LIFETIME ESTIMATION OF UPPER STAGES RE-ENTERING FROM GTO WITH DIFFERENT INCLINATIONS

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ABSTRACT

A study on the orbital lifetime estimation of upper stages with different inclinations is carried out through response surface technique. The combined influence of the luni-solar perturbations and drag on lifetime variations plays an important role. In order to extract information on the lifetime from orbital data, the estimation of area to mass ratio, initial perigee altitude and ballistic coefficient is performed. In this context, it is noticed that when an object undergoes orbital resonance, bifurcation between the observed and predicted trajectories takes place. However, in some scenarios, simulation of apogee and perigee altitude profiles can still be performed to estimate relevant parameters. To capture only secular changes in apogee and perigee altitudes, osculating elements can be converted to their respective mean values by utilizing a suitable theory. Furthermore, a study on the re-entry of the object is examined closely by estimating different ballistic coefficients to match the position at different epochs. The non-uniform change in the ballistic coefficient provides an indication of possible chaotic motion.

1. INTRODUCTION

One of the proposed criteria to ensure a stable space debris environment is to place the objects in orbits with limited lifetime of up to 25 years. It is well known that the time of launch during a day can have substantial effect in determining the orbital life of an object placed in a highly elliptic orbit like GTO (Geo-stationary Transfer Orbit) [13]. A spent stage in GTO is unique in the sense that its collision threat potential spreads across the gamut of operational orbit regimes- from LEO (Low Earth Orbit) to GEO (Geo-stationary Earth Orbit). So minimization of spent stage lifetime is one of the important objectives for space debris mitigation efforts. Unfortunately, some of the measures like de-boosting following the completion of mission, which have been contemplated for LEO lifetime reduction, may not be very suitable for GTO. Also, any measure for de-boosting compromises the payload capacity of the launch vehicle. Fortunately, interplay of natural forces like drag and lunar-solar perturbations can give rise to favorable conditions when the lifetime in GTO may be reduced to a level much lower than the one, which can be attained under the sole influence of the atmospheric drag [1], [2], [3], [4], [5], [6], [7]. In this perspective, the monitoring of the accelerated decay from GTO under the above favorable circumstance is an important activity to gather valuable knowledge so that for future missions, it can be incorporated into the planning of spent stage lifetime reduction without jeopardizing operational requirements.

In order to set up any orbital propagation, initial states are to be known at a chosen epoch. For any meaningful prediction of GTO lifetime through simulation, it is indeed essential that correct initial states are available, but at the same time, still higher order accuracy is warranted for initial perigee altitude.

Accordingly, in order to extract information on the lifetime of a GTO object from its orbital data, the estimation of initial perigee altitude as well as area to mass ratio is performed. Since, the effect of drag, acting on the spent stage in and around perigee, is reflected in the gradual but continual contraction of apogee, estimation is carried out by minimizing average apogee and perigee dispersions (between observation and prediction), through identification of initial perigee altitude and area to mass ratio. Numerically, it is more advantageous that the estimation of area to mass ratio and initial perigee altitude be performed with the coupling of an optimizer with *easy–to–calculate* approximations of the cost function. The optimizer evaluates the minimum value of this approximate cost function and then an accurate orbital propagator, with estimated parameters, simulates evolution till reentry.

In this context, one of the approximation approaches that are suitable for the estimation of area to mass ratio and initial perigee altitude is response surface technique. This technique replaces the cost function with simple functions, which are fitted predictions made with numerical orbital propagator at the set of observation epochs. The apogee and perigee values predicted by numerical orbital propagator with different values of initial perigee altitude and area to mass ratio at a set of observation epochs are fitted to constitute the response surfaces, which in this case literally, are apogee and perigee surfaces in initial perigee altitude and area to mass ratio space. It may be noted that the temporal

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variation of these surfaces reflects the dynamics of orbital motion. This methodology is successfully applied to predict orbital life of GSLV (Geo-synchronous Satellite Launch Vehicle, ISRO/India) spent upper stages [8], [9], [10]. The orbital lifetime GSLV-D1 spent stage was 640 days whereas for GSLV-D2 it was 658 days. However, the lifetime of GSLV-F01 spent stage was 1160 days. All these objects were around 19 degrees inclination. Against this backdrop it may be noted that orbital lifetime of GTO objects are very much dependent on initial state. The dependence of orbital lifetime in GTO on initial state is described in the next section.

