30th June 2013

SAR Formation Flying

Annex 7. Orbit Modelling and Analysis, Simulated Mission Planning

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Dr Li Qiao
Australian Centre for Space Engineering Research (ACSER)
University of New South Wales, Sydney, 2052, Australia

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1. EXECUTIVE SUMMARY

The Garada project is investigating the design of a satellite mission that uses a Synthetic Aperture Radar (SAR) as the primary imaging sensor. Work Package 7 “Satellite Orbit Models“ is focused on determining the optimum orbit for the mission and selecting the candidate launcher vehicle. This WP is led by Professor Chris Rizos and Research Associate Dr Li Qiao. This WP mainly supports and serves WP1 (Overall Design).

Orbit modelling is a complex task as it involves trade-offs between different mission and user requirement parameters. During the mission analysis conducted in WP1, the application of the Garada mission has been defined as flood mapping, bio-mass estimation and soil moisture measurement. It has been concluded that the key application that could be provided to deliver significant benefit to Australia is soil moisture monitoring over an agricultural area (in particular the Murray Darling Basin – MDB). The payload requirements therefore relate primarily to soil moisture mapping. MDB is highlighted as a key area of interest as the area produces one third of Australia’s agricultural output. A coverage revisit interval of 2-3 days would be required to satisfy the mission goal. The payload will revisit the target area at the same time of day on subsequent passes in order to monitor the soil moisture content. Since the required SAR antenna is large in size, the mission has a big power budget requirement. This implies that maximum access to sunlight is a crucial orbit requirement. The Garada satellite will therefore be inserted into a dawn-dusk orbit where the satellite stays in sunlight on a continuous basis.

To satisfy the mission requirements, a frozen, repeating, circular sun-synchronous orbit (SSO) is the best candidate orbit. The SSO is generally favoured for Earth observation satellites that need to be operated at a relatively constant altitude suitable for imaging/sensing instruments. The proposed Garada orbit is at an altitude of 612.98km with inclination 97.84 degrees. The orbit will repeat after 89 revolutions in 6 days, completing $Q = 14 + 5/6$ orbits per day. Since the soil moisture revisit requirement is 2-3 days, a two-satellite constellation is needed. It is desirable to place the Garada 1 and Garada 2 satellites at the same altitude to double the revisit frequency, and to maximise consistent near-simultaneous coverage. Two 6-day repeat SSO satellites can meet the MDB 3-day revisit requirement.

In reality the orbit will change from its nominal geometry due to a variety of orbital forces. Therefore WP7 analyses how much different the satellite trajectory will be relative to the reference orbit after a certain period of time. Sensitivity studies reveal that the Earth’s irregular gravity field has the largest impact on satellite orbital motion. The atmospheric drag is the second largest perturbing effect, affected by the space weather more than the satellite drag model. The above two orbital forces cause perturbations of a magnitude of tens of kilometres. The third body effect is relatively small, with a magnitude of some hundreds of metres. The solar radiation pressure effect has less than one hundred metres effect.

Another WP7 task is the selection of the launch system to place the Garada satellite(s) into the desired orbit. Since Australia does not have any launch systems it is necessary to survey the international launch market and consider various candidate launcher options. The most important factors to be considered are reliability, performance, suitability, and price. Other factors include availability and schedule, technology transfer safeguards, customer-provider relationship and partnership, as well as terms and conditions. The large antenna of Garada drives the mission towards a launch vehicle of the size of a Falcon-9. Falcon-9 is a rocket-powered spaceflight launch system designed and manufactured by Space Exploration Technologies (SpaceX), headquartered in Hawthorne, California. The launch cost is estimated to be of the order of US$49-54 million. The U.S. Delta IV-M and the European Ariane 5 could also be used to launch Garada, however with much higher launch cost (greater than US$100 million). Therefore the Falcon-9 is the favoured candidate for the launch system.

2. INTRODUCTION

The Garada project is an Earth Observation Satellite design using Synthetic Aperture Radar (SAR) as the primary imaging sensor. In Phase 0 of this mission, precisely determining the user requirements is critical to the whole project as well as the satellite orbit, as the orbit design will be subject to the constraints of the requirements.

In 2011, the application of the Garada mission had been defined as flood mapping, bio-mass estimation and soil moisture measurement by WP1. The three applications lead to different user requirements. Accordingly, to support WP1's analyses, WP7 explored using a small satellite constellation to achieve hourly revisit, as well as the lifetime of small satellites, revisit performance with various orbit types and satellite formation stability (see reports TK7.1 and TK7.2).

In early 2012, the key application was changed to soil moisture monitoring over an agricultural area (in particular the Murray Darling Basin – MDB). The payload requirements therefore relate primarily to soil moisture mapping. MDB is highlighted as a key area of interest as the area produces one third of Australia's agricultural output. A coverage revisit interval of 2-3 days would be required to satisfy the mission goal. The payload revisits the target area at the same time of day on subsequent passes in order to monitor the soil moisture content. Since the user requirement is specified and unambiguous, the subsequent research in WP7 has focused on the soil moisture application. The three main tasks of Work Package 7 are listed as:

1) To include a description of the soil moisture imaging requirements, and the analysis performed to select the final orbit parameters (see TK7.3). This task is performed by first analysing the mission, payload and satellite design requirements to determine if the mission is feasible. Trade-off studies are then performed in order to find a suitable orbit that satisfies the mission goals. The proposed Garada orbit is a circular, frozen repeating sun-synchronous (SSO), dawn-dusk orbit at an altitude of 613km and 6 days repeat cycle. A constellation of two 6-day SSO satellites could reduce the revisit time to 3 days. This SSO will satisfy the soil moisture application requirement; therefore it has been chosen as the preferred orbit for Garada.

2) To perform the orbit perturbation sensitive study to form a baseline for the orbit force model (see TK7.4). This task investigates the orbit force models for the specified orbit. The task is performed by sensitivity analysis including gravity, atmospheric drag, solar radiation pressure, etc., and the ephemeris comparison. The objective is to determine the relative importance of each orbit force and calculate the magnitude of their impacts.

3) To analyse how to choose the launch system to put Garada satellite(s) into the desired orbit (see TK7.5). The task investigates the launch vehicle selection to transport the Garada satellite into the desired orbit. The task is performed by presenting the launch vehicle selection criteria, and comparing the Garada mission characteristics to the candidate launcher performance.

This final report assembles the previous work, organised in six sections with appendix. Section 1 is the executive summary for WP7. Section 2 summarises the research work and introduces the organisation of this report. Section 3 analyses the user requirements with respect to satellite orbits design. Based on the requirements, Section 4 presents the methodology and process of orbit selection and determines the baseline of proposed orbit. Section 5 performs the sensitive study of the effect of various orbit forces on the orbit trajectory. Section 6 selects the candidate and back-up launcher vehicles based on the survey of the global launcher market. The appendix lists the launch vehicles in the global launch markets and the orbit lifetime analysis for small satellites.

3. MISSION APPLICATION REQUIREMENTS

3.1. Soil Moisture Monitoring of Murray Darling Basin (MDB)

It has been concluded that the key application that could be provided to deliver significant benefit to Australia is soil moisture monitoring over an agricultural area such as the Murray Darling Basin (MDB) (Figure 1). The payload requirements therefore relate primarily to soil moisture mapping. MDB is highlighted as a key area of interest as the area provides one third of Australia's agricultural produce. A coverage revisit interval of 2-3 days would be required to satisfy the mission goal. The payload will revisit the target area at the same time of day on subsequent passes to determine the moisture content of the soil in the coverage area. Priority will be given to Australian target areas and Australian clients.

The science requirements of soil moisture drive the selection of specific orbit parameters, which require: 1) same illumination for repetitive imaging, 2) revisit the target area during early morning, 3) image at the same altitude, 4) sufficient ground resolution, and 5) revisit 2-3 days. Mapping these requirements to orbit characteristics, the desired orbit should be a circular, frozen repeating sun-synchronous, dawn-dusk orbit. The revisit time can be achieved by designing a wider swath SAR and appropriate selection of the orbit repeating cycle.

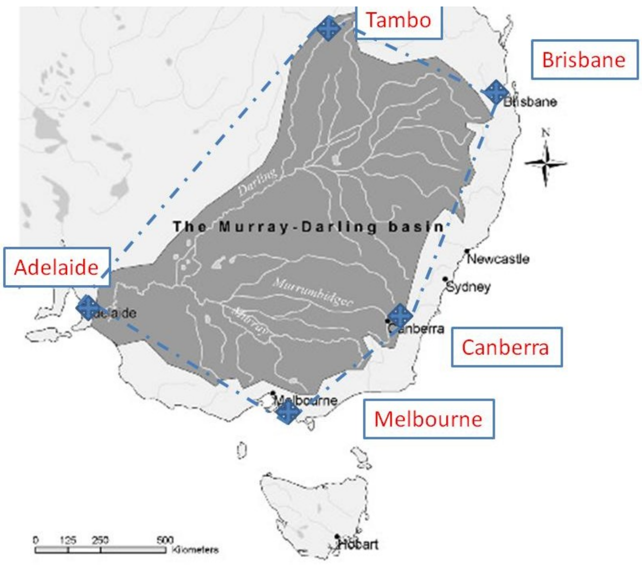

Figure 1 Murray Darling Basin.

3.2. Power System Requirements

The power subsystem requirements are not available at this stage. Since the required SAR antenna is large in size, the mission has a big power budget requirement, which implies maximum access to sunlight. Accordingly, the Garada satellite will be inserted into a dawn-dust orbit where the satellite stays in sunlight on a continuous basis.

3.3. Orbit Lifetime

The WP3.2 has established that the Garada SAR antenna will be very large. According to the latest satellite design from WP1, the gross mass is 2368.89kg and the height of Garada antenna is 15.6m when deployed and 7.8m when stowed. The diameter is 3.9m. The cross section is a trapezoid shape when deployed and a hexagon shape when stowed. An estimate of orbit life was performed using the “lifetime analysis tool” in STK to corroborate this assumption. The lifetime until de-orbit was calculated to be approximately 88 years. According to WP1, the lifetime of the Garada mission is expected to be 5 years, and hence the mission lifetime doesn’t depend on the orbit lifetime; but it will mostly depend on the manoeuvring fuel and other factors.

4. ORBIT SELECTION FOR GARADA

The orbit selection is almost entirely based on orbital mechanics. This section will provide some background on the subject, particularly the parameters that will be used to describe orbits.

4.1. Orbit Definition

The satellite orbit can be defined by the classical set of Keplerian parameters, referred to the vernal equinox inertial coordinates axes. In fact, the orbit modelling task is to find the optimal set of orbital parameters to meet the mission requirements. The six Keplerian parameters (2) are a e i Ω ω and ν. ν varies with time and the others are considered constant for a given orbit for the purposes of orbit design/analysis.

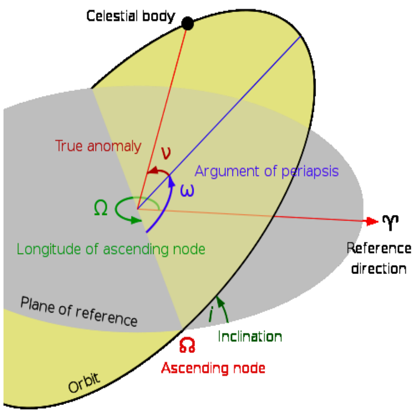

Figure 2 The Six Keplerian Elements[1].

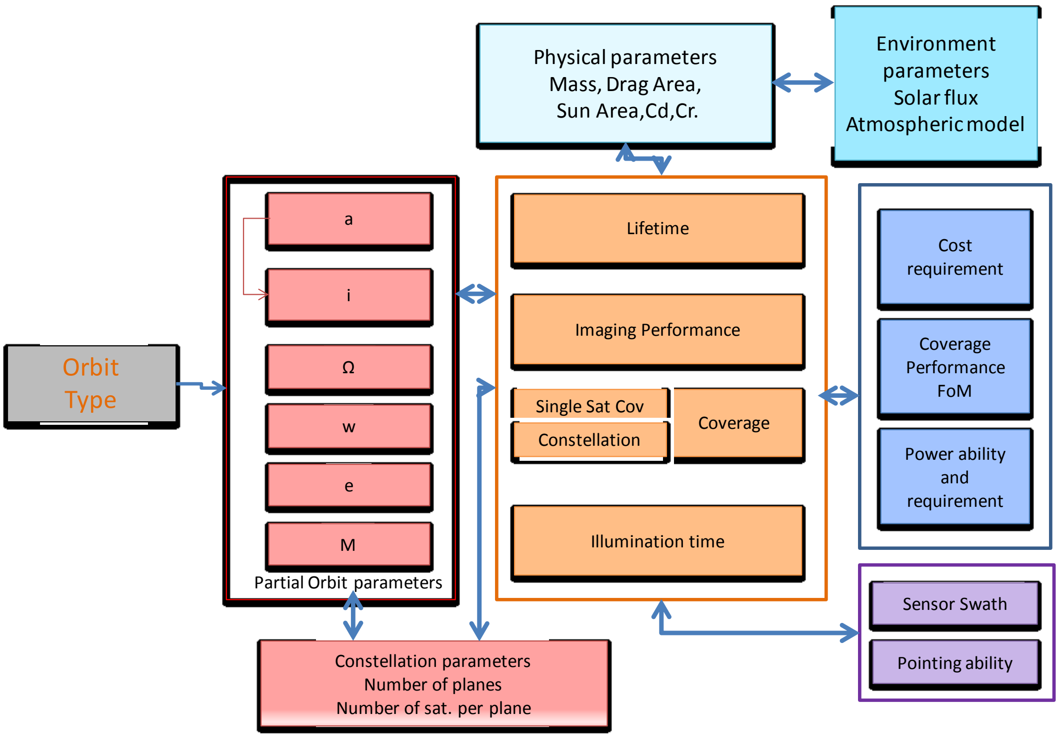

Figure 2 GARADA SSO design flow.

In the Garada design, the process could be presented by a simplified flow (see Figure 2). The choice of orbit plane is usually a compromise due to the balance of requirements. Till now, the lifetime and the coverage have been the two main issues. Illumination time is analysed briefly as the power supply requirements are not given. The ranges of original orbital altitude are determined according to the imaging performance, determined from WP 2. The coverage requirements are determined according to the mission overall design (WP 1).

4.2. Orbit Type

This section presents why choose repeating, frozen, sun-synchronous orbit for Garada mission.

4.2.1. Sun-Synchronous Orbits

The SSO is the most frequently used orbits for earth science missions [2]. The sun-synchronous orbit is generally favoured for Earth observation satellites that should be operated at a relatively constant altitude suitable for imaging/sensing instruments. Because of the deviations of the gravitational field of the Earth from that of a sphere that are quite significant at such relatively low altitudes, a strictly circular orbit is not possible. Very often a frozen orbit is selected that is slightly higher over the Southern Hemisphere than over the Northern Hemisphere.

Through careful consideration of the orbit perturbation force due to the oblate nature of the primary body a secular variation of the ascending node angle of a near-polar orbit can be induced without expulsion of propellant. As a result the orbit perturbations can be used to maintain the orbit plane in, for example, a near-perpendicular (or at any other angle) alignment to the sun-line throughout the full year of the primary body. Such orbits are SSOs[3].SSOs are typically near-circular Low-Earth Orbits (LEOs). It is normal practice to design a LEO in which the orbit period is synchronised with the rotation of the Earth surface over a given period, and a repeating ground-track is established. A repeating ground-track, together with the near-constant illumination conditions of the ground-track when observed from a SSO, enables repeat observations of a target over an extended period under similar illumination conditions[4].

The basic theory associated with how an orbital plane is perturbed as a result of the Earth's equatorial bulge is explained below. This bulge creates an out-of-plane gravitational force on the orbit causing the orbit to gyroscopically precess. The operative equation describing the rate at which the line of nodes moves due to this bulge is given by:

$$\dot{\Omega} = -\frac{3}{2}J_2\left(\frac{a_e}{a(1-e^2)}\right)^2\sqrt{\frac{\mu}{a^3}}\cos i \qquad (1)$$

$$\dot{\Omega} = [illegible]\frac{1}{(1-e^2)^2}\left(\frac{Re}{a}\right)^{\frac{7}{2}}\cos i \qquad (2)$$

When choosing $\dot{\Omega} = \frac{360°}{365.242199day} = 0.9856° / day$ i.e. the rate which equals that of the Earth moving on the orbit around the Sun, an SSO is obtained. Thus, at the equator, the satellite passes overhead at the same local time in each revolution.

