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Analysis of variations in orbital parameters of CubeSats

Master’s Thesis in Computer Engineering (30 ECTS)

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2 Introduction

The natural behaviour of a satellite in Low Earth orbit, from launch until decay, is of increasing interest as a large number of nanosatellites without a propulsion system have been launched since 2003. From the perspective of mission planning and nanosatellite design, it is vital to understand how the satellite lifespan is in correlation to its mass, shape, orbital parameters and to what extent different disturbing forces cause orbit perturbations.

The orbital behaviour of single bigger satellites has been studied before with very accurately measured parameters. However, there have been no similar studies concerning nanosatellites and no precise orbit data is available.

In this work, variations in orbital parameters of nanosatellites based on the CubeSat standard are investigated throughout their orbiting history with the intention to explore how these parameters change over time, correlate to each other, reflect in lifespan, and how different disturbing forces influence orbits. Publicly available historical orbit data in Two-Line Element (TLE) format is used as input information. Since the main purpose of TLE data is to provide rather imprecise orbital elements of satellites to radio amateurs and hobby astronomers for satellite tracking; the suitability of TLE-s for detailed orbital behaviour analysis will be evaluated.

Briefly, work plan can be split into the following stages:

- Evaluate which physical properties of CubeSats affect their behaviour in orbit;
- Create a database structure for CubeSats and their historical orbital information;
- Identify all satellites with associated properties that correspond to the CubeSat standard, have successfully reached the orbit and have TLE data publicly available (data mining);
- Fill in the database;
- Assess the quality of data (errors, uncertainties);
- Analyse the evolution and correlations of the orbital parameters (data analysis).

3 Background information

3.1 About CubeSats

Embrace of CubeSat standard is one of the most remarkable development in space technology during the 21st century. Satellites adopted to this standard are relatively inexpensive to build and send to orbit. Thereby many universities, enterprises, military unions and others have composed their own CubeSats for different purposes – mainly Earth observation, technology demonstration and education. Planned mission length is usually around 6-12 months. CubeSats average orbit altitude is between 200 and 1000 km.

All CubeSats (cube alike) satellites are either 1 unit (1U) satellites or have their height (and thereby also mass limit) multiplied by 1.5, 2 or 3 and are called accordingly: 1.5U, 2U or 3U. Figure 1 presents a typical example of one unit CubeSats. 1U CubeSat dimensions are 100x100x100 mm (with rail 100x100x113.5 mm) and mass up to 1,33 kg. 3U CubeSat dimensions are 100x100x300 mm (with rail 100x100x340) and mass up to 4 kg [1].

Figure 1. An example of a 1U CubeSat [2].

Several deployable parts may lie in CubeSats, usually a boom or wings with solar arrays that are expected to be deployed at some point during the mission. Mostly 1U CubeSats are built, but 3U-s are common as well. 1.5U-s and 2U-s are noticeably rarer. Mission's success rate is near 50% [3-4].

3.2 Factors affecting CubeSat orbit

Balance between centripetal force and gravitational force allows satellites to stay in orbit [5]. CubeSats (and other objects) orbiting velocity is in direct correlation of its current altitude: the lower the altitude, the higher the velocity (and orbital period) to balance strengthened gravitational force. Figure 2 illustrates what forces affect CubeSats while in orbit.

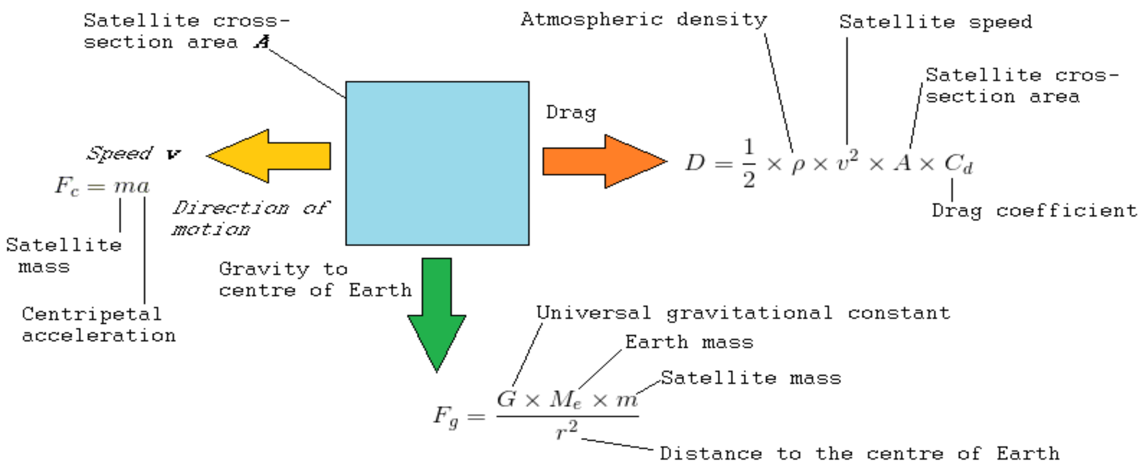

Figure 2. Forces acting on a satellite in a low Earth orbit [6].

A satellite in a nearly circular orbit has almost constant velocity and a satellite in a highly elliptical orbit has a higher orbital velocity in perigee than in apogee. When the satellite travels through atmosphere, it experiences a drag force in a direction opposite to the velocity vector. The atmospheric drag is largest in perigee, thus reducing the eccentricity of the orbit.

The bigger the satellite cross-section size towards motion direction, the bigger drag. Drag coefficient depends on the object shape. Circular cross-section area has a little bit smaller drag coefficient than a square, which in turn has smaller drag coefficient than objects with deployable parts [7].

In a circular orbit, radial acceleration a (m/s²) equals to object`s linear velocity along the circular path squared and divided by radius:

$$a = \frac{v^2}{r} \qquad (1)$$

where v (m/s) is object velocity and r (m) is a radius.

In a stable orbit, the satellite kinetic E_k (J) and potential E_p (J) energy sum is constant in every point of the orbit.

$$E = \frac{v^2}{2} - \frac{\mu}{r} = -\frac{\mu}{2a} \quad (2)$$

Component $v^2/2$ corresponds to kinetic energy and $-\mu/r$ to potential energy. μ (km^3/s^2) is gravitational parameter, a sum of universal gravitational constant G and Earth`s mass M multiplication ($G*M$). v (km/s) is satellite velocity vector at specific moment, a (km) is a length of semi-major axis, r (km) is current satellite distance from Earth mass centre point.

Satellite current orbital velocity v (km/s) at any point on elliptical orbit can be calculated as follows:

$$v = \sqrt{\mu\left(\frac{2}{r} - \frac{1}{a}\right)} \quad (3)$$

where μ (km^3/s^2) is gravitational parameter of the Earth, r (km) is current orbital radius of the satellite and a (km) is semi-major axis of the orbit.

Satellites orbital eccentricity is expressed mathematically with a decimal number between zero and one. 0 means perfectly circular orbit, values between 0 and 1 means elliptical orbit and 1 is a parabolic escape orbit. All CubeSats have relatively circular orbits so their eccentricity values are tilt strongly towards zero (eccentricity is important factor for Earth observation as it helps to hold a satellite longer above some part of Earth).

There are several disturbances affecting a CubeSat’s orbit. These factors can be divided into three main categories: non-perfect shapes of the Earth, third parties influence and artificial force (propulsion thruster) [8].

- Atmosphere density;
- Earth`s mass heterogeneous distribution;
- CubeSat shape;
- Celestial bodies gravitational forces;
- Pressure of the Sun radiation.

In Low-Earth orbit (LEO) (altitudes up to 2000 km) [9], atmosphere has the biggest influence on a satellite's orbit. The magnitude of this effect depends on the satellite mass, size and shape. CubeSats shape may vary due to deployed parts. Atmospheric particles cause the force of friction, which depends on altitude, solar activity and Sunlight/eclipse. The atmospheric density and gravitational force are smaller near poles than the equator, at same orbit altitude. During the 11-year solar cycle maximum, the atmosphere extends higher, so satellites experience higher drag.

The second major factor is Earth's oblateness. The effects of the motion of the Moon and other celestial bodies, and solar radiation pressure are several orders of magnitude smaller in Low-Earth orbit.

The Sun and Moon gravitational forces have a marginal effect on satellite orbit in month and year basis. [10]. There has been a simulation how strongly various disturbances affect satellites orbits [11]. Figure 3 shows the effect of different disturbing forces on a satellite in various altitudes.

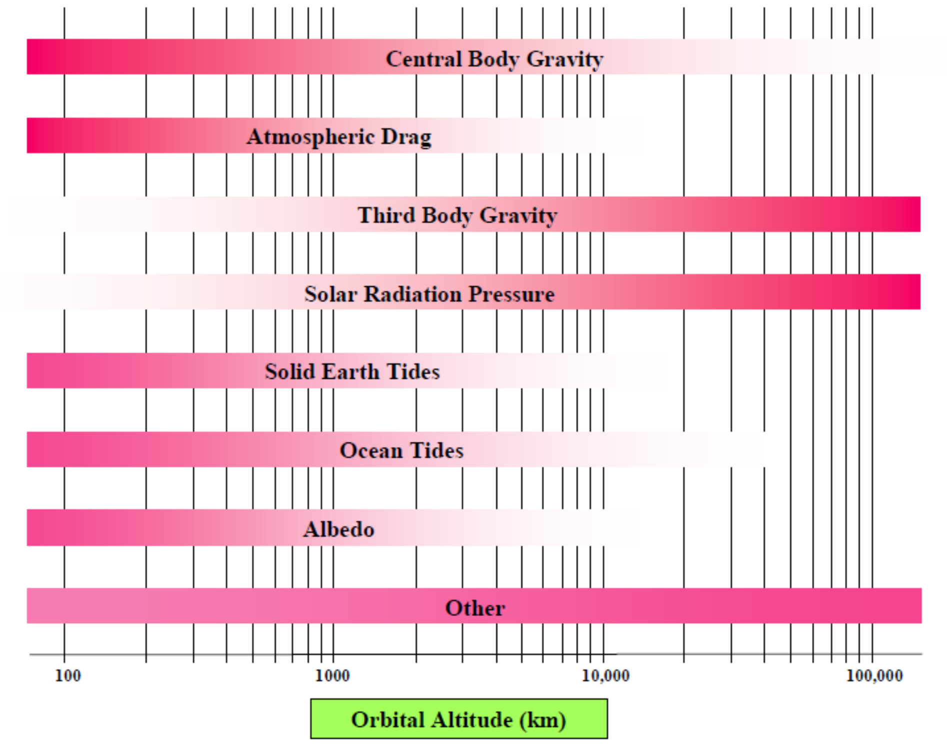

Figure 3. Approximate effect of different disturbance forces on a satellite orbit depending on the altitude. Note that specific accuracy requirements may extend the areas of applicability, and hence the faded colour bars [11].

Differences in Earth's mass distribution affect the ellipticity and inclination of orbits. Precession (similarly to the effect of the Sun) makes satellite orbit to twist slowly around Earth –orbit spin plane turns slowly in a way where ascending node (where satellite crosses equator from South to North Pole) shifts continuously to west. To keep a satellite in a stable orbit, thrusters are required for regular correction manoeuvres; however, most CubeSats do not have a propulsion system on board.

3.3 Single CubeSat lifespan

In Low-Earth orbit satellites experience orbital decay and their lifetimes are predicted by estimating drag forces from atmospheric models. The prediction of CubeSats lifetimes depends upon knowledge of several factors. Firstly, initial satellite orbital parameters must be known. Secondly, the satellite mass to cross-section area in direction of travel must be determined. Thirdly, atmospheric density and how this responds to space environmental parameters must also be predicted. [6]

Satellite mass and the cross-section area are directly related to its lifetime [12]. Every CubeSat average orbital altitude is declining over time (assuming no propulsion thrusters or other artificial methods are used) hence its lifetime is also related to the fact how high it was sent to orbit.

Satellite with launching altitude over 800 km is more expensive to be placed into orbit and is expected to have few hundred years of lifespan. Satellite with launching altitude around 300 km will decay in few months, but is remarkably cheaper to send up. Satellite with orbiting altitude between 500 and 650 km are preferred for missions with duration of several years. Satellite expected lifetime will stay there beneath 25 years, which is now compulsory for all satellites in new Low-Earth Orbit. Altitude less than 200 km is the matter of days or even hours when satellite decays. Figure 4 illustrates how satellite orbit dwindles over time until decay and figures 5 shows how fast satellite altitude declines in relation to its current altitude.

CubeSat lifetime can be considerably increased by making it heavier and/or reducing its cross-section area, or reduced greatly while sending in orbit during high solar activity [12].

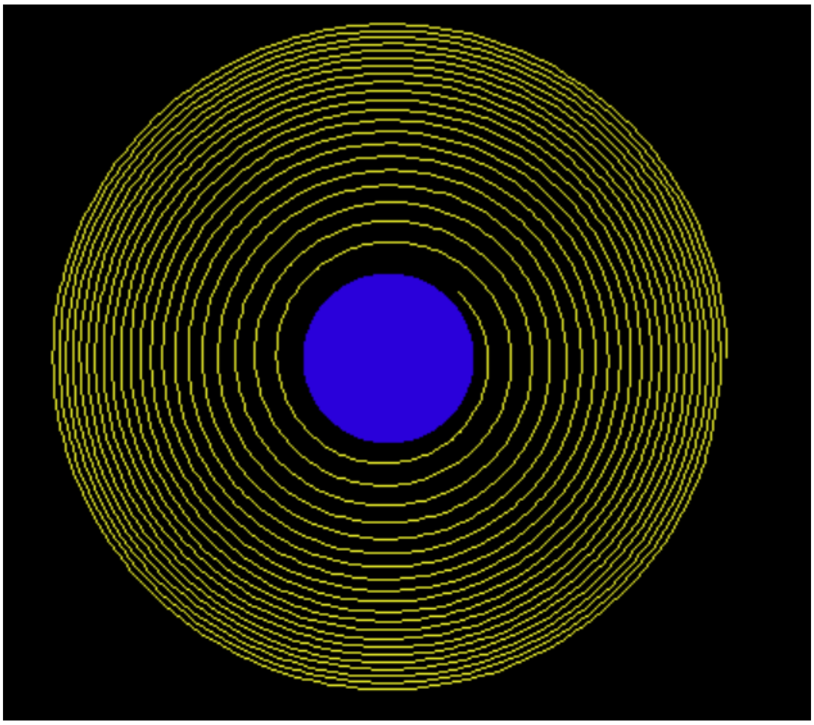

Figure 4. Diagrammatic view how satellite decays in Low-Earth orbit. In reality, many more occur as shown here [6].

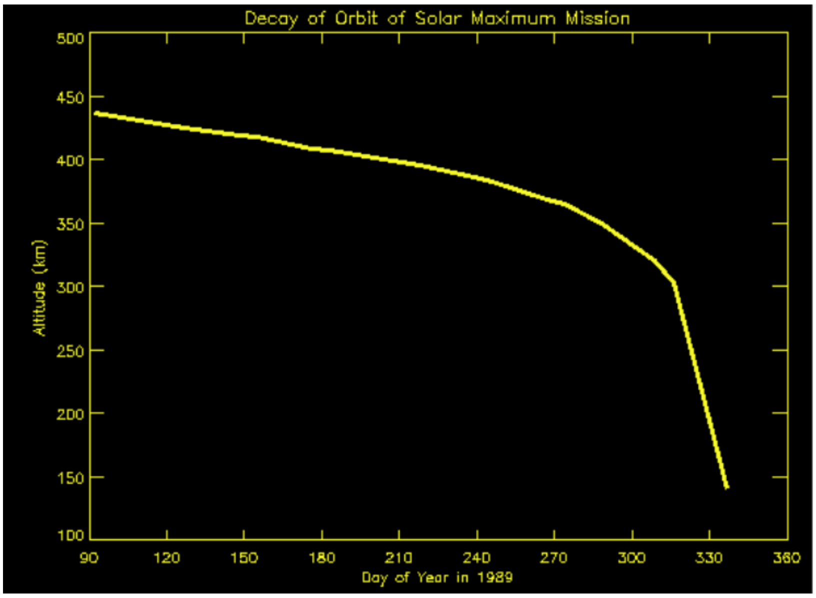

Figure 5. Actual decay curve for Solar maximum Mission satellite which re-entered the Earth`s atmosphere in December 1989. Satellite decays slowly at higher altitudes and very rapidly towards the end of its life [6].

4 Data and methodology

4.1 Two-Line Element Set

The plan is to use database for CubeSats and their historical orbital parameters in TLE format for later analysis.