2. RESONANCE IN GTO

The GTO is a highly eccentric orbit with the perigee normally at low altitude (180 km- 650 km) and the apogee near geo-stationary altitudes (36000 km). The combined influence of the luni-solar perturbations and drag can result in lifetime variations from a few months to several decades. The desired effect from the space debris point of view is a short lifetime. The long-term evolution of objects in GTO is either decay predominantly by luni-solar gravity effect or decay by combination of atmospheric drag and luni-solar gravity perturbations. While, the atmospheric drag reduces apogee altitudes with marginal changes in perigee altitudes, the cumulative effects of the Sun and the Moon on the satellite can result in either increase or decrease in the perigee altitude. The decay, which takes place predominantly by luni-solar gravity effect, drives the perigee below the reentry altitude while the orbit still remains highly elliptical (e.g. orbits with critical inclination, 63.5°). In contrast, the decay by combination of drag and luni-solar gravity is accompanied by phases with complex interplay of these perturbations. This happens in GTO.

Furthermore, it is also known that orbital evolutions in GTO are often very sensitive to initial conditions. Under certain conditions, GTO exhibits orbital resonance, which manifests in large secular variation in eccentricity [11]. Although orbital resonance can result in very long lifetime, it can also lead to a very short lifetime, which is indeed a desirable outcome for space debris mitigation measures. Against this backdrop, one of the interesting examples of resonance can be gleaned from the orbital evolution of an Ariane-V rocket body. This rocket body was launched on 20th December 2000 and it finally reentered on 31st May 2002. The orbital lifetime of this body is evaluated for different values of area to mass ratio. An ensemble of area to mass ratio between 0.004 m^2/kg and 0.006 m^2/kg is generated through uniform distribution.

The initial state for lifetime evaluation is the following:

Epoch	: 2000.973058 year
Apogee Altitude	: 36054.7402 km
Perigee Altitude	: 197.7343 km
Inclination	: 2.00293°
Argument of Perigee	: 179.02384°
Right Ascension of Ascending Node	: 252.22676°

Debris Assessment Software (DAS) from NASA is utilized to evaluate the orbital lifetime for a sample size of 100. The distribution of orbital lifetime is presented in Figure- 1.

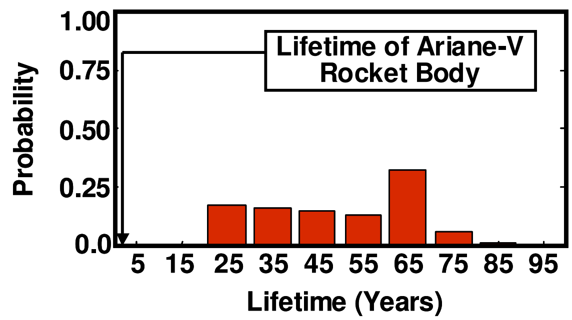

Figure-1: Orbital lifetime distribution for Ariane-V rocket body

It is evident that in all likelihood, the expected value for the orbital lifetime of Ariane-V rocket body is 25 years or more. The orbital evolution envelops of Ariane-V rocket body is also simulated with different values of initial perigee altitude and area to mass ratio. The dispersion bands on initial perigee altitude and area to mass ratio are (200 ± 10) km and (0.0055 ± 0.0007) m^2/kg respectively (Figure- 2). The bifurcation between the observed and predicted orbits takes place almost right after insertion.

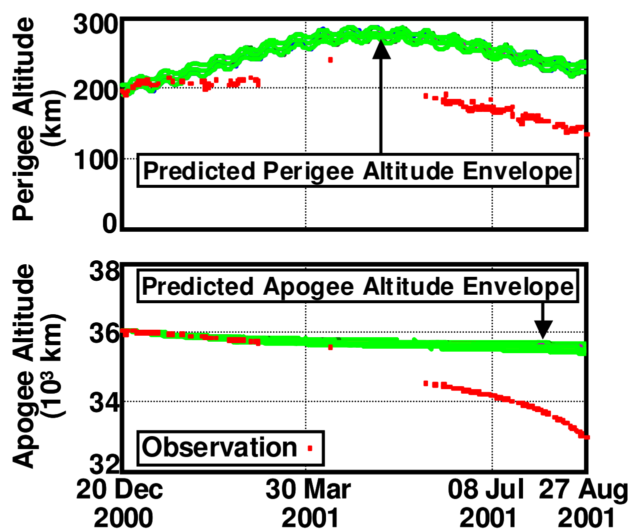

Figure-2: Orbital evolution envelopes for Ariane-V rocket body

On 20th Dec 2000, the Right Ascension of Sun (RAS) is about 270°. Since, the Ariane-V orbit is a very low inclination orbit and the sum of its Argument of Perigee (AOP) and Right Ascension of Ascending Node (RAAN) is around 90° (Figure- 3), during the first 2 months after insertion, the apsidal line remains almost collinear with the Sun-Earth line. This scenario gives rise to orbital resonance, which sets off initial bifurcation and subsequent reduction in orbital life.

