

LULEÅ UNIVERSITY OF TECHNOLOGY

MASTER THESIS

SPACEMASTER

Mission Concept for a Satellite Mission to Test Special Relativity

Author :
VOLKAN ANADOL

Supervisors:
DR. THILO SCHULDT
DR. NORMAN GÜRLEBECK

Examiner :
ASSOC. PROF. THOMAS KUHN

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Abstract

In 1905 Albert Einstein developed the theory of *Special Relativity*. This theory describes the relation between space and time and revolutionized the understanding of the universe. While the concept is generally accepted new experimental setups are constantly being developed to challenge the theory, but so far no contradictions have been found.

One of the postulates Einsteins theory of Relativity is based on states that the speed of light in vacuum is the highest possible velocity. Furthermore, it is demanded that the speed of light is independent of any chosen frame of reference. If an experiment would find a contradiction of these demands, the theory as such would have to be revised. To challenge the constancy of the speed of light the so-called *Kennedy Thorndike* experiment has been developed. A possible setup to conduct a Kennedy Thorndike experiment consists of comparing two independent clocks. Likewise experiments have been executed in laboratory environments. Within the scope of this work, the orbital requirements for the first space-based Kennedy Thorndike experiment called *BOOST* will be investigated.

BOOST consists of an iodine clock, which serves as a time reference, and an optical cavity, which serves as a length reference. The mechanisms of the two clocks are different and can therefore be employed to investigate possible deviations in the speed of light. While similar experiments have been performed on Earth, space offers many advantages for the setup. First, one orbit takes roughly 90 min for a satellite based experiment. In comparison with the 24 h duration on Earth it is obvious that a space-based experiment offers higher statistics. Additionally the optical clock stability has to be kept for shorter periods, increasing the sensitivity. Third, the velocity of the experimental setup is larger. This results in an increased experiment accuracy since any deviation in the speed of light would increase with increasing orbital velocity. A satellite planted in a Low Earth Orbit (LEO) travels with a velocity of roughly 7 km/s. Establishing an Earth-bound experiment that travels with a constant velocity of that order is impossible. Finally, space offers a very quiet environment where no disturbances, such as vibrations, act upon the experiment, which is practically unavoidable in a laboratory environment.

This thesis includes two main chapters. The chapter titled "Mission Level" exploits orbital candidates. Here, possible orbits are explained in detail and the associated advantages and problems are investigated. It also contains a discussion about ground visibility and downlink feasibility for each option. Finally, a nominal mission scenario is sketched. The other chapter is called "Sub-Systems". Within this chapter the subsystems of the spacecraft are examined.

To examine the possible orbits it is necessary to define criteria according to which the quality of the orbits can be determined. The first criterion reflects upon the scientific outcome of the mission. This is mainly governed by the achievable velocity and the orbital geometry. The second criterion discriminates according to the mission costs. These include the launch, orbital injection, de-orbiting, satellite development, and orbital maintenance. The final criteria defines the requirements in terms of mission feasibility and risks, e.g. radiation. The criteria definition is followed by explaining the mission objectives and requirements. Each requirement is then discussed in terms of feasibility.

The most important parameters, such as altitude, inclination, and the right ascension of the ascending node (RAAN), are discussed for each orbital option and an optimal range is picked. The optimal altitude depends on several factors, such as the decay rate, radiation concerns, experimental contributions, and eclipse duration. For the presented mission an altitude of 600 km seems to be the best fit. Alongside the optimal altitude possible de-orbiting scenarios are investigated. It is concluded that de-orbiting of the satellite is possible without any further external influence. Thus, no additional thrusters are required to de-orbit the satellite. The de-orbiting scenario has been simulated with systems tool kit (STK). From the simulation it can be concluded, that the satellite can be de-orbited within 25 years. This estimation meets the requirements set for the mission.

Another very important parameter is the accumulative eclipse duration per year for a given orbit. For this calculation it is necessary to know the relative positions and motion of the Earth and the Sun. From this the eclipse duration per orbit for different altitudes is gained.

Ground visibilities for orbital options are examined for two possible ground stations. The theory is based on the geometrical relation between the satellite and the ground stations. The results are in an agreement with the related STK simulations. Finally, both ground stations are found adequate to maintain the necessary contact between the satellite and the ground station.

In the trade-off section, orbit candidates are examined in more detail. Results from the previous sections with some additional issues such as the experiment sensitivities, radiation concern and thermal stability are discussed to conclude which candidate is the best for the mission. As a result of the trade-off, two scenarios are explained in the "Nominal Mission Scenario" section which covers a baseline scenario and a secondary scenario.

After selecting a baseline orbit, two sub-systems of the satellite are examined. In the section of "Attitude Control System (ACS)" where the question of "Which attitude control method is more suitable for the mission?" is tried to be answered. A trade-off among two common control methods those are 3-axis stabilization and spin stabilization is made. For making the trade-off possible external disturbances in space are estimated for two imaginary satellite bodies. Then, it is concluded that by a spin stabilization method maintaining the attitude is not feasible. Thus, the ACS should be built on the method of 3-axis stabilization.

As the second sub-system the possible power system of the satellite is examined. The total size and the weight of the solar arrays are estimated for two different power loads. Then, the battery capacity which will be sufficient for the power system budget is estimated together with the total mass of the batteries.

In the last section, a conclusion of the thesis work is made and the possible future works for the BOOST mission are stated.

Keywords: BOOST, Special Relativity, Kennedy Thorndike, Eclipse duration, Ground visibility, Downlink feasibility, 3-axis stabilization, Spin stabilized satellite, Solar array estimation, Battery size estimation.

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List of Symbols and Abbreviations

AC: Attitude Control
ACS: Attitude Control System
AOCS: Altitude and Orbit Control System
BOL: Beginning of Life
CE: Concurrent Engineering
CMB: Cosmic Microwave Background
DOD: Depth of Discharge
DLR: German Space Agency
EOL: End of Lifetime
GEO: Geostationary Orbit
HU: Humboldt University of Berlin
JD: Julian Date
JDN: Julian Day Number
KT: Kennedy-Thorndike
LEO: Low Earth Orbit
LUH: Leibniz University of Hannover
MM: Michelson–Morley
MLTAN: Mean Local Time of Ascending Node
PDC: Power Distribution and Conditioning
PL: Payload
RAAN: Right Ascension of the Ascending Node
RTG: Radioisotope Thermoelectric Generators
SR: Special relativity
SEM: Standard Error of Mean
SET: System Enabling Technologies
STK: Satellite Toolkit
SSO: Solar Synchronous Orbit
TCS: Thermal Control System
ZARM: Center of Applied Space Technology and Microgravity, University Bremen.

1 Introduction

Special Relativity (SR) is one of the key aspects of our understanding of physics. Albert Einstein's SR theory [1] is based on two main principles: The laws of physics are invariant in all inertial systems (non-accelerating frames of reference) and the speed of light in vacuum is the same for all observers, regardless of the motion of the light source. SR can be tested by three common experiments: Michelson–Morley (MM) experiments [2], Kennedy–Thorndike (KT) experiments [3] and Ives–Stilwell experiments [5].

A Michelson–Morley experiment can be conducted to test the dependence of the speed of light on the *direction* of the measuring device. It establishes the relation between longitudinal and transverse lengths of moving bodies. The experiment can be conducted by using a light source, a beam splitter, two mirrors and a detector. The beam splitter oriented at an angle of 45° relative to two mirrors which are placed at the same distance to the beam splitter. The initial idea of the experiment was to measure the speed of light in different directions in order to measure speed of the "ether"¹ relative to Earth. In the end, no sign of "ether" was found by the initial experiment conductors Michelson, Albert Abraham and Morley, Edward in 1887 [2].

An Ives–Stilwell experiment demonstrates *time dilation* contribution to the Doppler shift of light. In 1938, Herbert E. Ives and G. R. Stilwell conducted the first experiment and the result was the first direct quantitative confirmation of time dilation factor [6]. The experiment setup had a beam called as "Canal rays" (a mixture of hydrogen ions) which is accelerated through perforated plates charged with various voltages. The beam and its reflected image were simultaneously observed with the aid of a concave mirror offset 7° from the beam.

The Kennedy Thorndike experiment is the third type of procedure which is a promising concept for enhancing our current understanding of fundamental physics. KT experiments are a modified version of MM experiments and realized the first time by Kennedy, R. J. and Thorndike, E. M in 1932 [4]. By conducting a KT experiment, the dependence of the speed of light on the *velocity* of the measuring device can be tested. It establishes the relation between longitudinal lengths and the duration of time of moving bodies.

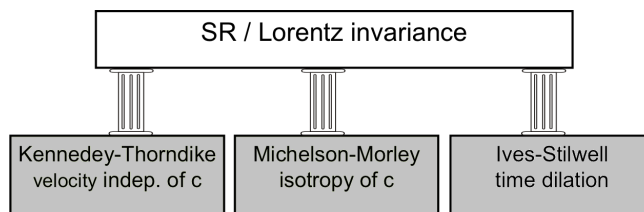


Figure 2: Three fundamental experiments to test Special Relativity

KT experiment are velocity based experiments, as can be seen in equation 1.0.1: Having a higher velocity results in a lower velocity dependence which corresponds to a higher accuracy in knowing

¹Ether was the hypothetical substance through which electromagnetic waves travel. [8]

that the speed of light remains constant. After the initial experiment by Kennedy and Thorndike, the experiment was conducted several times by other scientists. And the accuracy of these KT experiments have dramatically increased over the last decades [11] as shown in figure 3.

$$\frac{c(\theta, \vec{v})}{c_0} = 1 + (\beta - \alpha - 1) \frac{\vec{v}^2}{c_0^2} + \left(\frac{1}{2} - \beta + \sigma\right) \frac{\vec{v}^2}{c_0^2} \cdot \sin^2(\theta) \quad (1.0.1)$$

Equation 1.0.1 gives the light velocity dependence on the experiment's velocity and orientation. Where the speed of light is denoted as $c(\theta, \vec{v})$, the velocity of the frame as $\vec{v}$ and the orientation of the frame as θ . The first part of the right side of the equation $'(\beta - \alpha - 1)'$ corresponds to the KT coefficient, α_{KT} , and $(\frac{1}{2} - \beta + \sigma)$ corresponds to the MM coefficient, α_{MM} . In theory these coefficients should equal to zero. Actually "zero" is the target for the experiments. The current best results of two experiments are $4.8 \cdot 10^{-8}$ (Tobar, 2010) for the KT experiment and $4 \cdot 10^{-12}$ (Herrmann [7], 2009) for the MM experiment. Depending on the last (best) results of two experiments, today it can be wise to conduct a KT experiment in space to reach a higher improvement since the current best KT experiment result is order of four smaller than the best MM experiment result.

An experiment mounted upon a satellite that orbits Earth could be operated at much higher velocities than a similar experimental setup in a laboratory environment. Thus, a higher accuracy can be gained with a space-based setup due to the higher achievable velocities compared to ground-based experimental setups. Conducting a KT experiment takes longer with a ground-based setup than in a space bound experiment due to longer integration times in the same one day interval. This shorter-time-interval factor also results in a higher experiment accuracy. Because it is easier to provide stability along a duration of a LEO period (about 90 minutes) than a daily Earth rotation (24 hours). Finally, another advantage of operating a space-based experiment is the reduction of noise due to vibrations and other disturbances. First of all, space is a vacuum environment which is not easy to provide and maintain for a ground-based experiment. On Earth there are lots of noises due to human civilization such as passage of trucks or planes. Also gravity has an influence on the experimental setup. However, in space gravity field effects are suppressed due to the high altitude.

This thesis is part of development of a mission called BOOST with several partner universities from Germany to perform a KT experiment in space which would also be the first KT experiment in space.

BOOST is a small satellite mission and the experiment is basically based on comparing a length reference (a highly stable optical resonator) with a molecular frequency reference. Two clocks are compared to find a potential boost dependence of the velocity of light. By employing clocks with 10^{-15} frequency stability at orbit time (≈ 90 min) and by integration over 5000 orbits (2 years mission lifetime with 50% duty cycle) a high rate of improvement on the experiment accuracy is expected. To achieve the same number of cycles in a ground-based experiment, this requires an experiment over a time span of 15 years. As the result of the experiment, at least 100-fold improvement in measuring the Kennedy-Thorndike coefficient is targeted, compared to the current best terrestrial test [9].

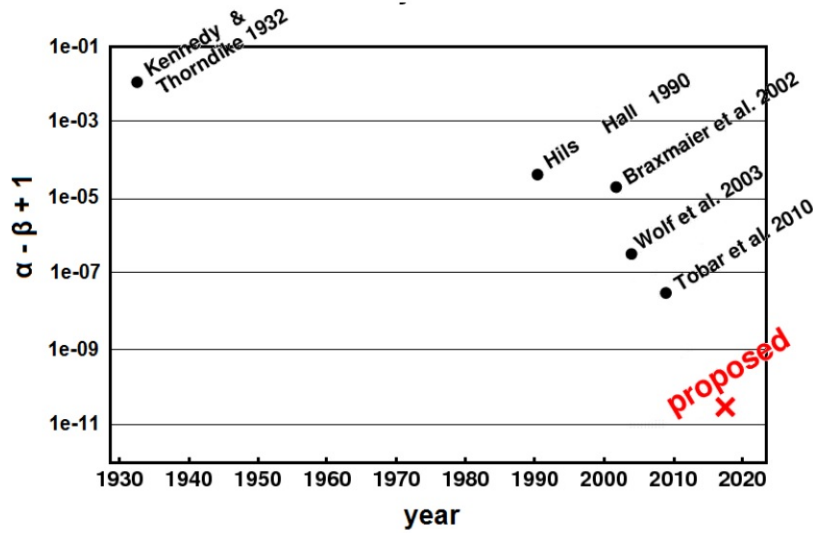


Figure 3: Former KT experiment results and proposed sensitivity for the BOOST space mission [11]

The BOOST mission aims at delivering answers to following open physics questions as showed in CE report [9]:

- What is the symmetry of space-time? Up to which accuracy is Special Relativity valid?
- Is there a deviation of the constancy of the light speed at a minuscule scale?
- What is the nature of space-time and which theories can (can not) describe it?
- How do matter and energy, space and time behave under the extreme conditions, e.g. short after the big bang?

Despite of seeking answers to fundamental questions in physics, BOOST will be able to address some open challenges in technological advancement, such as laser technologies and stabilization of optical components in a harsh (space) environment.

As part of BOOST some technologies will be demonstrated in space: a laser technology, an extreme thermal stability for the cavity and an unprecedented frequency stability for the optical setup.

Former Studies on the BOOST Project

The main input for this master thesis is the "Concurrent Engineering (CE) Study Report for BOOST" [9] by the System Enabling technologies (SET) department of DLR/Bremen. This document is a summary of scientific aims and technical challenges for the mission. In the report, while most subjects are addressed in a great detail some open issues remain including certain risks to the mission (also listed in the report at risk analysis section) which need to be minimized.

For BOOST as analyzed within this master thesis, the DLR compact satellite bus, which was the initial candidate bus for the mission, can not be taken as baseline due to programmatic constraints. Therefore the bus needs to be newly analyzed.

A bachelors thesis [10] about orbital analysis for the mission was written by Jonas Holtstiege from ZARM/ University of Bremen. The thesis is mostly based on experimental sensitivity estimation for different types of orbits and as an outcome several orbits are suggested. This thesis takes them as a baseline and tries to get more details and covers some missing aspects especially topics which are not mentioned in Jonas' thesis and in the CE study report.

Payload for the Mission

The payload is consisting of three divisions, optics, lasers and electronics. Iodine clock and optical cavity are optical units. The laser division has laser drivers and a laser beater. Electronic circuits like frequency controller and data progress units are in the electronics division (figure 4).