4.2.2. Circular and Frozen Orbit

SAR is a powerful remote sensing tool that has useful characteristics such as day-night, all-weather operation and good resolution. To provide a reliable imagery from the side scan radar, the SAR has to be maintained at a constant altitude, which means the appropriate orbit should be 1) circular orbit, i.e. its eccentricity $e \approx 0$ and 2) in spite of Earth's oblateness, the perigee remains fixed to a specific latitude. Consequently, the SAR candidate orbits are normally frozen and circular orbits.

A frozen orbit is characterised by no long-term changes in the orbital eccentricity and the argument of perigee. The design of frozen orbits involves selecting the correct value of eccentricity and argument of perigee, for a given semi-major axis and orbital inclination, which satisfies the following system of non-linear perturbation equations:

$$\frac{de}{dt} = \frac{3}{2}\frac{J_3 \mathrm{R}_{eq}^3}{p^3}(1-e^2) n \sin i \cos\omega (\frac{5}{4}\sin^2 i - 1) = 0 \tag{3}$$

$$\frac{d\omega}{dt} = \frac{3}{2}\frac{J_3 \mathrm{R}_{eq}^3}{p^3} n(2-\frac{5}{2}\sin^2 i) - \frac{3}{2}\frac{J_3 \mathrm{R}_{eq}^3 \sin\omega}{p^3 e \sin i} n \left\{ (\frac{5}{4}\sin^2 i - 1)\sin^2 i + e^2(1-\frac{35}{4}\sin^2 i \cos^2 i) \right\} \tag{4}$$

To simplify the equations, the orbit can be frozen by satisfying the following equations:

$$\begin{cases} e \approx 0 \\ \omega = \pm 90^\circ \end{cases} \tag{5}$$

This is implemented in the orbit control software used for the European ERS-1, ERS-2 and NASA's EOS satellite. Since the orbit is frozen, the perigee will not change and the altitude will be a function of latitude. This means that in spite of the Earth's oblateness, the perigee will remain fixed (in an average sense) at the northernmost latitude (essentially at the North Pole). It is noted that the electrical control is needed to maintain the frozen orbit and for inclination manoeuvre.

4.2.3. Repeating Ground Track Orbit

Repeating ground track is a useful characteristic that ensures that global coverage is complete and repeatable over a designated sampling period. Repeating orbit's sub-satellite track forms a closed curve on the Earth's surface. The repeating ground track equation is:

$$\frac{1}{T} = \frac{kM+m}{M} \tag{6}$$

For a circular orbit, N is the revolution of the orbit per day, which is given by:

$$N = k(\frac{m}{M}) = \frac{kM+m}{M} \tag{7}$$

T is the duration between two contiguous ascending nodes. For the SSO, T could be given by:

$$\begin{aligned} T &= T_0(1+1.5J_2(\frac{\mathrm{Re}}{a})^2(3-2.5\sin^2 i)) \\ T &\approx (0.9992 \qquad T_0 \end{aligned} \tag{8}$$

$$T_0 = 2\pi\sqrt{\frac{a^3}{\mu}} \approx 1.658669 \times 10^{-4} a_0^{\frac{3}{2}} \tag{9}$$

Most SAR applications have been based on a repeat-pass orbit scenario (Tsang and Jackson 2010). Numerator part of revolution M united in day is the orbit repeat cycle. It is the period of the repeat-pass interferometry.

4.3. Classical and Perturbed Orbit

Under the influence of the gravity pull of a large spherical body, the path in space followed by a satellite is a conic section with the central body at one focus. For remote sensing missions, a closed circular or elliptical orbit is favoured.

The shape of the orbit is described by its semi-major axis a and eccentricity, e .The orbit orientation with respect to the Earth is given by the inclination i (the angle between the orbit plane and the equator), the location of the ascending node Ω (the right ascension where the satellite crosses the equator heading north – RAAN), and the argument of perigee ω (the angle in the direction of satellite motion between the ascending node crossing and the point of closest approach). In the absence of disturbing forces, the orbit shape and orientation are constant.

The Earth is not a sphere but rather an oblate spheroid in which the radius at the equator is about 21km greater than at the poles. The elliptical path followed by the satellite is perturbed because the Earth's mass is not spherically symmetrical. The extra mass at the equator relative to the poles creates a torque on the satellite about the centre of the Earth, rotating the plane of the orbit about the polar axis. This results in a secular change in the location of the ascending node known as nodal regression. The regression rate of the orbit plane $\dot{\Omega}$ depends mainly on the altitude and inclination:

$$\dot{\Omega} = -\frac{3}{2} J_2 \frac{R^2}{a^2(1-e^2)^2} n \cos i \tag{10}$$

where

J_2 coefficient describing Earth oblateness (1.08263×10^{-3})

R the equatorial radius of Earth (approximately $6378.144 km$)

n the angular speed of a circular orbit ($n = \sqrt{\mu / a^3}$) and the orbit period $2\pi / n$

μ the gravitational parameter of the Earth ($3.986005 \times 10^5 km^3 s^{-2}$)

Equation (10) indicates that a SSO can be achieved by choosing the i according to h :

$$\cos i = -0.09890445 (\frac{\text{Re}}{\text{Re} + h})^{-3.5} \tag{11}$$

Earth oblateness also causes the line of apsides connecting the perigee and apogee to rotate in the orbit plane. This secular change in perigee location is given by:

$$\dot{\omega} = \frac{3}{2} \frac{J_2 R^2}{a^2 (1-e^2)^2} n (2 - \frac{5}{2} \sin^2 i) \tag{12}$$

In addition to oblateness effects, the Earth's northern and southern hemispheres are not equal causing a satellite to experience different forces during its orbit. The perturbation affects the argument of perigee at a rate that depends on the sine of ω :

$$\dot{\omega} = \frac{3}{2} \frac{J_3 R^3 \sin \omega}{a^3 (1-e^2)^3 \sin i} \cdot n \left\{ \left(\frac{5}{4} \sin^2 i - 1 \right) \left(\sin^2 i - e \cos^2 i \right) \right\} \tag{13}$$

J_3 coefficient describes the Earth's north/south asymmetry (-2.536414×10^{-6})

The argument of perigee in a polar orbit moves through $360°$ over tens of days due to J_2 , while the perturbation due to J_3 is considered long period. J_3 also causes a long period perturbation on the eccentricity given by:

$$\dot{e} = \frac{3}{2} \frac{J_3 R^3}{a^3 (1-e^2)^2} n \sin i \cos \omega (\frac{5}{4} \sin^2 i - 1) \tag{14}$$

The change in argument of perigee is undesirable in remote sensing missions – the platform altitude over a given site will change from pass to pass. In order to avoid this, a frozen orbit in which the eccentricity and perigee location are nearly constant has been proposed for a soil moisture monitoring mission. Equations (13) and(14) imply that an argument of perigee of $\pm 90°$ results, i.e. $\sin\omega = 0$, where $\omega_e = 0.00417807°/s$ is the angular speed of the Earth.

4.4. Eccentricity, Perigee Location, Inclination and LTAN Selection

Eccentricity (e), argument of perigee (ω) and inclination (i) are fixed by the requirements.

4.4.1. Eccentricity

A near-circular orbit is desired so that the antenna will view all areas of the Earth from approximately the same altitude, and thus with the same resolution and sensitivity. This implies a circular orbit. The secular and long period changes in eccentricity and argument of perigee are undesirable in extended remote sensing missions, otherwise the platform altitude over a given site will change from pass to pass. A frozen orbit in which the eccentricity and perigee location are nearly constant has been proposed for the Garada satellite orbit. An argument of perigee of $\pm 90°$ results in an $\dot{\iota}$ of zero. Then an eccentricity is chosen so that $\dot{\iota}$ due to J_2 and J_3 is zero. The frozen eccentricity e in a polar orbit is approximately 0.001 – approximately a circular orbit and the altitude of the platform then defines the semi-major axis.

4.4.2. Perigee Location

In a circular orbit the location of closest approach is not defined. A frozen orbit about the Earth requires an argument of perigee of $90°$. In this case $\dot{\iota}$ and there is no secular change in perigee location.

4.4.3. Inclination

Constant solar illumination at a target from one observation to the next is desired. The orbit that achieves this by maintaining a given sun orbit plane orientation is the SSO and is achieved by taking advantage of the Earth's oblate shape. For a given altitude, the inclination can be selected so that the nodal regression is equal to the apparent motion of the sun about the Earth (about $1°$ per day, eastward). If a certain local time of node crossing is desired, the orbit plane is oriented with the sun accordingly. The Earth rotates $360°$ in about $23h56\min$, or $15.042°/h$. For LEOs, the sun-synchronous inclination is between $90°$ and $100°$, satisfying the requirement to view the entire Earth.

This constant sun orbit plane orientation varies throughout the year. The Earth's orbit around the sun is not a circle and therefore the sun's apparent motion is not constant, through the precession of the orbit plane. In the spring and autumn this difference amounts to about $2°$, or $8\min$ of local time. Solar perturbation of the moon's orbit around the Earth causes a slight change in the orientation of the Earth's poles, contributing to variations in the sun orbit plane orientation with a period of 18.6 years, the combined effect amounting to about $4°$ twice each year.

4.4.4. Equator Crossing Time

As indicated in Section 3 the Garada satellite will be inserted into a dawn-dusk orbit in order to maximise solar power generation. Accordingly, the local time of node crossing (either ascending or descending) is specified.

A SSO allows the selection of a desired platform equator-crossing time. The sun orbit plane orientation corresponding to this desired time will be maintained throughout the mission, though small orbit adjustments may be required. In circular, inclined orbits, each ascending equator crossing occurs at the selected local crossing time and each descending crossing at this local time plus 12 hours. There are two options for the Garada orbit, the local time of ascending node (LTAN) is 6:00 in the morning or 18:00 toward evening.

It is assumed that the Garada orbit's LTAN is ascending at 6:00 in the morning, and descending at 18:00 toward evening, as it will meet one of the user's requirements that "6:00am is generally considered optimal for soil moisture monitoring, due to the thermal equilibrium between soil/air/vegetation and also the reduced capillary moisture raise in the top soil which happens during the night (explained by Dr. Rocco Panciera[*])".

4.5. Orbit Altitude

The orbital elements remaining to be selected are the altitude and node crossing location. For a repeating SSO, the choice of altitude determines the instrument coverage pattern and repeat cycle, instrument performance and satellite lifetime. Drag on the satellite determines the lower altitude bound, and launch vehicle capabilities and instrument performance set the upper bound. Other important altitude-dependent effects limiting the lower altitudes are atomic oxygen damage, wake currents, ionospheric plasma, and optical surface contamination. Natural ionising radiation is a significant constraint on higher altitudes.

For the Garada mission, the antenna being design by the Astrium team has specified the altitude range of 580-660km. At this stage the primary limiting factors are the revisit performance rather than atmospheric drag and launch vehicle performance. The number of orbits completed per day Q influences the location and sequence of all ground traces:

$$Q = \frac{86400}{P} \tag{15}$$

$$P = \frac{2\pi}{\omega} \tag{16}$$

$$\omega = \sqrt{\frac{\mu}{(\mathrm{Re}+h)^3}} \tag{17}$$

Altitude h of SSOs for Q is given by:

$$Q = \frac{86400}{2\pi}\sqrt{\frac{\mu}{(\mathrm{Re}+h)^3}} \tag{18}$$

It shows the corresponding values of Q and h when Q is set to integer. Since the radar specified orbit height range is from 580 to 660km, the I value is 15. Q is found by comparing the rotation of the Earth beneath the satellite with the motion of the orbit plane. In slightly less than 1 day the Earth rotates through 2π radians. In one orbit, the plane of the orbit moves eastward at the nodal regression rate [illegible]. For SSOs [illegible] is set to the approximately $1°/day$ eastward drift of the sun. To find the ground trace of the satellite, the rotation of the Earth is included to give the motion of the orbit plane relative to the Earth or longitude rate, positive west, [illegible], where [illegible] is the Earth's rotation rate ($7.292115856\times10^{-5} rad/s$, approximately $360°/day$). The angle that the orbit plane rotates through in one orbit relative to the Earth is the longitude rate multiplied by the time from one ascending

[*] Rocco Pancier is a Super Science Fellow at Cooperative Research Centre for Spatial Information (CRC-SI). He is expertise in soil moisture remote sensing. The requirements for the soil moisture application in this report are based on discussions with Rocco Pancier.

node crossing to the next, referred to as the nodal period P_n. This angle is known as the fundamental interval S, where $S = P_n\dot{\Omega}$. S is the longitude difference between one ground trace and the next.

For a SSO the number of orbit revolutions completed in 1 day Q is $2\pi/S$. To find the orbit giving a desired value of Q, the required nodal period is found using the relation:

$$P_n = 2\pi / Q\dot{\Omega} \tag{19}$$

where

$$P_n = P[1 - \frac{3}{2}J_2\frac{R^2}{a^2}(4\cos^2 i - 1)] \tag{20}$$

P is the classical orbit period $2\pi\sqrt{a^3/\mu}$. The semi-major axis corresponding to this nodal period then yields the sun-synchronous inclination.

Let Q be represented as:

$$Q = I(\frac{K}{D}) = \frac{N}{D} \tag{21}$$

where

Q	orbits per day
N	the number of orbits in the repeat cycle (i.e. number of revolutions to repeat)
D	the number of days in the cycle

The satellite is expected to repeat after a certain number of revolutions (N). An exactly repeating orbit, one in which the ground track of the satellite is retraced after a given period of time (D), is desired so that data can be consistently compared throughout the mission lifetime:

$$S = \frac{360°}{Q} = \omega_e P \tag{22}$$

S the fundamental interval at the equator

$$Si = \frac{S}{D} \tag{23}$$

Si S is general divided into D subintervals

Altitudes in the range of 580 to 660 km of several SSOs for different repeat cycles are shown in Table 1. Comparing columns D and Si indicates that fast repeating D requires a wider swath to cover the Si.

Table 1 Sun-synchronous orbit altitude vs. repeat cycles.

I	K	D	N	Q	h [km]	Si [km]
14	2	3	44	14.6667	665.964	910.7968
14	3	4	59	14.7500	639.351	679.2382
14	4	5	74	14.8000	623.503	541.5449
14	5	6	89	14.8333	612.987	450.2816
14	6	7	104	14.8571	605.500	385.337
14	7	8	119	14.8750	599.898	336.7652
14	8	9	134	14.8888	595.548	301.3162
14	7	9	133	14.7777	630.536	301.3162

14	9	10	149	14.9000	592.074	268.9601
14	9	11	163	14.8181	617.762	245.8593
14	10	11	164	14.9090	589.234	244.3601
14	11	12	179	14.9166	586.870	223.883
14	10	13	192	14.7692	633.245	208.7243
14	11	13	193	14.8461	608.953	207.6428
14	12	13	194	14.9230	584.871	206.5725
14	11	14	207	14.7857	628.022	193.5993
14	13	14	209	14.9285	583.158	191.7467
14	11	15	221	14.7333	644.654	181.3351
14	13	15	223	14.8666	602.511	179.7088
14	14	15	224	14.9333	581.675	179.7088
14	13	16	237	14.8125	619.555	169.0931
14	15	16	239	14.9375	580.378	167.6781

4.5.1. Swath and SSO Repeating Cycle

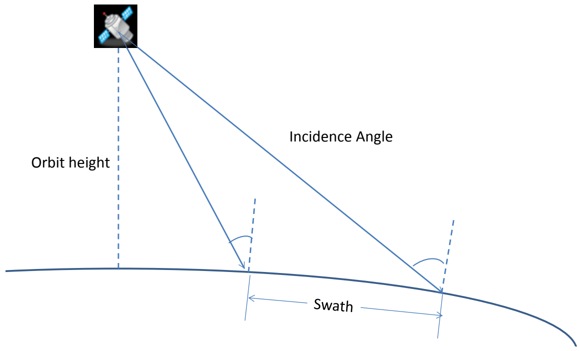

Figure 3 Geometry of swath, incidence angles and orbit height.

Garada is a satellite with an L-band SAR which allows not only conventional stripmap and ScanSAR modes but also a Spotlight mode with electric beam steering. To cover wide areas, Garada has the capability to view wide incidence angles of 8 to 40+ degree with electric beam steering, and the left- or right-looking by satellite manoeuvre from nominal look direction of nadir-looking.