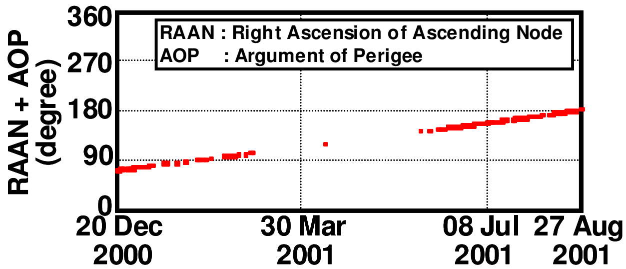

Figure-3: Movement of apsidal line for Ariane-V rocket body during the first 250 days after orbit insertion

3. RESPONSE SURFACE APPROXIMATION

The response surface approximation of a highly nonlinear dynamical system involves exploration of the relationships between several system parameters and one or more system states at a particular epoch. The response surface approximation to dynamical equations for motion is presented as an algorithm below:

(1) Prediction of several apogee and perigee profiles $h_a\left(t,S,h_{pi}\right)$ and $h_p\left(t,S,h_{pi}\right)$ through numerical orbital propagator where, S, called as reference area, is the ratio of effective area to mass, t_i's are time instants for observations and h_{pi} is the initial perigee altitude.

(2) A set of apogee and perigee surfaces $h_{ak}\left(S,h_{pi}\right)$ and $h_{pk}\left(S,h_{pi}\right)$ are constructed where, i = 1,2,…N and N is the number of observations.

Interestingly, while the temporal evolutions of apogee and perigee in GTO are governed by highly nonlinear physics, variations in orbital parameters, at any point of time, within a substantial portion of orbital lifetime, is, by and large, linear. Through ingenuous application of response surface approximation to orbital dynamics, it is possible to isolate the linearity in incremental departures from the non-linearity in global characteristics.

In this perspective, it is important to note that mean orbital elements without short periodic variations are very amenable to response surface approximation. Hence, appropriate mean elements should be adopted.

4. MEAN AND OSCULATING GTO ELEMENTS

The evolution of typical GTO orbit is simulated (Figure-4) with luni-solar gravity and the following initial state:

Epoch	: 2001.422044 year
Apogee Altitude	: 34541.939331 km
Perigee Altitude	: 190.571534 km
Inclination	: 1.76655°
Argument of Perigee	: 332.62787°
Right Ascension of Ascending Node	: 168.38215°
True Anomaly	: 27.27785°

It is evident from Figure-4 that appreciable changes in osculating apogee and perigee altitudes occur around perigee. Due to the dynamical effect of Earth's oblateness (represented by J_2) there is a decrease in osculating perigee altitude and simultaneous increase in osculating apogee altitude near perigee. On the other hand, mean apogee and perigee altitudes do not exhibit such kind of sharp changes. Hence to ensure gradual changes in apogee and perigee altitudes, osculating elements can be converted to their respective mean values by utilizing a theory, which considers only J_2 effect [12]. Smoothness in mean values is an attractive proposition for maintaining consistency in prediction.

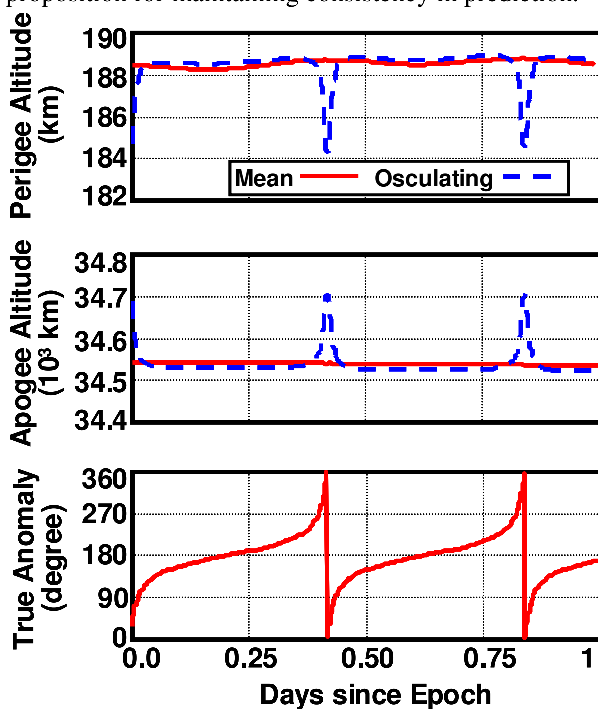

Figure-4: Variations in apogee and perigee altitudes (mean and osculating) of a typical GTO

Mean elements are not unique and depend on a particular formulation. It is essential that observed and predicted osculating elements be converted to their corresponding mean values by utilizing the identical scheme.