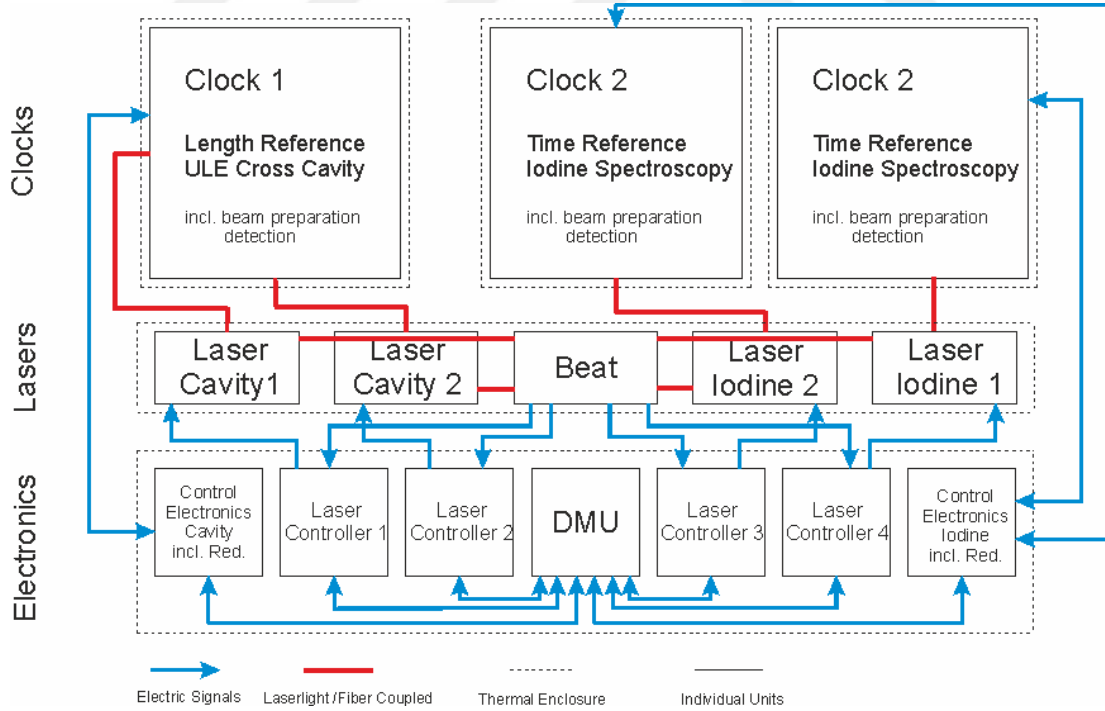


Figure 4: Functional diagram of the BOOST payload [9].

Every clock has its own thermal enclosures. The clocks are connected to control circuits in the electronics division. And laser lights will propagate over fiber cables from the laser producer devices to the clocks.

The most critical part of the payload is the cavity. It has to be in a thermal balance, because otherwise the experiment would be disturbed. To eliminate the thermal stability risk, a system requirement is stated: payload should be accommodated $\pm 5^{\circ}\text{C}$ around an absolute temperature in the range of $10\text{--}16^{\circ}\text{C}$. Thus, the satellite design should be able to provide a suitable thermal environment for the payload which is also an essential criteria for orbit selection. An eclipse free period has a lower temperature stability risk compared to a none-eclipse-free period. Because it is easier to balance the temperature in case of getting a certain amount of heat (by Sun) during the whole period. For the other case, heat flows to the satellite during a portion of the orbit and flows outward (space) for the rest of the period. As a result, thermal stability risk increases over long term for the second case.

Another important issue are disturbances like vibrations that may harm the experiment. Vibrations can be caused by various sources at the launch (dynamic loads by the launcher) or after the launch by reaction wheels or any other space dynamics, such as spinning attitude control or space disturbances, e.g. the gravity force of our planet and atmospheric drag.

Partners and Work Distribution

Three universities are in a collaboration for the mission. Those are ZARM/University Bremen, Humboldt University of Berlin (HU) and Leibniz University of Hannover (LUH). And all of them have been taking their parts depending on their expertise. Payload management, systems engineering, communication with DLR and defining system requirements tasks have been managed by ZARM.

The iodine clock is being designed by ZARM and the Humboldt University and the optical resonator (cavity) is being designed by ZARM and the Leibniz University. There is an Iodine clock in our System Enabling Technologies (SET) lab of DLR/Bremen which is already built and has a similar design, it just needed to be adapted for the mission. Unlike the iodine clock, the cavity is not yet built and is in the design phase.

Mission study: choice of orbit and spacecraft and cost estimation is within responsibility of ZARM. Instrument requirements are defined by HU and since they have considerable experience on laser systems. The team at the HU built several payloads for sounding rocket experiments which used similar types of laser systems to the one proposed for BOOST. They are also in charge of designing a laser system for the payload.

LUH is responsible for Electronics and Software for the payload and an error budget is being prepared by LUH.

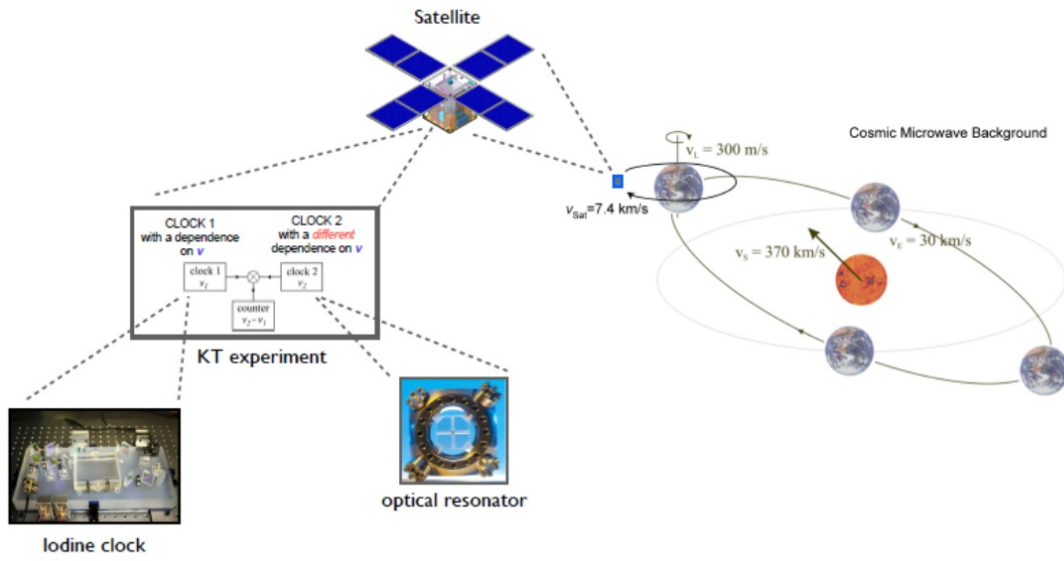


Figure 5: Overview of the BOOST mission.[9]

Airbus Defence and Space has been consulting the team on space aspects which are mostly related to the satellite aspect and has provided support for the mission costing. BOOST mission is currently at Phase 0 level and the team is preparing a proposal to submit in December 2016 for a national call by DLR. Then, if it is accepted, BOOST will step up for a Phase A/B1 study.

2 Mission Level

2.1 Overview

Selection of the orbit is a keystone for a space mission, as most of the other segments' depend on the orbit. In this section possible orbits and their advantages as well as disadvantages will be discussed. To define the best suited orbit three major criteria have to be considered:

- Scientific outcome: KT experiment sensitivity is different for every orbital plane, the reason for this is an angular relation between the inclination of the orbital plane and the velocity of the Sun, the Earth around the Sun and the rotational velocity of Earth. The Cosmic Microwave Background is referred to as the reference frame for the mission. A higher velocity of the spacecraft is beneficial for the outcome. However, there is not a radical orbital velocity difference between altitudes that may influence the outcome substantially².
- Cost: Includes launch cost of the satellite to space, orbital injection, deorbiting cost and satellite development cost. By sending a satellite to higher altitudes, launch and orbit injection costs rise. There are two ways of de-orbiting a satellite, a "free" de-orbiting where the satellite will become de-orbited without using any thrusters after the mission life time is passed and a controlled de-orbiting using thrusters which costs extra. Some orbits require special precautions like radiation shielding or necessity of using larger solar arrays. These increase the satellite development cost.
- Feasibility: A satellite has to be designed by taking into account the orbit. One orbit may offer a high scientific outcome but may not be feasible. For example, a satellite placed in an eclipse free Solar synchronous orbit (SSO) with about 1400 km altitude, may offer a higher scientific outcome (if we assume the experiment is only active in eclipse free periods), however that portion of space is in the inner Allen radiation belt. The high amount of present radiation may be hazardous to the satellite. In case of a "too low" orbit, the mission would be ended before the expected lifetime due to high atmospheric drag. Another concern for the feasibility is thermal stability. The thermal control system (TCS) should be designed with respect to the orbit. Upon selection of the orbit possible thermal environments during the mission life time should be considered in order to be capable to design a thermal control system that can meet the payload thermal requirements to provide a proper environment for the experiment.

²As an example: Orbital velocity of a 500 km orbit is 7616[m/s] and for a 600 km orbit it is 7561[m/s], the difference is only 0.73 %

2.1.1 Mission Objectives

Mission objectives are clearly defined by the BOOST team and listed in the CE study report as below [9]:

MI-OJ-0010	The mission BOOST shall contribute to extend the experimental foundation of Special Relativity by testing the symmetry of space-time under Lorentz transformations.
MI-OJ-0020	The mission BOOST shall improve the measurement of the KT coefficient by at least two orders of magnitude, compared to the current best (ground-based) results.
MI-OJ-0030	The mission BOOST shall demonstrate key technologies for the first time in space.

Table 1: Mission objectives

The first objective defines the main goal of the mission as conducting an experiment in space to test the theory of Special Relativity. The symmetry of space-time under Lorentz transformation will be tested by comparing two different clocks with a very high, never reached before, accuracy. The constancy of the speed of light will be measured up to a new limit that would enter Einstein's laws as a new parameter.

The second objective states how and at which quantity the main goal (the first objective) will be accomplished. An analysis will be made by comparing a rod length with the ticking rate of a clock as a function of velocity. The variation of the speed of light will be obtained with respect to a specific frame e.g. the Cosmic Microwave Background. By using advanced technologies especially advanced laser technology, BOOST shall be able to measure the KT coefficient with at least 100 times improvement from the current best experiment results which is conducted by Tobar in 2010 ($4 \cdot 10^{-8}$ in KT coefficient). However, only 77 times improvement [10] is calculated by Jonas Holtstiege. This issue should be re-checked later on.

The third objective is a secondary goal that states new high technologies such as laser technology and optical clocks will be demonstrated in space. Nowadays, high performance optical clocks are key units for many different aspects of space missions in fundamental physics, earth observation and navigation. BOOST will demonstrate such high technologies for the first time in space with a laser frequency stabilization to the 10^{-15} level and a thermal attenuation coefficient up to 10^8 .

2.1.2 Mission Requirements

There are five mission requirements defined by the team as follows [9]:

MI-DE-0010	The orbit shall ensure minimized orbit time and large velocity. Satellite velocity of at least 7 km/s and an orbit time not longer than 100 min.
MI-DE-0020	The orbit shall maximize the sensitivity in KT coefficient determination with respect to the CMB. At the same time the KT coefficient determination should be sensitive to other possibilities of preferred frames distributed over the entire sky.
MI-DE-0030	The mission lifetime shall be >2yrs.
MI-DE-0040	The inertial satellite velocity shall be determined with accuracy <5%.
MI-DE-0050	a)The spin rate shall not exceed 5 revolutions per minute with a spin rate stability AOCS. b)After several orbits a general maintenance of the payload might be required (re-calibration of the clocks). c)It is desirable to have at least 10 undisturbed orbits.

Table 2: Mission requirements

MI-DE-0010 states that the orbit should be as low as possible to achieve a high orbital velocity and a short orbit time in order to reach higher experiment accuracy. A higher velocity is a direct positive factor for the KT experiment accuracy. A shorter orbit yields a better clock frequency stabilization and therefore a better accuracy. Another advantage of having shorter orbits is improving the experimental statistics data by increasing the total number of orbits flown in the same mission life time. However, lower orbits come with increased eclipse times and it is still not decided whether the experiment will go on during eclipses or not. Besides considerations on lower power production from solar panels longer eclipse times pose a challenge upon the thermal stability which has to be provided.

According to Jonas' orbital analysis, increasing the orbit altitude by 100 km, increases the experiment accuracy by only a few percentages [10]. Thus, it can be concluded that changing the altitude by 100 km for the experiment sensitivity is not vital compared to other issues like eclipse duration and de-orbiting time.

Mission lifetime is expected to be 2 years and the satellite will be de-orbited within 25 years after the end of the scientific mission regarding to U.S. Government Orbital Debris Mitigation Standard Practices, and known as the 25 years rule [12].

The velocity of a satellite can be determined either by internal measurement or by theoretical assumptions. Measuring the velocity by an internal measurement is based on using satellite built-in sensors. The most common method makes use of a GPS receiver or an accelerometer. A GPS receiver may have about 10 meters position and 1 cm/s velocity accuracy for a Low Earth Orbit (LEO) satellite mission. Considering the Alcatel TOPSTAR 3000 GPS receiver, specifications given in table 3, and Viceroy GPS Receiver [15] from General Dynamics³ as examples, it can be concluded

³Accuracy: Autonomous Position: < 15 meters, Autonomous Velocity: < 0.05 meters/sec

that this requirement is feasible.

LEO	Position	10 m (3D-1 σ)
	Velocity	1 cm/s (3D-1 σ)
	Time Transfer with OCXO	200 ns (3 σ)
	Time Transfer with TCXO	1 μ s (3 σ)
	Orbital navigator full accuracy	2 orbital periods
GEO	Position	100 m (3D-1 σ)
	Velocity	2 cm/s (3D-1 σ)
	Time Transfer with OCXO	1 μ s (3 σ)
	Orbital navigator full accuracy	1 orbital period

Table 3: Accuracy of Alcatel TOPSTAR 3000 GPS receiver.[14]

Theoretical assumptions are essentially based on the altitude. Knowing the altitude alone enables to calculate the orbital velocity if the reasonable assumption is made that the other orbital parameters do not differ too much from the expectations, such as the eccentricity. If something unusual happens during orbit time the satellite may move away from its mission orbit which can not be estimated by theory, actual information is needed from satellite or from any other source.

The altitude can be determined from ground by analyzing received signals from the satellite. However, this is a non-preferred method and can be counted as a secondary data source for a space mission. The mission demands a continually high accuracy in the velocity measurement to assure the science case. Therefore relying on the indirect calculation which is based on theoretical assumptions is not sufficient.

MI-DE-005A puts a spin rate limit for a spin stabilized satellite to ensure centrifugal forces due to spin are negligible. Otherwise, a measurement error is expected to occur. Moreover, a spin can be a reason for vibrations and a higher spin rate results in a higher magnitude of vibrations. MI-DE-005B mentions that a re-calibration is necessary for the experiment to be sure clocks are stable and working simultaneously with each other. Any disturbance such as attitude control maneuvers can affect the experiment. Thus, MI-DE-005C introduces a period in which the experiment should not be disturbed by such reasons.

2.1.3 System Requirements

There are eight system requirements for the BOOST mission listed as a table below.

SY-IN-0010	The satellite shall enable a payload mass of at least 70 kg.
SY-IN-0020	The satellite shall enable a payload power of at least 95 W.
SY-IN-0030	The satellite shall enable a downlink capacity of $>294\text{Mb/day}$.
SY-IN-0040	The satellite bus shall ensure temperature stability $<+/-5^{\circ}\text{C}$ at orbit frequency around an absolute temperature in the range of $10\text{-}16^{\circ}\text{C}$ at specifically defined temperature reference points for the different payload units (valid during operation; launch relaxed).
SY-IN-0050	Vibrations/spacecraft drag on payload platform (cavity) shall be $<10^{-6}\text{g}$ for the frequencies in the frequency band TBD.
SY-IN-0060	The magnetic field variations amplitude at the outside of the iodine spectroscopy payload unit shall be $<10^{-2}\text{G}$.
SY-IN-0070	Data shall be taken in fast mode (diagnostics, 1 kHz) and slow mode (science, 10 Hz).
SY-IN-0080	Data storage onboard should allow for 4-8 GB.

Table 4: System requirements

The payload mass requirement is acceptable with a 70 kg mass. Today space technology is advanced enough to lift even tons of payload mass to space. By using a payload estimated mass, the total mass of the satellite can be estimated with multiplying by a factor of 3.3 [17]. This results an estimated mass of 231 kg which puts the mission satellite in *small satellite* class.

The payload requires 95 W power to operate properly. This amount is achievable for today space missions. Payload power can be easily provided by solar panels with a total area of about half a m^2 dedicated to payload. Today a solar panel provides about 220 W per m^2 [31].

SY-IN-0030 defines a downlink rate requirement for the BOOST mission which is found quite feasible and examined at section 2.3.4.

Temperature stability is one of the highest risks for the mission. To achieve the stability as given SY-IN-0040, thermal stability should be a key point at orbit selection phase. Thermal control system of the satellite shall aim to provide and keep a temperature environment about room temperature. The cavity unit of the payload has its own five layer thermal shields to keep it thermally stable. The payload is expected to have its own thermal control system with several heaters on certain positions of the payload and heat pipes to exhaust the unwanted heat.

SY-IN-0050 introduces a vibration (due to drag) limit for the mission. The drag force is a small force to disturb any of the satellite systems. However, it can be an issue for the payload if the payload carries sensitive elements. The feasibility of this requirement is checked at section 2.2.1.

The size of the data needs to be stored on-board which asked in SY-IN-0080 as 4-8 GB which is in usual range for today space missions. A mass memory unit with 16 GB size is in use for small storage requirement and 256 GB is commonly in use for a larger data storage.