The swath depends on altitude and tow incidence angles, referred to as inner incidence and outer incidence angles. Set the inner incidence angle to 8 degree (value from WP1); Figure 4 shows the swath with an outer incidence angle varying from 40 to 50 degree when the orbit height increases from 580 to 660km. The swath is in the range of 359.3 to 587.9km. Some of the values in Figure 4 are listed in Table 2. The 40 degree outer incidence angle is intended for the soil moisture application. The outer incidence angle can extend to 50 degree for other applications.

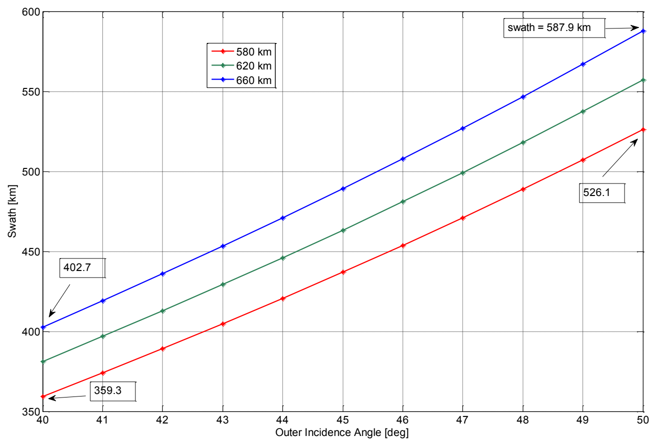

Figure 4 Swath vs. incidence angles when the orbit height varying from 580 to 660km.

Table 2 Swath [km] vs. out incidence angle at various orbit height (Inner Incidence angle = 8 deg).

Outer incidence angles [deg]	Orbit height [km]				
	580	600	620	640	660
40	359.397	370.366	381.253	392.058	402.783
50	526.132	541.784	557.298	572.675	587.918

4.5.2. Instrument Requirements and Orbit Selection

One factor in determining the length of the exact repeat cycle for the Garada orbits is set by the imaging instrument's swath. There are two options to choose for the orbit from Table 1:

1) Instrument is able to image the entire Earth (capable of global coverage).

Si is the widest length between the two adjacent ground tracks. With the given swath, the quickest repeat cycle could be found by comparing the swath and Si. A 370.3km swath width is accessible at 599.8km altitude. Compare the swath to the Si, the result is 8 implying that a minimum 8-day repeat is required to view the entire Earth. Thus, in this situation the altitude selection of SSOs is based entirely on just how wide the swath is.

2) Instrument is able to image the entire target area (full coverage)

The length between the two adjacent ground tracks varies with the latitude. Si is the widest as it presents the length on the equator (latitude = 0 degree). The length at a given latitude is $Si \times \cos(\text{latitude})$ which is narrower than Si. Only covering the target area implies a shorter repeating cycle with gaps in the equatorial area. In this situation the altitude selection of SSOs must consider the target area location.

Table 3 shows the coverage percentage over MDB with four SSO satellites. Because of the swath's constraints, the fastest repeating cycle for MDB for complete coverage is 6 days.

Table 3 SSOs and their coverage percentage over MDB.

SSO repeat cycle [day]	Height [km]	MDB coverage
6	612.987	100%
5	623.503	96.15%
4	639.351	78.70%
3	665.964	56.21%

With incidence angle 8 and 40 degree, a 6-day SSO imaging of the entire MDB is possible, see Figure 5. With incidence angle 8 and 50 degree, a 6-day SSO could also achieve full coverage over the whole Australian region, see Figure 6.

Figure 5 6-day SSO satellite's coverage over MDB (Incidence angle is 8 and 40 degree).

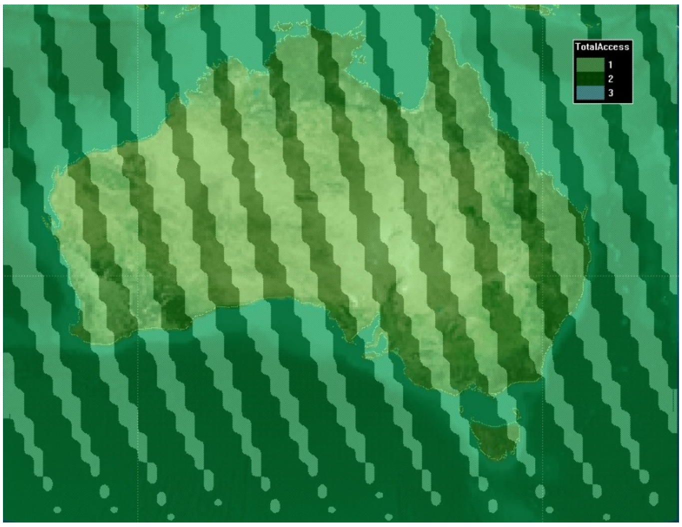

Figure 6 6-day SSO satellite's coverage over the whole Australian region (antenna incidence angle is 8 and 50 degree).

For the optimum 6-day repeating cycle SSO the inclination must then be 97.84 degree. It will repeat after 89 revolutions. It completes $Q = 14 + 5/6$ orbits per day. The fundamental interval is $6 \times Si$, or 2701.7km at the equator, so that the second ascending orbit trace lies $5 \times Si$ east of the first. In Figure 7, the location of the first six equatorial crossings is shown. Since the Q is not an integer, at least one orbit will cross in the interval between the first and second orbits. The fundamental interval is crossed once each day in a different location until after 6 days (89 orbits) the first orbit trace is repeated and the cycle begins again. It also indicates that by choosing a different value for Q, a completely different ground trace sequence would result.

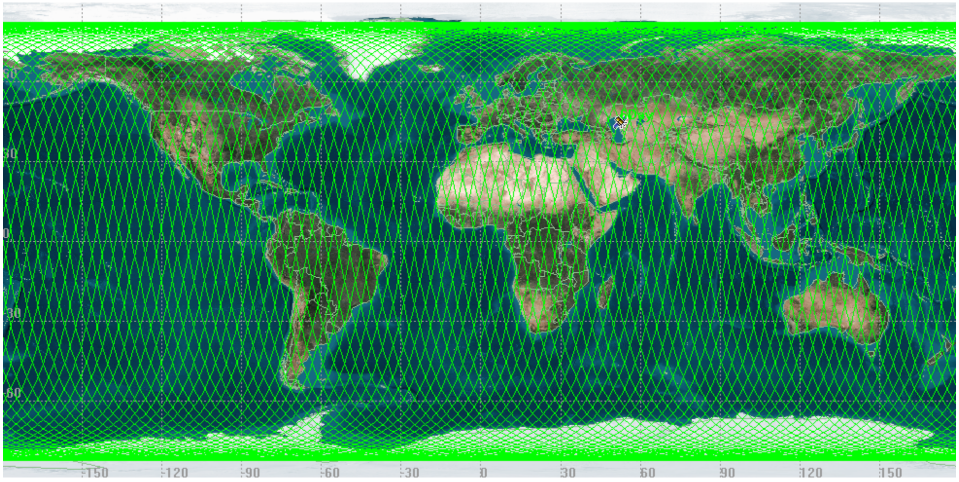

Figure 6 Ground track over 6 days.

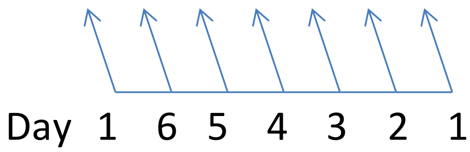

Figure 7 Coverage pattern at the equator.

Since the soil moisture revisit requirement is 2-3 days, one satellite could not achieve the goal and hence a satellite constellation is needed. It is desirable to place the Garada 1 and Garada 2 satellites at the same altitude (see Figure 8) to double the revisit frequency and to maximise consistent near-simultaneous coverage. Two 6-day repeat SSO satellites can meet the MDB 3 day revisit requirement (see Figure 9).

Figure 8 Geometry of Garada 1 and Garada 2.

Figure 9 Two 6-day SSO constellation coverage over MDB (antenna incidence angle is 8 and 40 degree).

3) Instrument is able to image most of the target area (partial coverage)

Table 3 shows that the minimum repeating cycle is 6 days to satisfy the requirement for full coverage of MDB. However this needs a minimum of two satellites in order to meet the 2-3 day revisit requirement. If accepting the assumption that "Less coverage with quick revisit is more suitable and would allow more accurate soil moisture predictions (explained by Rocco Pancier)", the 3-day SSO in Table 3 is the candidate orbit. It is possible to increase the revisit, but sacrificing full coverage (Figure 10). Before the launch of the second Garada satellite, the first Garada satellite could be put into the 3-day SSO to offer a quick revisit. Then Garada 1 satellite could transfer its orbit to a 6-day SSO and enable 3-day revisits in combination with the Garada 2 satellite.

Figure 10 3-day SSO coverage over MDB.

4.6. Proposed Orbit Summary

The proposed Garada orbit is a circular, frozen repeating sun-synchronous, dawn-dusk orbit, with an altitude of 613km and 6 days repeat cycle.

The latest physical parameters from the satellite design study:

- The gross mass is 2368.89kg
- The height is 15.6m when deployed in the orbit and 7.8m when stowed in the launch vehicle
- The diameter is 3.9m
- Drag Area ≈ 3.9^2 = 15.21m2
- Radiation Area ≈ 3.9*15.6m = 60.48m2

Soil moisture requirements:

- 2-3 days revisit at the Murray Daring Basin
- Partial access over the Murray Darling Basin
- Image the soil at dawn and dusk

Assumption:

- A large power budget requirement

The common characteristics of the proposed orbits are:

- Circular orbit
- Sun-synchronous, dawn-dusk
- Local time of ascending node: 6:00 am
- Longitude of first ascending node (decided by the ground station longitude)

Due to the large satellite size and mass design, it is proposed that the Garada 1 and 2 satellites be launched separately. A plan could be to launch Garada 1 into a 3-day SSO in Phase 1 before Garada 2 is launched. Then transfer the Garada 1 satellite into a lower 6-day SSO after Garada 2 is inserted into orbit. In terms of the classical orbit elements (Epoch 28 Jun 2012 02:00), the orbit is defined in Table 4.

Table 4 Garada satellite orbit characteristics.

	Garada 1	**Garada 2**
Phase 1 (Before Garada 2 is launched)	• Semi-major axis = 7044.1km • Height = 665.96km • Eccentricity ~=0 • Inclination = 98.05 degree • Argument of perigee = 90 degree • RAAN = 7.31 degree • True anomaly (determined by the launch) • Number of revolutions to repeat 44 • Approximate revolutions per day 14+2/3	
Phase 2 (After Garada 2 is launched and Garada 1 is transfer to 6-day SSO.)	• Semi-major axis = 6991.12km • Height = 612.98 km • Eccentricity ~= 0 • Inclination = 97.84 degree • Argument of perigee = 90 (frozen orbit) • RAAN = 7.31 degree • True anomaly (determined by the launch) • Approximate revolutions per day: 14+5/6 • Number of revolutions to repeat: 89	• Semi-major axis = 6991.12km • Height = 612.98 km • Eccentricity ~= 0 • Inclination = 97.84 degree • Argument of perigee = 90 (frozen orbit) • RAAN = 7.31 degree • Garada 1'sTrue anomaly +180(determined by the launch) • Approximate revolutions per day: 14+5/6 • Number of revolutions to repeat: 89

5. ORBIT FORCE ANALYSIS

In the realistic word, the orbit will tend to diverge from its nominal position (the reference orbit) due to the orbital force. A simple example is that the atmospheric drag could reduce the velocity of the satellite and decay the orbit. How much difference will the satellite trajectory be away from the reference one due to various orbit forces? What are the motions of the satellite under the influence of orbit forces such as gravity, atmospheric drag, third body gravity, etc? Among these forces, which impact significantly and which can be ignored in an accepted accuracy? In order to answer these questions, it is required to build a proper orbit propagator for satellite.

Generally, orbit propagator concerns the determination of the motion of a satellite over time. According to Newton laws, the motion of a body depends on its initial state and the force that act upon it over time. High fidelity propagators attempt to include all significant force models acting on the satellite; low fidelity propagators approximate the effects of some force while completely disregarding others. High fidelity propagators solve Newton’s laws suing numerical methods; low fidelity propagators tend to be analytic. Numerical propagator asks for more calculation requirement; analytic propagators are the fastest to use.

There are some common propagators for use: TowBody, J2Pertubation, J4 Pertubation, SGP4 and HPOP. In order to decide which propagator is appropriate for Garada, this section analysis the characteristics of Garada orbit. As a Low Earth Orbit satellite, the atmospheric drag impacts on it significantly, thus TowBody, J2 and J4 Pertubation which do not model atmospheric drag or solar or lunar gravitational forces are not suitable. SPG4 propagation cannot support accurate orbit modelling analysis due to its simplified model. Therefore, HPOP is adequate as it is a high fidelity numerical integration propagator and the aforementioned forces can be included. Then the report uses the STK/HPOP tool to perform the sensitivity study which describes the orbit propagator performances contributed by each orbit force so that to obtain a baseline to propagate an orbit at a certain level of accuracy.

The sensitivity study results reveal that the Earth’s gravity contributes the largest effect on satellite orbits. New updated gravity model causes limited difference, and the maximum degree and order is the main factor to be considered. Therefore, the gravity fields should not be truncated for precise operations. Solid tides and ocean tides contribute very small effects to orbits, and would be considered only for precise operations. The atmospheric drag is generally the second largest effect. The atmospheric drag is affected by the space weather more than that of the dag model. Therefore, it is important to model the space weather and choose an accurate space weather file. The above two forces are of a magnitude of decades of kilometres. The third body effect is relatively small with a magnitude of hundreds of metres. The solar radiation pressure effect is less than one hundred metres in the orbit repeat cycle. Integration contributes generally small unless the RK4(5) is chosen.

5.1. Garada Orbit

Garada orbit is a sun-synchronous orbit using the oblateness of the Earth's shape. An orbit that is close to being polar will be affected asymmetrically by the bulge at the equator. This asymmetry acts to slowly rotate the plane of the orbit about the axis of the Earth. When the inclination is suitably chosen, the motion of the orbit plane matches the motion of the sun across the sky. In other words, the plane of the orbit executes one full rotation about the axis of the Earth in one year. For these reasons, Garada orbit force model cannot be simplified as a two-body motion problem. The Earth oblateness is the main perturbation for Garada, represented by the dominant J2 non-spherical term of the earth's gravity field force.

If only Earth gravity is considered, the Garada orbit will have an exact 6-day repeat ground track. STK generated 2D map of its ground track (ascending passes) is given as Figure 1. But in the real world, due to other perturbations such as atmospheric drag, the ground track could not maintain its repeating characteristic as shown in Figure 12. It can be seen that the second dominant perturbation for Garada is therefore the atmospheric drag.

The baseline of the orbit force modelling is the Earth's gravity field plus the atmospheric drag. Other perturbations, such as the high-order Earth gravity, the third body and the solar radiation pressure influences will be described in this section.

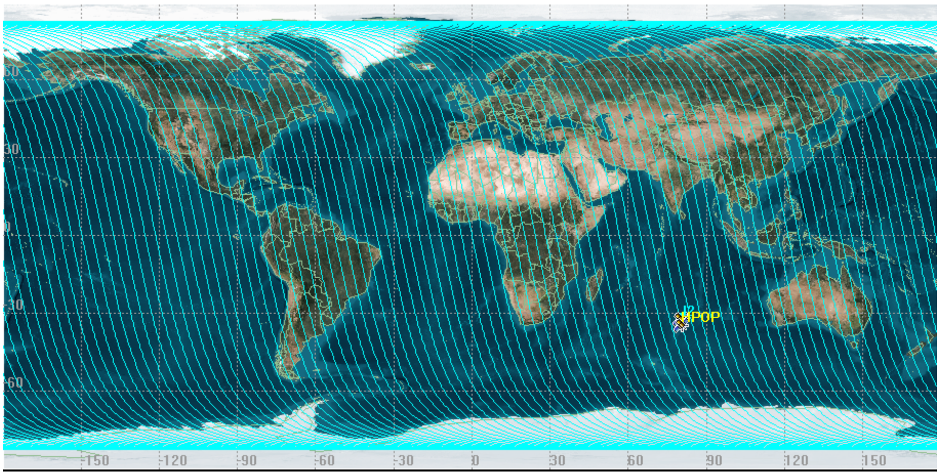

Figure 11 STK generated image of Garada's 6-day repeat ground track mission orbit.