5. RESPONSE SURFACE GENERATION

For successful implementation of response surface approximation, it is necessary that observed values are within predicted apogee and perigee envelopes. In order to simulate, apogee and perigee profiles one needs to have some idea about the physical characteristics of the space object of interest so that feasible values of area to mass ratio can be arrived at. In this context, it is stressed that while gravitational effects (of Earth) and luni-solar perturbations are relatively well characterized space object drag is very difficult to predict. This is because,

(a) Reference area, S (for drag force computation) is a function of geometric profile and body orientation (time varying) with respect to instantaneous velocity vector.
(b) Dispersion bounds on solar flux and geomagnetic index results in uncertainty in atmospheric density.

Hence, it is more appropriate to estimate an equivalent value of area to mass ratio (from orbital data), which incorporates the effects of uncertainties in geometric profile, body orientation and atmospheric property.

Ariane-V rocket body is idealized as a cylinder with length, l = 3.36 m and diameter, d = 3.96 m. The mass of rocket body is taken as 2700 kg. For a tumbling cylinder, there can be two extreme orientations which are [13],

(a) Rotation axis is perpendicular to velocity vector
(b) Rotation axis is parallel to velocity vector

So depending on body orientation with respect to instantaneous velocity vector, the area to mass ratio can vary between 0.0048 and 0.0062 (Figure- 5).

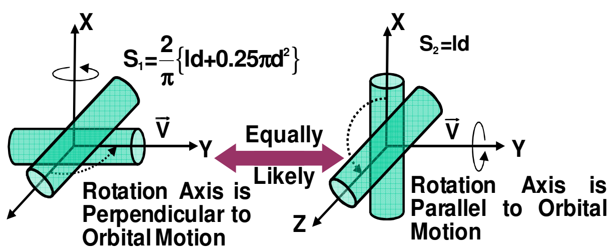

Figure-5: Extreme orientations of a tumbling cylinder

Furthermore, a constant average solar flux of $F_{10.7}$ = 160 W/m^2/Hz is taken for simulation (Figure- 6).

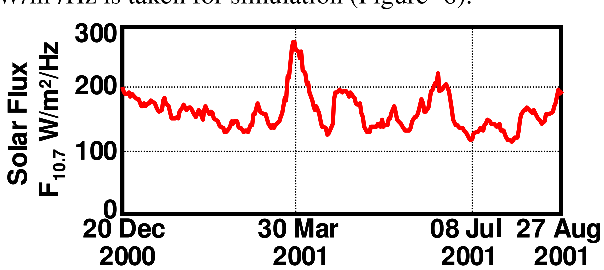

Figure-6: solar flux between 20 Dec '00 and 27 Aug '01

The post resonance envelopes of Ariane-V apogee and perigee altitudes are generated for 90 days from 03rd June 2001 by varying its area to mass ratio between 0.0048 and 0.0062 and its initial perigee altitude between 185 km and 195 km (Figure- 7). It is noticed that observed apogee and perigee altitude values are contained within the predicted envelopes.

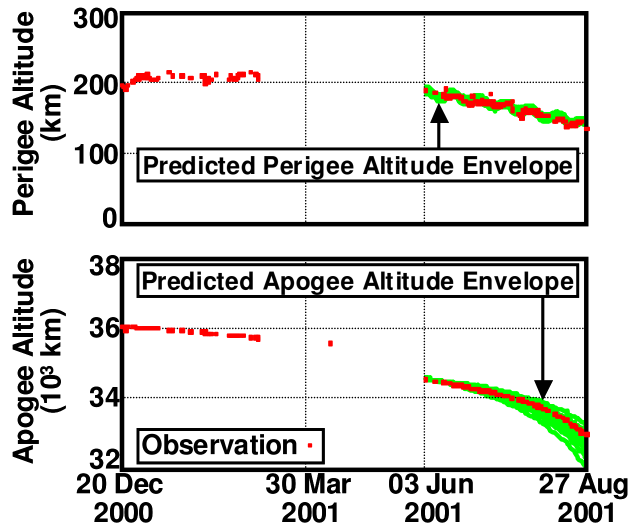

Figure-7: Orbital evolution envelopes for Ariane-V rocket body during 90 days from 03rd June '01.