TLE was first taken into use by (United States) Department of Defence (DoD) more than 25 years ago in order to predict satellite positions variations in particular way [13]. Different models (different equations) are applied on objects regarding to their altitude (velocity). North American Aerospace Defense Command (NORAD) uses TLE-s to maintain general orbital information about all resident space objects, including CubeSats, and classifies them as near-Earth (period < 225 minutes) or deep-space (period ≥ 255 minutes) objects. NORAD element sets consist of several orbital parameters generated at specific time moment and are expressed as mean values (obtained by removing periodic variations in a particular way [14]), but unfortunately does not contain any kind of accuracy information. TLE-s are public information and historical information about past TLE-s can be accessed.

NORAD also provides orbital collision warnings to satellite owners whenever risk appears. TLE epoch timestamp is given with an accuracy of a second, so there is a small uncertainty exactly when these calculated orbital parameters were valid – a typical CubeSat moves around 7 km in a second.

End users - from radio amateurs to enterprises are using TLE information among with orbital propagators and atmospheric models to track objects. There are many software programs that use TLE-s to predict the appearance and disappearance of a satellite for a specific location on Earth. Standard NORAD`s Simplified General Perturbations 4 (SGP4) orbital propagation model for near-Earth objects is implemented in most orbital tracking software packages. Simplified Deep Space Perturbations 4 (SDP4) model is used for objects in far space.

4.2 Data selection

The plan is to choose out objects that's corresponds to CubeSats standard from publicly available TLE database.

Apart from CubeSats, the spacecraft form, mass and orbit varies a lot. First CubeSats were sent to orbit in June 2003. During the past ten and a half years, over 150 CubeSats have been successfully launched to various orbits near Earth. Most CubeSats have an orbiting history up to two-three years and six have orbited more than ten years.

CubeSats provide an excellent opportunity to investigate how celestial forces trigger perturbations among objects with same or similar parameters. The great majority of CubeSats does not have propulsion systems and thus their orbits cannot be affected artificially. It is hard to find a CubeSat with a propulsion thruster and launching date before a year 2013, but starting 2013 they aren't so rare anymore. Some CubeSats have GPS receiver on-board.

CubeSat standard sets requirements for the mass centre of the satellite.

Not all cube alike satellites are eligible for analysis. Firstly, there are some "CubeSats", such as CUTE 1.7 + APD II [15], all MEPSI [16], BRITE satellites [17] and others that are considered as bogus CubeSats and thus are out of scope. They do look similar to a CubeSat, but actually do not meet the CubeSat standard. As one exception - three 1U spherical POPACS are included.

Secondly, all PocketQubes [18] (with dimensions of 5x5x5 cm) are also discarded for analysis since they have very short orbital history (first launch in November 2013 [19]).

Thirdly, NORAD does not return historical data for 26 CubeSats. These satellites are either military or were launched among military satellites. Therefore, they cannot be included into analysis. In total, 131 CubeSats are included in the analysis.

Some CubeSats may have remarkable deployable structures, such as big booms, solar arrays, and long antennas. For example, DELFI C3 (also known as OSCAR 64) is a 3U CubeSat that has 4x30 cm deployable solar arrays and 8 antennas; WE WISH is a 1U CubeSat with two side panels (~10x10 cm) deployable side panels, two antennas and a boom. Some CubeSats may be designed to have their booms pointing towards specific direction and thus their cross section area towards direction of motion may differ from expected. For example, QUAKESAT has a 7 m long telescoping boom, which is nadir pointing over the North Pole. These features can influence their orbits.

There are also CubeSats that are supposed to have noticeable outer parts, but have failed to deploy them. For example, DTUSAT 1 was launched in 2003 and was designed to deploy a 1.3 m boom, but the radio contact with satellite was not established and boom never deployed. Or PW-SAT, it was launched in February 2013 and supposed to deploy a 2 m long tail but at least so far - a bit more than a year later - it hasn't been deployed yet.

4.3 NORAD`s Two-Line Element Set Format

A NORAD two-line element set consist of two lines of data, both up to 69 characters long and can be used together with NORAD`s SGP4/SDP4 orbital model to determine the position and velocity of the satellite [20]. Figure 6 shows what type of character is valid for both columns in TLE and figure 9 shows an example of it.

```
1 NNNNNC NNNNNAAA NNNNN.NNNNNNNN +.NNNNNNNN +NNNNN-N +NNNNN-N N NNNNN
2 NNNNN NNN.NNNN NNN.NNNN NNNNNNN NNN.NNNN NNN.NNNN NN.NNNNNNNNNNNNNN
```

Figure 6. TLE format. Columns with a space or period can have no other characters. Column with "N" can have any number 0-9 or a space. Columns with "A" can have any character A-Z or a space. Column with "C" can only have a character representing the classification of the element set (normally either "U" for unclassified data or "S" for secret and publicly unavailable data). Columns with "+" can have a plus sign, a minus sign or a space. Columns with "-" can have a plus or minus sign [20].

Table 1. Two-Line Element Set Format Definition, Line 1

Field	*Column*	*Description*
1.1	01	Line Number, always "1"
1.2	03-07	NORAD/NASA Catalogue Number
1.3	08	Security Classification ("U" stands for unclassified, publicly available data)

1.4	10-11	International Designator (last two digits of launch year)
1.5	12-14	International Designator (launch number of the year)
1.6	15-17	International Designator (piece of the launch, “A” indicates payload, “B” the rocket booster or second payload, “C” designates the third object catalogued for launch)
1.7	19-20	Epoch Year (last two digits of year)
1.8	21-32	Epoch (day of the year and fractional portion of the day)
1.9	34-43	First Time Derivative of the Mean Motion (divided by two, in units of revolutions per day^2)
1.10	45-52	Second Time Derivative of Mean Motion (decimal point assumed, divided by six, in units of revolutions per day^3)
1.11	54-61	BSTAR drag term (decimal point assumed, a SGP4- type drag coefficient, a ballistic coefficient, in units of radii^{-1}
1.12	63	Ephemeris type (i. e. orbital model) used to generate the data. All disturbed element sets have value of zero and are generated using SGP4/SDP4 model.
1.13	65-68	Element set number (normally incremented every time a new element set is generated)
1.14	69	Checksum (Modulo 10) of the data on that line

Note that an epoch of “13001.00000000”, for example, corresponds to 0000 Universal Time (UT) on 01/01/2013. In other words, a midnight between 31/12/2012 and 01/01/2013. An epoch of “13000.00000000” actually corresponds to the beginning 31/12/2012.

However, fields 1.9 and 1.10 are not used by the SGP4 and SDP4 orbital models (only by the simpler SGP model) and, therefore, serve no real purpose. [20]

Field 1.10 and 1.11 have a bit different format than other fields. They use a modified exponential notation with an implied decimal point. This practise comes from Formula Translating System (FORTRAN, programming language, where all such numbers range from 0 to less than 1. First six columns of both fields represents the mantissa and last two digits an exponent. For instance, the value of "-12345-6" means -0.12345x10^{-6}. These fields can also be blank, representing a zero.

Bstar is a SGP4 drag-like coefficient. In aerodynamic theory objects have ballistic coefficient *BC*, which is the product of its drag coefficient C_D , cross-sectional area size *A* (m^2), and mass *m* (kg) as follows:

$$BC = m/C_DA \quad (4)$$

Bstar (Earth radii^{-1}) is an adjusted value of *BC*, using the reference atmospheric density ρ_o (kg/m^3).

$$Bstar = BC\rho_0/2 \quad (5)$$

To calculate field 1.14, individual numeric values have to be added, ignoring all letters, periods, plus signs and assigning a value of 1 to a minus sign. So letters, blanks, periods, plus signs = 0; minus signs = 1. The checksum is the last digit of that sum.

Table 2. Two-Line Element Set Format Definition, Line 2

Field	*Column*	*Description*
2.1	01	Line Number, always "2"
2.2	03-07	Satellite Number (same as in line 1)
2.3	09-16	Inclination [degrees, 0-180] (measured counter-clockwise from true East to true West), *i*
2.4	18-25	Right Ascension of the Ascending Node [degrees, 0-360], *Ω*
2.5	27-33	Eccentricity [0-1] (decimal point assumed), *e*
2.6	35-42	Argument of Perigee [degrees, 0-360], *ω*
2.7	44-51	Mean Anomaly [degrees, 0-360], *M*
2.8	53-63	Mean Motion [revolutions per day], *N*
2.9	64-68	Revolution number at epoch [revolutions]
2.10	69	Same as 1.4, Checksum (Modulo 10) of the data on that line

Line 2 consists primarily of mean element sets calculated using SGP4 or SDP4 orbital model. Six values (inclination, right ascension of the ascending node, eccentricity, argument of perigee, mean anomaly and mean motion), called orbit parameters and describing the geometry of the orbit, are required to orient an orbit in space and are all in line 2. There is also a seventh number, the epoch of the TLE in first line, which describes when all these six orbital parameters in line 2 were valid. Three numbers describe the size and shape of the orbit and satellite location on it. Other three numbers describe the orientation of orbit with reference to the Earth`s surface. These six parameters are show on figure 7.

Fields 2.4, 2.6, .2.7 all have units of degrees and can range from 0 to 360, field 2.3 units are degrees as well but ranges from 0 to 180. Field 2.5 is a unitless value with an assumed leading decimal point, for example “1234567” means 0.1234567. Field 2.8 is measured in revolutions per day.

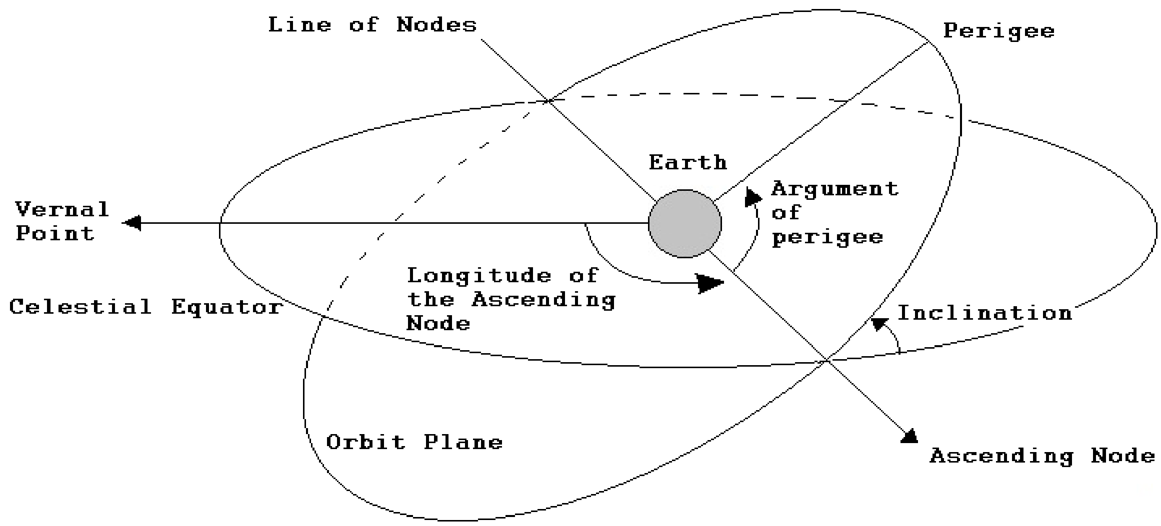

Figure 7. Orbital parameters to orient an orbit in space [21].

Eccentricity *e* shows how elliptical (oval) satellite orbit is and can have a value between 0 and 1 without an unit. “0” represents a perfectly circular orbit and “1” parabolic escape orbit. It is defined by the relationship:

$$e = \sqrt{1 - \frac{b^2}{a^2}} \qquad (6)$$

where *a* (km) is the semi-major axis and *b* (km) is the semi-minor axis.

Mean motion and eccentricity are raw values. Orbit semi-major axis *a* (km) can directly be computed from mean motion *N* (rev/day) with equation:

$$a = \frac{6.6228}{N^{\frac{2}{3}}} \qquad (7)$$

There is an inverse relationship between them – as the mean motion increases, the semi-major axis decreases. Orbit semi-minor axis *b* (km) can be calculated from semi-major axis *a* (km) and eccentricity as follows:

$$b = a\sqrt{1 - e^2} \qquad (8)$$

Knowing orbit semi-major axis, apogee and perigee can be derived.

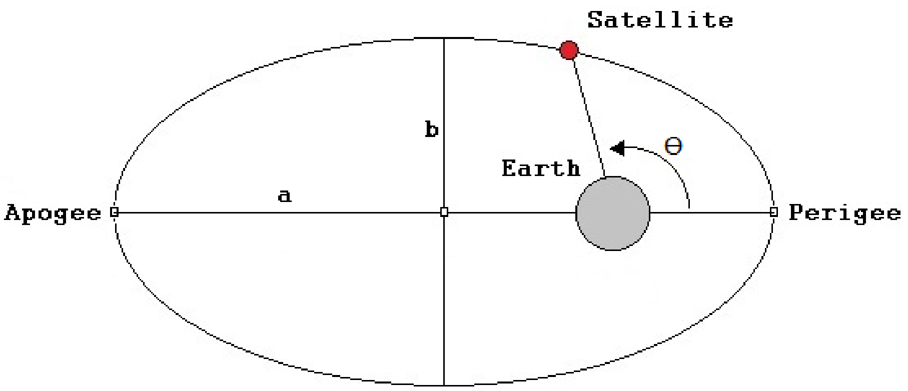

Figure 8. Eccentricity and true anomaly. Satellite on this specific drawing has an eccentricity value around 0.75 [21].

True anomaly *Θ* (deg) defines the location of the satellite within orbit; it is measured from the perigee position to the satellite. Mean anomaly *M* (deg) can be calculated from true anomaly by using an intermediate quantity called eccentric anomaly *E*. The equations are as follows:

$$cosE = \frac{e + cos\theta}{1 + ecos\theta} \qquad (9)$$

$$M = E - esinE \qquad (10)$$

In a circular orbit, true anomaly equals to mean anomaly.

In NORAD`s practice, a revolution begins when the satellite is at the ascending node of its orbit. Revolution is the period between successive ascending nodes. Rev 0 is the period from launch to first ascending node and Rev 1 begins when first ascending node is reached.

Any number less than maximum possible can be padded with either leading spaces or leading zeroes, for example an epoch can be represented as "13001.12345678" or "13 1.12345678" or an inclination can be represented as "56.1234" or "056.1234". In practice, leading zeroes are used for fields 1.5 and 1.8 and leading spaces elsewhere.

This work concentrates mostly to mean motion, eccentricity and inclination.

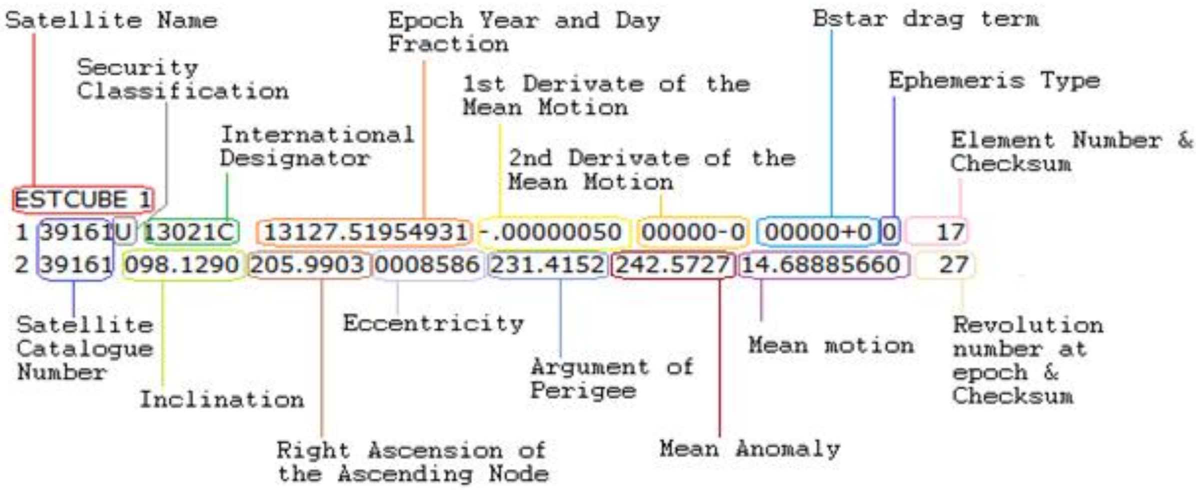

Figure 9. An example TLE (first TLE created for ESTCube-1). It was launched in 2013 and was on 21st launch on that particular year. First TLE for ESTCube was created on 7th of May (127th day of the year), 12:28:09 PM (0.51954931 portion of the day). First derivative (ballistic coefficient) is -0.00000050 revs/day^2, second derivative is 0.00000 rev/day^3. Drag term is 0.00000 radii^{-1}. Element number is 1 since it is the first TLE created for this object. Adding values in first line gives total 87 and in second line 207, so the checksum for both lines is 7.