2.2 Candidate Orbits for the Mission

In early discussions about the BOOST mission, a sun synchronous orbit (SSO) dawn(6:00)/dusk(18:00) with an altitude of 534 km is assumed. A SSO is a geocentric orbit that combines altitude and inclination in such a way that the satellite passes over equator line of the planet everyday at the same local solar time. A dawn SSO passes equator at 06:00 local time in morning while sun is rising at that location, and a dusk SSO is an opposite version which passes at sun setting time at 18:00 local time.

The altitude is calculated by estimating the orbital decay of a DLR compact satellite. Alternatively, a SSO 10:30 and an equatorial orbit are mentioned. Further assumptions are made by Holtstiege [10] and an altitude of 575 km is generally foreseen for the orbits.⁴

For an Earth orbit, earth's equatorial plane is taken as the reference plane, and the First Point of Aries is taken as the origin of longitude. In this case, the longitude is called the right ascension of the ascending node (RAAN). The angle is measured eastwards (or, as seen from the north, counter-clockwise) from the first point of Aries to the node. The mean Local Time of Ascending Node (MLTAN) is the local time of satellite when it passes equatorial plane, for example a dawn SSO passes equatorial line at 6:00 AM every day. In other words the satellite follows night/day termination line and enters night side⁵ from day side everyday on local time 6:00 AM. RAAN parameters are taken by assuming that the satellite mission orbit is on a specific date, which is the Coordinated Universal Time (UTC) of 10:00 AM, 1st of January, the year of 2020.⁶

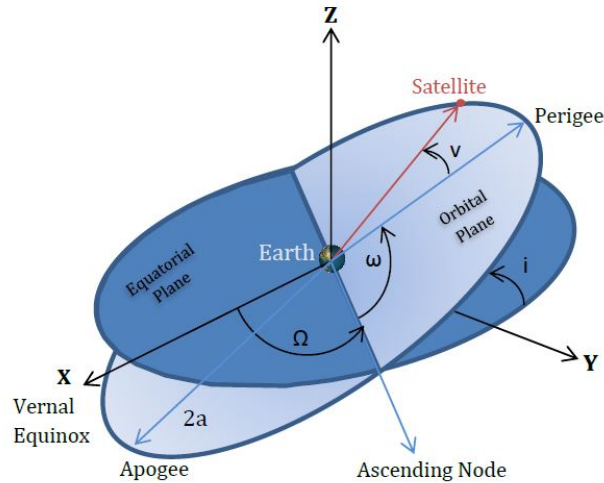


Figure 6: Common orbital parameters for a space mission. Where v is true anomaly, w is argument of periaapsis, Ω is longitude of ascending note, i is inclination and a is semi major axis [30].

⁴Assumption is not made for a specific satellite

⁵A satellite doesn't has to be in shadow over night side. It simply means, sub-satellite point is on the night side.

⁶This is valid for all orbits.

In this section a detailed analysis of the orbits listed in table 5 will be accomplished.

Orbit	Type	Inclination	Altitude	Period[min]	RAAN	MLTAN
#1	Circular	52	575	96.1	0	-
#2	Circular SSO	97	575	96.1	10.7	18:00
#3	Circular SSO	97	575	96.1	190.8	06:00
#4	Circular SSO	97	575	96.1	258.2	10:30
#5	Circular SSO	101.4	1400	113.6	190.3	06:00

Table 5: Candidate orbits

The period of an orbit can be calculated as follows⁷:

$$h = h_a = h_p \quad (2.2.1a)$$

$$a = h + R_e \quad (2.2.1b)$$

$$T = 2\pi \sqrt{a^3/\mu} \quad (2.2.1c)$$

where a is semi major axis which is summation of radius of earth (R_E) and altitude (h). Altitude in our case equals to height of perigee (h_p) and height of apogee (h_a), since eccentricity is zero. μ is the standard gravitational parameter of a celestial body. It is a product of the gravitational constant G and the mass M of celestial body.

The rotation of a satellite around the Earth is said to be posigrade when it rotates in the same direction as the rotation of the Earth, otherwise it is said to be retrograde orbit. A posigrade orbit ($i < 90$) RAAN shifts westward, for retrograde orbits ($i > 90$) RAAN shifts eastward and for an exact polar orbit ($i = 90$) it does not shift. The inclination of a solar synchronous orbit only depends on variable altitude since it is in retrograde and will shift due to non-spherical Earth perturbation (J_2). The inclination of a Sun synchronous orbit can be calculated as:

$$i = \arccos\left(\frac{a^2(1 - \epsilon^2)^2}{-3\pi J_2 R_E^2} \cdot \nu\right) \quad (2.2.2)$$

Where ν is $\frac{360}{365.256}$ deg/day as the angular velocity of the Earth's orbit around sun, ϵ is the eccentricity.

While a satellite orbits around Earth, some parameters vary slowly due to some disturbances. One

⁷only when orbit is not eccentric.

of the parameters is RAAN and some of the disturbances on RAAN can be listed as follows:

$$\Delta\Omega = -2.06474 \cdot 10^{14} \cdot a^{-7/2} \cdot \cos(i)(1 - \epsilon^2)^{-2} \quad (2.2.3a)$$

$$\Delta\Omega_{sun} = -0.00154 \cdot \cos(i)/n \quad (2.2.3b)$$

$$\Delta\Omega_{moon} = -0.00338 \cdot \cos(i)/n \quad (2.2.3c)$$

where $\Delta\Omega$ is called J2 influence due to Earth's oblateness. It can be accepted as a disturbance force acting on the satellite which pulls the satellite to south when the satellite is over the northern hemisphere and pulls it to northern when the satellite is over the southern hemisphere. B and C are gravitational influences by the Sun and by the Moon on a satellite and finally n is the number of revolution per day.

All three disturbances have the inclination as a parameter, as a $\cos(i)$ multiplication, this means when an orbit is exactly polar ($i=90$), all disturbances down to zero. Moreover, B and C have much less influence than J2 disturbance since those bodies are far away.

The altitude of the orbit is driven by the following parameters for the mission:

- Decay rate due to atmospheric drag
- De-orbiting method
- Radiation concern
- KT experiment accuracy with respect to orbital velocity
- Eclipse duration

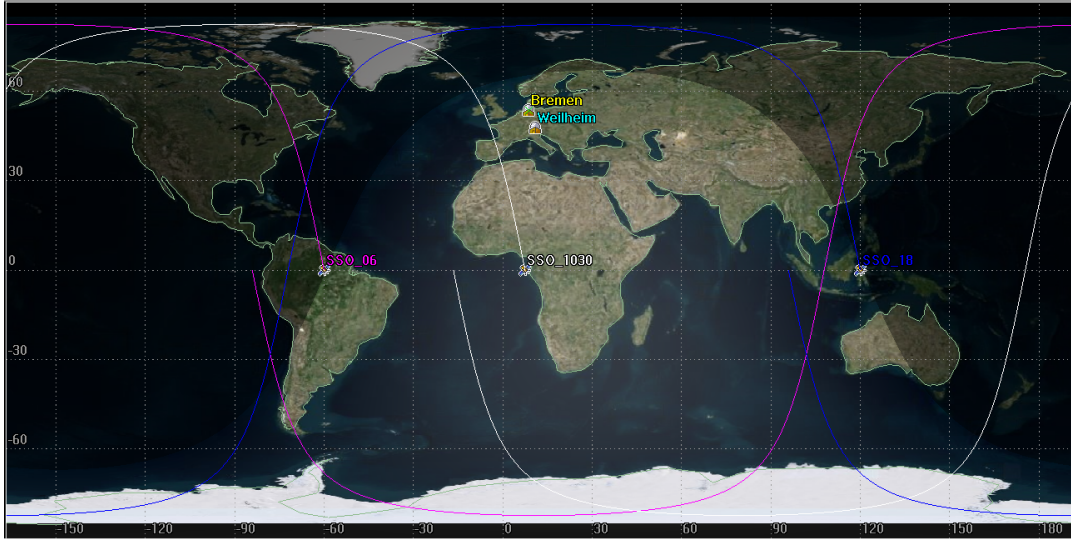


Figure 7: Representation of SSO orbits by Systems Tool Kit (STK) of Analytical Graphics. Blue, purple and white lines represent orbit #2, #3 and #4 respectively.

2.2.1 De-orbiting method and time

Today space debris is a serious issue and removing satellites safely from orbit is gaining importance. As stated before, a satellite shall be removed from its orbit during the 25 years following its mission. Two possibilities exist to meet this requirement. The main difference between these two methods is cost. The first one is a free removal where the satellite decays slowly by atmospheric drag and the second one is a costly method. Where the cost depends on mission altitude and how much propellant is needed to be added to the spacecraft mass before its launch and (relatively) the amount of propellant needed to be saved for orbital removal.

For the first method, the altitude needs to be chosen such that the satellite would enter lower atmosphere in 25 years due to active drag force on satellite. If a too low altitude is chosen the mission may end earlier than the desired mission life time. The second method is based on the using of satellite on-board thrusters to decrease perigee to such a low altitude like 180 km.⁸

The atmospheric drag force which adjusts orbital decay can be estimated by $D = \frac{1}{2}\rho v^2 A C_d$, where ρ is the local atmospheric density, v is the velocity of the spacecraft, A is the cross section of the object perpendicular to the motion and C_d is the drag coefficient which depends on the satellite physical design (mostly taken as 2.2 [24] to simplify estimations). When numbers are inserted for the BOOST mission as $2.21 \cdot 10^{-13}$ [17] for ρ , 7588 m/s as the velocity and 3 m² as area: $1.67 \cdot 10^{-7}$ G is reached which is about 8 times lower than the requirement SY-IN-0050 asks for, and makes the requirement feasible.

By applying the Newton's second law of motion, it ends up with:

$$a_d = -\frac{1}{2} \cdot \rho \cdot v^2 \frac{A \cdot C_d}{m} \quad (2.2.4)$$

where a_d is deceleration due to drag and m is spacecraft mass.

The inclination of the orbit and the radiation originating from the sun have to be taken into account when estimating the drag force upon the spacecraft. Inclination of the orbit plays a major role for determining the atmosphere density that the satellite would pass through. Polar orbits experience a lower atmospheric drag, due to a polar orbit's reduced time spent under the solar sub point (where solar heating causes the atmosphere to expand outward into space, thereby causing atmospheric density to rise). Therefore, a polar orbit requires a larger ballistic coefficient to match the same 25-year orbit lifetime constraint observed in an equatorial orbit case.

The solar radiation causes the satellites orbit to decay faster due to an increase of sun originated particles density and energy which eventually will hit the spacecraft. Solar activity is nearly periodic every eleven years [25], a future estimation can be seen in figure 8. The year of 2025 is a possible orbital injection year for BOOST and as can be expected from above graph solar activity will be high on that year. This factor may allow the BOOST satellite orbiting higher as can be seen in figure(9), if the ballistic coefficient is lower than 65 kg/m²).

The ballistic coefficient is a useful parameter for decay estimations which is described as $m/(C_d A)$. An object with a high ballistic coefficient is less sensitive to drag deceleration therefore has a higher orbit life time. A moderate (65 kg/m²) ballistic coefficient body should start its orbit somewhere

⁸180 km is a common altitude that re-entry is assumed when altitude drops lower than it.

between 600 km and 650 km regarding the estimation on figure(9) to have a 25 years long orbit life time.

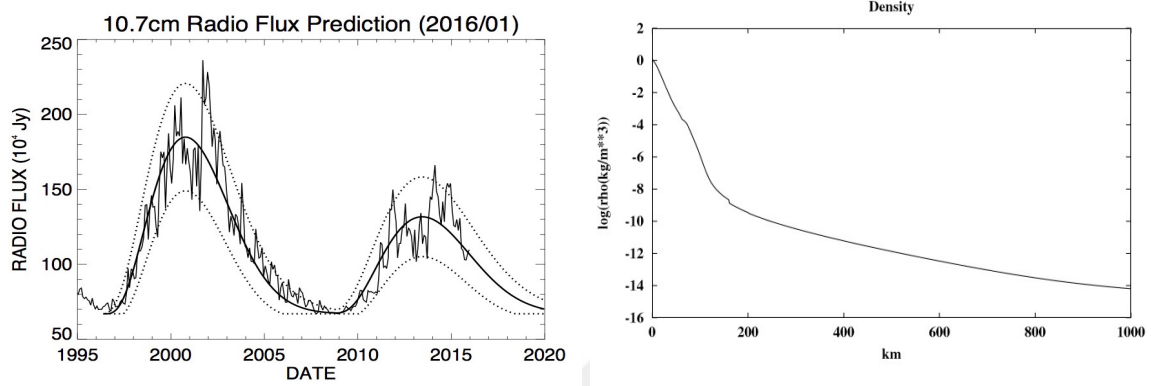


Figure 8: (a) Solar flux prediction [16] and (b) density of Earth's atmosphere [20]

Solar activity and inclination have a considerable effect below 600 km altitudes for orbital decay progress. The density of atmosphere drops to a very low amount at 600 km of 10^{-13}kg/m^3 which causes a weak drag force on a satellite. For the BOOST mission it can be assumed that mission altitude possibly would be about 600 km with a margin of 50 km to ensure a "free" de-orbiting. Optimum altitude should be defined when the satellite bus is specified or chosen since it is directly related.

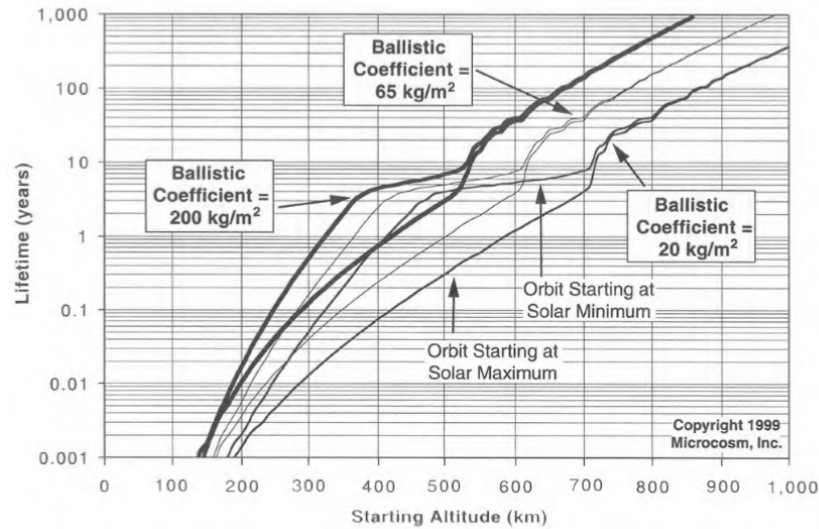


Figure 9: Orbit life time vs altitude [17]

A satellite can be disposed of by using thrusters. To achieve this, a certain amount of propellant mass is required to be saved. Total orbital velocity change (Δv) is a key parameter for estimating the amount of required propellant in order to achieve spacecraft disposal with a certain exhaust velocity (v_{exh}). There are two common methods to perform an orbit transfer of a satellite, the first one is the so-called Hohmann transfer [26] which requires two burns.

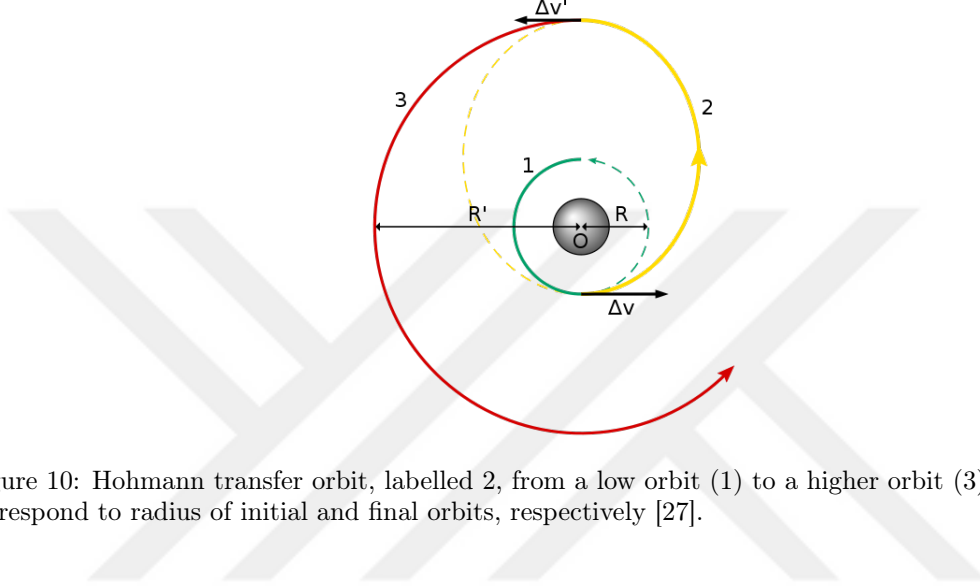


Figure 10: Hohmann transfer orbit, labelled 2, from a low orbit (1) to a higher orbit (3). R and R' correspond to radius of initial and final orbits, respectively [27].