6. COST FUNCTION CHARACTERISTICS

Average dispersions in apogee, F_a, is

$$F_a=\sqrt{\frac{\sum_{k=1}^{N}\left(h_{aobs}\left(t_i\right)-\left\{h_{ak}\left(S,h_{pi}\right)\right\}\right)^2}{N}} \quad (1)$$

Similarly, average dispersion in perigee, F_P, is defined as

$$F_p=\sqrt{\frac{\sum_{k=1}^{N}\left(h_{pobs}\left(t_i\right)-\left\{h_{pk}\left(S,h_{pi}\right)\right\}\right)^2}{N}} \quad (2)$$

Here, h_{aobs} and h_{pobs} are observed apogee and perigee altitudes. In order to investigate the characteristics of the cost function, average dispersions in are plotted as functions of initial perigee altitude and area to mass ratio (Figure- 8). It is noticed that there is a valley in average perigee dispersion surface. Moreover, this valley is almost flat at its base and runs nearly parallel to area to mass ratio axis. From this peculiarity in the surface topology, one can deduce that a suitable initial perigee altitude, which remains virtually independent of area to mass ratio, can minimize average perigee dispersion. The initial perigee altitude, which corresponds to the minimum perigee altitude dispersion, is identified as 189.16 km. This happens because evolution of perigee altitude during the 90 days from 03rd June 2001 is driven primarily by luni-solar gravity perturbations. At the same time, average apogee dispersion surface is also marked by another flat valley, which is at an angle with area to mass ratio axis. This indicates that the minimum apogee dispersion can be achieved through a particular linear combination of initial perigee altitude and area to mass ratio. But once, the initial perigee altitude corresponding to the minimum value of average perigee dispersion is identified; the area to mass ratio, which minimizes the average apogee dispersion, is uniquely estimated. If drag coefficient, C_D = 2.2, the area to mass ratio, which corresponds to the minimum apogee altitude dispersion, is 0.005 m^2/kg.

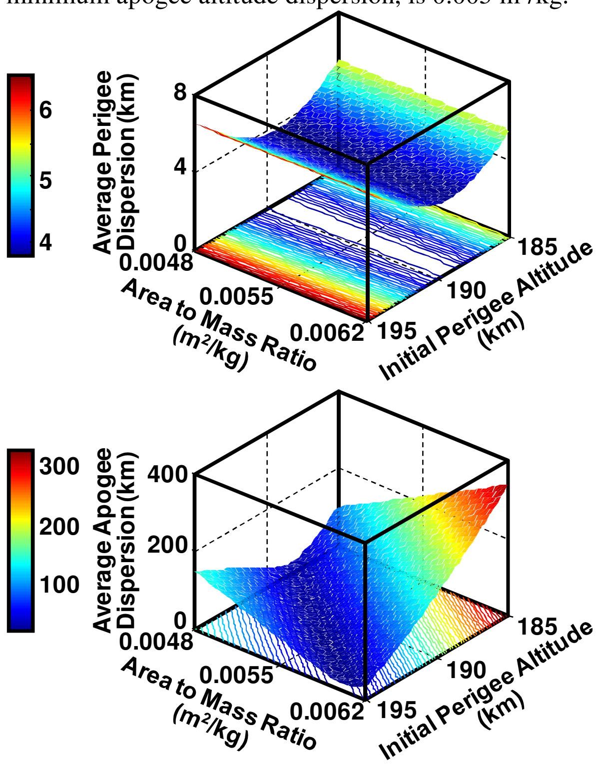

Figure- 8: Average perigee and apogee dispersion surface as a function of initial perigee altitude and area to mass ratio (with drag coefficient, C_d = 2.2).

It is also noticed that between 20th December 2000 and 04th March 2001 simulated (with luni-solar gravity) apogee and perigee altitudes do not follow the corresponding observed profiles (Figure- 2). However, during this period, the observed perigee altitude is about 205 km and it remains almost constant. On the other hand, observed apogee altitude profile registers a steady fall. The fall of apogee altitude along with invariant perigee altitude is a typical characteristic of drag dominated orbital evolution without luni-solar gravity. Hence, between 20th December 2000 and 04th March 2001, an approximate yet equivalent simulation of apogee and perigee altitude profiles is performed without considering luni-solar gravity so that the area to mass ratio, which leads to the minimum apogee altitude dispersion, can be estimated. Accordingly, between 20th December 2000 and 04th March 2001, the area to mass ratio, which corresponds to the minimum apogee altitude dispersion, is 0.006 m^2/kg (with C_D = 2.2).

7. ALTITUDE BASED DRAG COEFFICIENT

If the equivalent area to mass ratio is constant in absence of any change in rotational dynamics, then it is clear that a drag coefficient model, which is more consistent with physics, needs to be adopted to achieve still higher level of simulation fidelity. This is because; a spent stage in GTO descends to very low perigee altitudes where the mean molecular free path is comparable with its physical dimensions. The aerodynamic interactions between a body and its surrounding flow field depend on the degree of rarefaction, which is quantified by the ratio of molecular mean free path to characteristic length of the body. A GTO spent stage during its orbital life experiences differential flow regimes and consequently, its drag coefficient may change during its low perigee passages [13]. Molecular mean free path (for MSIS90 atmosphere) as a function of altitude is shown in Figure-9. Also, in the same figure, typical dependence of drag coefficient (for a sphere) on the degree of rarefaction is illustrated. Therefore, it might be more appropriate to consider the drag coefficient as a function of altitude so that more accurate estimation of lifetime can be made.