One way to calculate satellite altitude *h* (m) above the surface of Earth utilizes satellite mean motion value *N* (rev/day) and is based on idea finding radius from Earth centre point R_{orbit} (m) and later taking Earth own radius R_{earth} (m) off. For example, mean motion value of 14.68885660 gives an altitude of 675.38 km.

$$h = R_{orbit} - R_{earth} \qquad (11)$$

$$\frac{T^2}{R_{orbit}^3} = \frac{4 \times \Pi^2}{G \times M_{earth}} \rightarrow R_{orbit}^3 = [(T^2 \times G \times M_{earth})/(4 \times \Pi^2) \quad (12)$$

where T (s) is a period (86400 seconds divided by mean motion value), R_{orbit} (m) is radius from Earth`s centre point, $G$ ($Nm^2/kg^2$) is universal gravitational constant and $M_{earth}$(kg) is Earth`s mass.

4.4 NORAD Two-Line Element Set accuracy assessment

4.4.1 Propagation prediction

Unfortunately, TLE does not provide any kind of accuracy information. There have been studies exploring satellites location and propagation prediction based on values in element set. The accuracy of propagation predictions using two-line element sets and SGP4 model depends on how far away from element set epoch predictions are made [22]. As shown on figure 10, the error rate increases quite similarly in both directions, means there is not much difference if propagation was calculated towards past or future and on average, accuracy stays within few kilometres for few first days and then goes off to 15 kilometres already after 10 days. Error characteristics for satellites in similar orbits may be considerably different, so the error characteristics of each satellite should be determined independently.

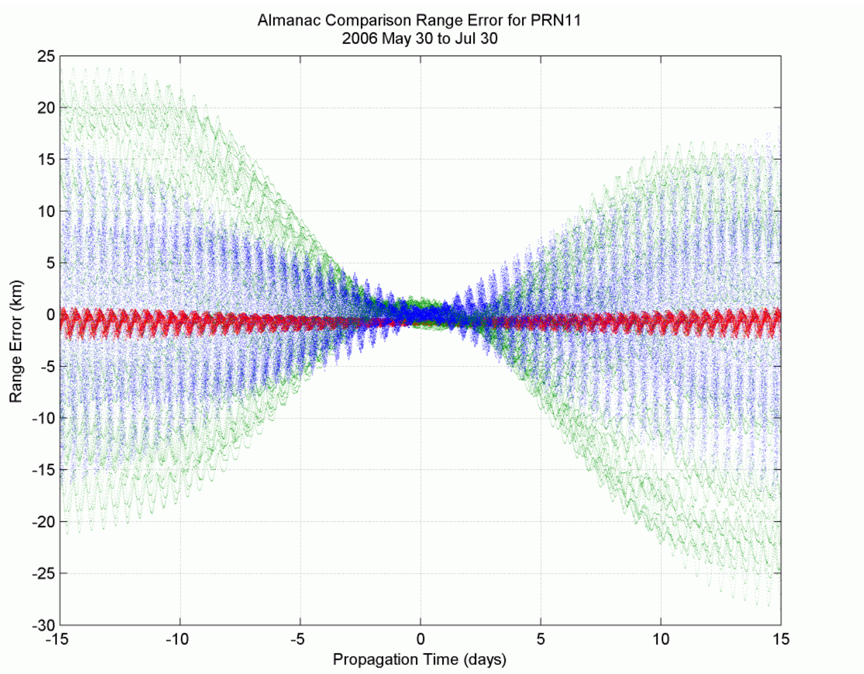

Figure 10. An example of prediction error rate for a satellite. Radial errors are shown in red, in-track errors in green and cross-track errors in blue [22].

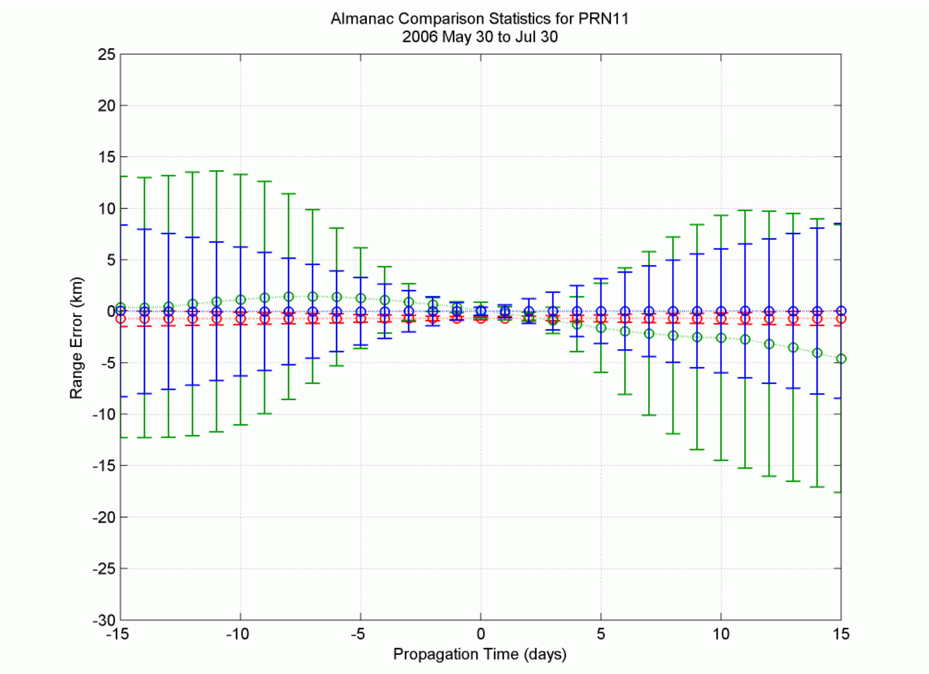

Figure 10. An example of prediction error amplitudes for a satellite. Radical errors are shown in red, in-track errors in green and cross-track errors in blue [22].

CelesTrak has very recently started to offer supplemental two-line element sets derived directly from owners/operators supplied data and are released one to two days later after they are generated. Postponed release offers a lot better accuracy.

According to GPS test (24 hours, 31 spacecraft were involved), predictions from TLEs were compared to NGA Precise Ephemerides. NORAD two-line element sets had an average error rate of 7.54 km (maximum ~32.45 km), supplemental sets 0.87 km and 2.37 km accordingly. In other test, comparison was against GLONASS Final Precise Ephemerides. Error rates were 3.30 km (maximum 9.39 km) and 0.20 km (maximum 0.54 km) accordingly [23].

This research does not concentrate on satellite exact location nor propagation prediction, but on historical orbital parameters behaviour over time. Therefore, values own relative precision in element-sets is important and accuracy not much. Values precision can be appraised by comparing sequential TLE values.

4.4.2 Relative precision

TLE-s created within first ten days after object launch are left out of the analysis. Within this period, there are rapid changes in TLE parameters. For example, without first ten days, 6.87% of all LIBERTAD 1 TLE-s (out of 3413) are towards positive altitude change (see figure 11). There are only 10 (out of 233) positive altitude change bigger than one meter (biggest 25 meters). Rest are less than a meter. Including first ten days, two positive changes in Libertad 1 altitude are more than a thousand meters (biggest 14.14 km).

Small positive altitude changes may be explained simply by radar measuring uncertainty and can therefore considered as normal behaviour. A bigger positive change may sometimes occur right after negative change to balance. POPACS 2 and 3 have similarly huge altitude increase within the first ten days. In this case, it is caused by huge (7-10 days) delay between two element sets.

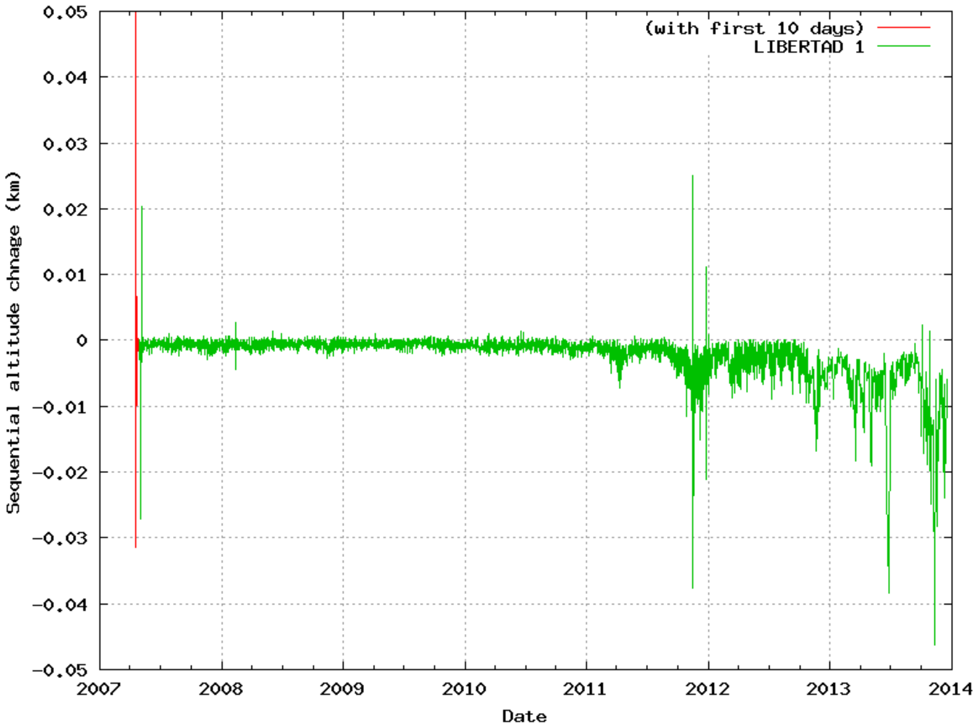

Figure 11. LIBERTAD 1 daily altitude changes since launch. First peak (out of scale) value is 14.14 km. Peaks close to each other but in opposite direction is most likely balancing effect.

4.5 Data quality assessment

Properties that were put to database aside the satellite name and NORAD catalogue number were: mass; type (unit); dimensions; general categorization by deployable parts; orbit type (eccentricity value) at start; orbit type by Jan 2014 (or until decay); launch date; decay date (if applicable); planned/actual mission length; information about attitude control; whether satellite has a GPS receiver or a propulsion system on board.

Information about CubeSats and their properties was collected from official mission sites, relevant web pages such as www.space-track.org, http://celestrak.com/ http://www.astronautix.com/craft/CubeSat.htm, http://space.skyrocket.de/index.html, https://directory.eoportal.org/web/eoportal/satellite-missions/ and some others.

Collecting information about CubeSats and their important properties turned out to be extremely time-consuming task. Even identifying CubeSats real names took a lot of effort as they were often called differently in various information sources and some have many nicknames. To unify the naming conventions, the NORAD naming format was taken into use for this thesis.

Official mission web pages often did not contain more usable information than satellite name, launch date and perhaps a photograph of the satellite or drew vision how it would look like in space during the mission. So most information had to be collected from other sources, which surprisingly often did not even agree on the satellite mass, not to mention other properties. By far, the hardest part was to identify mission status - whether deployable parts were actually deployed, what was actual and planned mission length. Some info was also collected from Facebook and Twitter.

All found CubeSats were entered into database with all TLEs and their status as end of 2013. Fortunately, CubeSat standard sets tiny variation range for mass and size. Launch orbit type was taken from first TLE and decay date from last TLE.

Collected information accuracy may vary over satellites, but can be generally evaluated as follows:

- very good or good accuracy - information is factually correct or cannot be far from truth: CubeSat name, NORAD catalogue number, type (unit), dimensions, deployable parts, orbit type at start and by January 2014 (or decay), launch date, decay date.
- average - several values over all CubeSats may be wrong: mass, propulsion thrusters presence, GPS receiver presence.
- poor accuracy - a good chance to be wrong (or information is unknown): actual/planned mission length, mission end date (whether satellite is just orbiting or is being actively operated), deployable parts possible release during the mission.

4.6 Data storing

Data was stored into regular, free licensed MySQL database located in *kratt.physic.ut.ee.* Data amount was known not to expand progressively. There was no need for any features provided by paid licenses either. Two tables were created, one for CubeSats and the other for all element-sets. Information about CubeSats was collected and inserted into database manually.

TLE data was collected from https://www.space-track.org. The website offers several ways to retrieve data: by search, query form (returns archived data by email) or Application Programming Interface (API, allows users to access data programmatically using custom, stable URLs with configurable parameters). In this case, where data was collected once, there was no actual reason to prefer API to search form. Search allows collecting complete historical data at once by the satellite catalogue number.

Collected data needed to be edited before inserting into the database: sometimes spacing was missing between mean motion and revolution number at epoch with checksum (see figure 12).

```
1 27842U 03031C   03182.84629643  .00000171  00000-0  10000-3 0    13
2 27842 098.7247 189.7521 0008088 285.9739 074.0497 14.20484086   167
1 27842U 03031C   03182.98717165 +.00000004 +00000-0 +22251-4 0 0004
2 27842 098.7237 189.8930 0008293 285.1077 074.9205 14.2048972100018
1 27842U 03031C   03183.48023672 +.00000006 +00000-0 +23366-4 0 0002
2 27842 098.7236 190.3792 0008292 283.6766 076.3487 14.2048977200025
1 27842U 03031C   03183.97330223  .00000006  00000-0  23366-4 0    30
2 27842 098.7254 190.8646 0008049 278.4568 081.5579 14.20487212   326
```

Figure 12. First two-line element sets created for DTUSAT. An example about missing spacing in end of second line.

Desire was to save whole TLE data into database not as one string, but to keep values in respective columns. Single CubeSat all historical TLE data were copied from the web page to a text redactor and edited before mass insertion into the database.

Using a macro, a space was set right after eight decimal points of mean motion value (where every mean motion value ended), year and portion of day was separated and second line was joined with first. Right after, all possible extra spacing was replaced with a single space using regular text redactor replace option. Without separating second line ending, revolution number with checksum would have been lost if case spacing was missing from a TLE. Because only eight decimal points would fit into mean motion column and rest would be dropped, not moved over to last column in database, which in this case would be just left empty. A complete historical TLE data of a CubeSat was loaded at once from a file, satellite by satellite. In total, over 141 thousand TLEs were collected.

5 Results

5.1 Data overview

In total, 131 CubeSats were put into database. Division by units is as follows: 80 1U, 32 3U, 14 1.5U and 5 2U. Mass varies from 800 g to 5800 g, Average launch altitude of all 131 CubeSats is 596.66 km and orbit inclination 72.38 degrees. Most CubeSats have been orbiting naturally, without artificial methods for orbit control.

Despite the fact that the CubeSat standard sets strict mass limits, some launch adapters are tested to actually allow greater mass, up to 6 kg for a 3U [24]. 1U-2U CubeSats have so far stuck with the mass limit set in the standard, but half of 3U-s does exceed 4 kg.

100 CubeSats were still orbiting in the end of 2013 and 31 have decayed. Mission actual length varies from zero (total 18, failed at start, usually the case for not being able to establish a radio contact) to over ten years (and still actively being operated). On average, mission actual length is a little bit over 10 months. Planned mission length is usually between 6-12 months, however in reality, successful missions tend be prolonged into many years.

Based on pure number comparison, an average launch eccentricity through all 131 CubeSats is 0.0097553 and by January 2014 (or last value before decay) average value is 0.0080499 (-0.0017053). Biggest single eccentricity value at launch was 0.0848817, lowest 0.0001222. By January 2014 (or last value before decay) these values were 0.0784560 (change -0.0064257) and 0.0002116 (change +0.0000894).

Table 3. Statistics about CubeSats launches in past ten and a half years (see also figure 13 and 14).

Launch number	Date	Cube Sats count	Average altitude (km)	Average orbit inclination (deg.)
1	30/06/2003	6	836	98.73
2	27/10/2005	3	706	98.18
3	21/02/2006	1	515	98.19
4	16/12/2006	1	427	40.02
5	17/04/2007	7	723	98.09
6	28/04/2008	5	639	97.98
7	19/05/2009	4	457	40.47
8	23/09/2009	4	729	98.33
9	20/05/2010	3	309	29.98
10	12/07/2010	2	639	98.15
11	12/08/2010	8	305	34.53
12	12/10/2011	1	869	19.98
13	28/10/2011	5	646	101.71
14	13/02/2012	7	886	69.48
15	21/07/2012	2	423	51.65
16	04/10/2012	3	423	51.65
17	25/02/2013	2	789	98.63
18	19/04/2013	5	546	64.89
19	21/04/2013	4	244	51.61
20	26/04/2013	3	653	98.07
21	07/05/2013	1	675	98.13
22	29/09/2013	3	888	81.01
23	19/11/2013	3	423	51.65
24	20/11/2013	29	508	40.52
25	21/11/2013	19	656	97.81

Figure 13. CubeSats launches with average launch altitude and number of satellites in launch (represented by circle size) during past eleven years. Launches with highest eccentricity values (>0.03) have big disparity between their perigee and average altitudes and are coloured in blue.