Total (ΔV) by a Hohmann transfer can be estimated as [26]:

$$\Delta v_{total} = \Delta v + \Delta v' = \sqrt{\mu} \left[\left| \left(\frac{2}{R} - \frac{1}{a_t} \right)^{\frac{1}{2}} - \left(\frac{1}{R} \right)^{\frac{1}{2}} \right| + \left| \left(\frac{2}{R'} - \frac{1}{a_t} \right)^{\frac{1}{2}} - \left(\frac{1}{R'} \right)^{\frac{1}{2}} \right| \right] \quad (2.2.5)$$

And the semi-major axis of the transfer orbit as:

$$a_t = (R + R')/2 \quad (2.2.6)$$

The second method for orbital transfer is the so called spiral transfer which is some less propellant efficient than Hohmann Transfer but offers a bit quicker transfer and the time of flight is also shorter. Low thrust spiral transfers are used when a low thrust engine such as an ion thruster or a solar sail is available where thrusters continuously add ΔV and require more total ΔV than Hohmann transfers.

Total velocity change (Δv_{total}) requirement for a spiral transfer can be formulated as follows⁹ where v_1 is the velocity of initial orbit and v_2 is the velocity of target orbit:

$$\Delta v_{total} = |v_1 - v_2| \quad (2.2.7)$$

⁹inclination is accepted as constant.

The required velocity for satellite disposal can be calculated with the formula below, where v is the orbital velocity, H_e is the target altitude and H_i is the initial altitude:

$$\Delta v_{disposal} \approx v \cdot \left[1 - \sqrt{\frac{2(R_e + H_e)}{2 \cdot R_e + H_e + H_i}} \right] \approx v \cdot \left[\frac{H_i - H_e}{4(R_e + H_e)} \right] \quad (2.2.8)$$

To perform a disposal H_e should be set to a low altitude about 180 km where the satellite can burn in a few days due to atmospheric friction.

Tsiolkovsky rocket equation is a fundamental theory to estimate Δv from propellant usage and vice versa. The equation can be found below, the exhaust velocity is v_{exh} , initial mass of rocket or spacecraft including propellant, m_0 and final mass without propellant, dry mass, m_f .

$$\Delta v = v_{exh} \cdot \ln \frac{m_0}{m_f} \quad (2.2.9)$$

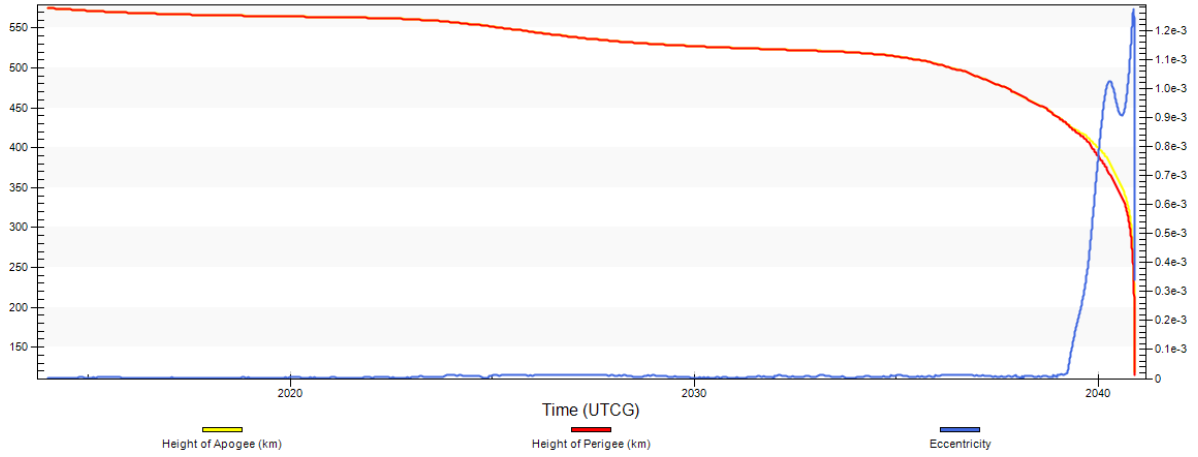


Figure 11: Decay progress of a SSO with a 575 km altitude (actually this orbit represents the candidate orbit two, which is going to be explained in the following section). The progress is simulated with a STK simulation.

By a STK simulation, satellite decay progress of a SSO is represented in figure 10. Mission starting date is taken as Jan of 2014 instead of the expected mission year, 2025. The reason of this is to reach a more realistic result by using actual data from the past. The year 2014 is reached by subtracting 2025 by 11 which equals to a solar cycle and a similar behaviour of the next solar cycle is assumed. The simulation shows the satellite will decay in NOV of 2040 which corresponds to about 27 years as mission life time. The simulation also reports a 11 meters per day altitude decay in the first days of the mission. After one year of mission, altitude is expected to decay about 3.7 km and after the mission life time (2 years) by 5.9 km. Since numbers are too small to create an influential difference for the orbital velocity, there is no need to make an altitude correction during the mission life time.

2.2.2 Eclipses

Earth shadow is a significant factor for a space mission. There are two types of shadows, Penumbra and Umbra. Penumbra shadow is a very weak shadow compared to Umbra. Penumbra can be accepted as a partial eclipse cause, however Umbra is a kind of exact shadow where there is no direct sunlight. Penumbra does not take a significant place in a space mission design and it is mostly negligible.

In absence of sun light, a spacecraft starts to cool down rapidly and even for a SSO satellite a thermal control system is needed. As can be expected, without sunlight solar panels can not produce power, thus for a power system design eclipse duration is an important parameter.

To estimate eclipses for a satellite mission, the positional relation between sun and earth is needed to be known and also the position of the Sun in our sky can be used for the same purpose. For our theoretical approach, we have to convert our time format (Gregorian calendar) to the Julian time format. Gregorian calendar is introduced in October 1582 as a replacement for Julian calendar and Gregorian calendar is the most common calendar that we use today.

The Julian Day Number (JDN) is the integer assigned to a whole solar day in the Julian day count starting from noon Greenwich Mean Time, with Julian day number 0 assigned to the day starting at noon on January 1, 4713 BC, proleptic Julian calendar [32].

Gregorian calendar date to Julian day conversion [32] can be done by the following formula¹⁰:

$$JDN = d + \lfloor \frac{153 \cdot m + 2}{5} \rfloor + 365 \cdot y + \lfloor \frac{y}{4} \rfloor - \lfloor \frac{y}{100} \rfloor + \lfloor \frac{y}{400} \rfloor - 32045 \quad (2.2.10)$$

$$\text{Julian Date(JD)} = JDN + \frac{hr - 12}{24} + \frac{min}{1440} + \frac{sec}{86400} \quad (2.2.11)$$

where d corresponds to day number, m to month number, y to Gregorian year, hr to hours, min to minutes and sec to seconds where all indexes for parameters start from one.

After time conversion, some parameters such as declination and right ascension of the Sun should be calculated step by step for estimating the beta sun angle which is a primary parameter for estimating eclipses.

Obliquity of the ecliptic plane (axis tilt) very slowly decays over centuries and can be estimated for a specific date:

$$\varepsilon = 23.439 - 0.0000004 * JD \quad (2.2.12)$$

Mean longitude of our sun in degrees can be estimated:

$$L = 280.46 + 0.9856474 * JD \quad (2.2.13)$$

Mean anomaly of our sun in degrees can be estimated:

$$g = 357.528 + 0.9856003 * JD \quad (2.2.14)$$

¹⁰Note the floor brackets: the largest integer less than or equal to the division result should be taken

The ecliptic (or celestial) longitude of the Sun is the angular distance along the ecliptic from the vernal equinox to the great circle through the Sun. It is measured eastwards (like the right ascension), but in degrees, between 0 and 360. The ecliptic longitude can be estimated in degrees:

$$Lambda = L + 1.915 \cdot \sin(g) + 0.02 \cdot \sin(2 \cdot g) \quad (2.2.15)$$

Declination of the Sun is the angular distance of the sun from north or south of the earth's equator. Actual declination of the sun can be found in degrees as following:

$$\psi = \arcsin(\sin(\varepsilon) \cdot \sin(lamda)) \quad (2.2.16)$$

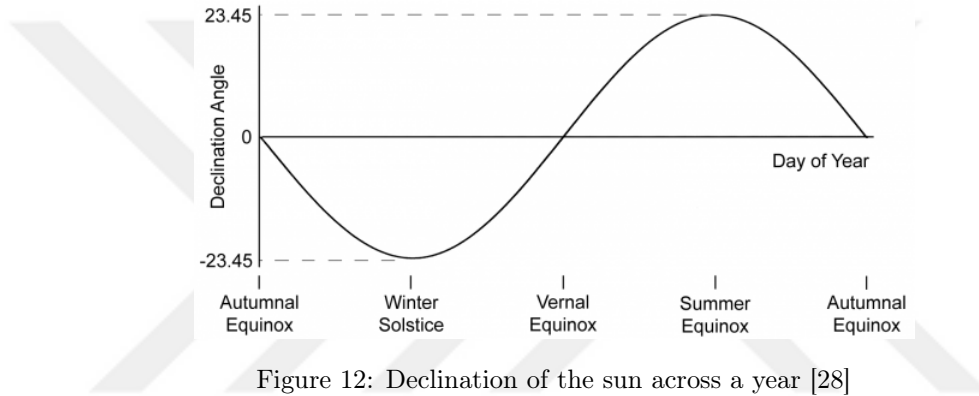


Figure 12: Declination of the sun across a year [28]

Right ascension of the Sun (RA_{sun}) in degrees can be estimated for a specific time:

$$X = \cos(Lambda) \quad (2.2.17a)$$

$$Y = \cos(\varepsilon) \cdot \sin(Lambda) \quad (2.2.17b)$$

$$RA_{sun} = \tan(Y/X) \quad (2.2.17c)$$

$$RA_{sun} = \begin{cases} RA_{sun} + 180 & \text{if } X < 0 \\ RA_{sun} + 360 & \text{if } Y < 0 \text{ and } X > 0 \end{cases}$$

The beta angle (β) is in use for determining the percentage of time an object such as a spacecraft in earth orbit spends in direct sun light. The beta angle is defined as the angle between the orbital plane of the spacecraft and the vector to the sun which can be seen in figure 13.

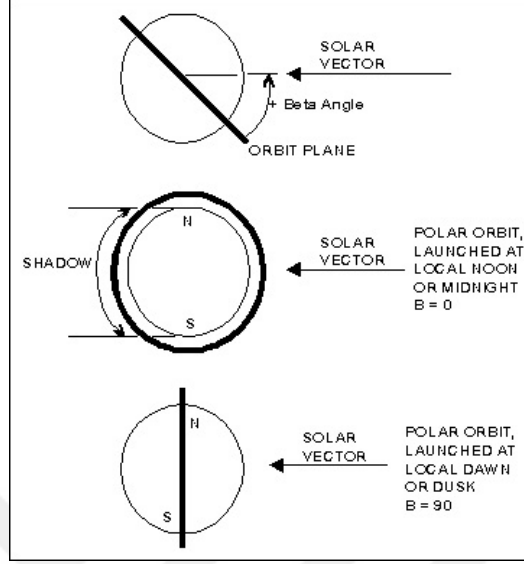


Figure 13: Beta sun angle with two extreme effects [29].

For a polar orbit launched in local noon or midnight, the resulting beta angle is 0 degrees which gives the maximum Earth shadowing to satellite. On the other hand, a polar orbit launched at local dawn or dusk has a 90 degrees of beta angle, which causes no eclipses for a certain time until a level of drift for the satellite has occurred due to external disturbances.

Beta sun angle, in degrees, can be estimated as shown below by using the previously calculated parameters from equation 2.2.10 to 2.2.17:

$$\beta = \arcsin((\cos(\psi)) \cdot \sin(i) \cdot \sin(RAAN - RA_{sun}) + \sin(\psi) \cdot \cos(i)) \quad (2.2.18)$$

For a circular orbit, a satellite is accepted as being in the shadow of the Earth when the angle between the orbit plane and the sun direction (β) is lower than the angle at eclipse entrance (β').

$$\beta' = \arcsin\left(\frac{R_e}{R_e + h}\right) \quad (2.2.19)$$

When β is smaller than β' , there is a eclipse period as T_{ecl} equals to $T \cdot F$, where F is the fraction of the eclipse:

$$F = \frac{1}{180^\circ} \arccos\left(\frac{\sqrt{h^2 + 2R_e h}}{(R_e + h)\cos(\beta)}\right) \quad (2.2.20)$$

As it can be seen in equation (2.2.20), the eclipse duration of a given orbit decreases with altitude. The actual dependency of the eclipse times on the altitude differs with the orbital type. The correlation between the duration of eclipses and the altitude is stronger for a SSO than for a non-SSO. The proof of this correlation can be seen at related orbit option vs altitude graphs.

Orbit option #5 is an eclipse free orbit and that is the main reason of it is being a candidate. If the experiment can not continue in eclipses, this orbit can have some advantage over others.

Theoretical results which are made via MATLAB from above equations (2.2.10-2.2.20) are compared below with a simulation called as Systems Tool Kit (STK) by Analytical Graphics Inc. (AGI). In the following different orbital options (except orbit option #5) with respect to eclipse durations are discussed.

2.2.2.1 Eclipses for orbit option #1

Orbit option #1, a circular orbit with 52° inclination and has eclipses most of the days over a year (341 days, see figure 14). Eclipse durations per day are represented by x-axis in the same figure which also show a maximum of 35.5 minutes eclipse duration for this orbit option over a period of 96.1 minutes.

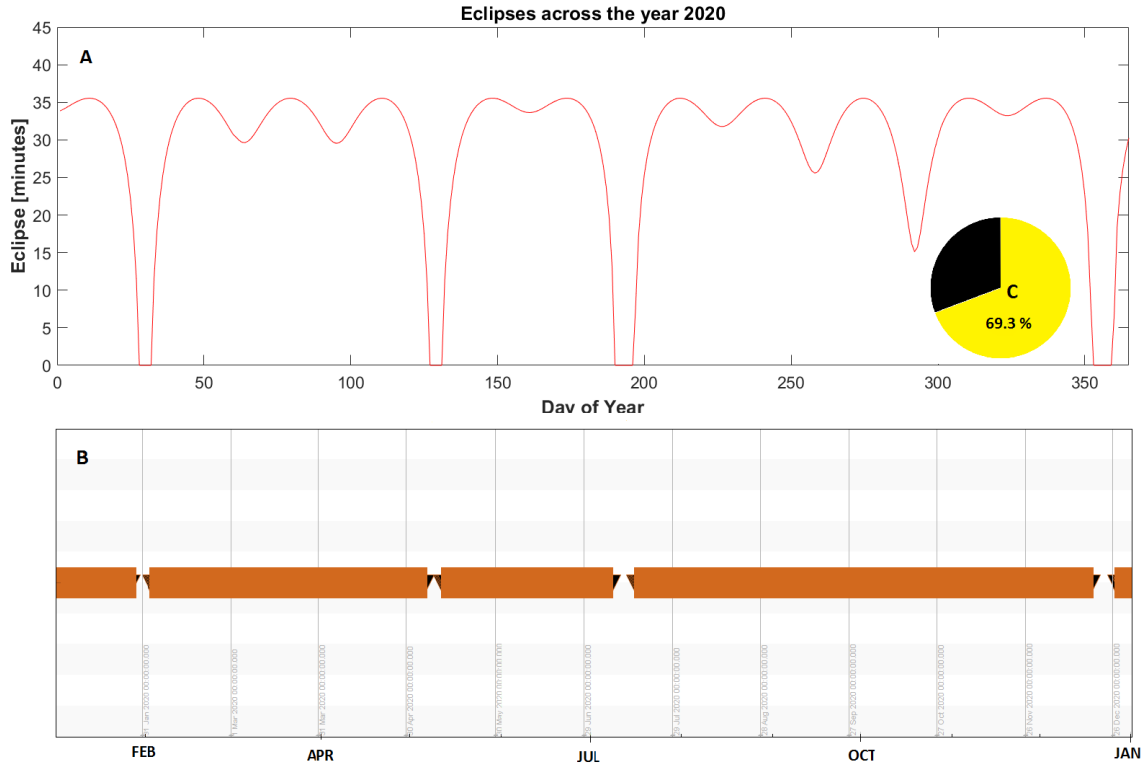


Figure 14: a) Eclipse duration per day for orbit #1 depicted over the duration of one year. b) Depiction of the eclipse occurrence which is simulated with STK for a year. c) Cumulative sunlight percentage over a year (simulated with STK).