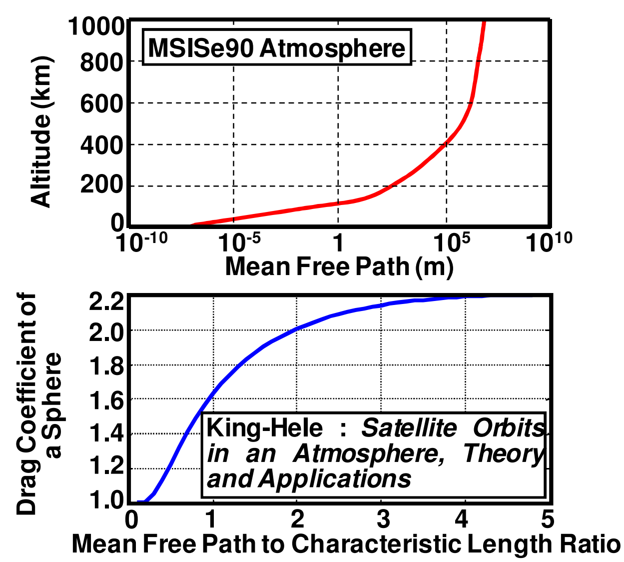

Figure- 9: Variation of molecular mean free path with altitude and drag coefficient with mean free path to characteristic length ratio

With C_D = 2.2, the estimated values of area to mass ratio are 0.006 m^2/kg and 0.005 m^2/kg, which correspond to average perigee altitude of 205 km and 150 km respectively. If it is assumed that the drag coefficient is 2.2 when the average perigee altitude is 205 km then an observational model for drag coefficient as a linear function of average perigee altitude is derived as (Average Perigee Altitude (km) + 125)/150. This model is utilized for the prediction of lifetime beyond 27th August 2001.

8. LIFETIME ESTIMATION

The orbital evolution of Ariane-V rocket body beyond 27th August 2001 (that is 250 days after the insertion on 20th December 2000) is predicted with 10 % variations over the estimated drag coefficient. The simulated and the observed apogee-perigee altitude profiles are shown in Figure- 10. It is noticed that the observed apogee altitude profile always remains within the simulated dispersion bounds. The predicted reentry is between 06th March 2002 and 01st July 2002. The actual date of reentry for Ariane-V rocket body is 31st May 2002.

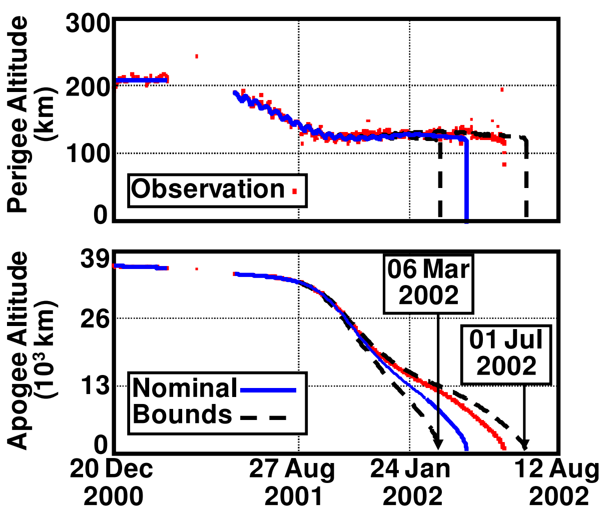

Figure- 10: Observed and simulated apogee-perigee altitude profiles during the lifetime of Ariane-V rocket body

9. PRESENCE OF CHAOTIC MOTION

The estimation of initial ballistic coefficient, **B** $\left(=\frac{m}{C_D A}\right)$ is made during the study from different TLE (Two-Line-Element) epochs from 03-Januaray-2001, 18:16:09 (UTC) to the last TLE epoch 31-May-2002, 16:40:02(UTC). The estimation of initial ballistic coefficient plays an important role in minimizing the cost function that involves mean perigee and apogee values. Six TLE epochs between January 2001 and May 2002, provided in Table 1 are utilized for detailed study. Table 1 provides the estimated values of **B**. A numerical propagator NPOE (Numerical Prediction of Orbital Events), is used for this purpose. Orbit propagation was carried out with NPOE using the force model which includes 36×36 Earth gravity model of GEM10B, atmospheric drag perturbations, lunar and solar gravity effects. MSIS90 density model with the monthly averaged values of solar flux ($F_{10.7}$) and geomagnetic activity (A_p) obtained from www.dxlc.com/solar website were utilized to compute the drag force.