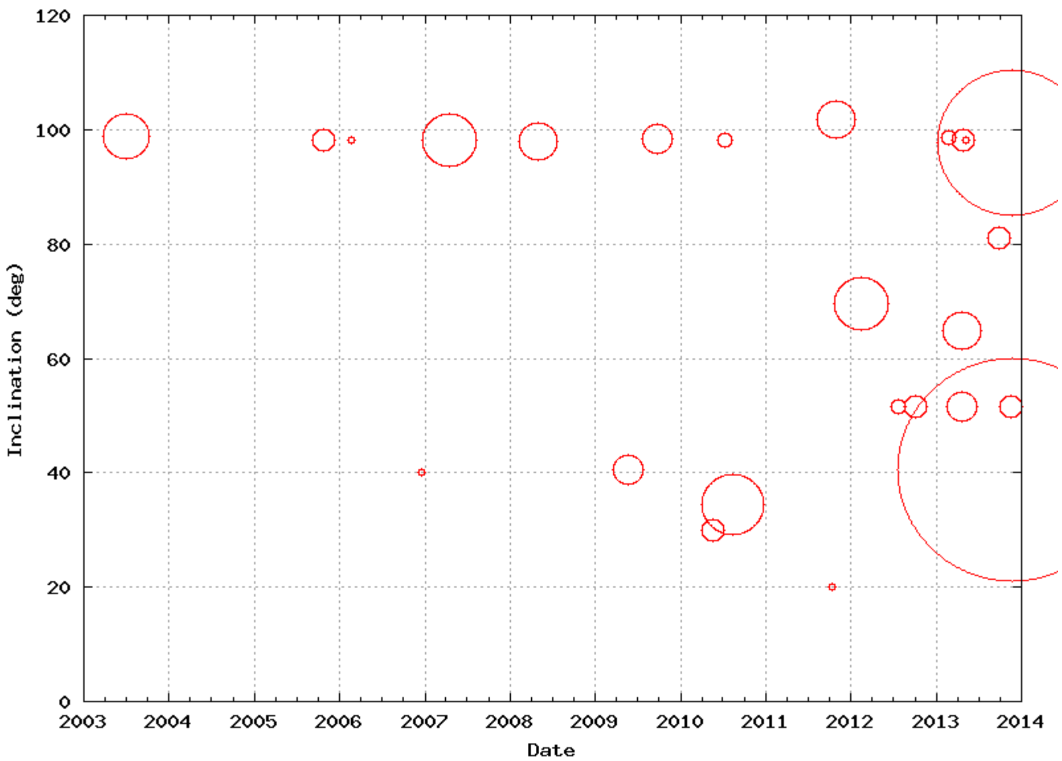

Figure 14. CubeSats launches with average launch inclination and number of satellites in launch (represented by circle size) during past eleven years. 0 degrees represent equator and 90 degrees a pole.

There are three orbit types used by CubeSats: near polar, International Space Station (near circular orbit with mean altitudes between 330 and 410 km and inclination a bit over 50 degrees) [25] and other.

Inclination near 90 degrees grants an opportunity to cover the whole Earth surface since satellite flies over both poles, crosswise over equator as Earth rotates.

Altitude around 600-700 kilometres and orbital inclination of 98 degrees is so-called Sun-synchronous orbit: precession around Earth is one full circle with a year, so the satellite is above the same specific point on Earth at the same time every day. This is important for remote sensing, for example to take pictures at same lighting conditions on every flyover.

Inclination less than 90 degrees means satellite is in an orbit less than polar and is in a prograde orbit – moving in an easterly directions and taking advantage of the Earth`s easterly rotation, so it takes less energy to send satellite into orbit.

The minimum energy needed to send a satellite to orbit would be an equatorial launch in a due easterly direction. Such launch will result satellite to have 0 degrees of inclination. A 90 degrees inclination means satellite crosses the Earth`s equator at a right angle and crosses both poles in one orbit. If inclination is greater than 90 degrees, the satellite is in less than a polar orbit – in retrograde orbit, where part of the satellite is in the opposite direction to Earth`s rotation. Figure 15 and 16 show respectively CubeSats mean motion relation to its eccentricity or inclination at launch.

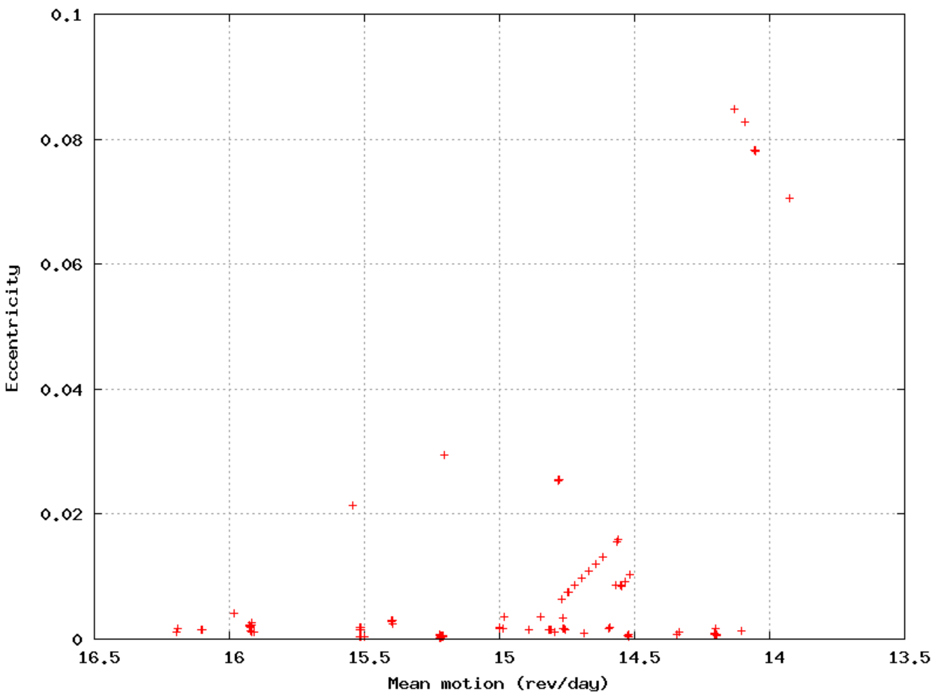

Figure 15. 131 CubeSats eccentricity relation to their mean motion after deployment. Note that eccentricity range is only 0-0.1 instead of 0-1.

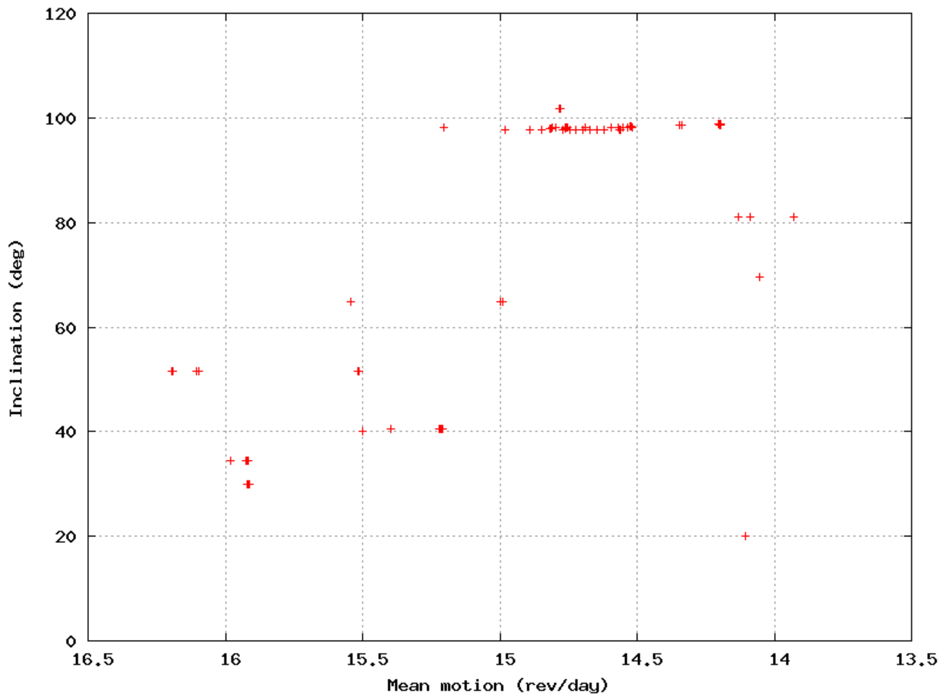

Figure 16. 131 CubeSats orbit inclination relation to their mean motion after deployment.

5.2 Topics and discussion

5.2.1 Two-Line Element creation frequency

All TLE-s published by North American Aerospace Defense Command have passed the checksum test plus format and range-checking tests [20]. Depending on the period, element sets are generated automatically as-needed basis. How frequently updates occur depends upon a number of factors such as the orbit type or maneuvering capability [26]. For example, a satellite in Low-Earth orbit would have its element sets updated several times a day because of somewhat unpredictable results of atmospheric drag as it varies its altitude and the maneuvering being performed. On the other hand, a satellite in a low-drag orbit and does not maneuver, won`t be updated as often. An object gets more frequent update if there is a prediction of a close approach with the operational payload. Special-interest objects, such as large object reentering the Earth`s atmosphere, usually get special treatment.

Overall TLE creation frequency have been cut approximately to half after Q4 2012, most probably due to assumed new radar implementation. There has been no official announcement about it. First six CubeSats annual average TLE creation frequency used to be around 500 in 2004-2011, but was around 261 in 2013. This is probably related to new radar increased measurement accuracy. TLE annual count for the satellite may be stable or vary remarkably.

Table 4. Statistics about TLE creation frequency for first six CubeSats.

Name	Unit	Mass (g)	TLE count Y2003*	TLE count Y2004	TLE count Y2005	TLE count Y2006	TLE count Y2007	TLE count Y2008
DTUSAT	1U	1000	181	499	509	488	496	501
CUTE-1	1U	1000	248	494	501	503	504	498
QUAKESAT	3U	4500	246	478	508	491	496	497
AUU CUBESAT	1U	1000	203	470	449	489	484	473
CANX 1	1U	1000	185	445	476	500	484	474
XI 4	1U	1000	181	491	495	492	487	496

Name	Unit	Mass (g)	TLE count Y2009	TLE count Y2010	TLE count Y2011	TLE count Y2012	TLE count Y2013**	Total:
DTUSAT	1U	1000	486	532	613	561	266	5132
CUTE-1	1U	1000	494	499	503	459	262	4965
QUAKESAT	3U	4500	501	536	638	574	263	5228
AUU CUBESAT	1U	1000	493	515	570	557	267	4970
CANX 1	1U	1000	485	487	502	452	256	4746
XI 4	1U	1000	501	502	500	462	253	4860

* Launch year, orbited half a year.

** TLE count is until 20th of Dec. Full year with new radar.

Out of first six CubeSats (all 1U-s with the mass of 1kg, except one 3U 4.5 kg), over 5200 TLE-s have been created throughout ten and a half years for 3U CubeSats and between 4750-5140 for 1U-s. Difference is too small to bring 3U size or shape forth.

If we compare results to another group that was launched in April 2008 with average launch altitude approximately 200 km lower (~630 km), with relatively low eccentricity and altitude decline, we get similar behaviour. Average annual TLE count for 1U-s (0.85 - 1.0 kg) in 2009-2011 is ~600 (highest 620) and ~300 in 2013. For 3U-s, the results was almost same: 2.2kg 3U average yearly TLE count was 573 from 2009 to 2011 and 306 in 2013. For 3U 3.5kg (with 4x 3U deployed solar arrays) numbers are 617 and 300 representatively.

There seems to be no noticeable difference between CubeSats size, shape and TLE creation frequency. Perhaps because of CubeSats variations are just too small for algorithms to make any difference. Other thing is that variation between very similar CubeSats are remarkable (see table 4), so TLE creation frequency is somewhat specific for every CubeSat.

If we compare results with a group (launched in May 2009 and decayed within 2.5 years) with very low average launch altitude (~450 km), but similarly with relatively low eccentricity, the result is surprising. No noticeable difference between 3U 5kg and 1U 1kg CubeSat as expected, but average TLE count is only ~550 (533 is average for satellites over 800 km) in 2010-2011. Therefore, satellites with average altitude near 400 km does not differ much from those above 800 km from algorithm perspective. Probably because CubeSats have relatively circular orbits and atmospheric drag is similarly predictable.

If we compare results with satellites with relatively high eccentricity, there is a big difference. For example, E-ST@R`s average altitude at launch was ~886 km and 22 months later it had lost 214 km of its average altitude. TLE total count for first 12 months is 903 and 929 for last 12 months. Again, seems altitude change have had no remarkable effect, but low perigee causes more unpredictable results of atmospheric drag and therefore element-sets gets updated more often.

5.2.2 Single CubeSat parameters analysis

DTUSAT is typical 1U CubeSats without remarkable deployable parts and since no connection has ever been made with it after launch. It has been orbiting naturally all the time, over ten and a half years by now. It has relatively circular orbit, average eccentricity value only 0.0009 (average trough all CubeSats was ~0.009 at launch) and inclination a bit over 98 degrees. During this period, its mean motion, eccentricity and inclination have changed minimally, only +0.0106 revolutions per day (-3.6 km of average altitude), -0.00003 of eccentricity and -0.0356 degrees in inclination. Figure 17-23 show DTUSAT orbital parameters change over its whole orbiting history.

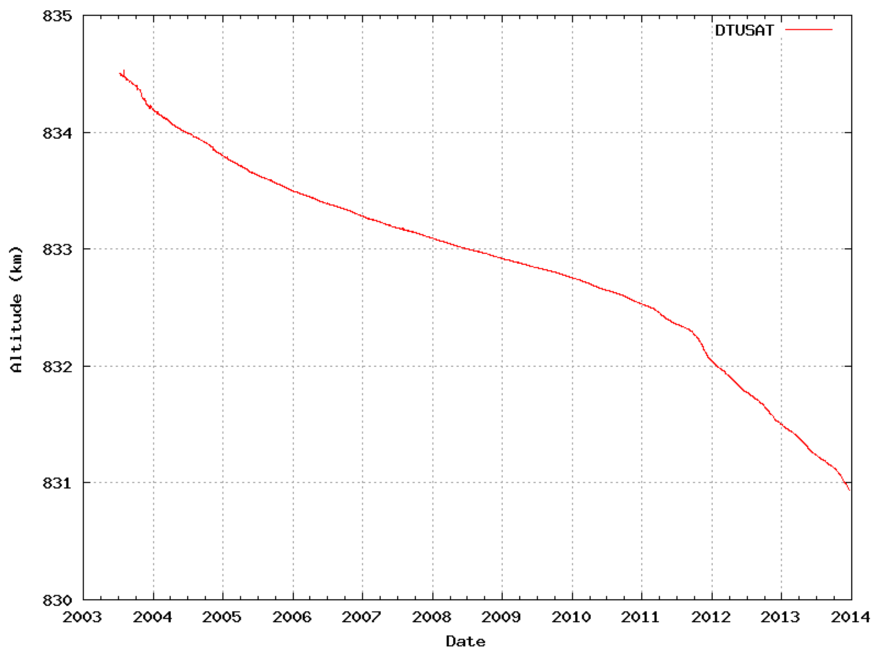

Figure 17. DTUSAT altitude change from launch during past ten and a half years.

According to simulation with NASA Debris Assessment Software (http://orbitaldebris.jsc.nasa.gov/mitigate/das.html), DTUSAT lifespan is expected to be near 300 years and its average altitude should be 832.5 km by year 2014, but it is a bit less than that – 831 km according to TLE-s. (see Appendix A).

Mean motion is in direct correlation to time and altitude for all CubeSats. Over time, mean motion steadily increases and altitude decreases. So time can be replaced with mean motion or altitude and graphs would look like very similar.

Since the DTUSAT orbit is comparatively high, it loses altitude very slowly and therefore its lifespan can really go beyond many tens of years. Altitude change for the currently observed time is not a straight line as one might expect due to disturbing forces. Slope is greater before 2004 and after 2010. Solar activity was relatively high before 2004 and after 2010, when the solar cycle started to move towards its highpoints again.

During whole DTUSAT orbital history, factors causing perturbations in DTUSAT parameters have all stayed practically the same. Except the Sun activity, which results in atmosphere density (extends higher when Sun is more active) and more photons will collide with a satellite due to increased solar radiation pressure. The effect from increased solar radiation pressure is very small, but effect from extended atmosphere is great since atmosphere density has the biggest effect on satellite motion in Low-Earth orbit.

