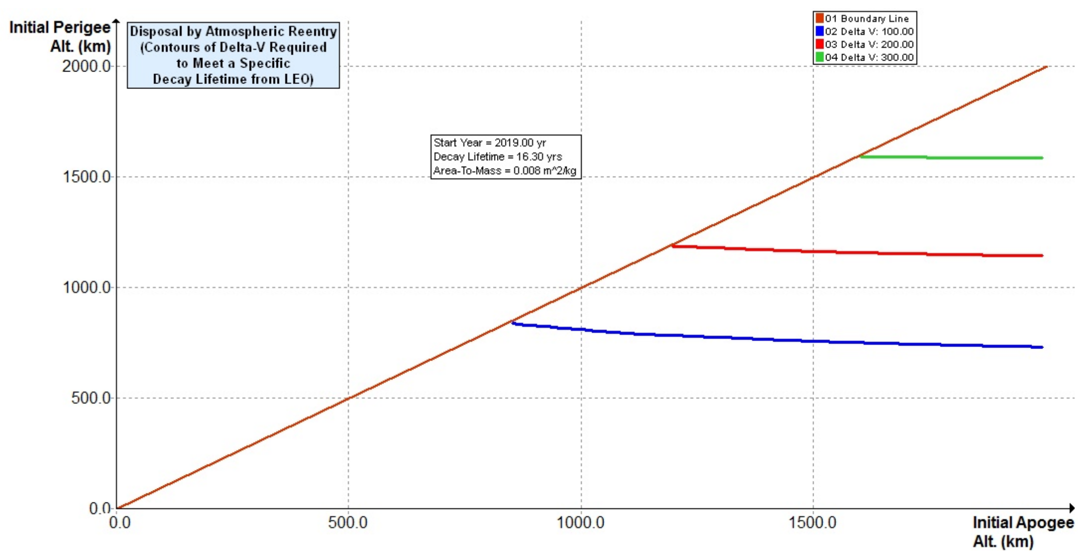

Figure 4. Disposal by Atmospheric Reentry

b) Orbit Evolution Analysis

These utilities assist in the assessment of compliance with Requirement, the (postmission disposal of space structures). Two utilities are available within this group. Apogee/Perigee Altitude History which is used to graph apogee and perigee altitudes over object lifetime as shown in Figure (5) and Orbit Lifetime/Dwell Time which calculates the Orbital Lifetime and LEO Dwell Time of an object in a specified orbit. For KufaSat it is found that the calculated Orbital Lifetime from year 2019 as a start year equal to 16.356 years. The year that the object either reentered or exceeded the propagation time limit is 2035.

Table .2 shows the effect of varying the area to mass ratio on orbital lifetime of KufaSat with assuming 2019 is the start year.

Table.2 Area to mass ratio vs. Orbital lifetime

Area to mass ratio	Orbital lifetime	Last year of propagation
0.005	26.683	2045
0.006	24.241	2043
0.0076923	**16.350**	**2035**
0.009	15.086	2034
0.010	14.286	2033
0.015	6.779	2025
0.020	5.235	2024
0.030	4.233	2023
0.100	2.689	2021
0.500	1.884	2020

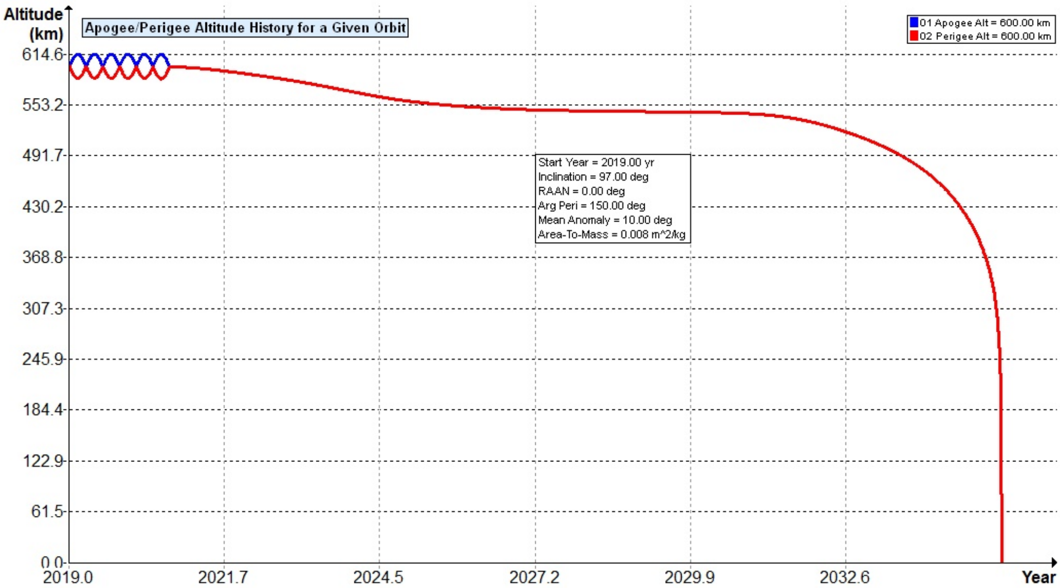

Figure 5. Apogee/Perigee Altitude History for a Given Orbit

c) Delta-V for Postmission Maneuver

These utilities calculate the change in velocity required for maneuvers needed to achieve atmospheric reentry within 25 years. Two utilities are available within this group.