S.No	TLE Epoch (UTC) (DD/MM/YY)	ballistic coefficient (kg/m^2)
1	03/01/2001, 18:16:09	95.5
2	03/03/2001, 12:16:56	153.5
3	03/10/2001, 15:58:56	26
4	03/01/2002, 04:05:34	75.3
5	03/04/2002, 16:10:15	124.94
6	28/05/2002, 04:52:17	96.8

Table-1: Computed values of initial ballistic coefficients (match TLE 28-May-2002, 10:51:54 (UTC))

A detailed study, using NPOE is carried out with the last 3 days TLEs (15 epochs) to estimate the ballistic coefficient from one epoch to match the location of the next epoch. The results are presented in Table 2. They provide a reasonable good match between computed and observed values of latitude, longitude, altitude, mean perigee and mean apogee heights. It may be noticed that for first 3 epochs the ballistic coefficient (kg/m^2) varies between 51.5 and 98.8. For the next 8 epochs, it is between 65 and 176.4. Since the mean perigee altitude does not vary significantly for these 8 epochs, so m/C_D will not change appreciably, therefore the effective area of rocket body must have decreased near perigee, where drag force is the maximum and is mainly responsible for the orbital decay. Further, the value of ballistic coefficient decreases to 57.8, then increases to 98 and finally decreases to 62.1, which provide the last observed orbital parameters. It suggests that during the *third* day before the day of re-entry, the rocket body has toppled and has different effective areas near perigee passage during each orbit. It suggests that the motion must have been chaotic.

A similar study has carried out for the last 11 TLEs of GSLV-F01/cryostage (CS) rocket body [13], which provides the details of the parameters observed as well as computed for these TLE epochs. This study also shows the existence of chaotic motion during the last few days of orbital life.

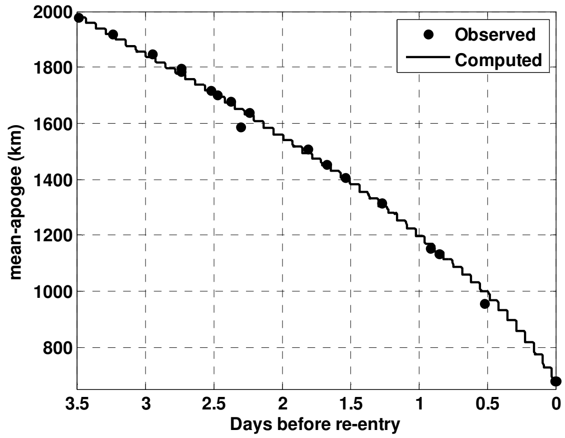

Figure-11: Comparison between the observed and computed mean-apogee values

Figure-11 show a comparison between observed mean apogee and computed mean-apogee values for these 15 epochs. This figure shows a good agreement between the predicted and observed mean-apogee values.

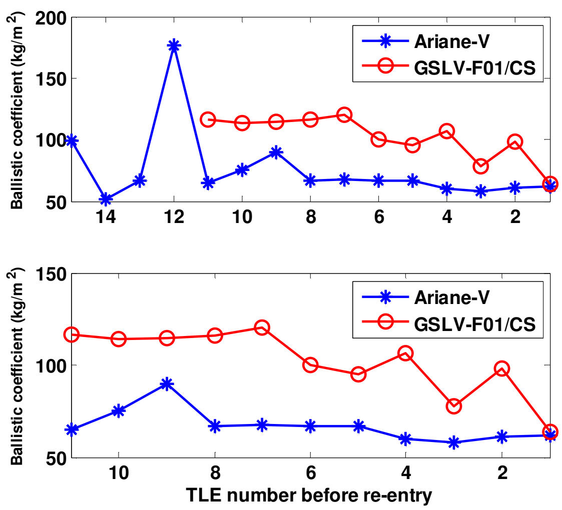

Figure-12: Variation of ballistic coefficient with TLEs from the last observation

Figure-12 shows the variation of ballistic coefficient with TLE of Ariane-V and GSLV-F01/CS. It is evident from the figure that between 11th TLE to 7th TLE from the last TLE, the ballistic coefficient **B**, of Ariane-V varies between 65 and 90, whereas for GSLV-F01/CS it is between 114 and 120. For the remaining six TLEs the value of **B** for Ariane-V varies between 57 and 62 whereas for GSLV-F01/CS it is between 63 and 107. Like Ariane-V the GSLV-F01/CS rocket body suggests that during the last *one* day, it has toppled and has different effective areas near perigee passage during each orbit, which indicates the presence of chaotic motion during the last one day of is journey.