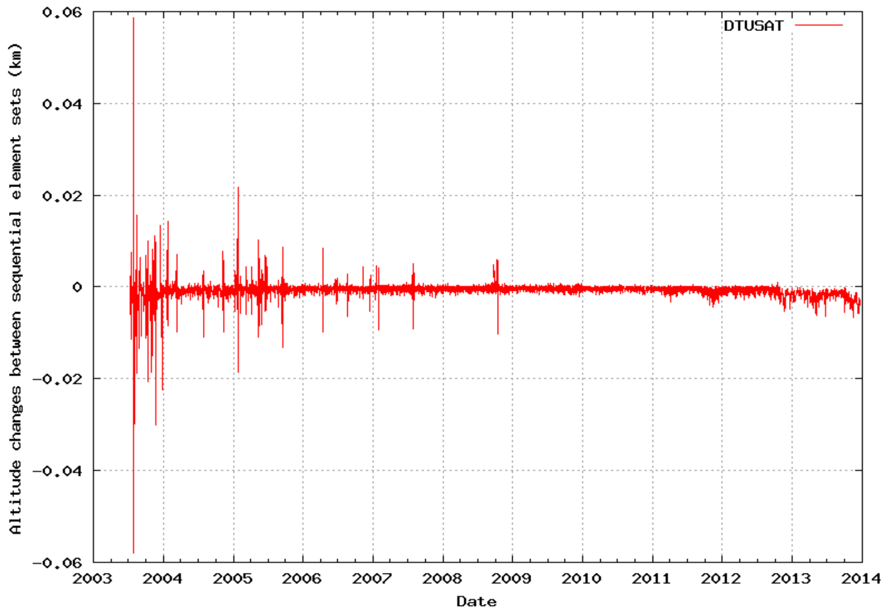

Figure 18. DTUSAT altitude changes between sequential TLE-s from launch during past ten and half years. Peaks with roughly same amplitude, next to each other and in opposite direction is most likely balancing effect. During some periods: before 2004 and after 2010, the satellite has lost more altitude on daily basis as on other periods.

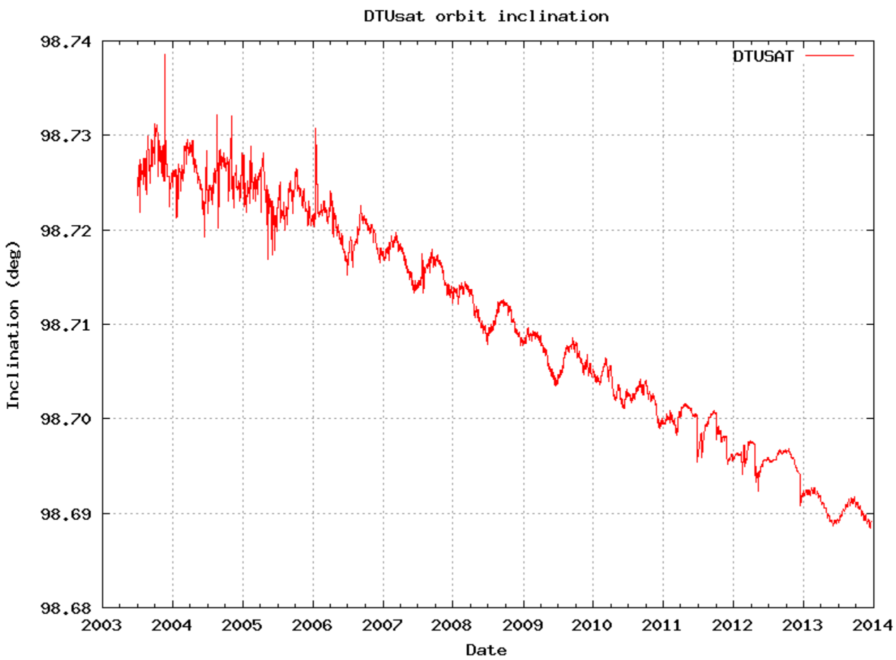

Figure 19. DTUSAT Inclination from launch during past ten and a half years. Inclination is slowly decreasing.

Periodical movement is most likely caused by combination of forces that affect satellite orbit (e. g. variations in Earth's gravity).

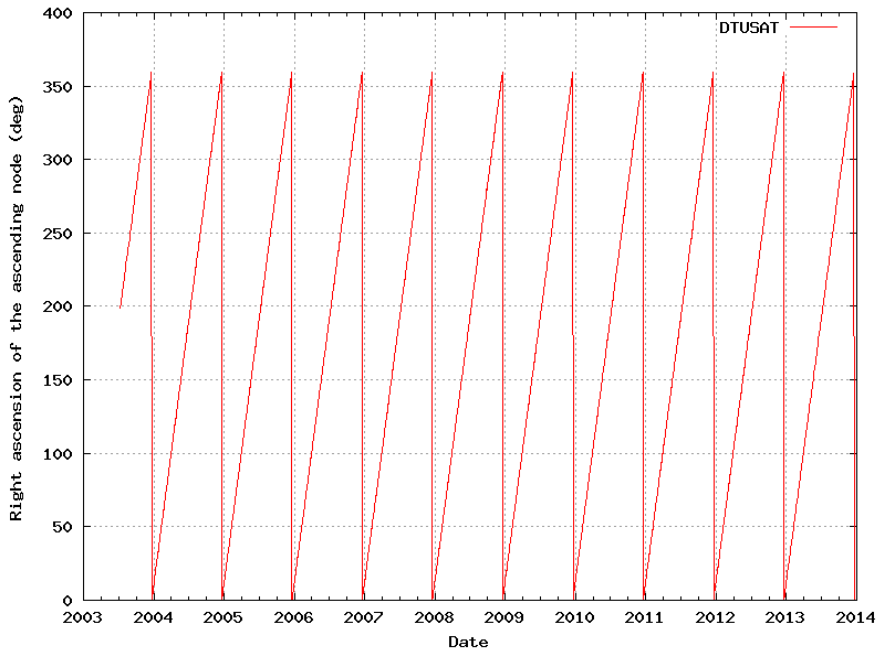

Figure 20. DTUSAT right ascension of the ascending node during past ten and a half years. After a year, when Earth has made a full revolution around the Sun, this parameter has turned 360 degrees and then starts over from zero (start of astronomical year is zero point).

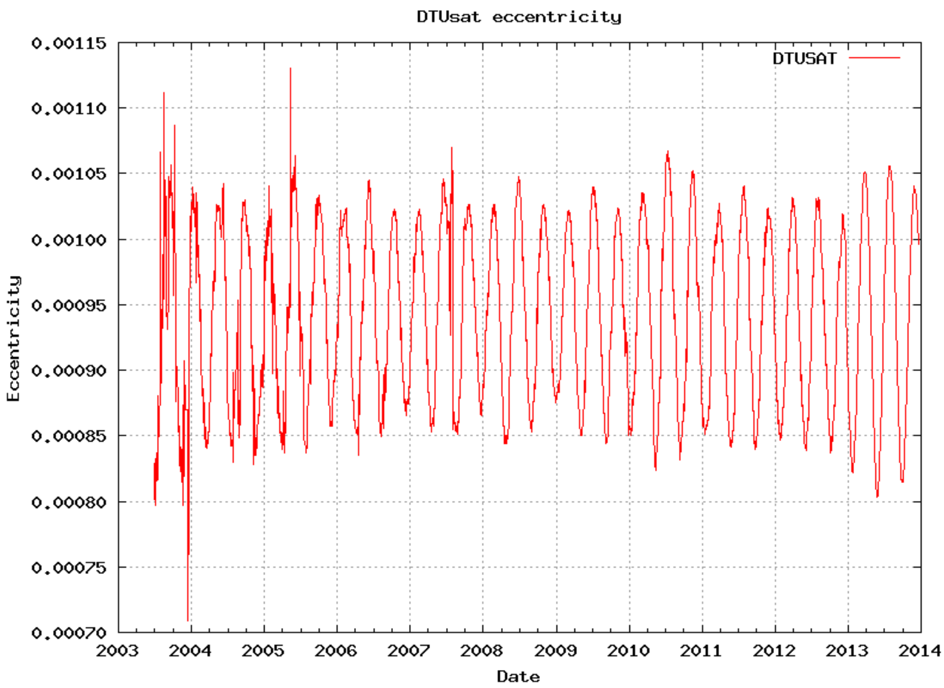

Figure 21. DTUSAT eccentricity from launch during past ten and a half years. Mean motion changed minimally during this period. It makes roughly four fluctuations in a year.

Eccentricity correlation to mean motion should be a straight line - over time, eccentricity consistently reduces and mean motion raises. However, if satellite has very low eccentricity, then on circular orbit Earth's oblateness may cause fluctuation in eccentricity.

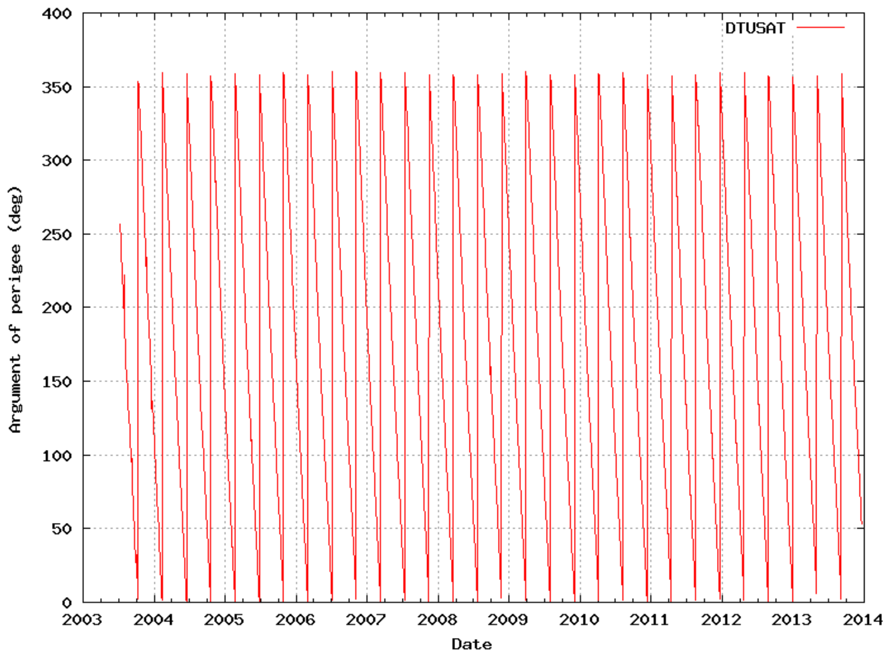

Figure 22. DTUSAT argument of perigee from launch during past ten years. It makes approximately three full rounds in a year.

Argument of perigee shows perigee point location in relation to moving Earth. If satellite had a stable elliptical orbit (high eccentricity), the argument of perigee would stand still. Instead, it has a non-perfect and unstable circular orbit (very low eccentricity). As eccentricity fluctuates, its argument of perigee rotates.

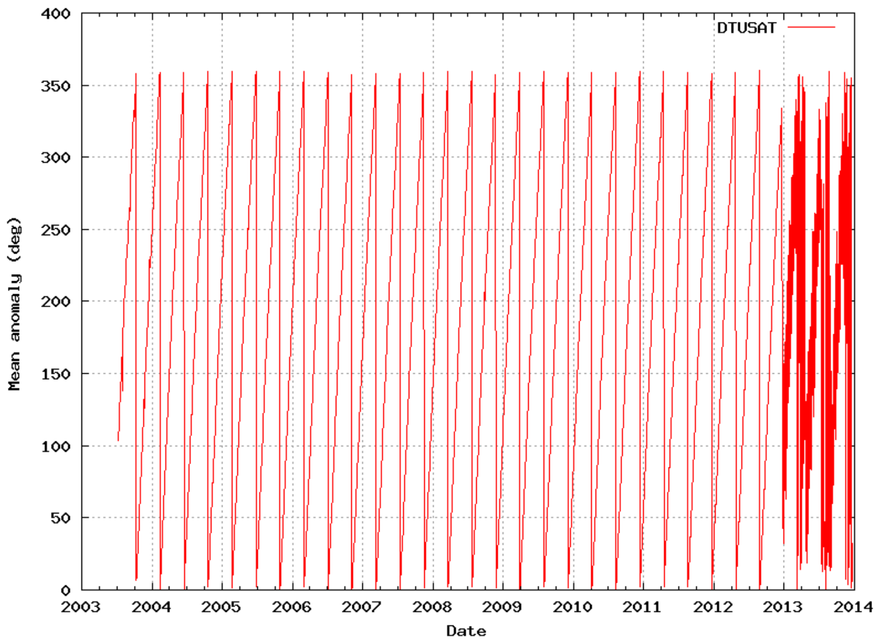

Figure 23. DTUSAT mean anomaly from launch during past ten and a half years. Mean anomaly is measured in reference to perigee, so mean anomaly directly depends on argument of perigee.

Since Q4 2012, probably new radar has been taken into use, which returns measurements with higher rate and therefore affects graph visual appearance. New radar effect can also be seen on figures 29, 30, 33, 34, 35. Figure 24 and 25 show how CubeSats altitude change in low orbits.

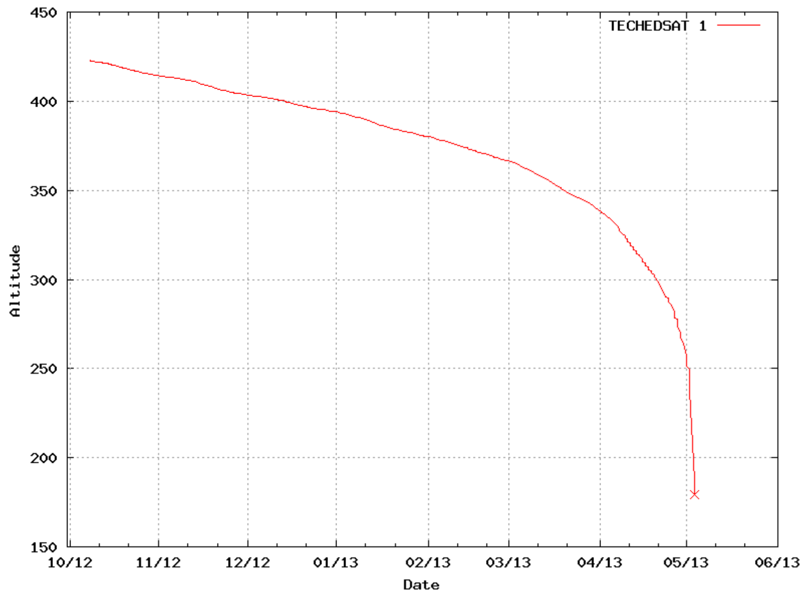

Figure 24. An example 1U CubeSat altitude change from launch until decay on low orbit. As expected (see also figure 5), on lower altitudes satellite lost altitude successively faster until it decayed below 200 km.

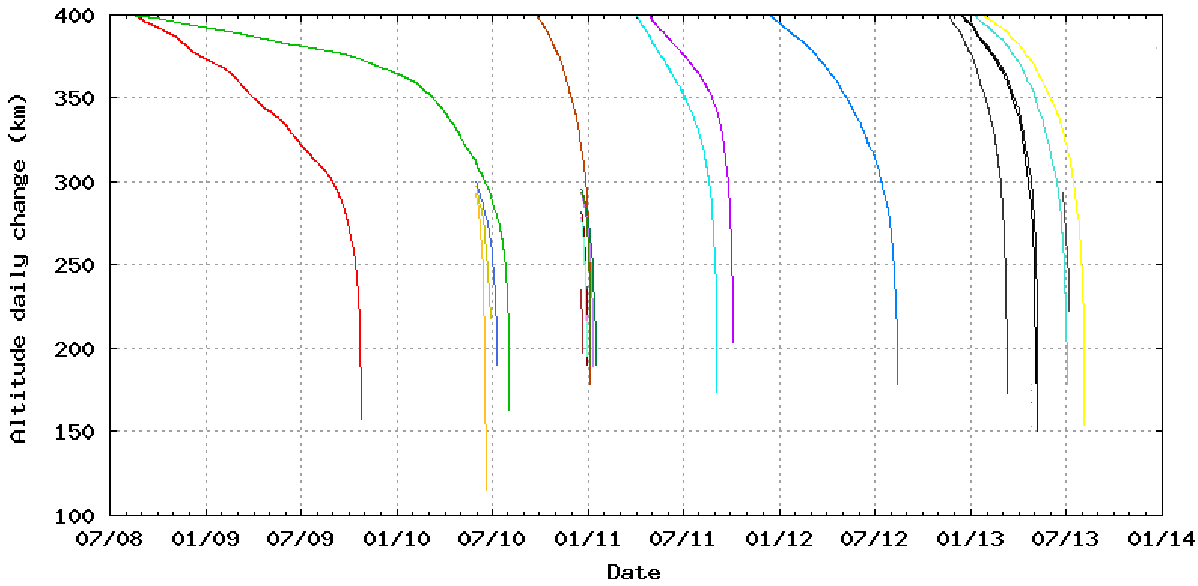

Figure 25. All CubeSats that have ever been below 400 km with their average altitudes until decay. On average, CubeSats decay within hours when they fall below 200 km.