The total eclipse duration over a year is about 30% of the time per year, which is depicted in figure 14c. In a space mission design, the most important points are the worst case and the best case

scenarios (mostly the worst case scenarios for design purposes) rather than looking at the percentage of sunlight over a year. The percentage can be useful when comparing two different orbits which have similar amount of maximum eclipse duration per period. For example among two different orbits which have equal amount of maximum eclipse durations, the highest sun exposed orbit over a year (the highest cumulative sunlight) would offer more power input to the solar panels, however may heat the satellite too much and may cause overheating or power gain can be lost for extra cooling reasons. In short terms, "the percentage" is a useful parameters for orbital trade-offs.

The worst case, with respect to eclipses for this orbit option is 35.5 minutes being in the shadow of earth. The satellite should be able to keep temperatures of satellite units in a certain boundary of operating temperatures. Moreover, the power system should be able to feed the satellite at least 35.5 minutes long in shadow phases.

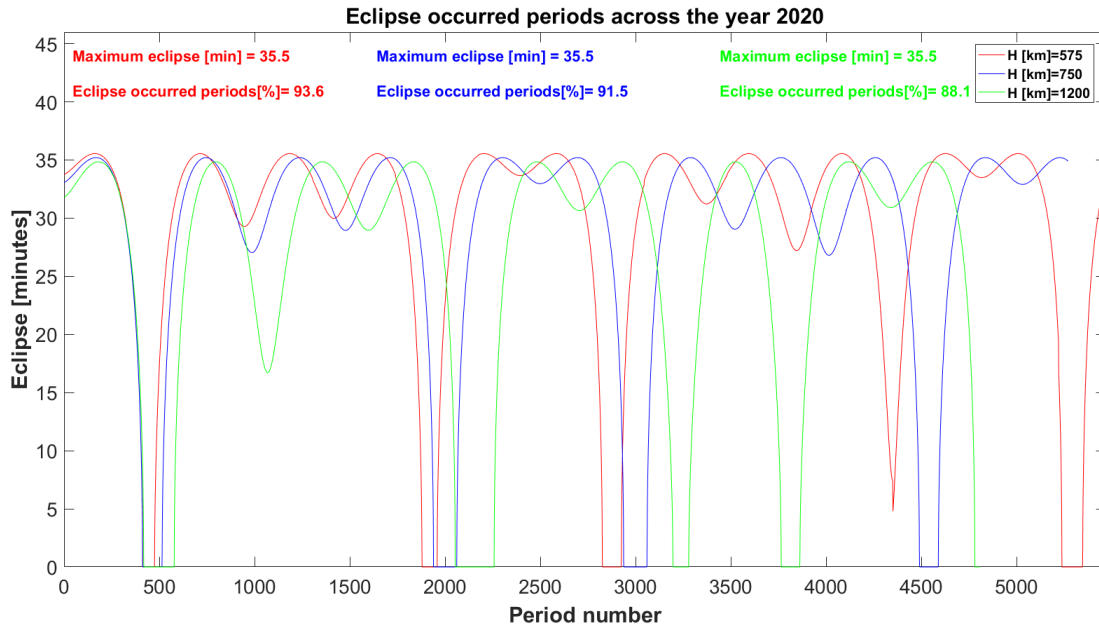


Figure 15: Eclipses vs altitude for orbit #1 across a year. The percentage of eclipse occurred periods over the total period number can be seen in the figure with the maximum eclipse duration for each period.

Figure 15 represents orbit option #1 in red and two different altitude orbits with the same inclination, 52 degrees. The blue orbit has a 750 km altitude with a 99.6 minutes period time and green has a 1200 km with a 109.2 minutes period time. Cumulative sunlight for the three orbits over a year which are obtained with STK are 69.1%, 71.1% and 76%, respectively.

There is an inverse relationship between eclipse duration and altitude, however the correlation is weak for this orbit type. Each of the three orbits has about 35 minutes maximum eclipse duration although having significantly different altitudes. By making a relationship between maximum eclipse

durations with period times, it can be concluded that the percentages of maximum eclipse duration over a period are the following: 36.9% for the first orbit (red), 35.3% for the second orbit (blue) and 31.8% for the last orbit (green).

As a conclusion, it can be said that for a 52° inclination orbit, altitude does not play a significant role regarding eclipse times.

2.2.2.2 Eclipses for orbit option #2

Orbit #2 is a SSO which passes the equatorial line at 18:00 local time. This so-called *dusk* orbit, has the maximum benefit of being a solar synchronous orbit regarding with minimum eclipses. Only certain portion of a year eclipses occur as an arc shape and it reaches up to 21.8 minutes at the peak (figure 16).

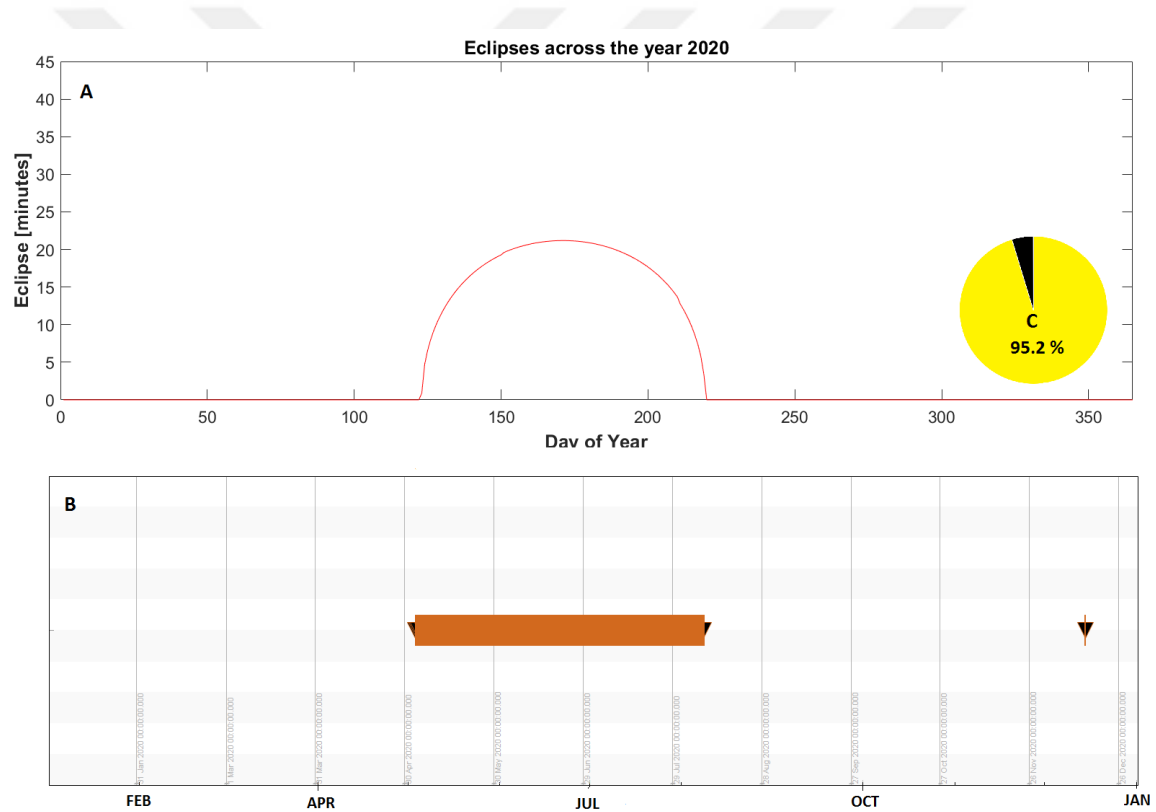


Figure 16: a) Eclipse duration per day for orbit #2 depicted over the duration of one year. b) Depiction of the eclipse occurrence which is simulated with STK for a year. The triangle at December represents a partial eclipse, caused by the Moon c) Cumulative sunlight percentage over a year (simulated with STK).

The cumulative sunlight for orbit option #2 is 95.2% as can be seen in figure 16c. This orbit has a much higher sunlight rate than orbit option #1 which has 69.3% (figure 14). The maximum eclipse duration is also much lower (with 21.8 minutes) than in orbit #1 (35.5 minutes).

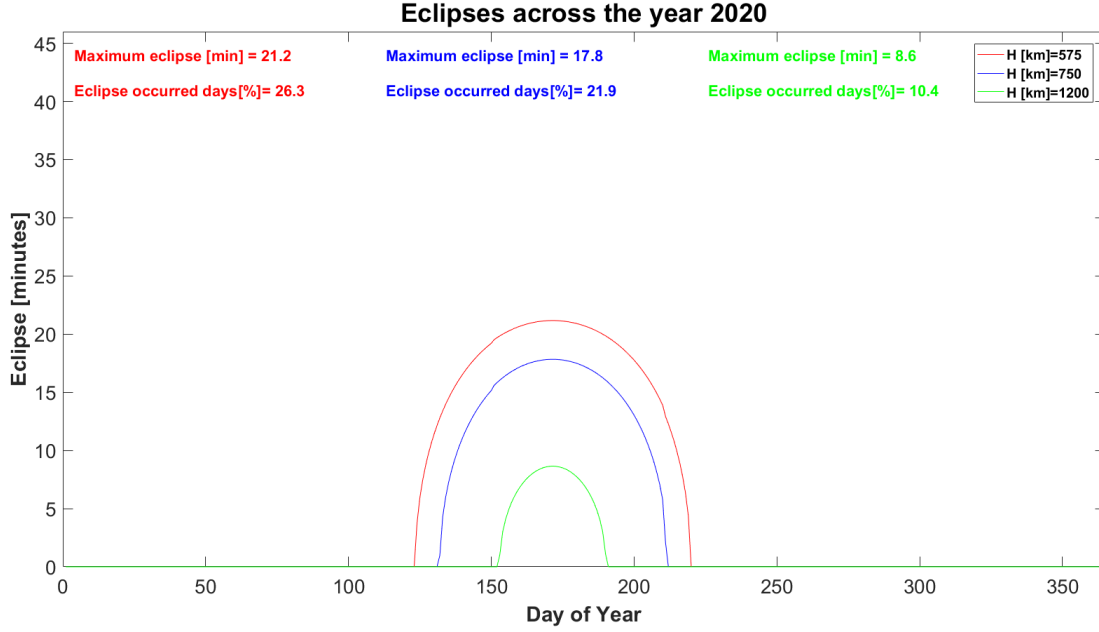


Figure 17: Eclipse duration vs altitude for orbit #2 across a year. The percentage of the eclipse occurred days over a year can also be seen in the figure.

Figure 17 represents orbit option #2 in red and two different altitude modified versions of the same orbit with blue color for a 750 km altitude orbit and finally with green color for a 1200 km altitude orbit. The maximum eclipse durations are: 21.2 minutes for the red (default) orbit, 17.8 minutes for the blue orbit and 8.6 minutes for the green orbit. The percentage of maximum eclipse duration over a period are the following: 22.6% for the first orbit (red), 17.8% for the second orbit (blue) and 7.8% for the third orbit (green). Existence of a strong correlation between maximum eclipse duration with altitude for this orbit type can be concluded by previous percentages.

Over a year, orbit #2 offers lots of eclipse free days with a rate of 73.7%. And this percentage also dramatically increases by increasing the altitude. As can be seen in figure 17, the percentage of total eclipse occurred days goes down to 10.4% for a 1200 km orbit.

2.2.2.3 Eclipses for orbit option #3

Orbit #3 is a SSO which passes the equatorial line at 06:00 local time. This so-called *dawn* orbit, which also has the maximum benefit of being a solar synchronous orbit like the previous orbit (option #2) regarding with eclipses.

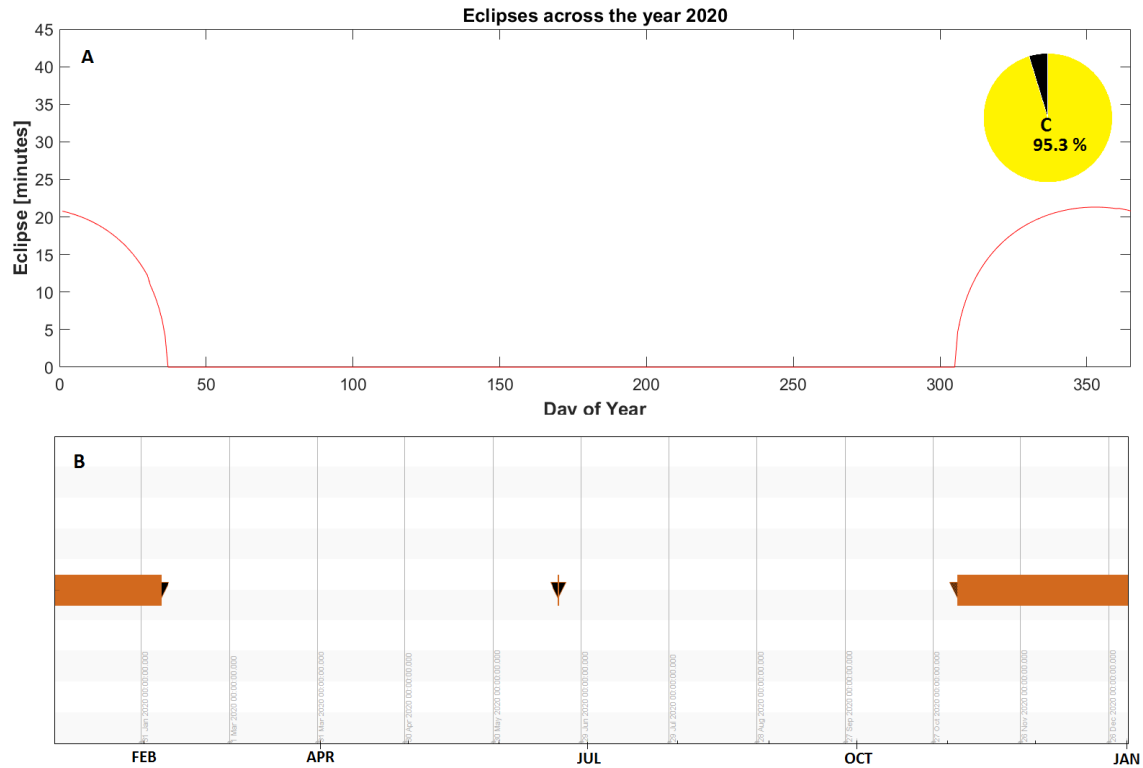


Figure 18: Eclipse duration by day for orbit #3 depicted over the duration of one year. b) Depiction of the eclipse occurrence which is simulated with STK for a year. The triangle at June represents a partial eclipse, caused by the Moon. c) Cumulative sunlight percentage over a year (simulated with STK).

As can be seen in figure 18A, the maximum eclipse duration is about 21 minutes for orbit option #3 which is equal to the maximum eclipse duration for orbit option #2 due to having the same orbital geometry. A satellite with this orbit is expected to have direct sunlight for 95.3% time of a year, see figure 18c.

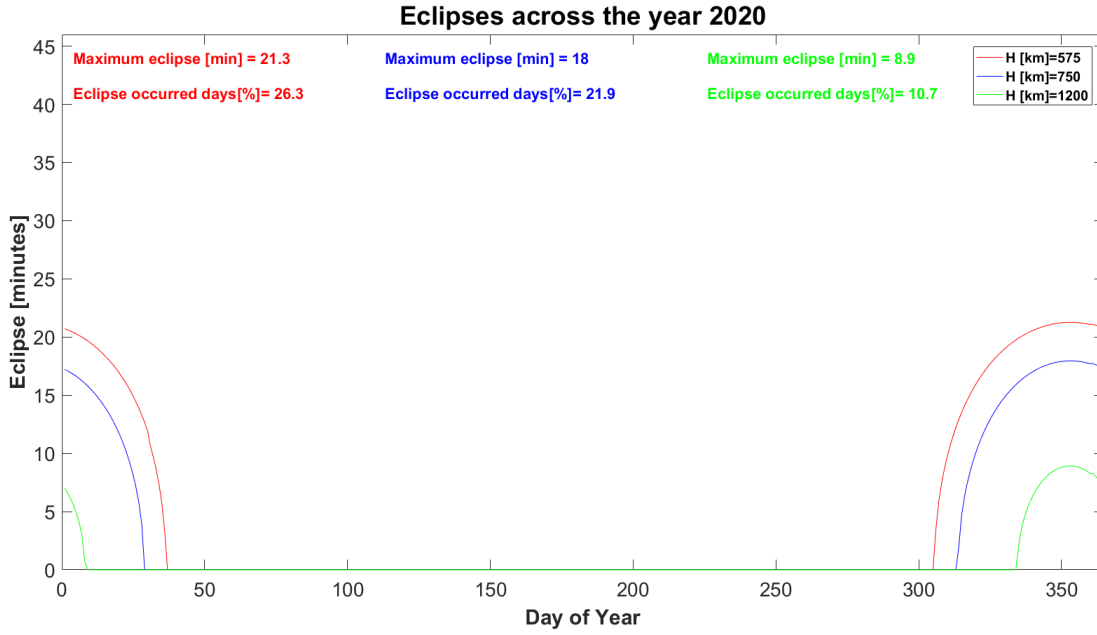


Figure 19: Eclipse duration vs altitude for orbit #3 across a year. The percentage of the eclipse occurred days over a year can also be seen in the figure.