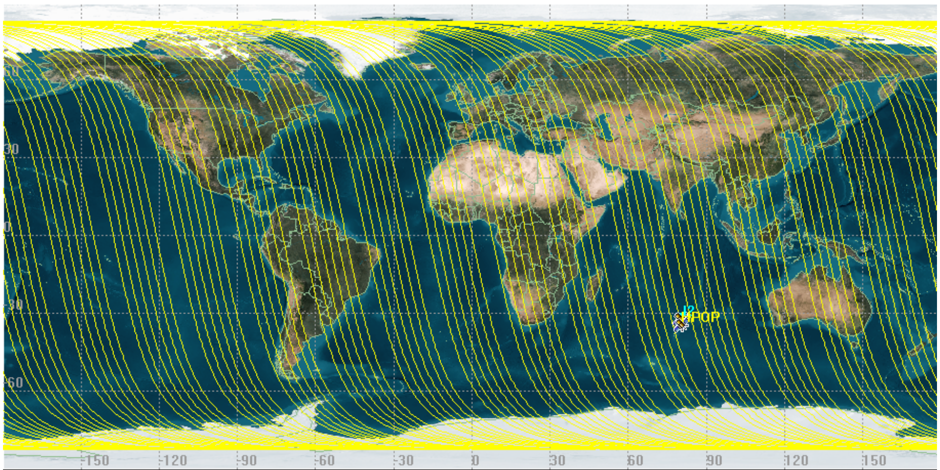

Figure 12 STK generated image of Garada's 6-day ground track affected by atmospheric drag.

5.2. High Precision Orbit Propagator

For any space mission one of the fundamental questions is what observations and processes are needed to achieve a certain level of accuracy on a particular satellite, now and at a future time. This requires orbit propagation using accurate orbit modelling.

Table 5 is a list of propagators for satellites available with a short description of each. As discussed in 5.1, Earth gravity and atmospheric drag are the two basic perturbations for Garada orbit. Consequently, the two-body, J2 and J4 cannot provide sufficient accuracy as they do not model atmospheric drag or solar or lunar gravitational forces. SGP4 is a historical propagator. It considers secular and periodic variations due to earth oblateness, solar and lunar gravitational effects, gravitational resonance effects and orbital decay using a simple drag model. As the drag model is relatively simple the SPG4 propagation cannot support accurate orbit modelling analysis. The High-Precision Orbit Propagator (HPOP) uses numerical integration of the differential equations of motion to generate an ephemeris. Several different force modelling effects can be included, including a full gravitational field model (based upon spherical harmonics), third-body gravity, atmospheric drag and solar radiation pressure.

Table 5 Common propagators and descriptions.

Propagator	Description
Two Body	Considers only the force of gravity from the earth, modelled as a point mass.
J2 Perturbation	The J2 Perturbation (first-order) propagator accounts for secular variations in the orbit elements due to earth oblateness.
J4 Perturbation	The J4 Perturbation (second-order) propagator accounts for secular variations in the orbit elements due to earth oblateness.
HPOP	The High-Precision Orbit Propagator (HPOP) can handle circular, elliptical, parabolic and hyperbolic orbits at distances ranging from the surface of the earth to the orbit of the moon and beyond.
SGP4	The Simplified General Perturbations (SGP4) propagator, a standard AFSPACECOM propagator, is used with two-line mean element (TLE) sets.

Because there are many parameter settings available for users, a precise description of the force model environment can be specified, and a highly precise orbit ephemeris can be generated. Different force model parameter settings make HPOP the most accurate STK propagator, however this high precision is not without costs: (1) the user is responsible for choosing force model settings appropriate to the situation being modelled; and (2) ephemeris generation takes more computational time and effort than analytical propagation (which simply evaluates a formula).

5.3. The Standard HPOP Settings

In order to compare the results while varying force model settings, the standard set of default parameters in Table 6 are used. The coordinate frame is always the J2000 frame.

Table 6 Default settings for force model.

Item	Value
Time	Start Time: 20 Dec 2012 01:00:00.000 UTC Stop Time: 26 Dec 2012 01:00:00.000 UTC Elapsed Time: 6 days (equals the repeat cycle) Step Size: 60 sec
Coordinate System	J2000:X and Z axes point toward mean vernal equinox and mean rotation axes of earth at 1 January 2000 12:00 UTC.
Orbit parameters (This propagator uses the orbital elements to set the state at epoch)	Semi-major Axis: 6991.12km Eccentricity: 7.40217e-016 Inclination:97.8436 deg Argument of Perigee: 0 deg RAAN:179.229 deg True Anomaly:2.48481e-017 deg
Mass, drag area, radiation area	Mass = 2368.89kg Drag Area $\approx$ 3.9^2 = 15.21m2 Radiation Area $\approx$ 3.9*15.6m = 60.48m2
Central body Gravity	WGS84_EGM96 21×21

	Solid Tide: Permanent tide only Use Ocean Tides: no
Atmospheric Drag	C_D = 2.2 Area/Mass Ratio ≈ 0.006m2/kg Atm. Density Model: jacchia Roberts SolarFlux/GeoMag Daily F10.7 150 Average F10.7 150 Geomagnetic Index Kp 3.0
Solar radiation pressure(SRP)	Use Spherical Model C_R = 1.0 Area/Mass Ratio ≈ 0.026m2/kg SRP Model: Spherical Shadow Model: Dual cone Use Boundary Mitigation: No
integrator	RK7(8) Step Size Control: Relative error Error Tolerance:e-14 Min Step Size: 1 sec Max Step Size: 86400 sec

The orbit propagated results using default settings are ephemeris information in the format of the position and velocity in the J2000 coordinate system, marked as $\vec{r}_0 = \lfloor x \quad y \quad z \quad v_x \quad v_y \quad v_z \rfloor_0$.When changing the parameter settings, the propagated results are $\vec{r}_i = \lfloor x \quad y \quad z \quad v_x \quad v_y \quad v_z \rfloor_i$. The difference is used to investigate the influence:

$$\Delta\vec{r} = \vec{r}_i - \vec{r}_0 \tag{24}$$

$$difference = \|\Delta\vec{r}\| \tag{25}$$

The following section will study the differences due to such factors as central body gravity, atmospheric drag, solar radiation pressure, and third body gravity and propagator integration.

5.4. Central Body Gravity

The motion of a satellite is influenced by the gravity field of multiple bodies. For the Earth orbiting satellite, the central body is the Earth. The Earth oblateness's, or bulge at the equator, causes a twisting force on satellite orbits that change various orbital elements over time. The central body gravity forces are mainly determined by gravity models, often defined in terms of series of spherical harmonic coefficients, with some maximum degree and order. Besides, the central body effect, gravity could also include solid tides and ocean tides.

5.4.1. Gravity Models

Gravity model is a file containing the central body geopotential model coefficients. Differences between gravity models are mainly reflected in their maximum degrees and orders. For example, EGM96 model contains a full set of coefficients to degree and order 360, namely 360×360; while the improved EGM2008 is 2159×2159. The standard setting of 21×21 may not be the most accuracy model but may be a suitable balance between the accuracy and the computational cost. If using the same coefficient degree and order, the differences between gravity models (listed in Table 7) are presented in Figure 13. It can be seen that the differences are relatively small, with 160m in 6 days (the orbit repeat cycle).

Table 7 Gravity models.

Gravity model	Description
EGM96	Earth Gravitational Model 1996, a geopotential model of the earth consisting of spherical harmonic coefficients complete to degree and order 360.
EGM2008	This gravitational model is complete to spherical harmonic degree and order 2159, and contains additional coefficients extending to degree 2190 and order 2159.
GGM01	GRACE Gravity Model 01. This model is based upon a preliminary analysis of 111 days of in-flight data gathered during the commissioning phase of the Gravity Recovery And Climate Experiment (GRACE) mission, which was launched on March 17, 2002.
WGS84_EGM96	Use the EGM96 coefficients with the WGS84 ellipsoid shape.
WGS84	The World Geodetic System is a standard for use in cartography, geodesy, and navigation. It comprises a standard coordinate frame for the earth, a standard spheroidal reference surface for raw altitude data, and a gravitational equipotential surface (the geoid) that defines the nominal sea level.
GGM01C	Improved earth gravity field model from GRACE.
JGM3	Joint Earth Gravity Modes denoted JGM1, JGM2, GUM3 developed by NASA's Goddard Space Flight Center in cooperation with universities and private companies.

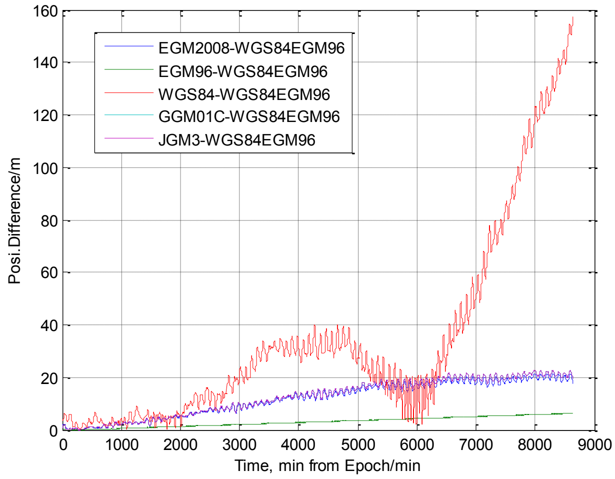

Figure 13 Differences vary with gravity model.

5.4.2. Maximum Degrees and Orders

The accuracy of a specific gravity model is determined by the maximum degree and the maximum order of geopotential coefficients to be included for central body gravity computations. The range of these values is from 0 to 90, depending on the gravity model. For instance, the standard gravity model in HPOP uses 21 ×21. Many applications use reduced gravity field orders to speed up computational processing. Figure 14-7 present the difference only considering the zonal harmonic terms, referred to as un-squared truncation. Figure 18-11 show the difference varying with the complete gravity field degree and order, referred to as squared truncation. Non-square truncations contribute half of the difference, and when the degree goes up to 7, the differences cannot be reduced significantly. While the complete gravity field is used, the 5×5 squared truncations are responsible for differences within 4000m in 6 days, the 12×12 and above make differences within 4000m, and the 17×17 and above make differences within 1400m in 6 days.

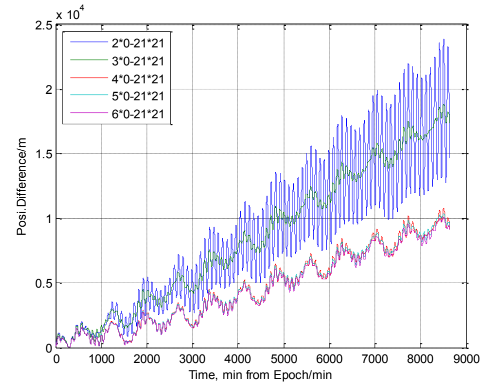

Figure 14 Difference vs. gravity field with the non-square truncation (from 2×0 to 6×0).

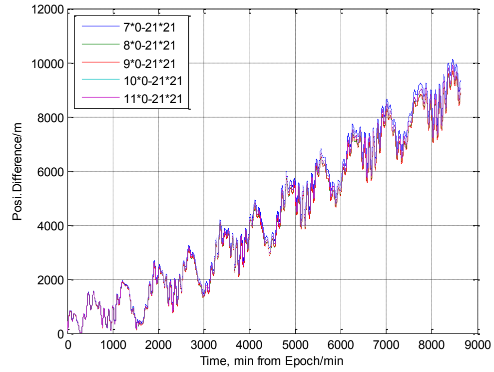

Figure 15 Difference vs. gravity field with the non-square truncation (from 7×0 to 11×0).

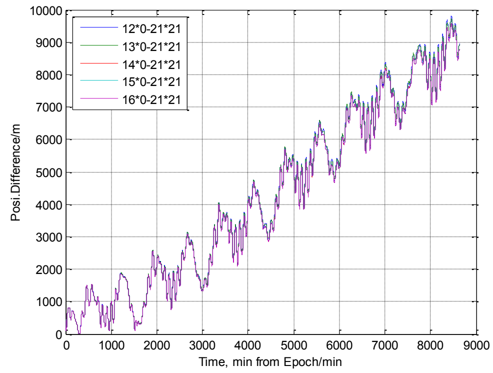

Figure 16 Difference vs. gravity field with the non-square truncation (from 12×0 to 16×0).

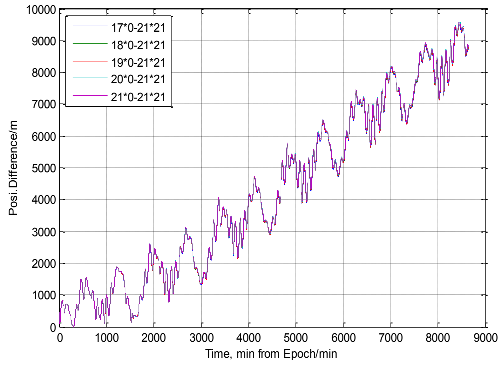

Figure 17 Difference vs. gravity field with the non-square truncation (from 17×0 to 21×0).

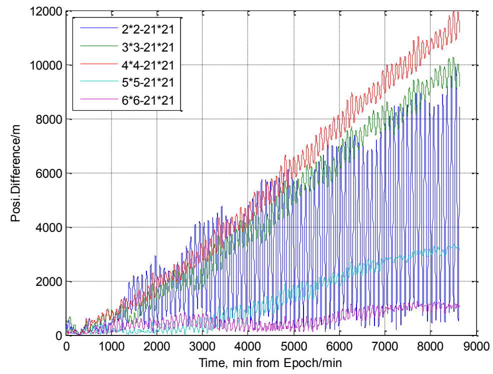

Figure 18 Difference vs. gravity field degree and order (from 2×2 to 6×6).

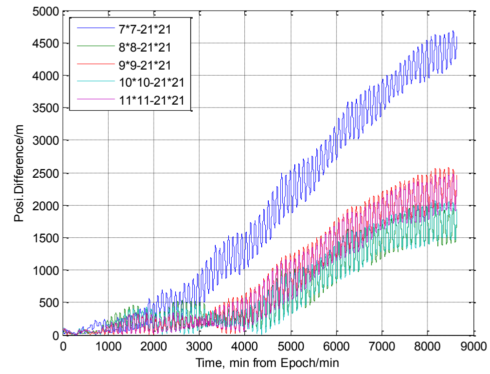

Figure 19 Difference vs. gravity field degree and order (from 7×7 to 11×11).

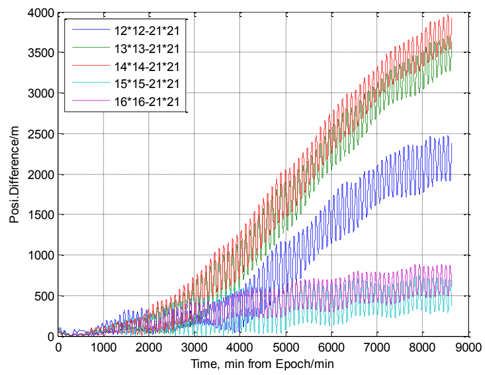

Figure 20 Difference vs. gravity field degree and order (from 12×12 to 16×16).

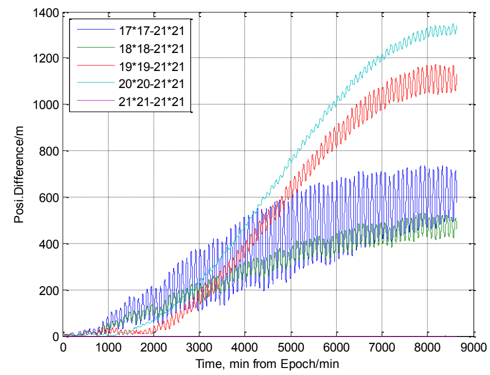

Figure 21 Difference vs. gravity field degree and order (from 17×17 to 21×21).

5.4.3. Solid Tide

The solid tide is the perturbation of the gravity field caused by the effects of solid tides. The standard setting includes only the permanent solid tides, which means includes only the permanent or time-independent tidal contribution of the solid tide model. Besides the permanent solid tides, there are other solid tide modelling contributions. Figure 22 shows the differences between the non-solid tides, the full tides and the permanent solid tides.