5.2.3 Eccentricity relation to mean motion.

Figure 21 showed how eccentricity was fluctuating when CubeSat mean motion was practically same and eccentricity value was very low. How eccentricity, argument of perigee and mean anomaly correlate to mean motion if CubeSat has relatively high eccentricity?

Figure 26-30 show how satellite mean motion is in relation to its eccentricity, argument of perigee, mean anomaly or inclination in case satellite eccentricity is >0.05.

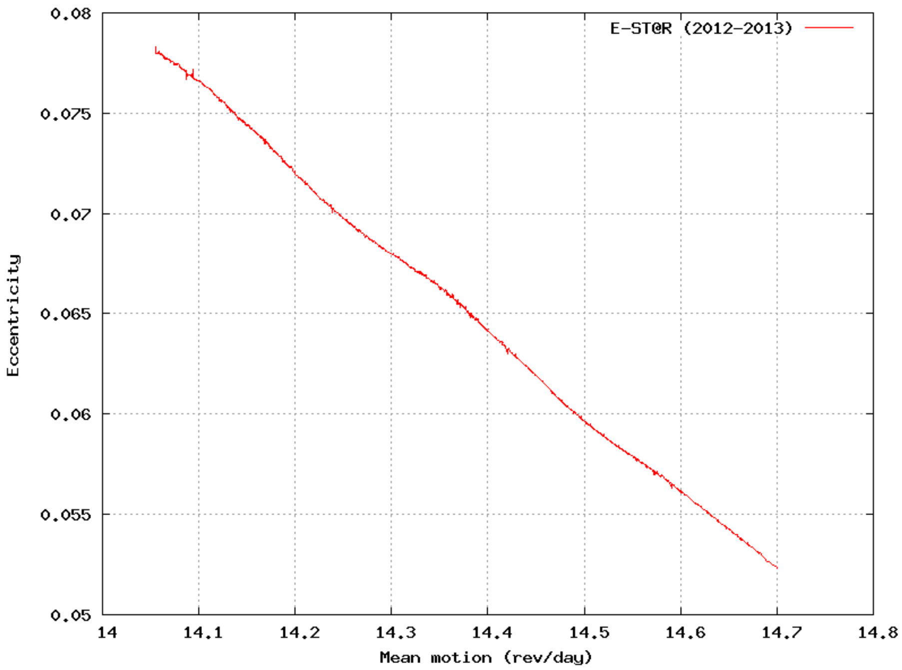

Figure 26. An example 1U CubeSat eccentricity correlation to mean motion (high eccentricity). When mean motion value changes drastically (E-ST@R launch altitude was 886 km and it declined by 214 km during 22 months), eccentricity declines remarkably and is in linear correlation to mean motion or time.

If a CubeSat has elliptical orbit, air density is highest near perigee and lowest near apogee. Every time satellite passes perigee, it decelerates and as a result - decreases apogee. So by nature, satellite reduces its eccentricity (and raises mean motion) over time.

Two satellites with same mean motion value but with different eccentricity have the same average altitude. Satellite with higher eccentricity has its perigee closer to the Earth and air density may be substantially different there. Therefore, a satellite with greater eccentricity loses its average altitude faster.

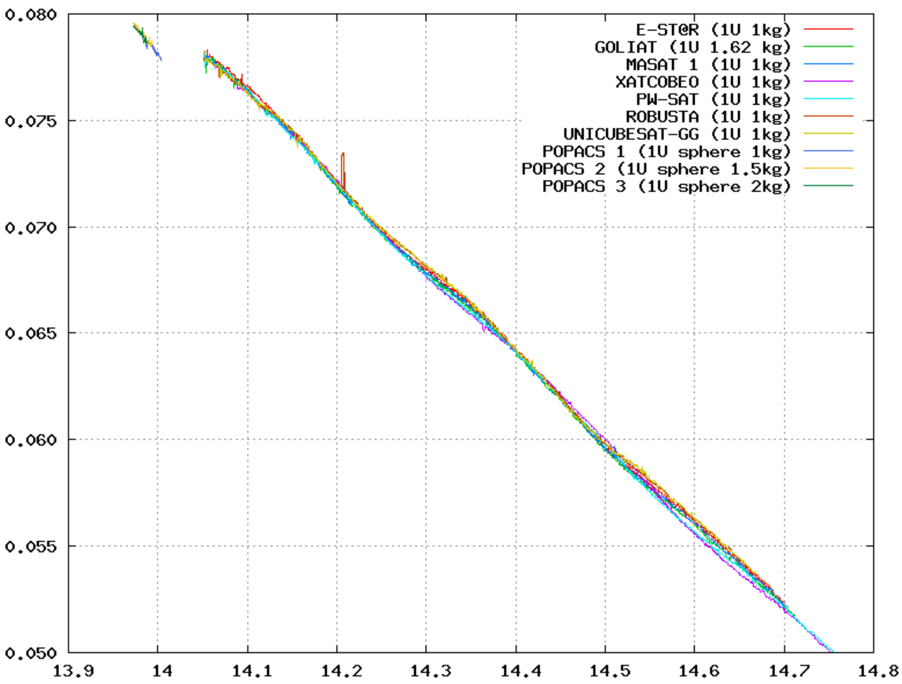

Figure 27. All CubeSats with eccentricity values over 0.05 and their eccentricity correlation to mean motion.

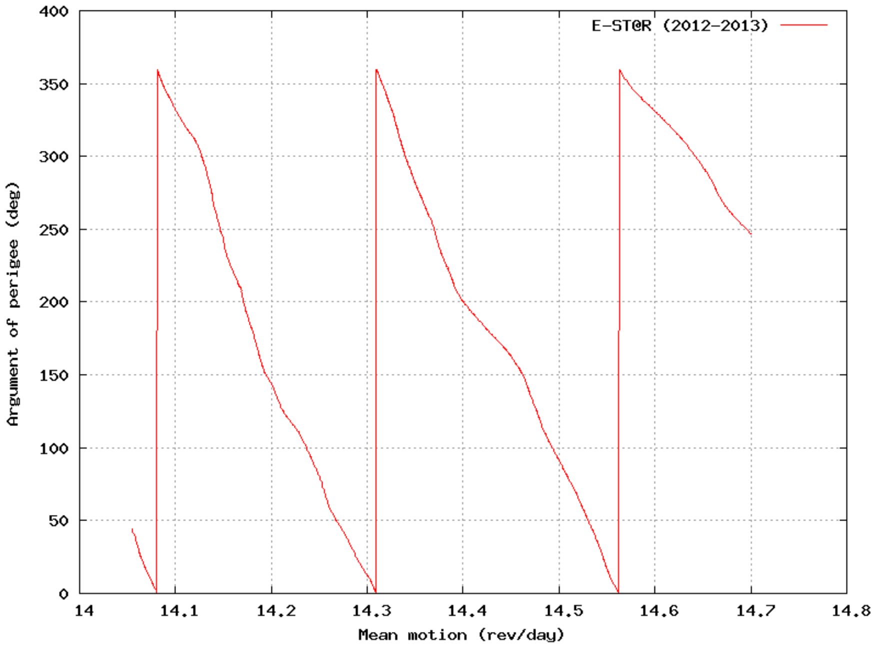

Figure 28. An example 1U CubeSat argument of perigee correlation to mean motion (high eccentricity). Relation is similar to figure 22.

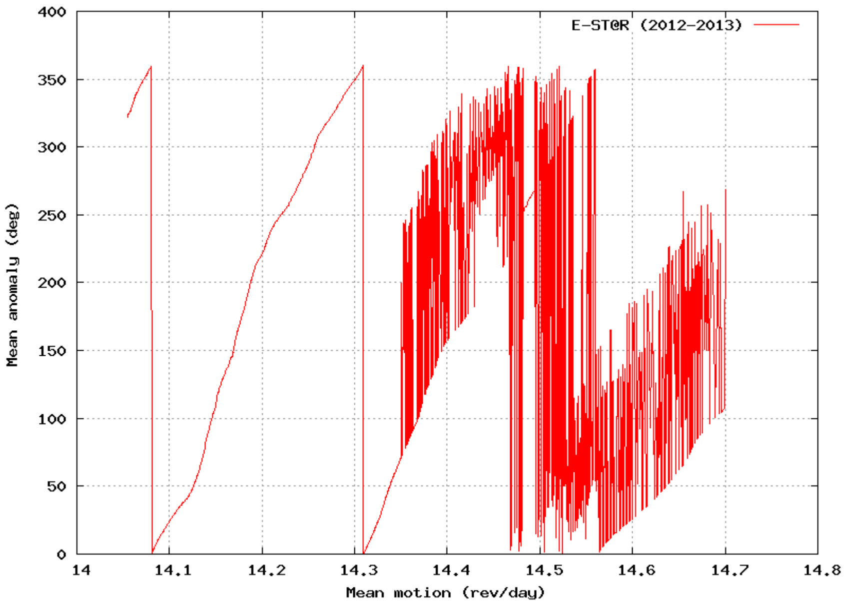

Figure 29. An example 1U CubeSat mean anomaly correlation to mean motion (high eccentricity). Relation is similar to figure 23.

5.2.4 Inclination correlation to mean motion

Figure 19 showed how inclination close to 98 degrees was fluctuating downward slowly when CubeSat mean motion and eccentricity (very low) values practically did not change over time. How very low inclination (and near average eccentricity) affects CubeSat mean motion?

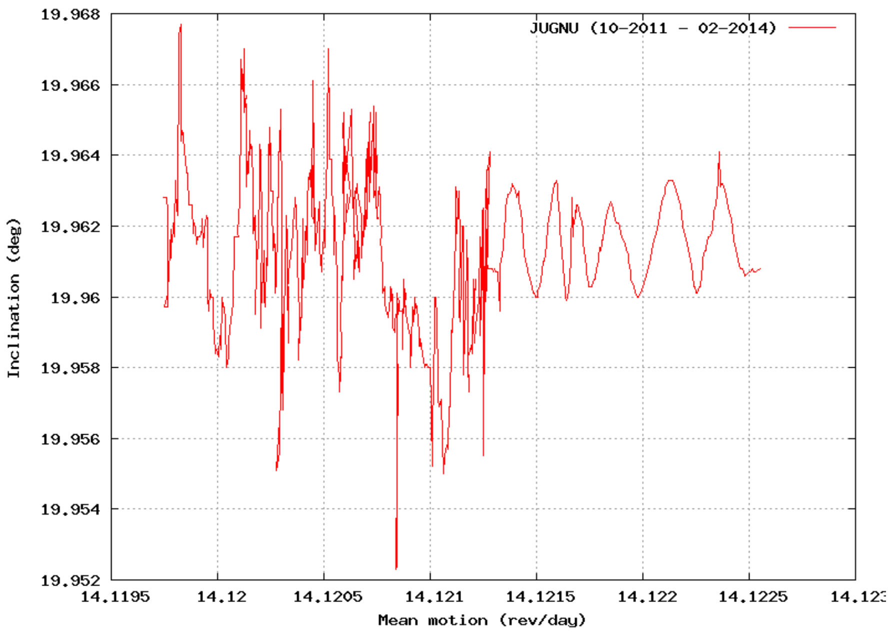

Figure 30. An example 3U CubeSat inclination correlation to mean motion. JUGNU is 3U CubeSat with launch altitude over 800 km, near average eccentricity value of 0.0013, but lowest inclination (19.9x degrees) among all CubeSats. During the period over 2 years and 4 months, it has lost 6.9 km in average altitude and increased mean motion by 0.02 revolutions per day.

Inclination has stayed same instead of tiny diminishing. Probably because timeframe is too small. Lower inclination seems not to affect average mean motion considerably, but has minor effect. DTUSAT (see figure 17) lost a bit over 1 km of its average altitude whereas JUGNU lost a bit less than 7 km on same period, probably due to higher gravitational force.

By nature, satellites should slowly reduce their inclination towards zero. When booster rocket set it to orbit on higher inclination, then part of deceleration force goes to inclination reduction so in very far future inclination could become zero (in reality they decay long before).

Seems with the new radar appearance, CubeSats orbital parameters behaviour from Q4 2012 and onwards can be claimed with greater certainty because of improved precision. As can be seen from Figures 29, 30, 33, 34, 35, error amplitude has greatly improved.

5.2.6 Multiple CubeSat analysis

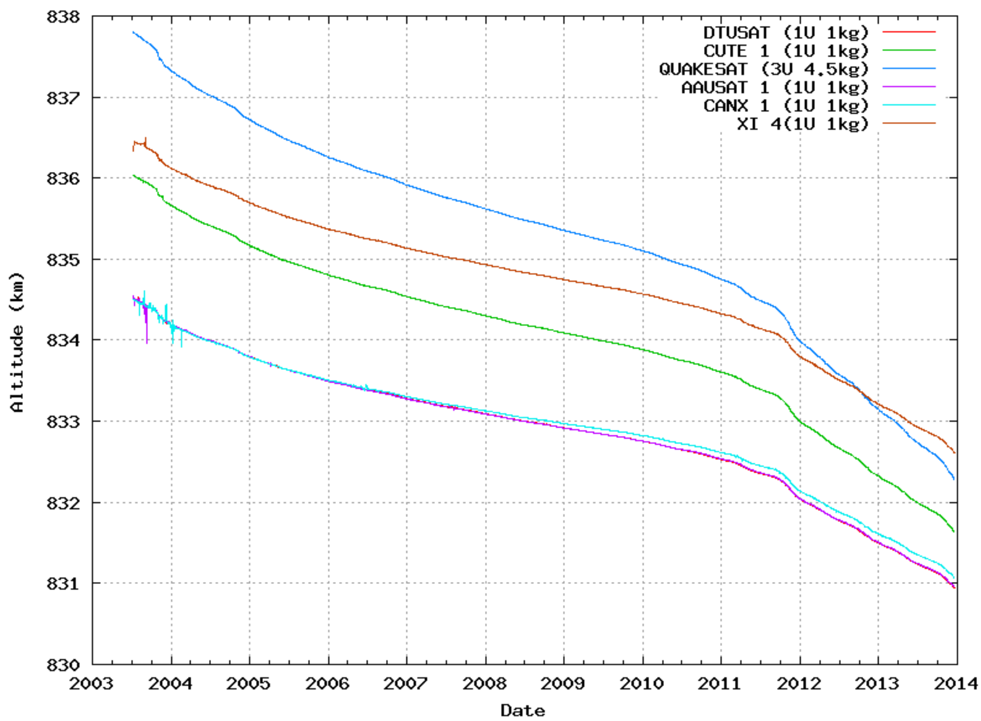

Figure 31. First six CubeSats and their altitude changes throughout orbital history. Note that AAUSAT 1 and DTUSAT coincide.

As can be seen from figure 31, CubeSats from same launch seems to act similarly. The way that a particular CubeSat is deployed out of the launch adapter seems to have small difference in starting altitude. Interesting is the fact that two satellites behaviour coincide over so long period.

In general, either way, if all CubeSats have their 1U side towards direction of motion or they rotate, a satellite with heavier mass should lose its altitude a bit slower. If they move 1U side in front, satellite with bigger mass have greater inertia with same aerodynamic resistance. If they rotate, we have almost same situation - for 3U-s mass is greater and aerodynamic resistance a little bit bigger higher.

On this particular case, QUAKESAT can be considered as special. It has four outer solar arrays (~10x30 cm) and a boom, which forces it to move in fixed position in relation to Earth. Therefore, instead of losing its altitude slower, it loses it faster. Between 2005 and 2010, when solar activity was comparatively low, QUAKESAT did not lose its average altitude slower. During the periods when Sun activity was on its high points, QUAKESAT was a lot more affected from raised atmosphere density.