Altitude preference has the same influence on eclipse times for orbit #3 and orbit #2. The difference between orbit option #2 and #3 with respect to the eclipses are eclipse occurrence dates. Orbit #3 reaches its maximum eclipse in winter solstice and orbit #2 reaches it in summer solstice (figure 16 and 18). From our mission perspective, orbit #3 offers the same outcome as orbit #2 since the orbital plane is same and only the RAAN parameter is different. Simulations of the theory and via STK also supports this behaviour. As a consequence of this, these two orbits can be examined together with respect to eclipses.

2.2.2.4 Eclipses for orbit option #4

Orbit #4 is also a SSO, however having a different local time (MLTAN) which result in losing the main advantage of being a solar synchronous orbit. A 10:30 MLTAN causes misalignment with the sun terminator, as a result of this it has eclipses every day of a year.

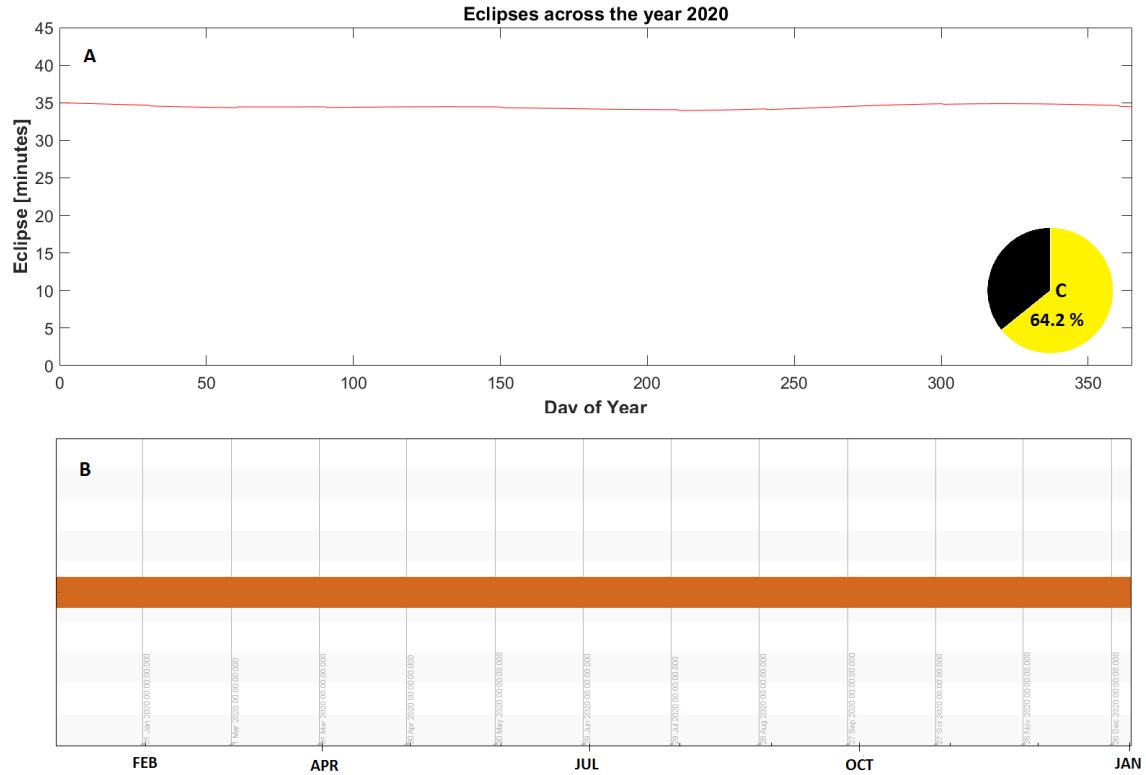


Figure 20: a) Eclipse duration by day for orbit #4 across a year. b) Eclipses over a year by STK. c) Cumulative sunlight percentage over a year (simulated by STK).

As it can be seen in figure 20, even the duration of the eclipses are much longer than other SSO candidates (#2 and #3 with about 35 minutes). An advantage of orbit #4 is being a "stable" orbit. An orbit with an almost constant eclipse time per orbit may be preferred due to the achievable thermal stability.

A more or less constant thermal environment over a long term results in a satellite temperature varying similarly every orbit. For example, the satellite temperature would be varying like a sinusoid over a day and this behaviour would be very similar for the next days, if there is no active thermal control system in use.

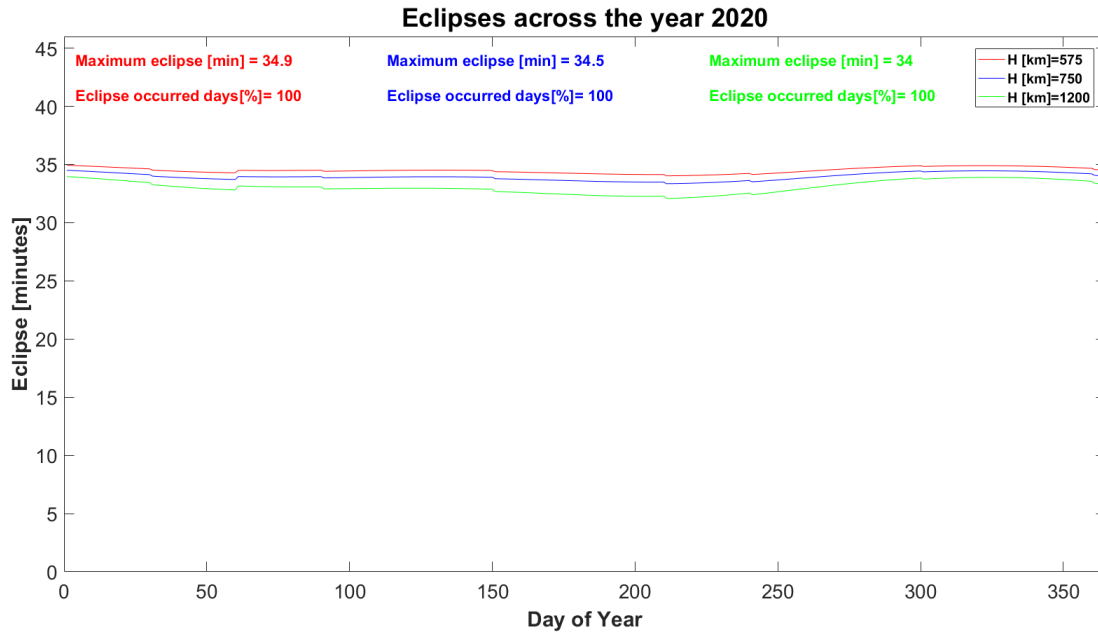


Figure 21: Eclipse duration vs altitude for orbit #4 across a year. The percentage of the eclipse occurred days over a year can also be seen in the figure.

The altitude parameter almost has no influence on the eclipse duration for orbit option #4, see figure 21. There is only one minute difference (from 35 minutes to 34 minutes) between default (red) orbit and 1200 km altitude version of same orbit (green).

2.3 Ground Visibility and Downlink Feasibility

2.3.1 Introduction

In this section, ground visibility estimations are made and discussed for two possible ground station locations in Germany: One located in Bremen (54°N, 8°E) and one located in Weilheim (47°N, 11°E). DLR has a proper ground station facility at Weilheim with five different antennas, with sizes between 4.5 and 30 meters [21] which is an advantage over the Bremen ground station. Other two ground station options in Germany can be Darmstadt and Oberpfaffenhofen where DLR also has ground facilities. However, the main purpose of this section is estimating ground visibility for Bremen and Weilheim rather than making a full ground station comparison.

Satellite communication works in a dual way, one is called uplink from the ground segment (e.g. a ground station or a mission control center) where telecommands are sent to the space segment (e.g. a satellite). And the other one is called downlink to get telemetry from the space segment to the ground segment.

Telemetry divides into two categories: payload and bus telemetry. Payload telemetry consist of science data for the BOOST mission which is needed to be transferred with a rate of about 300 Mb/day. Bus telemetry carries data that has information about actual orbital parameters and situation of subsystems.

Basically ground visibility determines how much contact duration is possible with a satellite if there is not any other limitation such as antenna signal power decay due to range¹¹.

2.3.2 Theory

The theory of ground visibility is based on geometrical relationships between the satellite and several points on the Earth such as the center of the Earth, horizon point, sub-satellite point and target point, which in our case represents a ground station. A depiction of the satellite surrounding Earth and the according variables is shown in figure 22 [17].

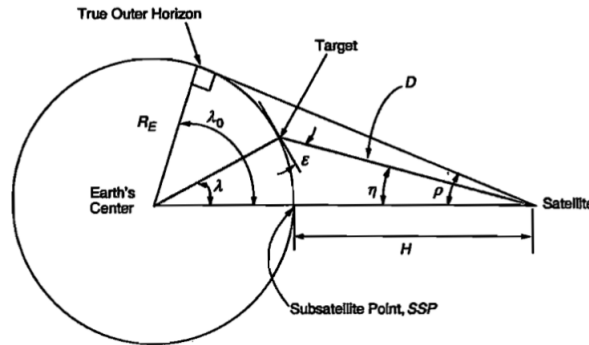


Figure 22: Angular relationship between a satellite, a target and the center of the Earth. [17]

¹¹For a LEO satellite, range usually is not a problem. However signal quality may decrease proportional to the distance.

The parameters in the above figure and the time of view for the ground location can be obtained using equations from 2.3.1 to 2.3.10 [17].

The radius of the Earth (R_e) is 6371 km and if an altitude h is chosen, the Earth angular radius in degrees for the satellite is:

$$p = \arcsin\left(\frac{R_E}{R_E + h}\right) \quad (2.3.1)$$

A ground station antenna has a specific angle which is called elevation angle (ε) between the lowest possible horizontal operating line and the horizon. Obstacles on the surface and the antenna design may reduce the maximal angle. The reduction is mostly about 5° , which I am going to use in the following descriptions. When the angular radius p and the minimum elevation angle are known, the maximum nadir angle in degrees can be obtained by:

$$\eta_{max} = \arcsin(\sin(p) \cdot \cos(\varepsilon_{min})) \quad (2.3.2)$$

The maximum Earth central angle in degrees is given by:

$$\lambda_{max} = 90^\circ - \varepsilon_{min} - \eta_{max} \quad (2.3.3)$$

The maximal distance between the satellite and the target can be calculated according to:

$$D_{max} = R_e \cdot \frac{\sin(\lambda_{max})}{\sin(\eta_{max})} \quad (2.3.4)$$

Instantaneous longitude of the ascending node of the satellite ($Long_n$) which is read from STK simulation (Figure 7) and taken as 109.5° for orbit#1, 120° for orbit#2, 300° for orbit#3 and #5, 7.5° for orbit#4. Difference in longitude between the ground station and pole is represented by Δ_l . Lat_p represents the latitude pole of the orbit and inclination with i .

$$Lat_p = 90^\circ - i \quad (2.3.5a)$$

$$Long_p = Long_n - 90^\circ \quad (2.3.5b)$$

$$\Delta_l = Long_{gs} - Long_p \quad (2.3.5c)$$

The minimum Earth central angle in degrees is given by:

$$\lambda_{min} = \arcsin(\sin(Lat_p) \cdot \sin(Lat_g) + \cos(Lat_p) \cdot \cos(Lat_g) \cdot \cos(\Delta_l)) \quad (2.3.6)$$

And the minimum nadir angle in degrees by:

$$\eta_{min} = \arctan\left(\frac{\sin(p) \cdot \sin(\lambda_{min})}{1 - \sin(p) \cdot \cos(\lambda_{min})}\right) \quad (2.3.7)$$

The actual distance between the satellite and the target is:

$$D = R_e \cdot \frac{\sin(\lambda_{min})}{\sin(\eta_{min})} \quad (2.3.8)$$

The maximum possible elevation is a check parameter which defines the maximum ground antenna elevation angle. If the antenna elevation angle is higher than the maximum possible elevation angle, contact would not be possible. The maximum possible elevation can be estimated as:

$$\varepsilon_{max} = \arccos(\sin(\eta_{min}) \cdot \frac{a}{R_e}) \quad (2.3.9)$$

where a is the semi-major axis.

Time in view [min] of the satellite by the ground station can be estimated by the formula below for one period which has above parameters:

$$Time_{view} = \frac{T}{180^\circ} \cdot \arccos\left(\frac{\cos(\lambda_{max})}{\cos(\lambda_{min})}\right) \quad (2.3.10)$$

2.3.3 Comparison of theoretical results and simulation

A simulation depends on theories, estimations, and expectations which are gathered by investigating statistical data. The theory and the simulation of STK have several issues that can cause some deviation from the actual ground visibility:

- The minimal elevation angle of the satellite antenna is not taken into account neither in the theoretic assumption nor in the STK simulation. Every antenna has a minimum elevation angle that limits its connectivity coverage degree. Thus, disregarding the elevation angle of the satellite antenna creates a deviation as a higher visibility time than the reality.
- The theory accepts the Earth as a flat plane. As consequence the horizontal range is elongated, therefore a higher visibility than the reality is achieved.
- Weather circumstances are not included which would cause less visibility time.

Despite all of deviations given above, results are expected to be close with the reality since deviation sources are minor factors. The acceptance of Earth surface as flat for such a low altitude orbit with respect to ground visibility is not a major issue. On the other hand, it would be an important factor for an Earth observation mission which requires a higher land scanning accuracy.

The main distortion by weather for space communication corresponds to a decay in signal quality which causes a lower rate of communication rate and also increases the power load on the satellite communication module.

The second and the third ones are negligible, only the first one is an important factor and should be considered at the later phases of the project.

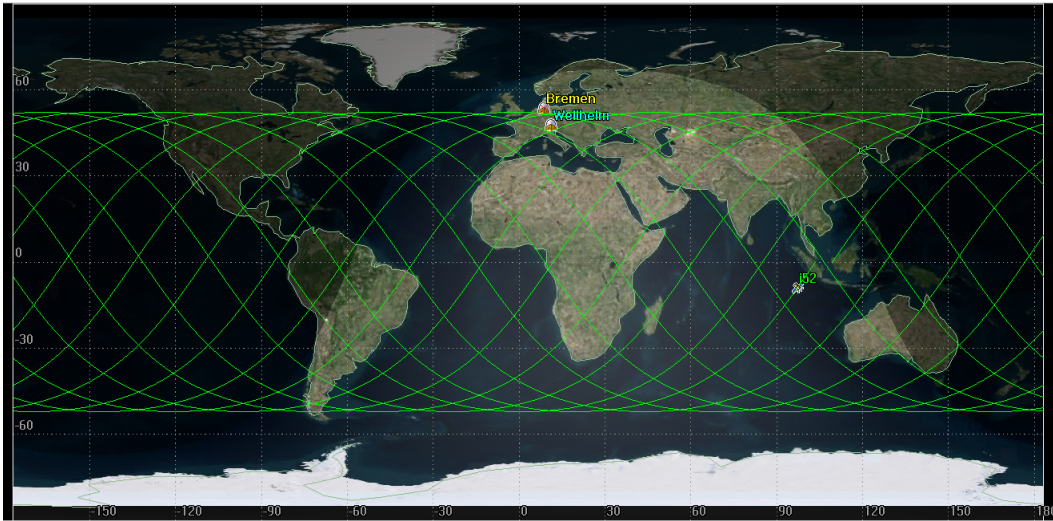


Figure 23: Simulated ground track of Orbit option #1 for one day with STK. Bremen is marked with yellow color and Weilheim with turquoise color.

Ground contact simulation with STK is made for the duration of a year for all orbit options. The estimation of daily contact duration is reached by dividing the total contact duration for a year by 365 days.

Inclination is an important parameter for calculating the ground track. There is a direct relation with inclination and ground track maximum and minimum latitudes. A satellite travels between north and south latitudes which is equal to its inclination. In reality a satellite can not pass over the same spot without making a maneuver due to disturbances like J2. J2 causes a tiny shift to the west, and the whole pattern has a slow variation of displacement to the west. The amount of shift is also related to the inclination. High inclination orbits suffering less J2 disturbance. J2 disturbance is included for both theory and STK simulations. A visualisation of the shift can be seen in figure 24.