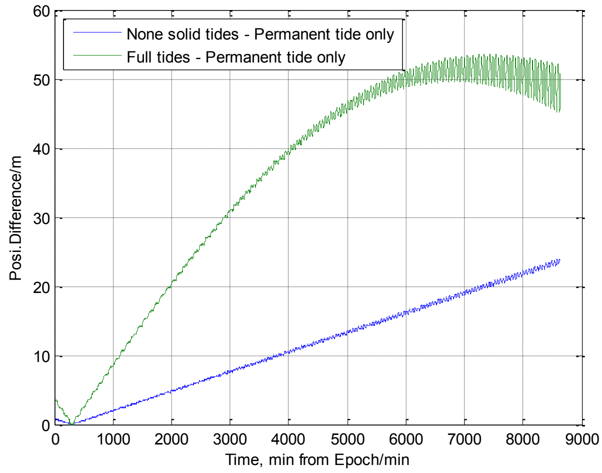

Figure 22 Difference vs. solid tides.

If the solid tides are not modelled, the differences are less than 60m in 6 days. If the full tides are modelled, the differences are within 30m in 6 days. This effect is much less, compared to the gravity models and degree/order settings.

5.4.4. Ocean Tide

Like the solid tide contribution, the ocean tide contribution is a time-consuming computation, as it computes geopotential variations of up to degree and order of 30, for over 200 tide constituents. Coefficients for the ocean tide model, based on the TOPEX mission, are provided in STK. The file contains over 1900 contributions to the geopotential field. The standard setting in HPOP is no ocean tides, as ocean tide influence is relatively small for most satellite orbit applications.

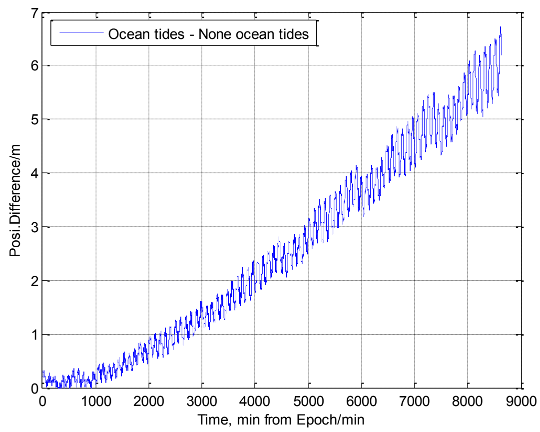

Figure 23 Difference vs. ocean tide.

The difference caused by ocean tides is less than 7m in 6 days, less than 1m in 1 day, as shown in Figure 23. Therefore, the ocean tides only need to be modelled in the case that the very precise force models (with error magnitude of <1 m) are required.

5.5. Atmospheric Drag

As discussed in Section 5.2, the atmospheric drag is the largest uncertainty when determining orbits of low altitude satellites. Figure 24 shows the differences when the atmospheric drag is not taken into account in the force model. The difference is 80km in 6 days, about 1500m in the first day.

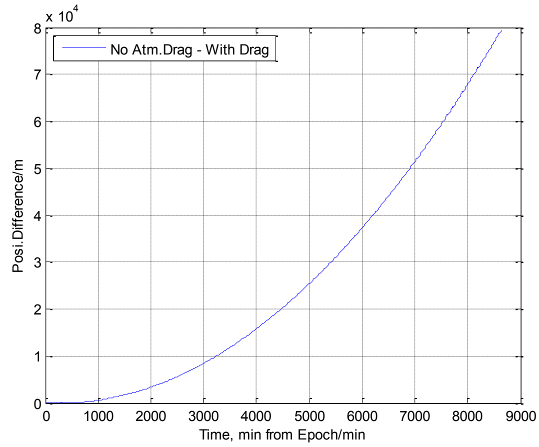

Figure 24 Differences without atmospheric drag.

The atmospheric drag is the most elusive of the force models. The drag force model provides a variety of options for modelling the atmospheric density used in the computation of atmospheric drag accelerations on the spacecraft. Consider the basic acceleration equation:

$$\vec{a}_{drag} = -\frac{1}{2}\rho \frac{c_D A}{m} v_{rel}^2 \frac{\vec{v}_{rel}}{|\vec{v}_{rel}|} \qquad (26)$$

where the density ρ typically depends on the atmospheric model and the space weather characteristics which are represented as three indices: Daily F10.7, Average F10.7 and Geomagnetic Index (Kp).

5.5.1. Atmospheric Density Models

Table 8 lists all the atmospheric density models provided in STK/HPOP, with a short description of each model.

Table 8 Atmospheric density models.

Model	Description
1976 Standard	A table look-up model based on the satellite's altitude, with a valid range of 86km - 1000 km.
Harris-Priester	Takes into account a 10.7 cm solar flux level and diurnal bulge. Valid range of 0 - 1000 km.
Jacchia 1970	The predecessor to the Jacchia 1971 model. Valid range is 90 km - 2500 km.
Jacchia 1971	Computes atmospheric density based on the composition of the atmosphere, which depends on the satellite's altitude as well as a divisional and seasonal variation. Valid range is 100km - 2500 km.
Jacchia 1960	An earlier model by Jacchia that uses the solar cycle to predict a value for the F10.7 cm flux and accounts for the effects of the diurnal bulge.
Jacchia-Roberts	Similar to Jacchia 1971 but uses analytical methods to improve performance.
CIRA 1972	Empirical model of atmospheric temperature and densities as recommended by the Committee on Space Research (COSPAR). Similar to the Jacchia 1971 model but uses numeric integration rather than interpolating polynomials for some quantities.

MSIS 1986	Empirical density model developed by Hedin based on satellite data. Finds the total density by accounting for the contribution of N2, O, O2, He, Ar and H. 1986 version, valid range of 90-1000 km.
MSISE 1990	Empirical density model developed by Hedin based on satellite data. Finds the total density by accounting for the contribution of N2, O, O2, He, Ar and H. 1990 version, valid range of 0-1000 km.
NRLMSISE 2000	Empirical density model developed by the US Naval Research Laboratory based on satellite data. Finds the total density by accounting for the contribution of N2, O, O2, He, Ar and H. Includes anomalous oxygen above 500 km. 2000 version, valid range of 0-1000 km.

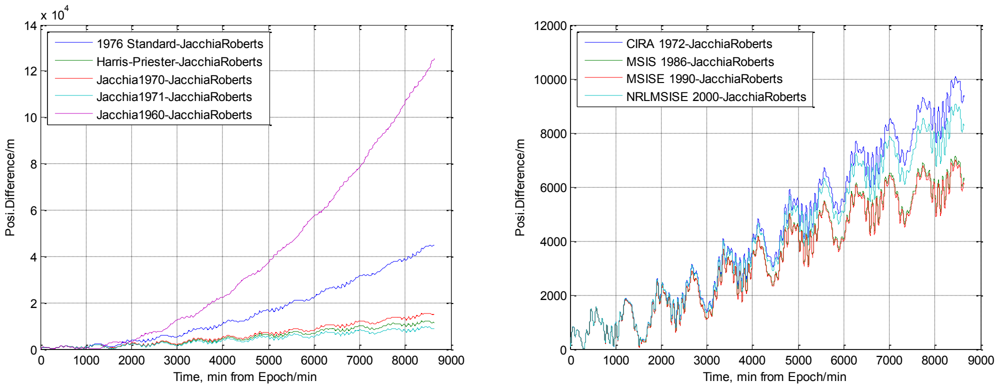

Figure 25 Differences vs. atmospheric density models.

The two plots in Figure 25 show the differences varying with atmospheric density models when the solarFlux/geomag with the standard settings are used. The default model is Jacchia-Roberts. The left plot shows the differences varying with 1976 Standard, Harris Priester, Jacchia 1970, Jacchia1971 and Jacchia1960. It can be seen that the 1976 Standard and Harris Priester do not agree with most of the models. The Jacchia 70, 71 and 60 are relatively old models compared to the models in the right figure. The newer models – CIRA1972, MSIS 1986, MSIS 1990 and NRLMSISE2000 – exhibit less difference. Using the newer models, the differences caused by the drag model selection is within 10km in 6 days, about 2000m in one day.

5.5.2. SolarFlux/GeoMag

Solar flux file is a text file containing solar flux and geomagnetic indices. A flux file contains flux data (Ap, Kp, F10.7, and Average F10.7) for each date. The F10.7 index is a measure of the noise level generated by the sun at a wavelength of 10.7cm at the earth's orbit. The global daily value of this index is measured at local noon at the Pentictin Radio Observatory in Canada. Figure 26 presents the differences varying with F10.7 (assume Daily F10.7 = Average F10.7, the standard F10.7 value = 150).

The geomagnetic index Kp is a quasi-logarithmic index of geomagnetic activity relative to an assumed quiet day curve for the recording site. Kp is a code from 0-9 that characterises magnetic activity (0 being the least active field and 9 the most active field) over a 3 hour period. Figure 27 shows the differences vary with Kp (the stand Kp value =3). Figure 26 and Figure 27 verify that the atmospheric drag is quite sensitive to space weather with a significant difference, with a magnitude of kilometres.

Garada

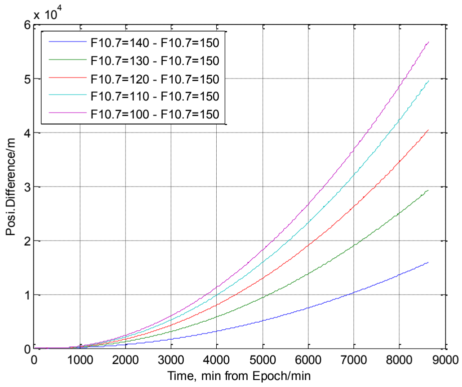

Figure 26 Differences vs. Ap F10.7.

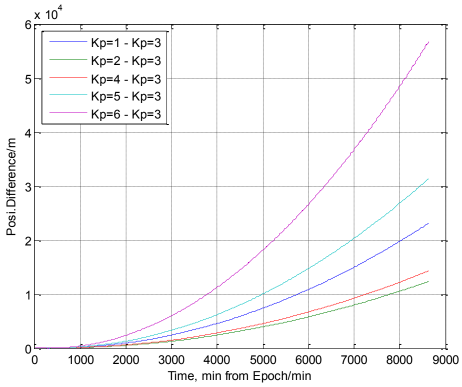

Figure 27 Differences vs. geomagnetic index Kp.

These data can also be input using the existing solar flux files. There are three types of solar flux files that can be used with STK: the Schatten Predicts, the Space Weather files and the FluxGeoMag. Schatten Predicts files are used for long-term predictions. The file contains predicted values of the monthly mean 10.7cm solar radiation flux ($F_{10.7}$) and geomagnetic index (A_p). The Space Weather file contains daily observed solar flux and geomagnetic indices, and approximately 10 years of predicted data. The stkFluxGeoMag file has been replaced by the Space Weather format. Three files are selected for comparison: SolFlx_Schatten.dat, SpaceWeather-All-v1.2.txt and stkNewFluxGeoMag.fxm. Figure 28 shows the differences varying with the selected file where the standard one is SolFlxSchatten. It can be seen that the solar flux impacts the difference heavily with a magnitude of kilometres.

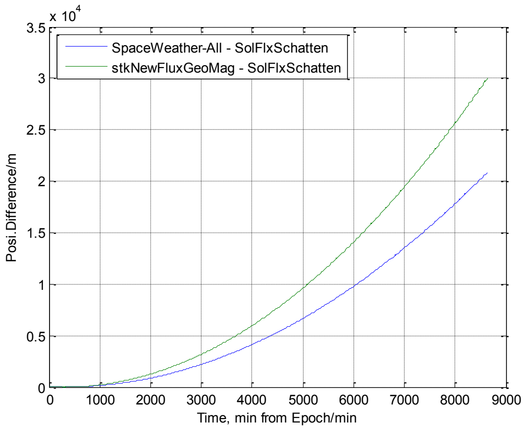

Figure 28 Difference vs. solarflux/geomag data.

5.6. Third Body Gravity

In addition to Earth gravity, the effects of gravity from a third body can be modelled. The ephemeris source for third bodies in HPOP will be inferred from the Gravitational Source settings.

In general, the inclusion of solar and lunar third body gravity contributions for Earth orbiting satellites is sufficient for accuracy in the most demanding applications. The standard settings include moon and sun, Figure 29 shows the results of comparing the standard to the below settings: 1) moon only,2) sun only, 3) no third body, 4) with moon, sun and Jupiter, and 5) with moon, sun, Jupiter and Venus. Generally, as Garada is in Low Earth Orbit, the third body gravity forces make differences within 600m in 6 days. From the aspect of third body influences, the moon has a larger impact than the sun, and the planets Jupiter and the Venus can be neglected.

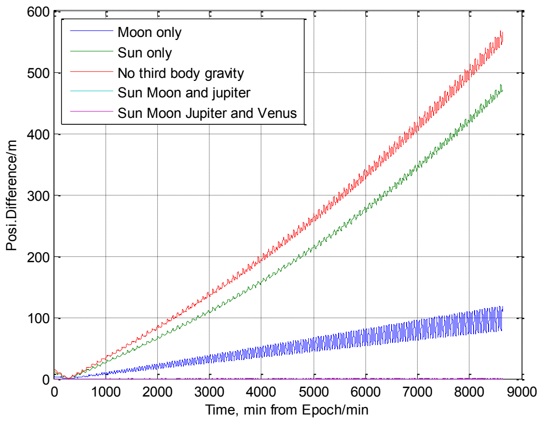

Figure 29 Differences vs. third body gravity.

5.7. Solar Radiation Pressure

The solar radiation pressure is determined by the satellite area exposed to the sun, satellite solar radiation coefficient, radiation model and shadow model. Figure 30 verifies that the solar radiation pressure is relatively a small effect with the difference within 60m in 6 days. Shadow model is used to determine the lighting condition of the satellite. The types when shadow is used in STK are listed in Table 9 with short descriptions. The differences using shadow models are plotted in Figure 31. The standard setting for solar radiation pressure uses a dual cone solar radiation shadow.

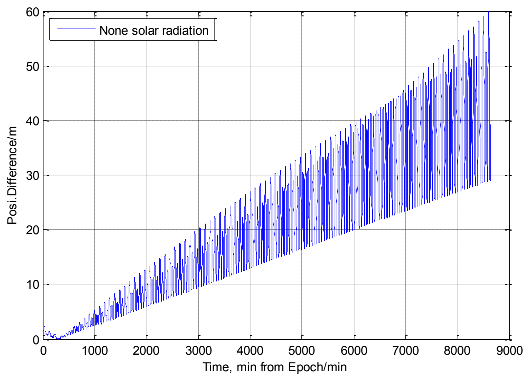

Figure 30 Difference without solar radiation pressure.

Table 9 Solar shadow model.

Model	Description
None.	Choosing this option turns off all shadowing of the satellite.
Cylindrical	The cylindrical model assumes the sun to be at infinite distance so that all light coming from the sun moves in a direction parallel to the sun to satellite vector.
Dual Cone	The dual cone model uses the actual size and distance of the sun to model regions of full, partial (penumbra) and zero (umbra) sunlight. The visible fraction of the solar disk is used to compute the acceleration during penumbra.

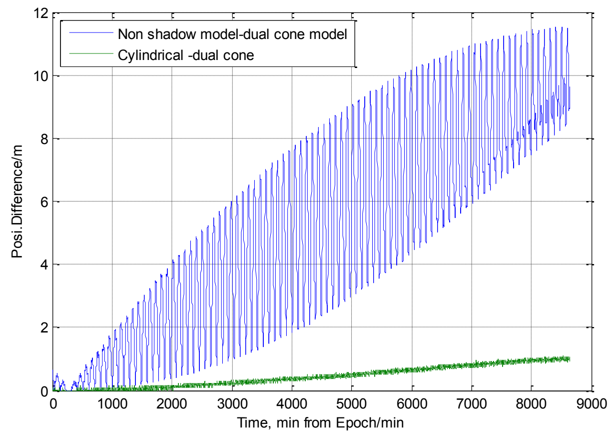

Figure 31 Differences vs. solar radiation settings.