S.No	TLE Epoch From – To (UTC)	latitude (deg) Observed [Computed]	longitude (deg) Observed [Computed]
1	28-May-2002, 04:52:17 to 28-May-2002,10:51:54	0.020 [0.019]	27.238 [27.225]
2	28-May-2002, 10:51:54 to 28-May-2002, 17:50:28	-0.021 [-0.018]	280.169 [279.504]
3	28-May-/2002, 17:50:28 to 28-May-2002, 23:02:19	0.013 [-0.030]	200.856 [200.397]
4	28-May-2002, 23:02:19 to 29-May-2002, 04:12:11	0.007 [0.006]	121.772 [121.817]
5	29-May-2002, 04:12:11 to 29-May-2002, 05:24:17	-2.109 [-2.109]	11.573 [11.390]
6	29-May-2002, 05:24:17 to 29-May-2002, 07:37:28	-0.015 [-0.015]	68.840 [68.803]
7	29-May-2002, 07:37:28 to 29-May-2002, 10:55:01	-0.734 [-0.745]	358.125 [358.140]
8	29-May-2002, 10:55:01 to 29-May-2002, 21:10:49	0.061 [0.009]	223.011 221.682]
9	29-May-2002, 21:10:49 to 30-May-2002, 03:50:51	0.002 [-0.002]	119.296 [119.184]
10	30-May-2002, 03:50:51 to 30-May-2002, 10:15:50	-1.248 [-1.266]	344.618 [344.001]
11	30-May-2002, 10:15:50 to 30-May-2002, 18:45:35	0.985 [0.997]	279.110 [278.898]
12	30-May-2002, 18:45:35 to 30-May-2002, 20:13:36	-0.017 [-0.010]	227.903 [227.843]
13	30-May-2002, 20:13:36 to 31-May-2002, 04:13:04	0.005 [-0.040]	105.352 [104.527]
14	31-May-2002, 04:13:37 to 31-May-2002, 16:39:55	0.0785 [0.0783]	275.849 [275.843]
15	31-May-2002, 16:39:55 to 31-May-2002, 16:40:02	275.905 [276.241]	663.715 [661.469]

S.No	mean perigee (km) Observed [Computed]	mean apogee (km) Observed [Computed]	altitude (km) Observed [Computed]	**B** (kg/m^2)
1	125.598 [119.386]	1917.376 [1917.344]	1811.901 [1808.851]	98.8
2	121.752 [125.806]	1849.955 [1849.944]	1772.348 [1782.477]	51.5
3	122.988 [121.866]	1784.626 [1784.625]	1729.762 [1728.804]	66.7
4	124.137 [114.497]	1718.705 [1718.757]	1680.926 [1679.791]	176.4
5	119.568 [123.904]	1702.200 1702.321]	1011.282 [1011.102]	65.0
6	121.535 [120.290]	1680.194 [1680.135]	1653.342 [1653.716]	74.98
7	1629.830 [1630.323]	1640.737 [1640.677]	121.825 [120.560]	90.1
8	1502.140 1504.540]	1509.748 1509.673]	115.452 122.047]	66.7
9	1404.430 [1404.552]	1405.182 [1405.233]	122.150 [121.125]	67.5

10	1169.663 [1167.111]	1315.150249 [1315.1939]	119.701 [121.465]	67.1
11	1112.434 [1112.462]	1153.880 [1154.935]	122.963 [119.656]	66.8
12	1126.557 [1125.8008]	1131.698 [1131.663]	120.541 [122.987]	60.1
13	945.072 [944.154]	956.502 [956.688]	114.000 [119.672]	57.8
14	660.678 [660.308]	679.495 [679.150]	82.998 [82.466]	61.30
15	99.998 [98.001]	275.905 [276.241]	680.184 [679.494]	62.1

(UTC-Coordinate Time Reference Frame)

Table -2: Comparison between observed and estimated parameters of ARIANE-V

10. CONCLUSIONS

The applicability of the methodology to estimate lifetime of an object reentering from highly eccentric orbit is extended to include low inclination GTO which is going under orbital resonance. This methodology relies upon response surface approximation, which replaces the dynamics of apogee-perigee evolution. In this context, the importance of mean elements to smoothen out short periodic perturbations is also emphasized. The initial perigee altitude and ballistic coefficient are estimated by minimizing average dispersions between predicted and observed apogee-perigee profiles. Additionally, an observational model for drag coefficient as a linear function of average perigee altitude is derived to predict orbital lifetime more accurately. The methodology is successfully utilized to predict the reentry of Ariane-V rocket body. The unpredictable change in ballistic coefficient, which is an indication of chaotic motion, is also observed. The differences in physical characteristics (length to diameter ratio) of Ariane-V and GSLV rocket bodies are reflected in the nature of changes in ballistic coefficients of these two bodies and orbital lifetime.

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