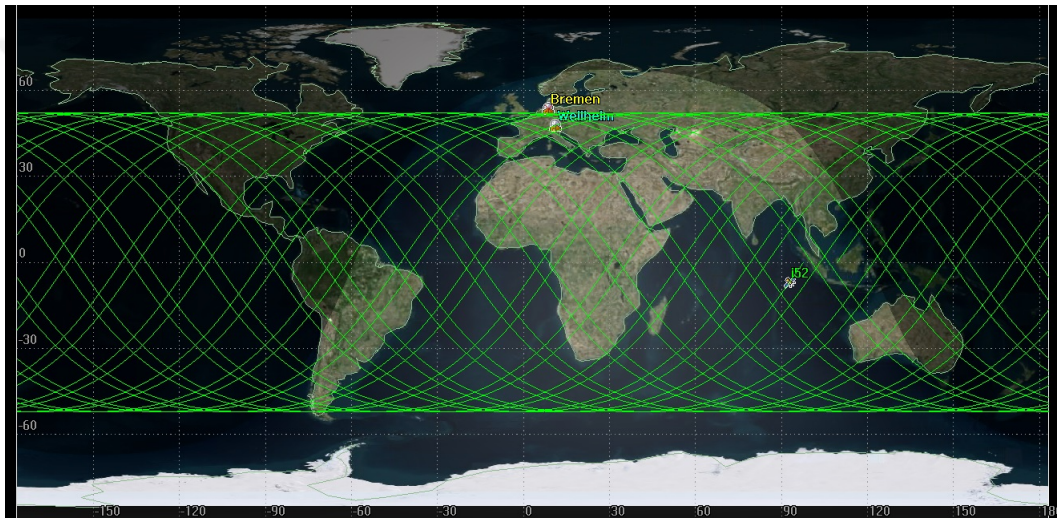


Figure 24: Orbit #1 ground track for one day and the next day from figure 23 by STK. The orbit shifted to the west due to J2 disturbance.

Orbit #1 with 52° inclination does not pass over higher latitudes than 52°N . However, Bremen has a 53° latitude. Thus, a lower visibility can be expected for Bremen than any other location between 52°N and 52°S . Because there will not be any contact between the ground station and the satellite from Bremen to northern regions.

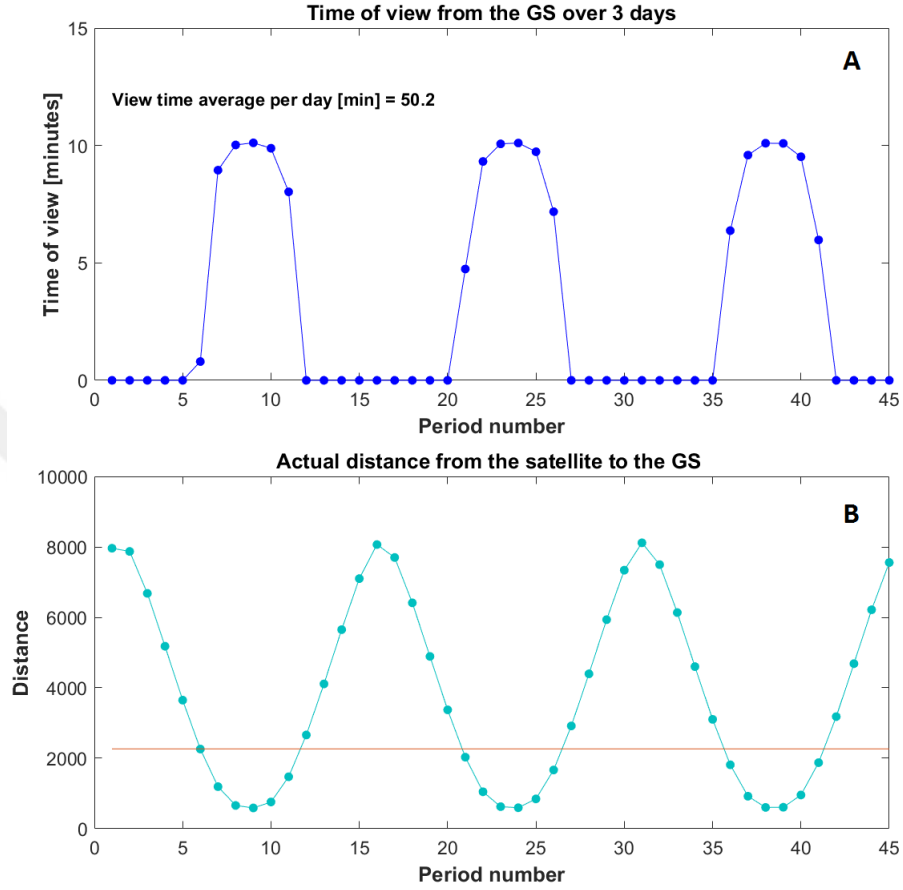


Figure 25: A: Orbit #1 ground visibility for Bremen over 3 days, dots represent visibility duration for every period. For example, during the first five periods the satellite is not visible from the ground station at all. The 6th period has about 1 minutes visibility duration and 7th period has about 9 minutes. B: Orbit #1 actual distance between ground station (Bremen) and the satellite, red line represents maximum theoretical visibility distance.

The average visibility duration is 50.2 minutes for orbit #1 from Bremen. The STK simulation calculated the visibility time to be 50.3 minutes which is more or less equal with the result from the theory. The red line in such as in figure 25 represents the maximum distance for which ground contact is still possible. When the satellite distance is lower than the red line, it means the satellite is visible from the ground station, such as in periods between eight and eleven in figure 25b.

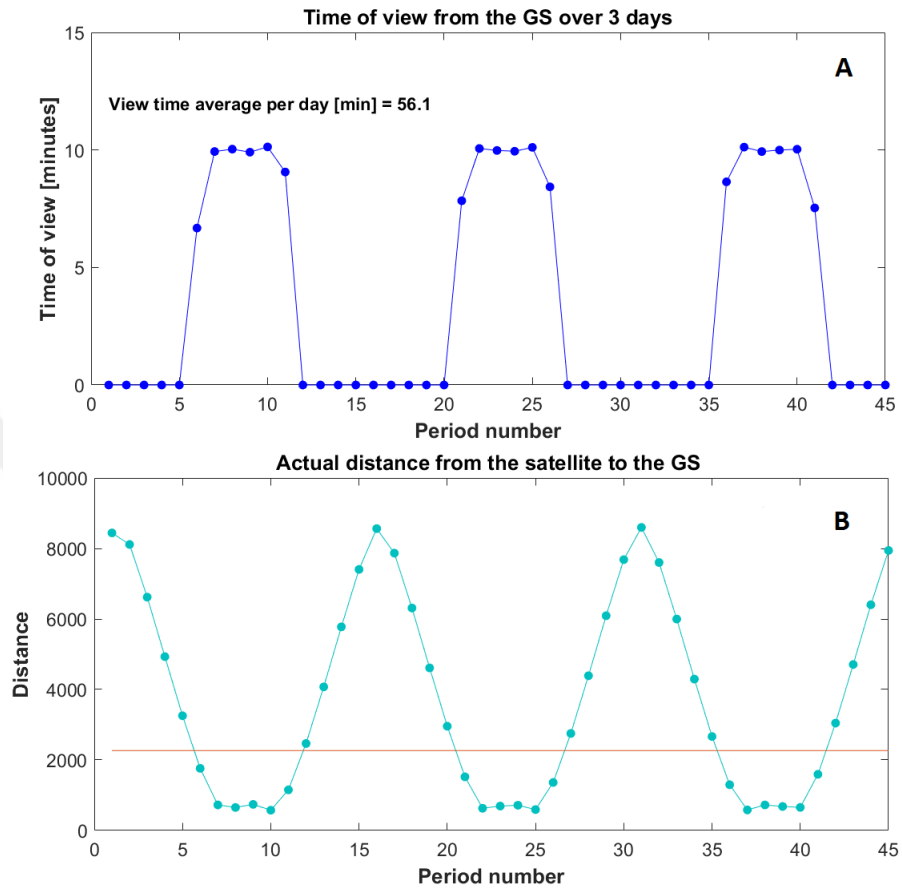


Figure 26: A: Orbit #1 ground visibility for Weilheim over 3 days, dots represent visibility duration for every period. Orbit #1 actual distance between ground station (Weilheim) and the satellite, red line represents maximum theoretical visibility distance.

Weilheim ground station offers a higher visibility time for the satellite than Bremen (about 12%). The satellite is visible from Weilheim for 56.1 minutes per orbit. STK simulations also support this with a 57.3 minutes duration. The deviation of the theory from the STK simulation is about 2% which can be accepted as a small value.

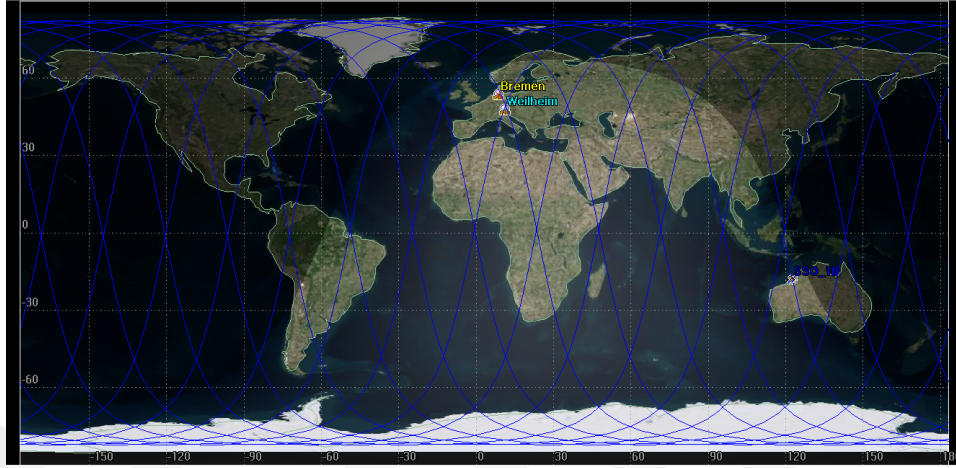


Figure 27: Orbit #2 (SSO18) ground track over one day

Orbit #2 ,#3 and #4 are polar orbits with an inclination of 97° and a 575 km altitude. Thus, those orbits have an equal amount of visibility due to having the same orbital geometry. As explained before, higher inclination orbits have a higher ground contact duration for higher altitude locations than lower altitude locations. As a result of this, a longer visibility duration is reached (as expected) for these orbits for Bremen than for the ground station at Weilheim.

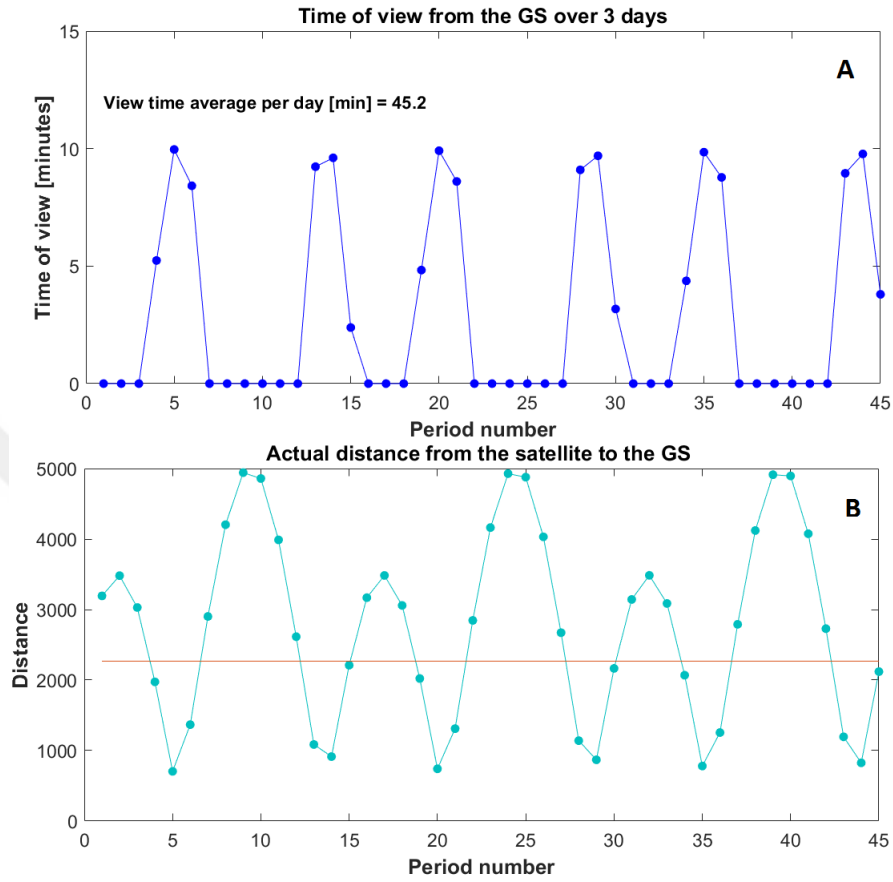


Figure 28: A: Orbit #2 ground visibility for Bremen over 3 days, dots represent visibility duration for every period. B: Orbit #2 actual distance between ground station (Bremen) and the satellite, red line represents maximum theoretical visibility distance.

Orbit option #2, #3 and #4 can offer about 45 minutes ground visibility for Bremen, figure 27a. The STK simulation for a ground access shows 47 minutes which is close with a deviation of about 4%.

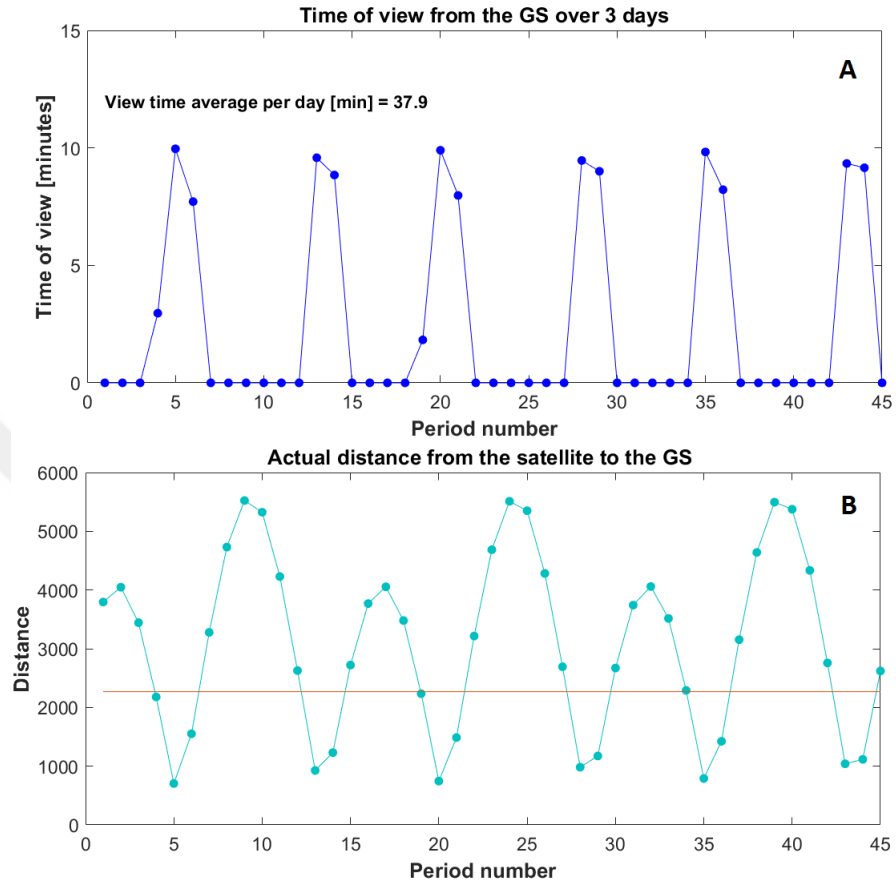


Figure 29: A: Orbit #2 ground visibility for Weilheim over 3 days, dots represent visibility duration for every period. B: Orbit #2 actual distance between ground station (Weilheim) and the satellite, red line represents maximum theoretical visibility distance.

As can be seen in figure 29A, 38 minutes of ground visibility is possible for orbit options #2, #3 and #4. The result has a small deviation about 4% from the STK simulation which shows a 40 minutes long visibility.