5.8. Propagator Integrator

The integrator is configured by defining the formulation of the equations of motion and the numerical integration technique to be used during orbit propagation. Available integration techniques include the Runge-Kutta-Fehlberg method of order4-5 (RK 4(5)) and order7-8(RK 7(8)), the Burlirsch-Stoer method and the Gauss-Jackson method of order12. RK 4(5) has no error control for the integration step size; RK 7(8) has 8th order error control for the integration step size. RK7 (8) allows good accuracy but results in increased computational requirements for the HPOP model. RK4 (5) has less computational requirement than RK7 (8), however its accuracy is significantly reduced.

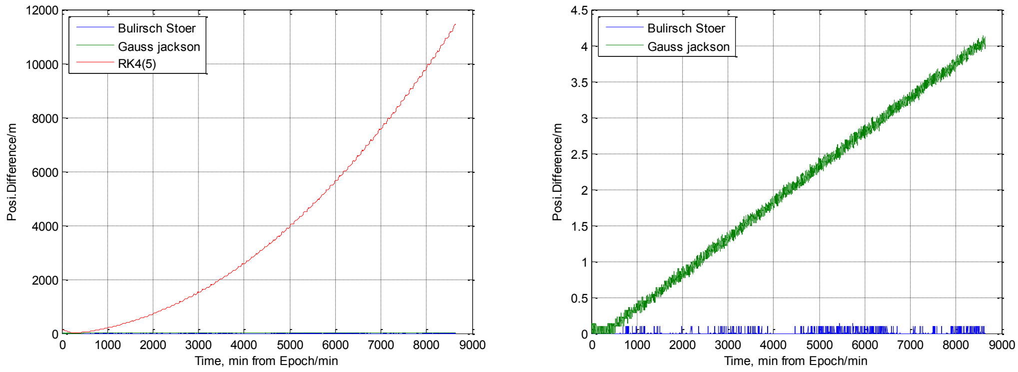

Figure 32 Differences vs. integrator.

In Figure 32, the left plot shows that RK 4(5) causes an increasing error up to 60km in 6 days, and less than 2km in 1 day. The right figure zooms in the differences caused by Burlirsch-Stoer and Gauss-Jackson which display quite small differences compared to RK 7(8).

5.9. Orbit Force Analysis Summary

Based on the orbit force sensitivity study for Garada, the magnitude of each orbit force impacts in the orbit cycle and some conclusions can be drawn in Table 10.

Table 10 The magnitude of the orbit force impacts

<table>
<tr><td colspan="3">Force Type/ Force parameters</td><td>Error Magnitude</td><td>Conclusions</td></tr>
<tr><td rowspan="7">Central body graivity</td><td colspan="2">Gravity Model</td><td>160m</td><td rowspan="7">The central body gravity contributes the largest effect on satellite orbits. New updated gravity model causes limited differences; and the maximum degree and order is the main factor to be considered. Therefore, the gravity fields should not be truncated for precise operations. Solid tides and ocean tides contribute very small effects to orbits, and would be considered only for precise operations.</td></tr>
<tr><td rowspan="4">Degree and order</td><td>2×2~6×6</td><td>12000m</td></tr>
<tr><td>7×7~11×11</td><td>5000m</td></tr>
<tr><td>12×12~16×16</td><td>4000m</td></tr>
<tr><td>17×17~21×21</td><td>1400m</td></tr>
<tr><td colspan="2">Solid tides</td><td>60m</td></tr>
<tr><td colspan="2">Ocean tides</td><td>7m</td></tr>
<tr><td rowspan="2">Atmospheric drag</td><td colspan="2">Drag model</td><td>12000m (the new models)</td><td rowspan="2">The atmospheric drag is generally the second largest effect. The atmospheric drag is affected by the space weather more than that of the drag model. Therefore, it is important to model the space weather and to choose an accurate space weather file. Note that new Earth gravity models and new atmospheric models are continually being improved.</td></tr>
<tr><td colspan="2">Space weather file</td><td>35000m</td></tr>
<tr><td colspan="3">Third body gravity</td><td>600m</td><td>The third body effect is of a magnitude of hundreds of metres in the orbit repeat cycle</td></tr>
<tr><td colspan="3">Solar radiation pressure</td><td>60m</td><td>The solar radiation pressure is a small force.</td></tr>
<tr><td colspan="3">Propagator integrator</td><td>12000m (with RK4(5))
4.5m (without RK4(5))</td><td>Integration techniques contribute generally small errors to the propagation process unless the RK 4(5) is chosen.</td></tr>
</table>

6. LAUNCH VEHICLE SELECTION

This section first presents the criteria to be used for evaluating launch vehicle options. The reference comes from two primary sources [5]. One source is the results of a user satisfaction survey conducted by a U.S. launch vehicle service provider. The other is derived from eight interviews of commercial communication satellite owners and operators. These sources identify the factors users considered most important in evaluating launch vehicles. While the particular requirements and resources of satellite owners and operators ultimately determine the launch vehicle selection, the survey reveals several common factors, with vehicle reliability, performance, suitability, and price topping the list. Other factors are availability and schedule, technology transfer safeguards, user relationship and partnership, as well as terms and conditions. The final decision must be made on the basis of not one but many factors, making trade-offs to achieve an optimal satisfaction of technical, programmatic, financial and contractual factors.

According to WP1's overall mission design and WP3's antenna design, the Garada antenna will be very large, and this large antenna will drive the mission towards a launch vehicle with the size of a Falcon-9. This is therefore selected as the candidate launcher vehicle. Falcon-9 is a rocket-powered spaceflight launch system designed and manufactured by Space Exploration Technologies (SpaceX www.spacex.com), headquartered in Hawthorne, California. It can take a 7451kg payload into a 600km sun-synchronous orbit. The launch cost is around US$49-54 million.

Launch procurers rarely confine themselves to a single launcher but prefer to diversify their choices. Thus the Garada mission should consider backup launcher options. This section has sought other candidate launch vehicles listed in the "International reference guide to space launch systems (4th edition)"[11]. A Matlab-based software was developed for the orbit selection according to the Garada satellite dimensions: 1) diameter, 2) height, and 3) mass. The U.S. Delta IV-M and the European Ariane 5 could also be used to launch Garada. These two launchers have good reliability and performance; however their costs are much higher than the Falcon-9 (greater than US$100 million). It should be noted that launch service prices depend on mission specific services and options, the terms and conditions of the contract (such as payment schedule, insurance, etc.), market conditions at the time of purchase, and a variety of other factors. Hence the price of a launch vehicle may be negotiated. China's Long March rockets and Russian rockets are cheaper than American rockets, but certain payloads may not be permitted to be launched by China or Russia. The Falcon-9 comparably priced to the Chinese and Russian rockets, hence the Falcon-9 is the top candidate for the launch system.

6.1. Launch Vehicle Selection Criteria

While established launch companies in the United States, France, Russia, and China work to introduce increasingly capable versions of their rockets, new player such as Japan, India and Israel continue to make headway in the development of their own launchers. This section lists seven selection factors often used for launch system selection.

6.1.1. Reliability of Launchers

A launcher's reliability should be such that there is a low risk of technical failure based on a history of prior mission success. This is one of the most important factors to be considered in evaluating a launch vehicle option.

Launcher reliability is critical so as to maximise the chances that payloads will reach orbit. In the case of a new satellite venture, a launch failure could substantially delay the time for deployment and operation. For a commercial mission, technical conservatism typically prevails over other factors. For government or private technology demonstration missions whose failures would not significantly affect a program of business, there may be an inclination to weigh reliability and cost of launchers more equally.

Users tend to place great emphasis on whether a launch vehicle is “proven”, that is, that it has a good record of launch successes. The success history of launcher components also needs to be considered as some indicated their willingness to fly payloads on new vehicle models using components with good records of success. It should be noted that of the world’s current launch vehicle families, 75% have had at least one failure in the first three flights.

Choosing a vehicle with high reliability translates into reduced insurance rates for users.

6.1.2. Performance and Suitability of Launchers

Launch vehicle performance and suitability to carry the satellites is one of the most important factors in the evaluation of launchers.

Performance of vehicle refers to its capability of lifting a certain payload mass to a desired altitude and its ability to insert it into the proper orbit. Launching a satellite into space but failing to deliver it into the correct orbit would effectively render it useless. Suitability refers to both the vehicle’s compatibility with various types of payloads and its payload margins. A vehicle with wide margins is often desirable because more changes can then be made to the satellite design without affecting the satellite’s ability to be transported on that vehicle.

The payload weight a vehicle can carry is a big factor. For instance, paying for a large vehicle could offset the costs of having to miniaturise satellite components in order to ensure the satellite fits on smaller vehicles.

6.1.3. Launcher Price

The price of a launch vehicle is one of the main factors in launcher selection. In some cases launch prices are variable/uncertain and subject to negotiation. Launch service prices depend on mission specific services and options, the terms and conditions of the contract (such as payment schedule, issuance, etc.), market conditions at the time of purchase, and a variety of other factors. Therefore, price ranges shown in 8.1 should be considered as approximate values only. The responsible business development organisation should be contacted directly for price quotes.

6.1.4. Availability and Schedule

In some cases it is important to choose a launcher whose availability is compatible with the desired launch schedules. For instance the Garada mission has two satellites. If it is required to launch the two satellites within a short period of each other, it is important to find a launch provider (or providers) that could meet the requirements. Consideration includes which launch vehicles can launch several times per year or can meet a demanding timetable, whether to use more than one provider, and the turnaround times and abilities to satisfy owner’s requests to change a launch date. Considerations also include that some launcher providers sometimes give priority to government needs; and launching with other spacecraft has further cost saving potential. However launching single satellites has more control over launch schedule.

6.1.5. Technology Transfer Safeguards

As the recommenced launch vehicle is the Falcon 9[6], the U.S technology transfer safeguards become a major factor in the evaluation of U.S. launch companies. Before a U.S. launch company can discuss the technical details of a business deal with a foreign satellite owner, it must obtain a marketing licence from the U.S. State Department. The launch company needs to get a government licence Technical Assistance Agreement to work with a foreign company on matters such as integrating the company's payload onto the vehicle. These licences can take several months to procure. As a result working with a U.S. launch provider presents many difficulties for an overseas satellite owner. Whether the U.S. launch provider will be able to secure the appropriate licences and whether the licensing process will affect their ability to launch when desired needs consideration.

6.1.6. User Relations and Partnerships

The quality of the relationship established with a launch service provider also has an influence on vehicle selection. Professional providers will be sensitive and respond to the customer's needs. It is critical that a good working relationship be established during both the negotiations and procurement stages. The ease of communications with launch providers over national and cultural divides is also important. Good rapport between the satellite's manufacturer and potential launch provider is also desirable.

Repeat business can enable both the satellite operator and launch provider to offer each other mutual benefits. Such partnerships can allow the partners to offer each other preferred prices for products and services. The potential for engaging in future collaborative work with the launch company is often a major consideration for satellite owners as well.

6.1.7. Terms and Conditions

Terms and conditions include issues such as payment schedule, payload integration and launch schedule, liability, and contract termination. The issue of liability is particularly important as satellite owners expect a launch company to share the financial risk associated with a launch failure. Some customers expect a launch company to offer a replacement launch at little or no cost, share in the loss of revenue due to their satellite's inability to reach orbit, and/or shoulder the cost of higher insurance premiums on future launches.

6.2. Satellite Launch System

A launch system includes the launch vehicle, the launch pad/launch site and other infrastructure.

6.2.1. Space Rocket Launch Sites

Several countries have the capability to design and build satellites but are unable to launch them, instead relying on foreign launch services. Here is a list of countries with an independent capability to place satellite in orbit, including production of the necessary launch vehicle. These countries are Russian, U.S., France, Japan, China, U.K., India, Israel, Ukraine, Iran, South Korea and North Korea. Australia has developed her own launchers, but has not had a successful launch of SSO satellites. Garada will choose an overseas launch site. Figure 33 shows the space rocket launch sites over the world[7]: Cape Canaveral & Vandenberg (USA), Baikonur (administered by the Russian Federation), Plesetsk (Russia), Kourou (French Guiana), Tanegashima (Japan), Jiuquan and Xichang (China) and Sriharikota Island (India).

1 - Vandenberg	7 - Hammaguir	12 - Palmachim	17 - Xichang
2 - Edwards	8 - Torrejon	13 - San Marco	18 - Taiyuan
3 - Wallops Island	9 - Andoya	14 - Baikonur	19 - Svobodny
4 - Cape Canaveral	10 - Plesetsk	15 - Sriharikota	20 - Kagoshima
5 - Kourou	11 - Kapustin Yar	16 - Jiuquan	21 - Tanegashima
6 - Alcantara			22 - Woomera

Figure 33 Space rocket launch sites.

The initial inclination of an orbit is constrained to be greater than or equal to the launch latitude. It is easy to launch into an inclination higher than a launch site's latitude. At the extremes, a launch site located on the equator can launch directly into any desired inclination, while a hypothetical launch site at the North or South Pole would only be able to launch into polar orbits. As the Garada satellite is intended for an SSO, it can in theory be launched from any launch site.

6.2.2. Space Rocket Launch Vehicle

Space launch vehicles can place a certain maximum payload mass into orbit at a given altitude. This payload mass consists of the spacecraft structure and systems, instruments, and on-board manoeuvring fuel. Appendix 8.1 lists the vehicles and their general performance characteristics such as payload mass to orbit, cost, first flight and launch site. There are total of 69 launchers listed and the U.S. has the largest number of launch vehicles (see Figure 34).

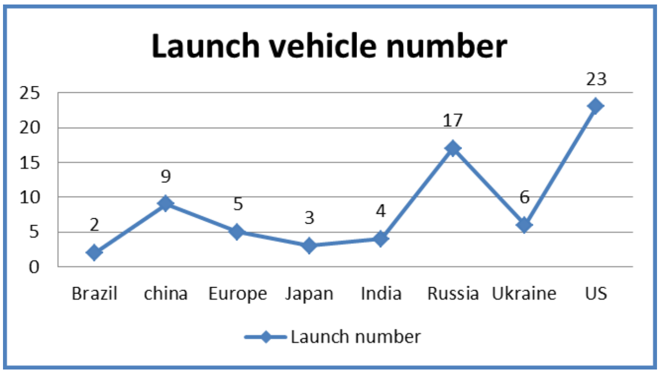

Figure 34 Launch vehicle number of each space force.

An important reference for the selection of launch system is the User's Guide. It is a planning document which is provided for potential and current users, and is not intended for detailed design use. In general the document presents the frequently-reported characteristics of launch vehicles, including:

1) History of the launch vehicle (the nation or space agency responsible for the launch, and the company or consortium that manufactures and launches the vehicle).
2) Vehicle overview (structure, propulsion, avionics, etc.),

3) Facilities overview (headquarter, space launchpad, test facility, government outreach and legal affairs).
4) General performance capability (performance capability for LEO/polar/SSO, separation accuracy, mission accuracy).
5) General payload information (payload fairing description such as size and shape, separation, collision avoidance, payload thermal, humidity, cleanliness, launch and fight environments).
6) Launch operations (launch control organisation, spacecraft transport to launch site, plans and schedules).
7) Safety (safety requirements, hazardous system and waivers).

Of the above seven points, 4) and 5) relate to the vital technical performance of the launch.

6.2.3. Falcon-9 Performance

Garada CAD modelling indicates that the satellite can be accommodated in a Falcon-9 launcher. It is a rocket-powered spaceflight launch system designed and manufactured by Space Exploration Technologies (SpaceX www.spacex.com), headquartered in Hawthorne, California. The base Falcon-9 is a two-stage, LOX/RP-1 (Liquid oxygen/Rocket Propellant -1) powered launch vehicle. It is currently the only active rocket of the Falcon rocket family. Falcon-9 v1.0 is 54.3m in height, 3.6m in diameter and 333,400kg in mass. First launch of Falcon-9 was from Cape Canaveral on June 4, 2010. As of March 2013, SpaceX has made five launches of the Falcon-9 since 2010, and all five have successfully delivered their payloads to LEO. Five main characteristics of Falcon-9 are described in sections 1) to 6).

1) Reliability of Falcon-9

After the successful launch of the CRS-2 mission on March 1, 2013, Falcon 9 v1.0 boasts a perfect record - five successful launches in five attempts. Future launches of the rocket will be in the v1.1 configuration. Falcon-9 has triple redundant flight computers and inertial navigation, with a GPS overlay for additional orbit insertion accuracy [8].