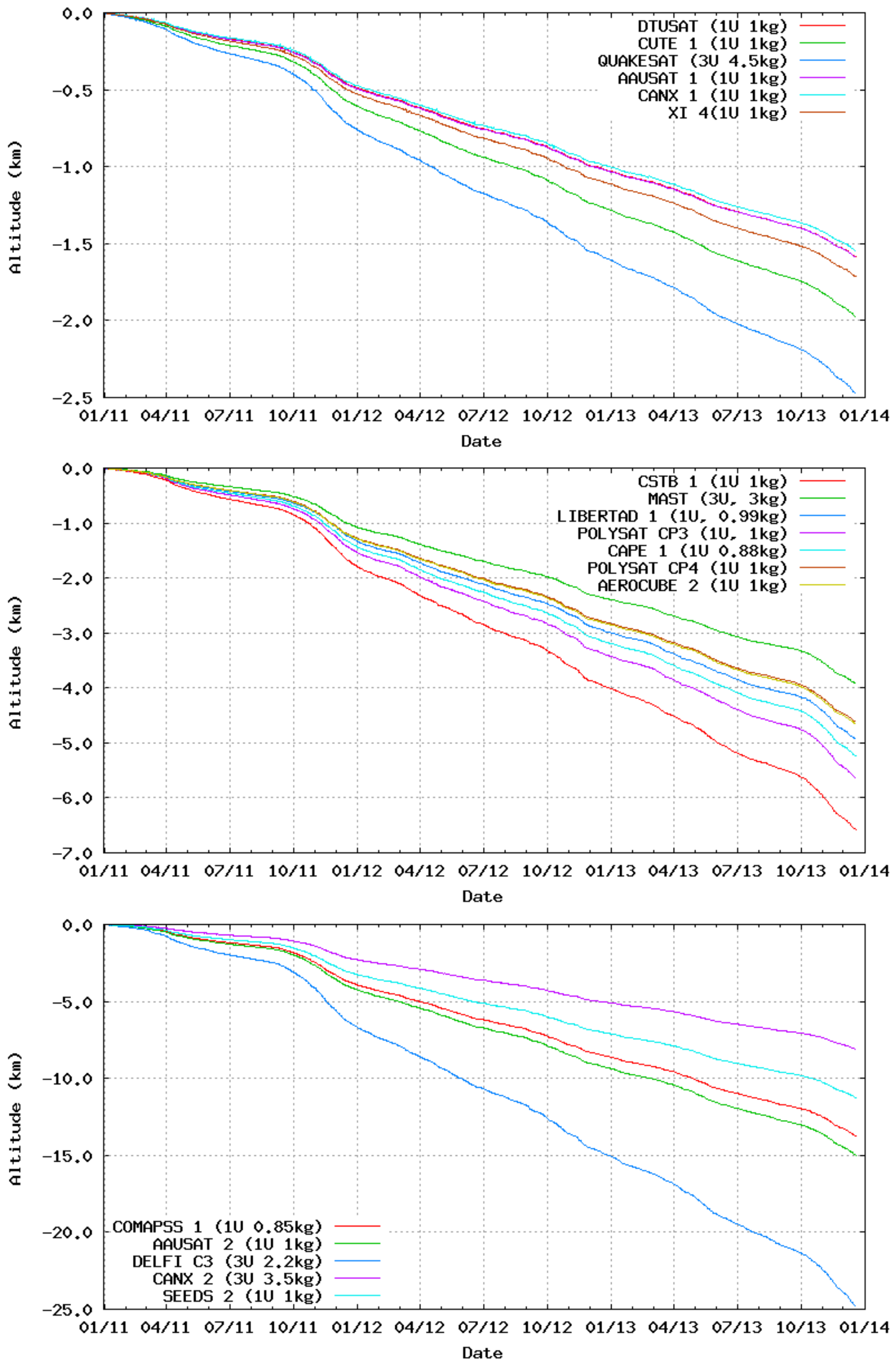

Figure 32. Three launches before year 2012: 30/06/2003, 17/4/2007, 28/04/2008 with average launch altitudes 833, 724, 633 km respectively.

As can be seen from figures 32-35, CubeSats from different launches behave similarly. On figure 32, all these launches have similar non-linear behaviour and amplitude depends on mean orbiting altitude. By computing perspective, 1U and 3U mass relation to its volume is same. However, satellite shape may have small effect: DELFI C3, similarly to QUAKESAT, has several antennas and four solar arrays (~10x30 cm). CUTE 1 has also a side panel. These satellites have lost altitude a bit more.

CANX 1, AAUSAT 1, CSTB 1, MAST, CANX 2, SEEDS 2 are without remarkable outer parts. MAST and CANX 2 are both losing its altitude a bit slower as expected. They have greater mass and no remarkable deployed parts. Considering average planned mission length, CubeSat size, shape or mass is not very important (if the satellite is not going to orbit on very low altitudes and/or with high eccentricity). One thing to remember is that actual mission lengths tend to prolong into years.

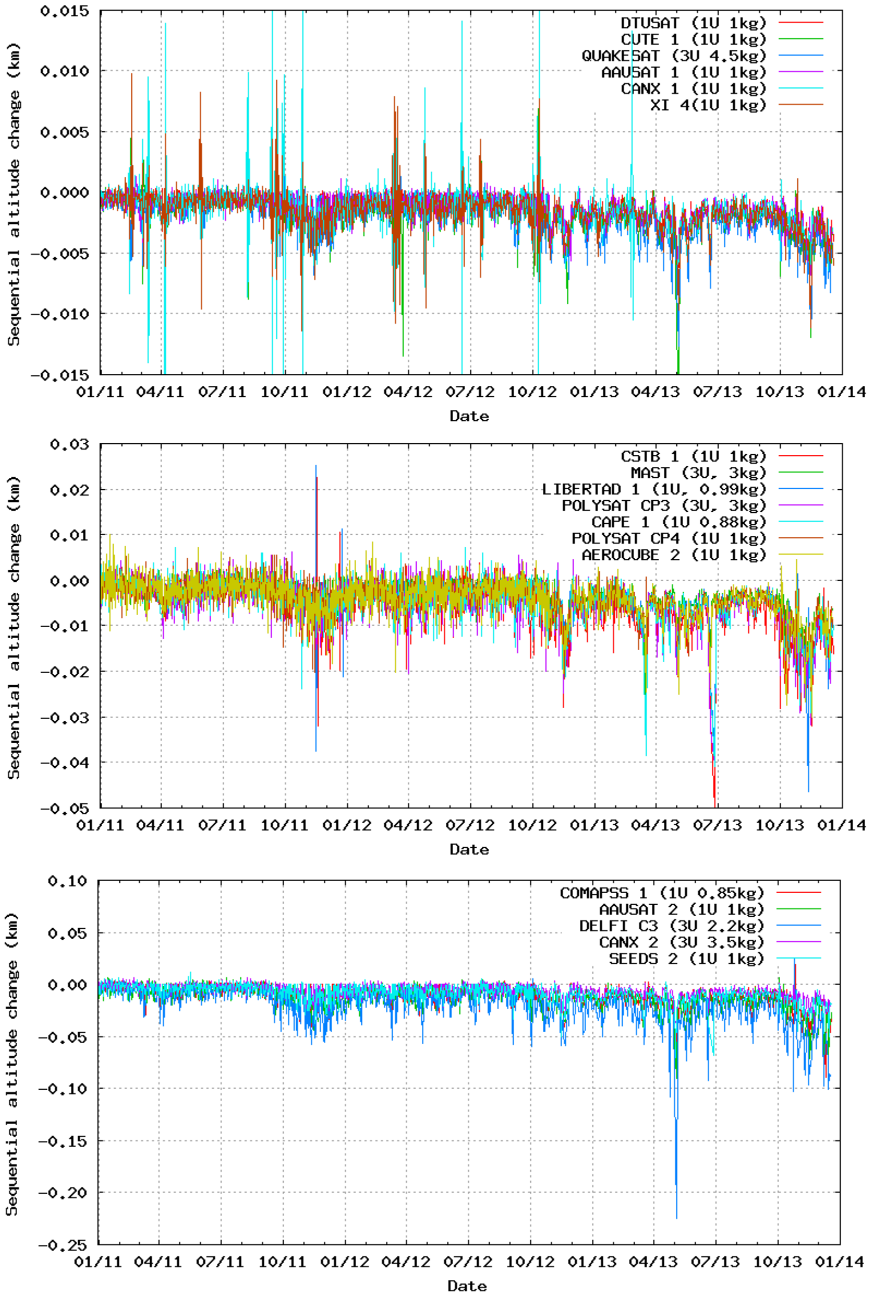

Figure 33. Sequential altitude changes from TLE-s. Three launches before year 2012: 30/06/2003, 17/4/2007, 28/04/2008 with average launch altitudes 833, 724, 633 km respectively.

All these launches have similar behaviour, but again amplitude depends on mean orbiting altitude.

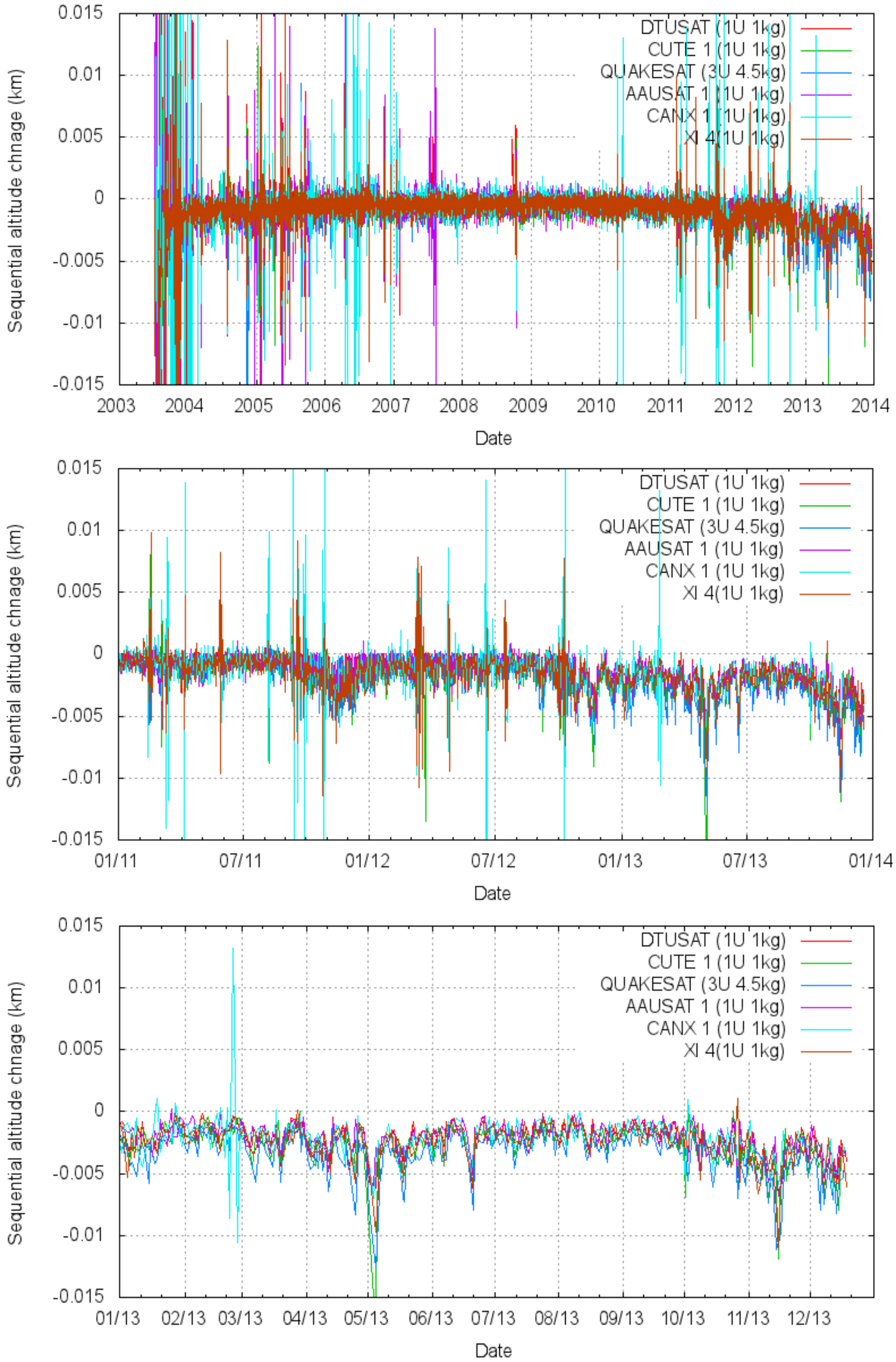

Figure 34. First CubeSats in orbit and their behaviour over time. Note that time scale is varying.

In shorter and longer time frame, there are similarities. Uppermost graph is almost with the length of full 11-year solar cycle. On all graphs, last third is much more restless and the pattern is somewhat similar. There is also a similar greater spike approximately after one third on time scale of last two graphs.

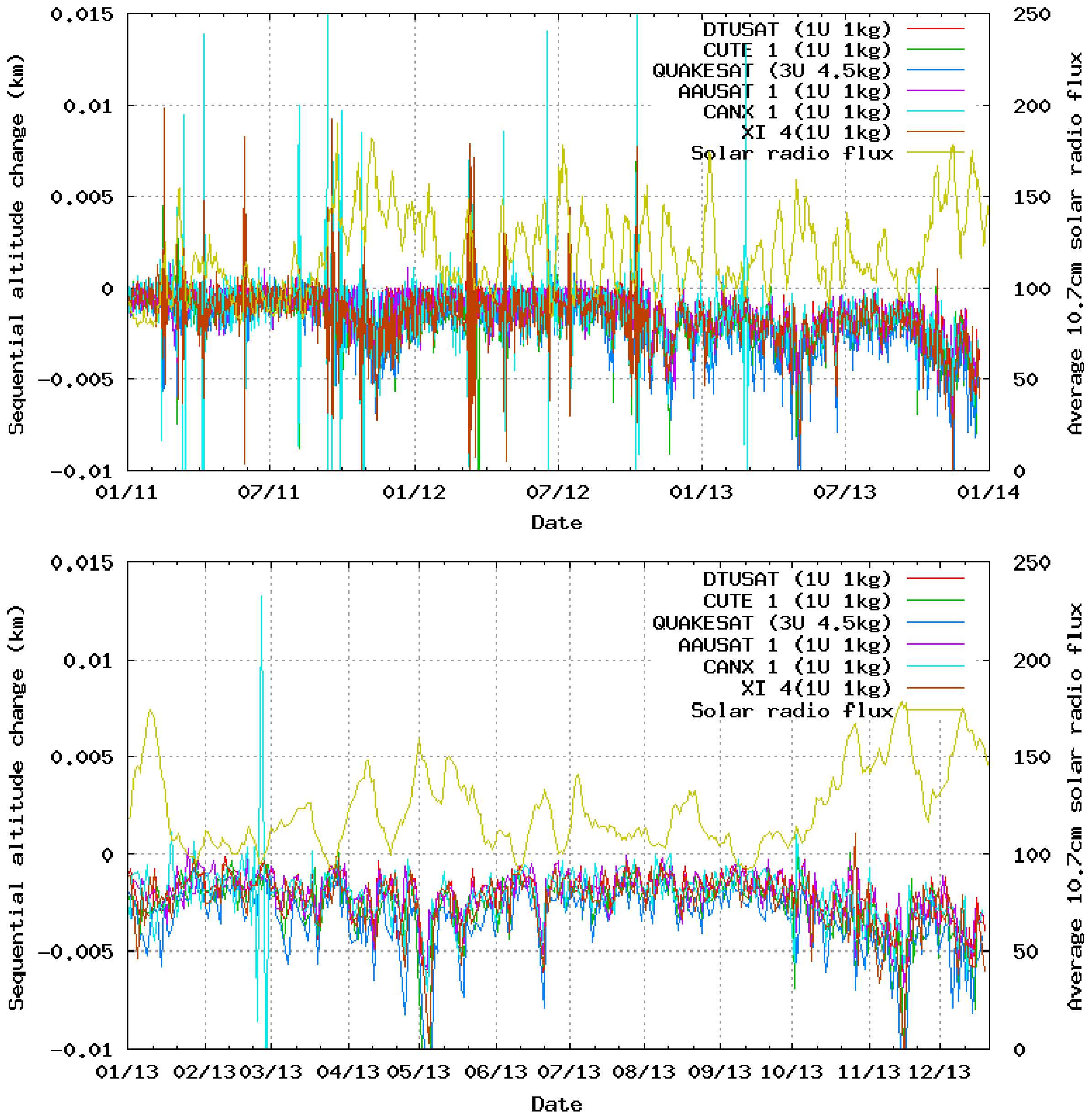

Figure 35. First CubeSats in orbit and their behaviour over time in relation to solar activity. Information about solar activity was taken from http://www.swpc.noaa.gov/Data/index.html.

There seems to be a link between satellite altitude change (lifetime) and solar activity. Current 11-year solar cycle is not as intensive as previous cycles and in the future, cycles may reach remarkably higher flux values and thus its effect on CubeSats is going to be greater. This should especially kept in mind if satellite is going to be placed on high eccentric and/or low altitude orbit.

Figures 36 and 37 show how much on average CubeSats have lost their altitude in daily basis in 2010, when solar activity was relatively small.