1. Delta-V for Decay Orbit Given Orbital Lifetime: plots area-to-mass ratio contour points corresponding to the Delta-V required moving an object, with a specified orbital lifetime, from an initial circular LEO orbit to a decay orbit. The plot allows exploring the cost of deorbiting a vehicle. Figure (6) represent the output of this utility for KufaSat.
2. Delta-V for Decay Orbit Given Area-To-Mass: plots lifetime contour points corresponding to the Delta-V required moving an object, with a specified area-to-mass ratio, from an initial circular LEO orbit to a decay orbit. The plot allows exploring the cost of deorbiting a vehicle over a range of decay lifetimes. Figure (7) represent the output of this utility for KufaSat.

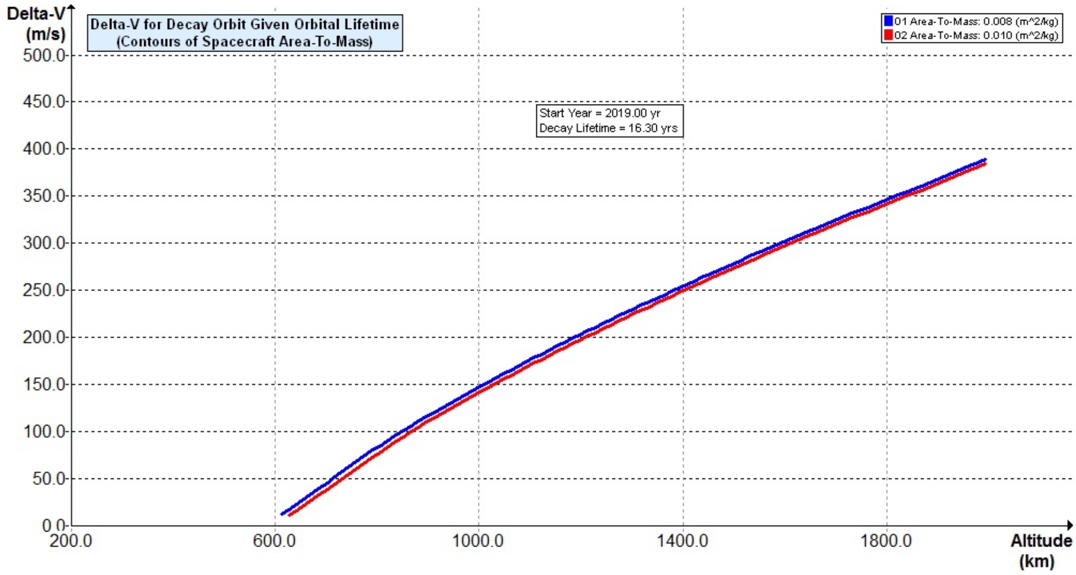

Figure 6. Delta-V for Decay Orbit Given Orbital Lifetime

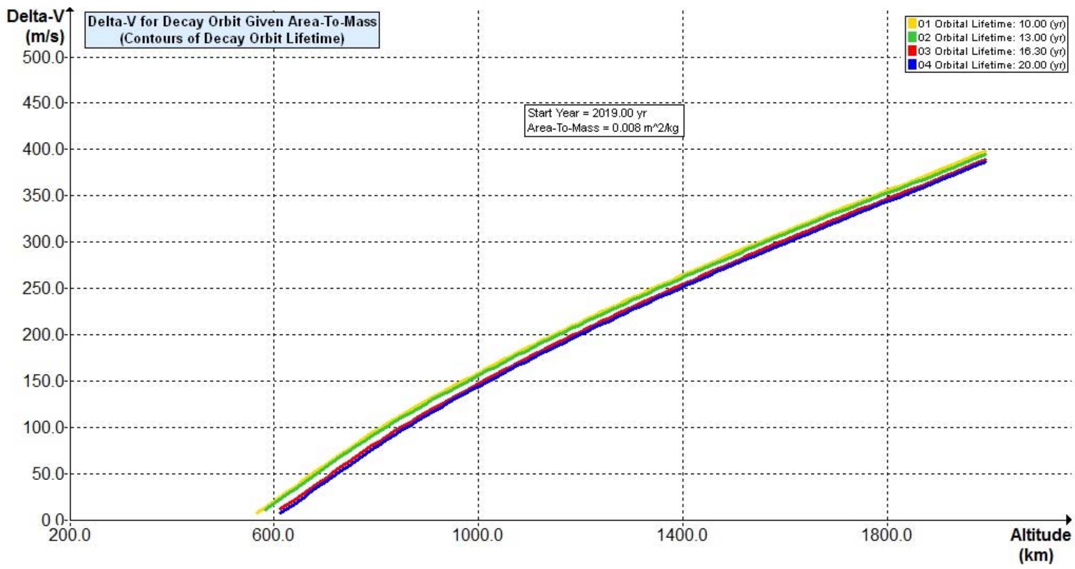

Figure 7. Delta-V for Decay Orbit Given Area-To-Mass

CONCLUSIONS

Space debris mitigation is required for each space mission to ensure of compliance with requirements and standards in addition to ensure that the spacecraft can resist the space debris environment during the mission lifetime. One of important aspect is estimation and limitation the probability of collision with known objects during orbital lifetime which cause loss of control to prevent post-mission disposal. KufaSat mission analysis was conducted using the Debris Assessment Software (DAS). By this software the probability of collisions with known objects during orbital lifetime of KufaSat was determined.

The prediction of orbital lifetime is not accurate, because the parameters which have an effect on the prediction can be varied so there is a range of expected life time values. The results indicate that the orbital lifetime of KufaSat can be reduced by increasing its area to mass ratio. By using orbital parameters and engineering specifications of KufaSat as inputs to DAS it has been verified that the orbital lifetime will not be longer than 25 years after completion of mission.

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