2) Launch Vehicle Lift Capability of Falcon-9

Figure 35 and Table 11 shows the performance for launching into an SSO. A typical payload in the Falcon-9 class is below 6800kg in LEO, while it is below 5300kg for an SSO. The Garada satellite mass is below 3000kg, and the Falcon-9 could launch 7451kg into a 600km SSO. Therefore Falcon-9 meets the launch requirements.

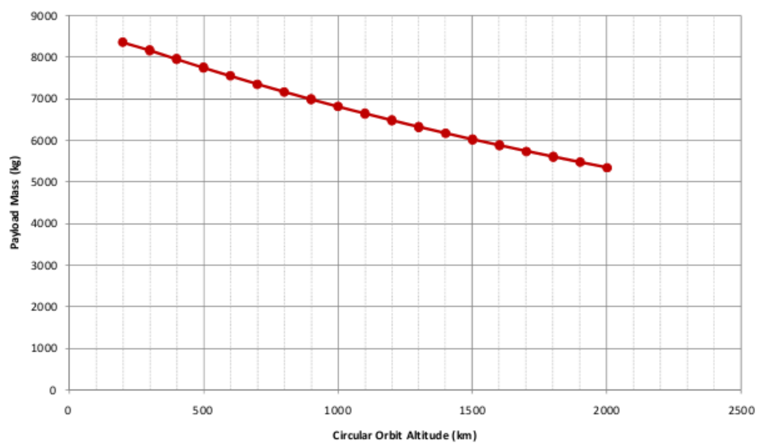

Figure 35 Falcon 9 Block 2 performance for SSO.

Table 11 Falcon 9 Block 2 performance for SSO.

Circular Orbit Altitude (km)	Inclination (degrees)	Payload Mass (kg)
200	96.3	8351
300	96.7	8159
400	97.0	7949
500	97.4	7742
600	97.8	7541
700	98.2	7348
800	98.6	7162
900	99.0	6981
1000	99.5	6807

Circular Orbit Altitude (km)	Inclination (degrees)	Payload Mass (kg)
1100	99.9	6639
1200	100.4	6476
1300	100.9	6319
1400	101.4	6166
1500	102.0	6017
1600	102.5	5874
1700	103.1	5735
1800	103.7	5600
1900	104.3	5468
2000	104.9	5340

3) Fairing Size of Falcon-9

Garada antenna is about 4m in diameter and 15.6m in height. Due to the folding design, the height in the fairing is 7.8m. Figure 36 shows the standard Falcon-9 fairing, and the Garada satellite could be accommodated in this fairing.

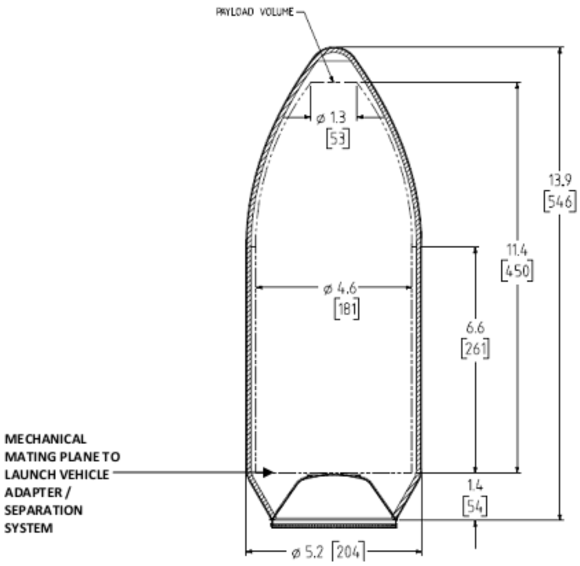

Figure 36 Falcon 9 standard fairing and dynamic envelop, metres [inches].

4) Acceleration and RF Environment of Falcon-9

During flight, the payload will experience a range of axial and lateral accelerations. Axial acceleration is determined by the vehicle thrust history and drag, while maximum lateral acceleration is primarily determined by wind gusts, engine gimbal manoeuvres, first stage engine shutdowns, and other short-duration events. The design load factors provided are expected to be conservative for a payload with the following basic characteristics: a fundamental bending mode greater than 10Hz, a fundamental axial mode greater than 25Hz, and a mass between 1360 to 9070kg. Actual spacecraft loads, accelerations, and deflections are a function of both the launch vehicle and payload structural dynamic properties and can only be accurately determined via a coupled loads analysis.

The Radio Frequency (RF) environment must be quiet enough to ensure that spacecraft materials or components sensitive to RF interference are compatible with both the launch pad environment and the RF environment during flight. The spacecraft RF characteristics should satisfy the limitations shown in Figure 37.

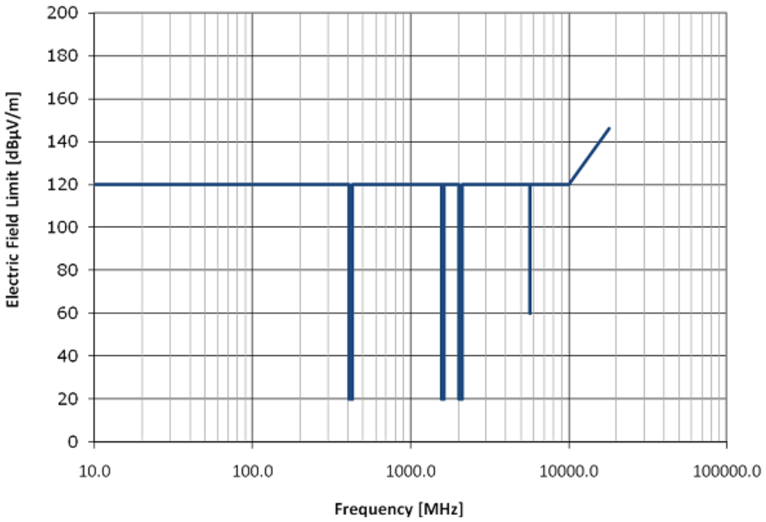

Figure 37 Falcon 9 worst case radiated environment.

5) Launch Sites of Falcon-9

As of November 2012, Launch Complex 40 at Cape Canaveral Air Force Station is the only active Falcon-9 launch site. A second site for polar-orbit launches is under development at SLC-4 of Vandenberg Air Force Base. A third site, intended solely for commercial launches, is currently being analysed, with possible locations in Texas, Florida and Puerto Rico [9].

6) Launch Integration Process of Falcon-9

The standard launch integration process stars from contract signing. For Falcon-9, 18 months or more are typically required from contract to final launch. A standard launch process is shown below.

Before launch	
18 months or more:	Contract signing and authority to proceed: ○ Estimated payload mass, volume, mission, operations and interface requirement ○ Safety information ○ Mission analysis summary provided to the user
16 months	Final payload design, including: mass, volume, structural characteristics, mission, operations, and interface requirements
4 months	Payload readiness review for range safety: ○ Launch site operations plan ○ Hazard analyses
3 months	Verification: ○ Review of payload test data verifying compatibility with launch environments ○ Coupled payload and launch loads analysis completed ○ Mission safety approval

4-6 weeks	System readiness review Pre-shipment review Verify launch site, range, regulatory agencies, launch vehicle, payload, people and paper are all in place and ready to begin launch campaign
2-4 weeks	Payload arrival at launch location
8-9 days	Payload encapsulation and mate to launch vehicle
7 days	Flight readiness Review
1 day	Launch readiness review
After launch	
4 hours	Post launch reports (quick look)
4 weeks	Post launch report (final report)

6.3. Backup Launch Vehicles

Launch procurers rarely confine themselves to a single launcher, but prefer to diversify their choices. Thus, Garada mission will consider a backup launcher in spite of attractiveness of Falcon-9[10] which is recommend by Astrium. In order to find other candidate launch vehicles for the Garada mission, this section has studied launch vehicles listed in the “International reference guide to space launch systems (4th edition)”[11].

By comparison, U.S. Delta IV-M [12] and European Ariane 5 [13] are also capable of launching the Garada satellite(s). Delta IV Medium can lift 6832kg to an SSO, and the Ariane5G can lift 9500kg to an SSO. Both of these launchers have good reliability and performance, however the costs are higher than for the Falcon-9. The Delta IV-M costs US$138 million to launch up to 11700 kg to LEO, while the Ariane 5 costs US$180 million to launch up to 16000kg to LEO [14]. As mentioned in section 6.1.3, the responsible business development organisation should be contacted directly for actual price quotes.

7. ANALYSIS TOOLS

Satellite Tool Kit (STK) and MATLAB were used to plan and evaluate the satellite orbit.

STK is a leading commercial off-the-shelf analysis tool used by the aerospace industry. Specifically, a scripting environment where Matlab and STK are used in combination was developed. Using the orbit propagator in the STK, coupled to a STK/Matlab interface, a software tool was developed to analyse the performance of the proposed orbit design. In this mode the STK software development kits are used as an "engine". Matlab uses the COM interface capability of the STK/Integration module to send Connect commands directly to STK.

All orbital calculations in Section 4 have been performed using Matlab software and verified using STK/Coverage. STK/Coverage analyses when and how well regions on or above the Earth's surface are covered by mission assets (e.g. SAR). Simply, it determines which area on the ground can be seen from a satellite flying over terrain.The perturbation analysis tool used in Section 5 is STK/HPOP, which is one of the legacy programmes used to study orbit propagation. HPOP propagates the orbit by numerically integrating the equations of motion. HPOP allows various different force modelling effects to be included as well as permitting the use of different numerical integration algorithms. This sensitivity study describes the orbit propagator performance using different force modelling settings. This research reveals the differences contributed by each orbit force so that to obtain a baseline to propagate an orbit at a certain level of accuracy. In Section 6, Matlab-based software was developed for launcher vehicle selection according to the Garada satellite diameter, height and mass.

One of the products of WP7 is a set of software tools which could be utilised in orbit modelling, coverage analysis, and launch vehicle selection, etc., for the future Earth satellite projects.

8. APPENDIX

8.1. List of Launch Vehicles

This list is re-organised from Reference [11].

Nation	Vehicle	Performance (kg)			Cost (million)	First Flight	Launch Site(s)
		Leo Maximum	SSO	GTO			
Brazil	VLS-1	380	80		$8	1997	Alcantara
	VLM	100	18		$4	TBD	Alcantara
China	LM-2C,2C/SD,2C/CTS	4400	1600	1400	$20-25	1975	Taiyuan Jiuquan Xichang
	LM-2E,2E/ETS	9500		3500	?	1990	Jiuquan Xichang
	LM-3	?	?	1500	$35-40	1984	Xichang
	LM-3A	6000	?	2600	$45-55	1994	Xichang
	LM-3B	11200	6000	5100	$50-70	1996	Xichang
	LM-3C	9100	?	3800	?	?	Xichang
	LM-4B	?	2800	-	$25-35	1999	Taiyuan Jiuquan
Europe	Vega	?	1395	-	$20	2006	CSG(Kourou)
	Ariane 5G	?	9500	6700	$125-155	1996	CSG(Kourou)
	Ariane 5ECA	?	?	10050	$125-155	2002	CSG(Kourou)
	Ariane 5ES	?	?	7575	$125-155	2005	CSG(Kourou)
	Ariane 5ECB	?	?	12000	$125-155	TBD	CSG(Kourou)
Japan	H-IIA 202	9940	4350	4100	$70	2001	Tanegashima
	H-IIA 204	?	?	5800	$83	?	Tanegashima
	M-V	1900	960	1280	$557	1997	Tanegashima
India	PSLV	3700	1350	1050	$15-17	1993	Satish Dhawan
	GSLV Mark I	5000	2000	1900	$35	2001	Satish Dhawan
	GSLV Mark II	5000	2000	2100	$35	2005?	Satish Dhawan
Russia	Angara 1.1	2000	?	--	?	TBD	Plesetsk
	Angara 1.2	3700	?	--	?	TBD	Plesetsk
	Angara A3	14000	?	2500	?	2006	Plesetsk
	Angara A5	24500	?	6400	?	2600	Plesetsk
	Kosmos 3M	1500	775	--	$12	1967	Plesetsk Kapustin, Yar
	Proton K/Block DM	19760	3620	4930	Negotiable	1967	Baikonur
	Proton M/Breeze M	21000	?	5500	Negotiable	2001	Baikonur
	Rockot	1950	1000	--	$12-15	1994	Plesetsk
	Shtil-1	140	--	--	$1.4-2.1	1998	Delphin Submarine

	Shtil-2	220	200	--	3-4.5	TBD	Kalmar Submarine
	Volna	180	40	--	1-1.5	2004	Kalmar submarine
	Star-1	632	167	--	$9	1993	Svobodny Plesetsk
	Strela	1560	700	1660	$10.5	2003	Svobodny Baikonur
	Soyuz U	7000	4300	1660	$30-50	1973	Baikonur Plesetsk
	Soyus FG	7000	4300	--	$30-50	2001	Baikonur Plesetsk
	Molniya M	3700	1500	--	$30-40	1960	Plesetsk
	Dnepr-1	--	300	--	$8-11	1999	Baikonur
Ukraine	Cyclone 2	3350	?	--	$20-25	1967	Baikonur
	Cyclone 3	4100	--	--	$20-25	1977	Blesetsk
	Cyclone 2K	2750	1500	?	?	2004	Baikonur
	Cyclone 4	5860	3800	1560	?	2006	Alcantara
	Zenit 2	13920	4900	--	?	1985	Baikonur
	Zenit 3SL/3SLB	--	--	6066	Negotiable	1999	See launch Odyssey, Baikonur
USA	Athena I	820	360	--	$40-50	1995	Cape Canaveral, Kodiak
	Athena II	2065	1165	590	$40-50	1998	Cape Canaveral, Kodiak
	Atlas IIAS	8618	--	3179	Negotiable	1993	Cape Canaveral, Vandenberg
	Atlas IIIA	8640	--	4037	Negotiable	2000	Cape Canaveral
	Atlas IIIB	10759	--	4119	Negotiable	2002	Cape Canaveral, Kodiak
	Atlas V 400	12500	--	4950	Negotiable	2002	Cape Canaveral
	Atlas V 500	20652	--	8670	Negotiable	2003	Cape Canaveral
	Delta II	5120	3186	1841	Negotiable	1990	Cape Canaveral, Vandenberg
	Delta IV Medium	8870	6832	3934	Negotiable	2002	Cape Canaveral, Vandenberg
	Delta IV Medium+	13327	10863	6400	Negotiable	2002	Cape Canaveral, Vandenberg

	Delta IV Heavy	23260	19665	12369	Negotiable	2004	Cape Canaveral, Vandenberg
	Falcon I	668	408	--	$5.9	2004	Cape Canaveral, Vandenberg
	Falcon V	5040	3173	1500	$12	2005	Cape Canaveral, Vandenberg
	K-1	4600	1250	1570	$17	?	Woonera, Nevada Test Site
	Minotaur	607	317	--	$12-20	2000	Vandenberg, Others
	Pegasus XL	443	190	--	$15-25	1994	Vandenberg, Wallops, Cape Canaveral, Others
	Commercial Taurus	1370	720	495	$25-47	1998	Vandenberg, Others
	Taurus XL	1590	860	557	$25-47	2004	Vandenberg, Others
	Titan II	1900	1100	--	$30-40	1988	Vandenberg
	Titan IVB	21680	--	--	$350-450	1997	CapeCanavral, Vandenberg
	Space Shuttle	28800	--	--	$450-750	1981	Kennedy Space Center
	Scorpius	314	125	--	$2.9	2006	Vandenberg, Others
	Falcon 9	6620	5300	--	$ 49-54 [15]	2006	Vandenberg

--: not applicable, not present

? : information not available or data shown is uncertain

Cost: launch service price or cost information was requested from the responsible organisation for each launch system. If the information was not provided, an estimated cost or price range is provide by the Office of the Federal Aviation Administration Associate Administrator for Commercial Space Transportation based on open source data. In some cases launch prices are too variable or too uncertain to provide an estimate, in which case only “negotiable” is indicated. Prices and costs are listed using the currency in which the value was originally quoted. A conversion to U.S. dollars is attempted if the values were not quoted in dollars.

8.2. Small Satellite Orbit Lifetime Analysis

As indicated in Section 2, WP7 studies the orbit lifetime analysis for small satellite as WP1 considered using small satellite constellation for flood mapping. This section will describe the study results. This section presents a definition of the magnitude of each of these factors, and attempts to estimate the uncertainty in lifetime.

The prediction of satellite lifetimes depends upon a knowledge of the initial satellite orbital parameters, the satellite mass to cross-sectional area (in the direction of travel), and a knowledge of the upper atmospheric density and how this responds to space environmental parameters which must also be predicted.