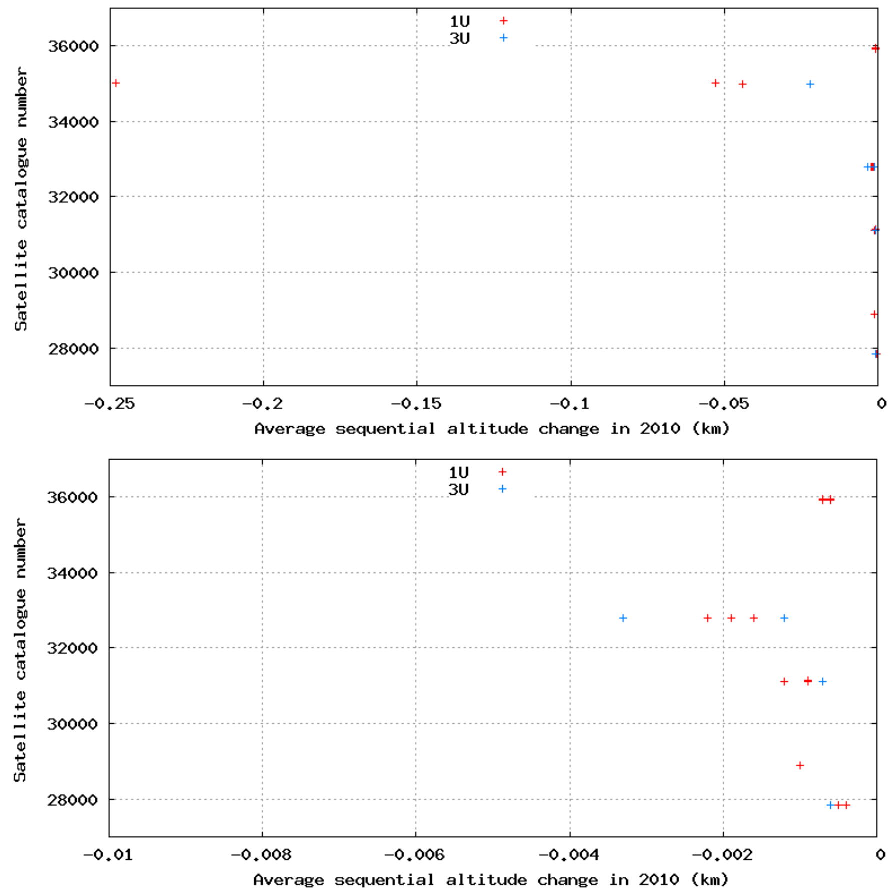

Figure 36. All CubeSats that were orbiting full astronomical year of 2010 with their average sequential altitude changes throughout the year.

Satellites from same launches (total 6 groups) tend to behave very similarly. CubeSats with highest altitude change have inclination of ~40 degrees (all others ~98 degrees) and near average eccentricity. Similarly to figure 30, lower inclination do have an effect on average altitude, but it is minor.

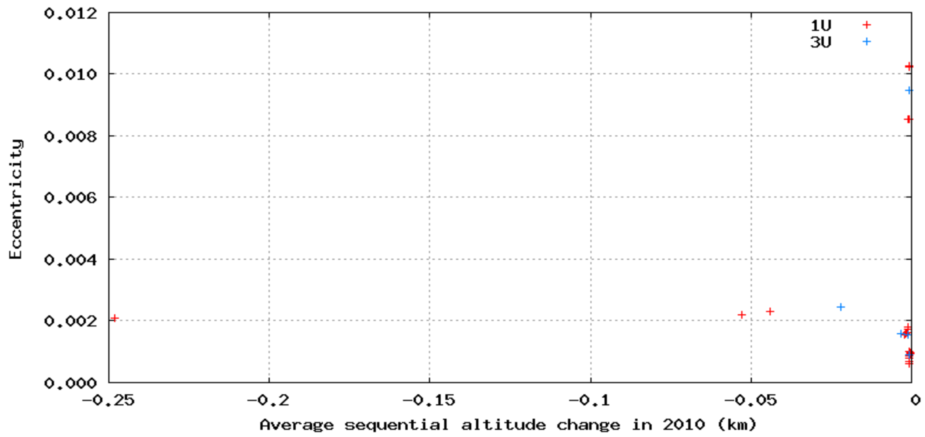

Figure 37. All CubeSats that were orbiting full astronomical year of 2010 with their average sequential altitude changes throughout the year.

CubeSats average launch eccentricity was ~0.010. Seems current eccentricity values near average or below does not affect satellite altitude much, but it does (and a lot) if eccentricity values goes near 0.05 and over (see Figure 26, 27). Launch 13.02.2012 average eccentricity was ~0.08 and 22 months later a bit less than 0.05. CubeSats with less than 0.03 eccentricities lose their eccentric value remarkably slower.

6 Conclusion

This work investigated changes in CubeSats orbital parameters to find out how these parameters change over time, what relations exist between them, how they affect satellite lifetime and how various disturbing forces influence orbits. CubeSat standard has become very popular and interest increases. From mission planning and satellite perspective, it is vital to understand how satellite lifetime is related to its mass, size, shape, orbital parameters and how big effect various disturbing forces have on satellites orbits.

Publicly available historical orbital parameters in TLE format were used as input. This work lasted approximately a year and has been very interesting. It consisted among other things a data collection, processing, quality assessment, work with database and data analyses. Historical orbital data for 131 CubeSats as end of 2013 was used in analysis.

The most important results of the work are:

- CubeSats average altitudes does not decline smoothly in time, but are greatly affected by solar activity. All CubeSats have similar spikes in their average altitude lines, but the amplitude depends slightly on satellite mass, size, shape and greatly from current altitude. The lower the current altitude, the bigger effect. So satellites lifetime depends greatly from launch time.
- Concerning estimated and actual mission lengths, CubeSat mass, size and shape is not important unless satellite is sent to very low orbits (~ 400 km or lower).
- Satellites with greater eccentricity (>0.05) lose their average altitude very fast irrespective of their mass, size or shape. Eccentricity <0.03 means satellites loses altitude a lot slower. Therefore, planned eccentricity must be considered carefully.
- The effect from Earth`s mass heterogeneous distribution is relatively modest. Considering average actual or planned mission lengths, inclination is not important.

Researches have been concentrated on satellites location and propagation prediction so far, but not on orbit own defined parameters. In this work, first time, orbit defined parameters were in focus. This study could go on from here by choosing satellites with very accurately measured orbital parameters (e. g. GOCE, ENVISAT, ERS-2, ERS-1 from European Space Agency) and assess TLE orbit accuracy.

7 Kokkuvõte

Kuupsatelliitide orbitaalparameetrites toimuvate muutuste analüüs

Indrek Ploom

Kokkuvõte

Käesolev töö uurib variatsioone kuupsatelliitide orbitaalparameetrites, et selgitada kuidas orbitaalparameetrid ajas muutuvad, millised omavahelised seosed on parameetrite vahel, kuidas nad mõjutavad satelliidi eluiga ning kuidas erinevad häirivad jõud mõjutavad orbiite.

Töö tulemused on suunatud rahvusvahelisele/ülemaailmsele nanosatelliitide arendajate huvigruppidele. Kuupsatelliidi standard on osutunud populaarseks ja huvi järjest kasvab. Missiooni planeerimise ja satelliidi disaini seisukohast on väga oluline teada seda, kuidas satelliidi eluiga on seotud tema massi, suuruse, kuju, orbitaalparameetritega ning kuivõrd suure mõjuga erinevad häirivad jõud orbiidis hälbeid tekitavad.

Kuupsatelliitide orbitaalparameetrite käitumist ei ole varem uuritud, küll aga on varasemalt uuritud üksikuid suuremaid satelliite väga täpselt mõõdetud parameetritega. Käesolev töö ei tee ise mõõtmisi vaid kasutab massiliselt ära vähem täpsemaid ja vabalt kättesaadavaid ajaloolisi orbiidi andmeid *Two-Line Element (TLE)* formaadis, mis oma esialgses lähenduses on mõeldud raadioamatööridele ja hobi-astronoomidele. *TLE*-s sisalduvate värskete hetke-orbiidiandmete ja erinevates tarkvarades rakendatud liikumismudelitega on võimalik ennustada objekti liikumist ning suunata oma antennid suhtluseks õigeaegselt vajalikus suunas. Töö uurib ka *TLE*-de sobivust orbitaalkäitumise analüüsiks ja tulemuste kasutatavust tegeliku missiooni planeerimise ja disaini juures.

Töö käik on olnud väga huvitav ja kestnud umbes aasta, See on hõlmanud muuhulgas andmete kogumist, töötlust, kvaliteedi hindamist. Tööd (MySQL) andmebaasiga – andmete jaoks sobiliku struktuuri loomist, andmetega täitmist ja sobival kujul andmete esitamist. Andmeanalüüsi, mille osana kasutati ka GNUPlot nimelist tarkvara orbitaalparameetritele graafilise vormi andmiseks.

Kokku koguti kõigi 131 kuupsatelliidi ajaloolised orbiidiparameetrid, koguarvus üle 141 tuhande, mis olid orbiidile saadetud ennem aastat 2014 ja kõrgusevahemikus natuke alla 900 km kuni ära põlemiseni madalatel kõrgustel. Lisaks orbiidiparameetritele koguti kokku ka satelliitide olulised tunnused, alates massist ja suurusest kuni informatsioonini missiooni olukorra üle, mis mõjutavad satelliidi orbiidi kahanemise kiirust.

Töö olulisemad tulemused ja järeldused on:

- Kuupsatelliitide keskmine kõrgus ei vähene ajas sujuvalt sirgena vaid on Päikese aktiivsusest tugevalt mõjutatud. Otsene Päikese kiirgusrõhk on väheoluline faktor, küll aga omab tõusnud Päikese aktiivsus suurt ülekantud mõju tänu paisunud ja kõrgemale ulatuvale atmosfäärile. Kõikidel satelliitidel on sarnased jõnksud keskmise kõrguse sirges, mis teatud määral sõltuvad satelliidi massist ja kujust ja suuremal määral hetke kõrgusest: mida madalamala on satelliit, seda suurem on amplituud. Seetõttu madalamale orbiidile saadetud satelliidi eluiga sõltub märgatavalt sellest, millises punktis ollakse Päikese 11-aastases tsüklis. Arvestada tuleb ka seda, et tsükliti on Päikse aktiivsuse suurus erinev ning et käesolev tsükkel on olnud siiani väiksema aktiivsusega kui paar eelnevat.
- Kui arvestada keskmist planeeritud (ja ka tegelikku) missiooni pikkust, siis üldjuhul ei ole satelliidi mass, suurus ja kuju olulised. Need faktorid muutuvad aga oluliseks väga madalate (~400 km või alla) orbiitide puhul.
- Ekstsentrilisusesse tuleb suhtuda väga tõsiselt ja seda ka kõige kõrgemate keskmise stardikõrgusega satelliitide puhul sõltumata massist, kujust või suurusest. Ekstsentrilisus juba üle 0.05 põhjustab väga suurt keskmise kõrguse kadu ajas. Satelliidid ekstsentrilisusega 0.03 või vähem kaotavad kõrgust märgatavalt vähem.
- Maa massi ebaühtlasest jaotusest tingitud mõju on tagasihoidlik. Arvestades keskmist planeeritud või tegelikku missiooni pikkust, siis orbiidi tasandi kalde nurk on väheoluline.

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Appensix A: Orbital simulation

Figures 38 and 39 show a simulation about “DTUSAT” expected altitude decline.

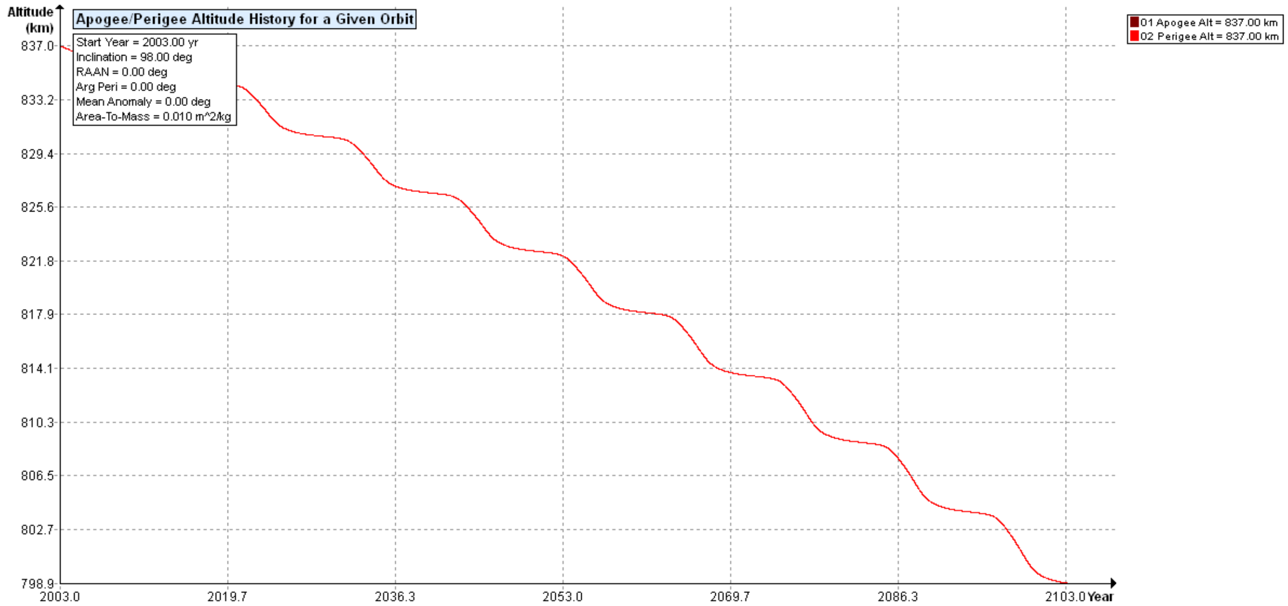

Figure 38. DTUSAT simulated altitude in hundred years. Used software: Debris Assessment Software.

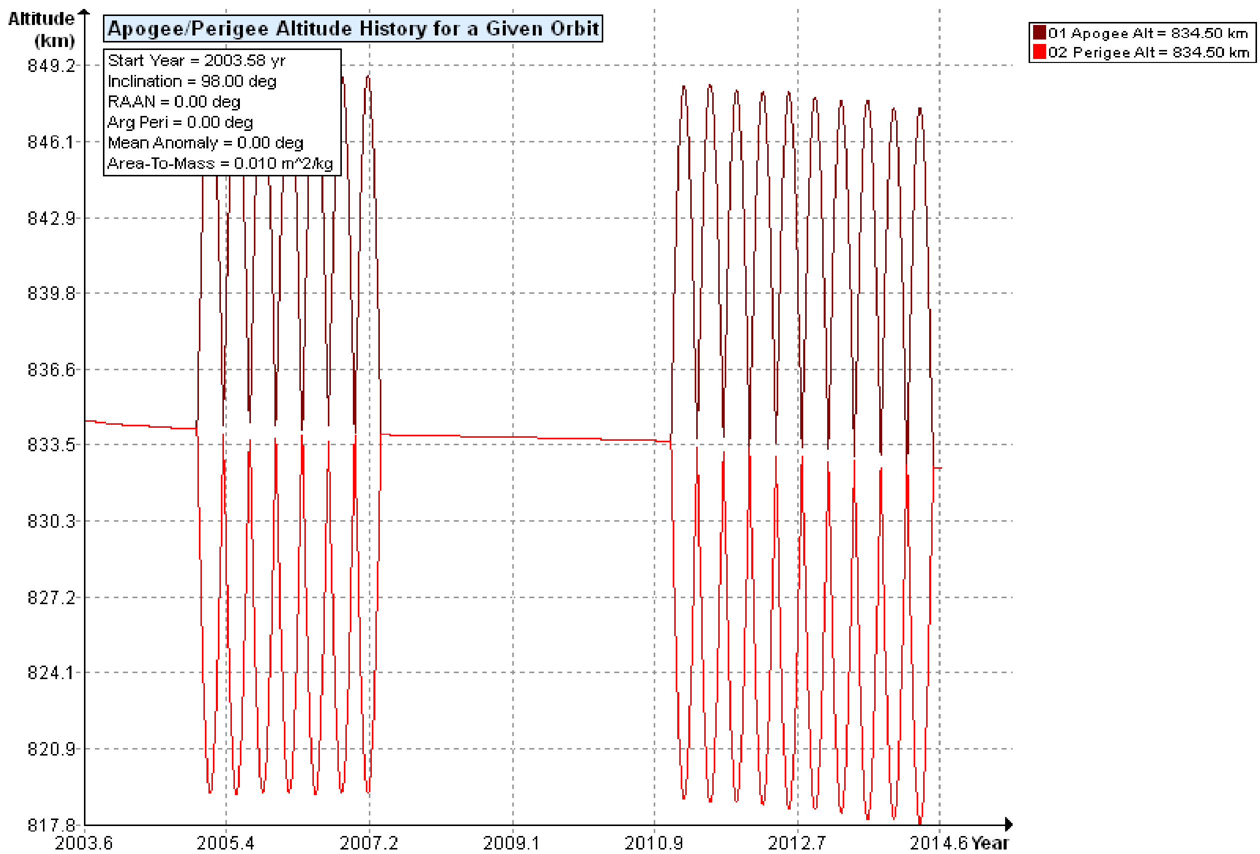

Figure 39. DTUSAT simulated altitude by 2014.. Used software: Debris Assessment Software.

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