8.2.1. Method for Lifetime Prediction

The reason for the computation of orbit lifetime is challenging as there is much uncertainty in the relevant parameters. STK is utilised to predict the satellite lifetime, based on examination of such issues as initial orbital parameters, atmospheric density model, and satellite physical characteristics such as drag coefficient, mass, area and solar flux. STK has lifetime commands that can be executed through connect. The lifetime module can be set up and perform calculations that predict the lifetime of a satellite. User inputs include the satellite's physical characteristics as well as solar flux and planetary geomagnetic index information. Note that vehicle attitude stabilisation is not considered in this research phase.

8.2.2. Sensitivity of Lifetime to Design Parameters

1) Sensitivity of lifetime to initial satellite orbital parameters

The initial state of the satellite is described by Keplerian elements: semi-major axis of the orbit, eccentricity, orbital inclination, right ascension of the ascending node(RAAN), argument of perigee and mean anomaly (a_r , e_r , i_r , Ω_r , ω_r M_r ,). The two orbital elements, a_r and e_r describe the size and shape of the orbit. The three elements i_r , Ω_r and ω_r describe the orientation of the orbit. The orbital element M_r describes the location of the satellite.

Altitude is the main factor impacting lifetime as this parameter is closely related to atmosphere density. There are more air molecules near the surface of the Earth than higher in the atmosphere. Therefore, high altitude satellites have longer lifetime than lower altitude satellites. In Figure 38, as the satellite altitude changes from 400 to 600km, the lifetime increases from 2 to 20 years.

The eccentricity affects the lifetime because it impacts on the altitude perigee and the atmospheric drag (see Figure 39). For the Garada Satellite Mission, the orbit is designed to be a circular Low Earth Orbit (LEO). The orbit eccentricity has to be controlled and kept close to 0.

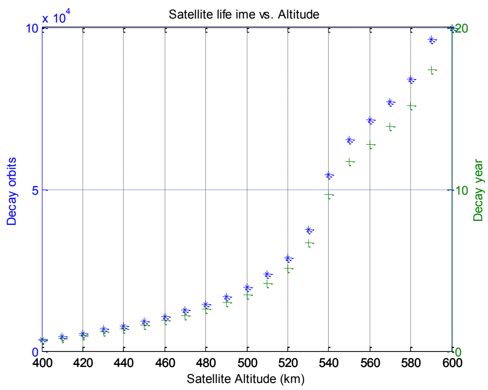

Figure 38 Satellite lifetime vs. satellite altitude a_r .

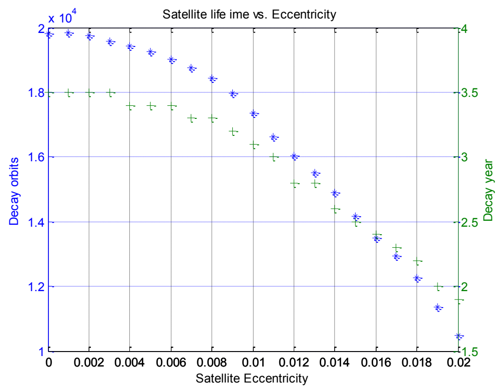

Figure 39 Satellite lifetime vs. eccentricity e_r

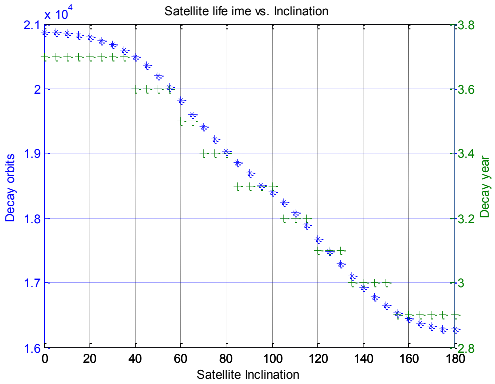

Figure 40 Satellite lifetime vs. inclination i_r

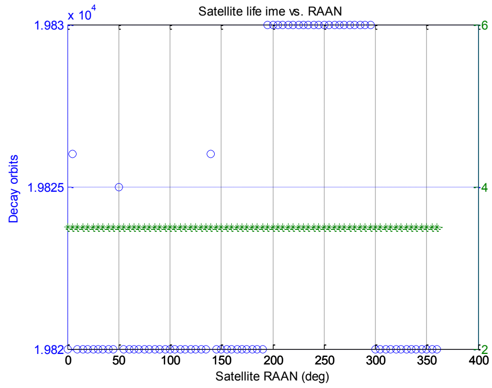

Figure 41 Satellite lifetime vs. RAAN Ω_r

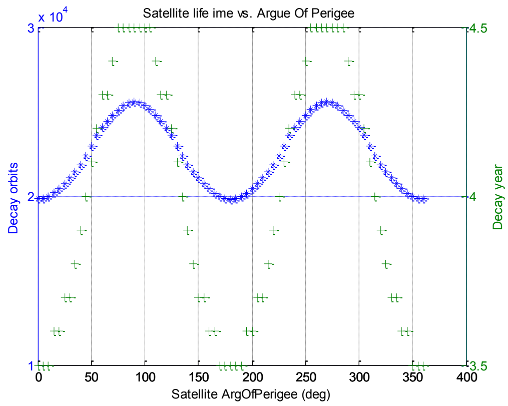

Figure 42 Satellite lifetime vs. argument of perigee. ω_r

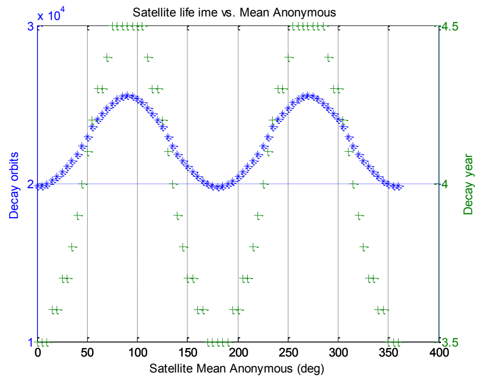

Figure 43 lifetime vs. mean anomaly M_r

The orbit's inclination i_r is the angle between the plane of the satellite orbit and the Earth's equatorial plane. Inclination affects the lifetime as atmospheric density varies with latitude. However from the view of system design, inclination is selected to satisfy the coverage requirement. For instance, polar orbiting satellites which have an inclination of 90 degree provides a global view of Earth, and an orbit with approximately 57 degree inclination provides coverage of Australia and New Zealand.

Ω_r, ω_r and M_r affect lifetime only slightly (shown in Figure 41, Figure 42, Figure 43, respectively) because they are elements that describe the angular position of the satellite and they do not affect the satellite's altitude.

2) Sensitivity of lifetime to satellite physical characteristics and solar flux

LEO satellites have physical lifetimes determined almost entirely by their interaction with the atmosphere.

The drag equation (27) essentially shows that the drag force on any object is proportional to the density of the fluid and proportional to the square of the relative speed between the object and the fluid.

$$\boldsymbol{F}_d = -\frac{\rho v^2 c_d A}{2}\hat{v} \tag{27}$$

The drag coefficient C_d is a dimensionless quantity that is used to quantify the drag or resistance of an object in a fluid environment such as air or water. It is used in the drag equation, where a lower drag coefficient indicates the object will have less aerodynamic or hydrodynamic drag. The drag coefficient is always associated with a particular surface area usually called drag area, defined as the mean cross-sectional area of the satellite perpendicular to its direction of travel. Figure 44 and Figure 45 show the lifetime change with drag coefficient and drag area. When C_d changes from 0.1 to 2.2 (for satellite drag coefficient, C_d usually taken to be between 2.0 and 2.2), the lifetime reduces from 20 to 4 years. When the drag area varies from $0.1m^2$ to $0.5m^2$, lifetime reduces from 13 to 3 years. Results verify that lifetime is strongly related to drag coefficient and drag area.

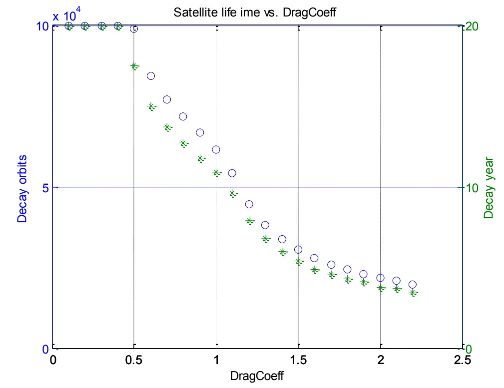

Figure 44 lifetime vs. drag coefficient.

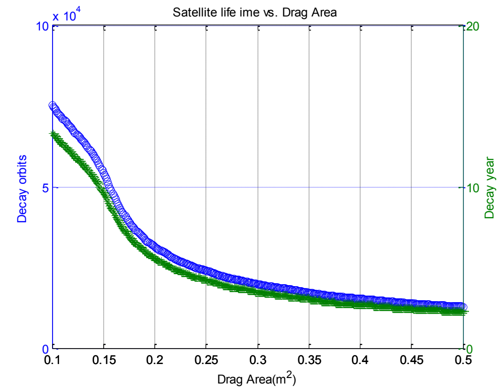

Figure 45 lifetime vs. drag area.

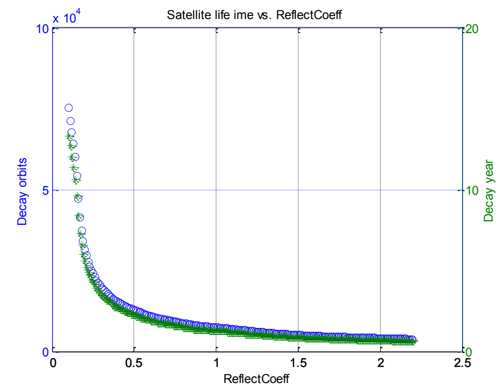

Figure 46 lifetime vs. reflect coefficient.

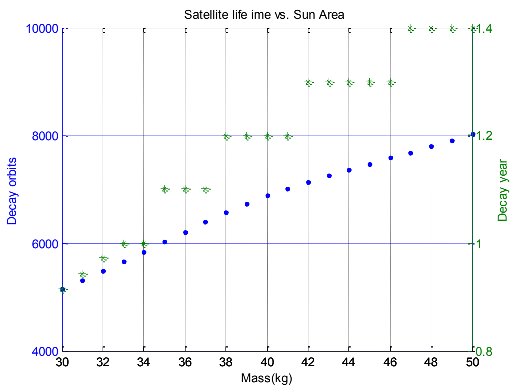

Figure 47 lifetime vs. mass.

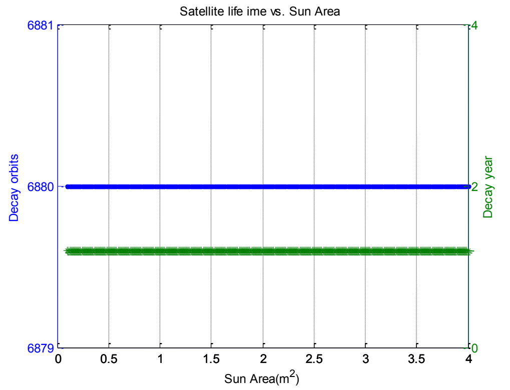

Figure 48 Lifetime vs. sun area.

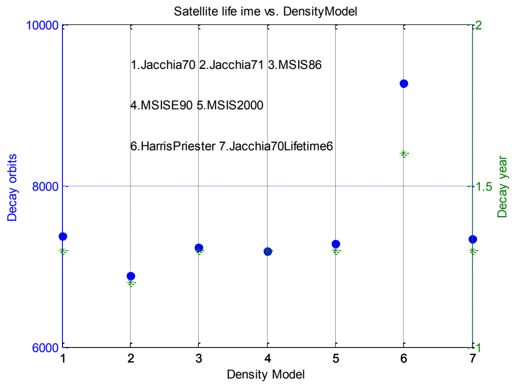

Figure 49 Lifetime vs. density model.

Mass to area ratio (*mass* to cross-sectional area in the direction of travel) directly affects the drag magnitude. Figure 47 shows that as the satellite mass changes from 30 to 50kg, the lifetime changes from less than 1 year to 1.4 year. In a real satellite system, the mass may be a function of time. That is, the mass of satellite will reduce over time due to fuel consumption. Note that the change of mass is not considered here.

Sun area barely affects the lifetime of the satellite as shown in Figure 48; this indicates that radiation pressure effects can be neglected in orbit lifetime calculations.

Figure 49 shows that the calculated lifetime varies with different atmospheric density model. The Jacchia 71 model has the shortest lifetime, while the Harris Priester model calculates the longest lifetime. In STK, the drag force model provides seven options for modelling the atmospheric density used in the computation of lifetime. These atmospheric density models are described in Table 8:

3) Sensitivity of lifetime to area and altitude

According to the single sensitivity study shown in the previous two sections, area and altitude are the two key factors for lifetime prediction. The satellite parameters used in the study are shown in Table 12. Lifetime with varying area from 0.2 to 1m^2 and satellite altitude varying from 400 to 600km are plotted in Figure 50 and Figure 51 where the lifetime unit is orbits; Figure 52 and Figure 53 where the lifetime unit is years.

Table 12 The study satellite initial orbital parameters and physical characteristics.

Inclination	60°	Mass	40kg
RAAN	0°	DragCoeff	2.2
Argument of Perigee	0°	ReflectCoeff	1.1
Eccentricity	0°	SunArea	3.3
Mean anomaly	0°	Rotate	On

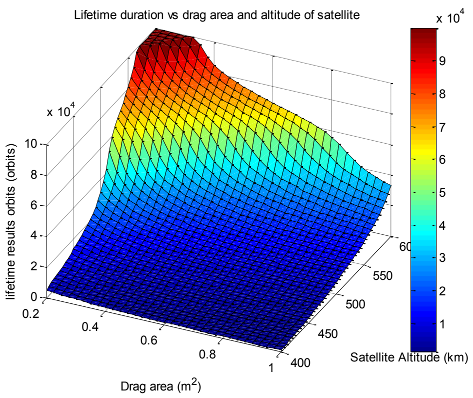

Figure 50 3D plot of lifetime duration vs. drag area and altitude of satellite (lifetime unit is orbits).

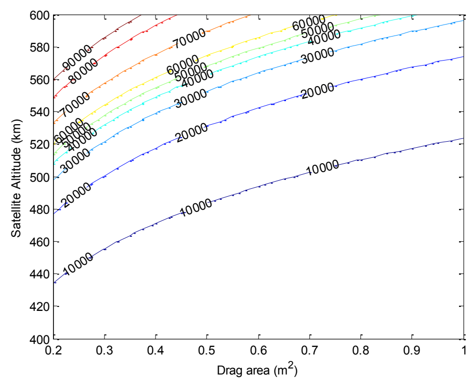

Figure 51 Contour plot of lifetime duration vs. drag area and altitude of satellite (lifetime unit is orbits).

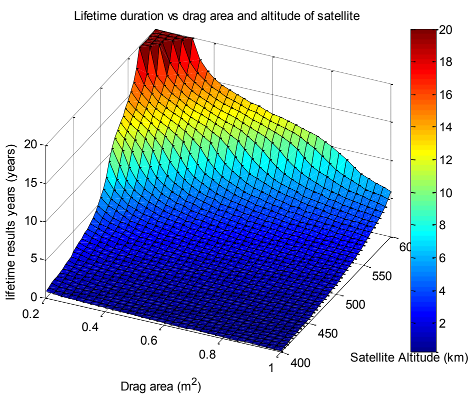

Figure 52 Lifetime duration vs. drag area and altitude of satellite (lifetime unit is years).

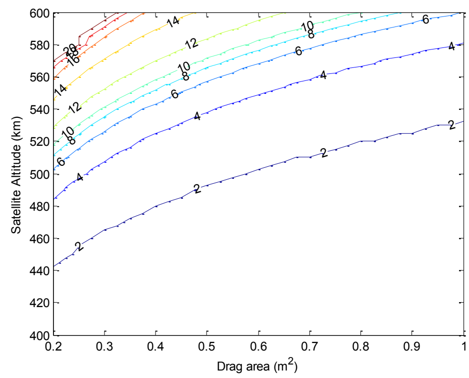

Figure 53 Contour plot of lifetime duration vs. drag area and altitude of satellite (lifetime unit is year).

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