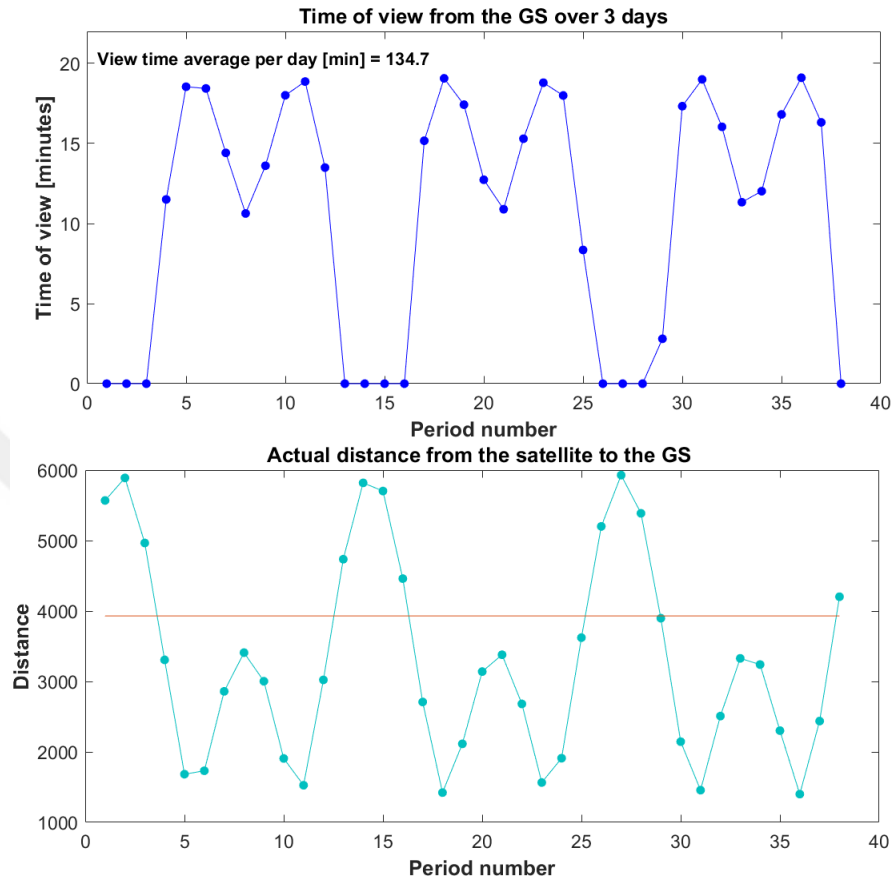


Figure 30: A: Orbit #5 ground visibility for Bremen over 3 days, dots represent visibility duration for every period. B: Orbit #5 actual distance between ground station (Bremen) and the satellite, red line represents maximum theoretical visibility distance.

Orbit #5 is a polar orbit with 1400 km altitude where the altitude is a primary parameter for ground visibility estimation. As can be seen at figure 30a, an average of 134 minutes are estimated for this orbit which is a quite long duration that makes it possible to transfer even a huge amount of data. There are more periods which allow the contact than non-contact periods. A day corresponds to about thirteen periods for this orbit and only four of them offer no visibility for the ground station. The rest of periods of the day (nine periods) consist of visibilities which are between ten to twenty minutes long for each passage near the ground station. Figure 30B shows a different maximum visibility distance by about 4000 km, which is much more than the previous ranges (2250 km, figure 29). The reason of this is that the satellite orbits farther around the Earth and making the line of sight longer. The STK simulation of the orbit #5 estimated a 138 minutes visibility which is pretty close to the theoretical estimation.

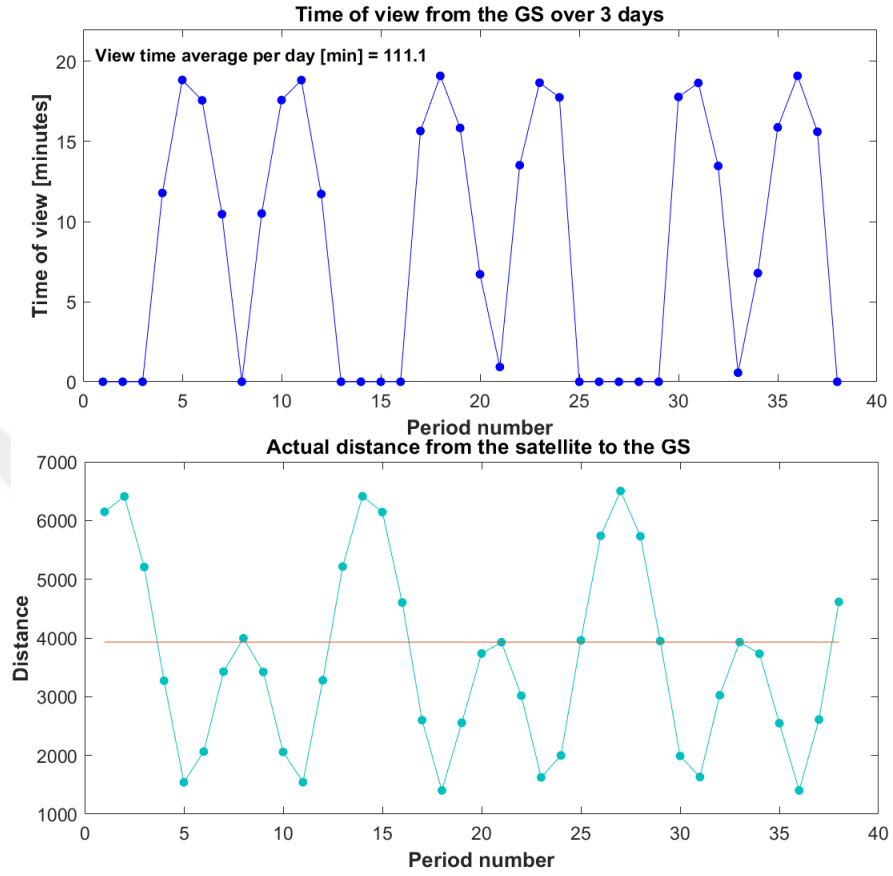


Figure 31: A: Orbit #5 ground visibility for Weilheim over 3 days, dots represent visibility duration for every period. B: Orbit #5 actual distance between ground station (Weilheim) and the satellite, red line represents maximum theoretical visibility distance.

Orbit #5 shows the same behaviour as the other SSO candidates by offering a shorter duration for the Weilheim ground station (which is logical since they are polar orbits) than a station at Bremen. The estimated visibility by the STK simulation is 114 minutes which is also very close to 111 minutes, figure 31a. It can be concluded that the theory offered reasonable results.

2.3.4 Downlink Feasibility

In space communication, there are lots of factors that have to be taken into account, such as antenna sizes of the ground facility and the satellite, weather circumstances, operating frequency, signal power and internal or external noises. Both the ground segment and the space segment are affected by those factors.

One of the system requirements, SY-IN-0080, is directly related with communication. The satellite and the ground station are required to deliver a downlink capacity of at least 294 Megabytes per day. Based on the calculated visibility times from the previous subsection, it can be estimated that much of downlink rate can meet with the mission requirement. Following basic method can be used:

$$RequiredAccess = \frac{Downlinkdatasize}{Downlinkspeed} [sec/day] \quad (2.3.11)$$

Communication science uses bits as units. Thus, the Megabytes have to be converted to bits first: one byte is 8 bits, so 294 Mbytes is 2352 Mbits. For a more realistic access time determination, a reduction of 20% is bestowed onto the visibility times.

Orbit	Visibility raw [min]	Access Time [min]	Required Downlink Speed [Mbps]
#1	50.3	40.2	0.97
#2,#3 and #4	47	37.6	1.04
#5	138	110.4	0.35

Table 6: Required Minimum Downlink Speed for a Ground station at Bremen

Orbit	Visibility raw [min]	Access Time [min]	Required Downlink Speed [Mbps]
#1	57.8	46.3	0.84
#2,#3 and #4	40.2	32.1	1.21
#5	114	91.2	0.43

Table 7: Required Minimum Downlink Speed for the Ground station at Weilheim

A satellite communication receiver or sender is capable of working with many channels simultaneously with a certain frequency and transmission rate. X band (8-12 GHz) and C band (4-8 GHz) are common frequencies for science (payload) data transmissions and S band (2-4 GHz) has been used for support of all TT&C services. The main reason for this selection is directional proportionality of frequency and transmission rate. The higher the frequency, the faster the transmission rate. An S-band link may provide about 1 Mbits per second, while an X-band link may provide several hundreds of Mbits per second. It can be concluded that, by using at least 1 Mbits rate of two channels via S-band or at least one channel of 2 Mbits via C-band the satellite payload downlink requirement is quite feasible.

2.4 Trade-off discussion

2.4.1 Overview

In this section several trade-offs are made for the mission candidate orbits. Every candidate has its own advantages and disadvantages over others. An overview of orbits with trade-off related issues is represented below in the table 8. Orbit option #2 and #3 are treated together again since they are offering same circumstances for mission related issues.

Orbit	Max Eclipses Duration [min]	Eclipse Free days over year	Cumulative Sunlight over year [%]	Access Time per day ¹² [min]	Sensitivity ¹³	Contradictions with Requirements
#1	35.5	23	69	46.2	$6.59 \cdot 10^{-10}$	-
#2 and #3	21.8	227	95.2	32.1	$7.03 \cdot 10^{-10}$	-
#4	35	0	64.2	32.1	$7.02 \cdot 10^{-10}$	-
#5	0	365	100	91.2	$8.77 \cdot 10^{-10}$	X

Table 8: An overview for the orbits

The trade off is made for following issues:

- Eclipse times
- Experiment Sensitivity
- Ground access time
- Orbit and development cost of the satellite
- Consistency with the mission requirements
- Mission environment such as radiation
- Thermal stability easiness

2.4.2 Trade-Off: Eclipse Times

Eclipse duration, especially highest expected duration, is a key factor of designing a thermal control system. The orbital period is about 96 minutes for all orbit options, except orbit #5. A satellite with orbit #1 or #4 have to stay in a dark cool environment about 35 minutes and about 61 minutes it is exposed to sunlight. On the other hand, a satellite with orbit option #2 and #3 has to stay only about 21.8 minutes in shadow. However, this amount of difference may not be enough to consider a different thermal control system but surely enough to induce changes in design parameters of the TCS and the structure of the satellite.

¹²Ground station is considered as DLR Weilheim ground station.

¹³Sensitivities as standard error of the mean for the three directions is calculated by taking the average of results from J. Holtstieges' thesis. And lower is better.

Orbit #1 has a similar maximum eclipse time as orbit option #4. However, it even offers more cumulative sunlight over a year with a 69.3% than orbit #4 which offers 64.2%. The main issue with orbit #4 is having eclipses every single day of a year.

Another issue is eclipse occurrence frequency. If a satellite experiences short eclipse times on a certain orbit, but if there is not enough non-eclipse time to recover its internal temperature balance, in long term its thermal instability might arise. Orbit option #1 has an unstable eclipse frequency, the eclipse duration changes every orbit. Orbit #4 has an almost perfect eclipse frequency with a constant occurrence. Having a constant eclipse time everyday is not preferred over an eclipse free rich orbit such as orbit #2 or #3. Because if there is a possibility of thermal dis-stability every orbit eventually the thermal instability risk may rise over time or may occur unexpectedly.

2.4.3 Trade-off: Sensitivity

Orbit #2 and #3 have similar sensitivity expectations. As it can be seen in figure 32, the KT sensitivity is strongly dependant on the inclination angle and altitude, respectively orbital velocity. As the orbit gets higher, the velocity decreases and this results in a sensitivity increase. The first four orbit options have the same orbital velocity (7.57 km/s) with an altitude of 575 km. And orbit #5 has a 1400 km altitude which results in a 7.16 km/s orbital velocity.

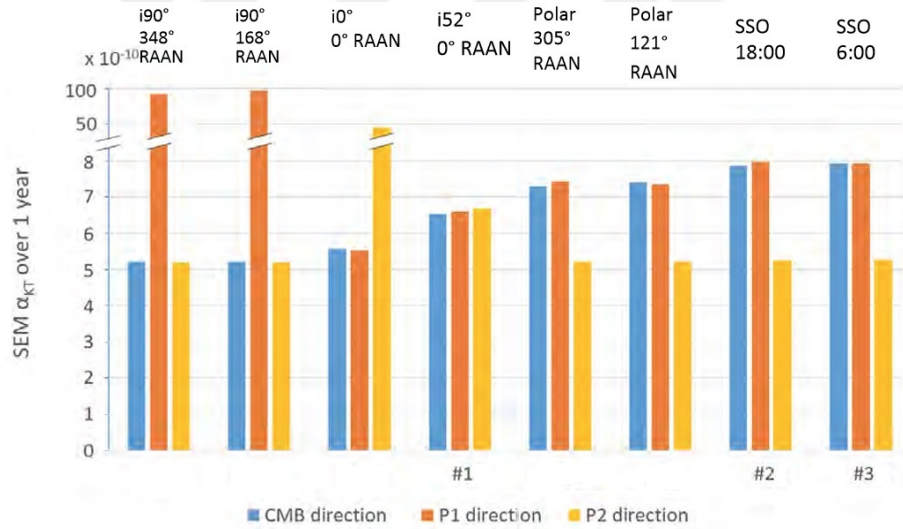


Figure 32: Experiment sensitivities as standard error of the mean (SEM) for the first three orbits and some other orbits [10].

There are three directions of measurement for KT coefficient: Cosmic Microwave Background (CMB) and two orthogonal vectors to CMB as P1 and P2 in figure 32. A lower SEM corresponds to a high experiment accuracy. A preferable BOOST mission orbit regarding to sensitivity should have a low SEM for all directions. The main reason for this is that the BOOST mission aims to take measurements in all three directions. Thus, an orbit with 90 degrees inclination or an equatorial orbit is not a good choose for the mission, see figure 32.

Orbit #1 has similar SEM rates for each directions. Orbits #2 and #3 have almost the same SEM rates compared with each other. And the SEM rates in CMB and in P1 directions are pretty close. However, for the P2 direction SEM is low as $5.2 \cdot 10^{-10}$. Orbit #4 has very similar SEM rates with orbit #2 and #3. Orbit #5 has a higher SEM rate than all other orbital options. The reason for this is having much higher altitude than the others with 1400 km rather than 575 km.

To sum up, orbits #1 to #4 offer good sensitivities. Orbit #1 has a bit better rate than for the others (from #2 to #4) with respect to all three directions but the others are acceptable too.

2.4.4 Other Issues

Orbit option #5 does not comply with a requirement set for the satellite mission which asks for at most 100 minutes period time. By using the equation 2.2.1 we can reach an altitude of about 750 km as the highest possible altitude, which meets the requirement. However, this requirement is a soft one and not much critical for the mission.

Eclipse durations for a 750 km SSO18 orbit are shown in figure 17. There is no radical difference between the orbit with the default altitude of #2 and the 750 km version. However, this new orbital option can enter as a candidate orbit for the mission from now on and will be treated as orbit option #5.1 which is MI-DE-0010 adapted version of orbit #5.

A secondary concern about orbit #5 is radiation. At an altitude of 1000 km, the inner Van Allen radiation belt starts. This environment is very harmful and will have to ensue precautions, such as heavy radiation shields. Moreover, the radiation might even render the experiment unfeasible since it is very sensitive to radiation and requires additional shielding at this altitude. Radiation concern will also increase the mission cost due to extra radiation tests, radiation shield design and extra work load. The satellite mass would be higher with heavy radiation shields which causes a higher mission cost.

Updating the orbit to version 5.1 appeased the radiation concern of version 5. However, in the process, the advantage of being eclipse free is lost. Orbit #5.1 is a similar to orbit options #2 or #3. The main difference between these two is the altitude, 575 km and 750 km respectively.

Orbit cost includes cost of orbital injection & maintenance cost and launch vehicle cost is also included. A polar orbit costs slightly more than lower inclination orbits. The main reason for this is that launch sites are mostly located at lower latitudes and launch vehicle need to fly longer therefore more maneuvers are required. Altitude of the satellite is a more important parameter than the inclination. Injection of a satellite further, would increase the cost due to the launch cost. Additionally, more satellite propellant is required to insert the satellite to its final mission orbit.

All of the candidate orbits offer a good ground visibility. Any ground station in Germany would offer a similar visibility for this mission and can meet telemetry data requirement. Thus, ground station selection should be made depending on ground station facility equipments, cost and agreement terms with the service provider.

Ground station access time is not a critical issue for the BOOST mission. Payload telemetry size (≈ 300 Mbytes/day) is a moderate size for a space mission, a 2 Mbits/s downlink rate is enough for all candidates. Today space communication technology is enough advanced to provide even much higher

data link rates such as 100 Mbit/s by a X-band frequency.

Rather than other candidates, orbit options #5 and #5.1 need to meet a special requirement which is the de-orbiting maneuver. The altitude of orbit options #1 to #4 are chosen regarding the principle of "free orbital decay" after mission life time. With (at least for now) the optimum altitude for a free decay 575 km, orbits #5 and #5.1 are required to do following maneuvers:

Orbits	Required Disposal Delta V [m/s]	Required ΔV for Orbital Transfer [m/s]		Required Propellant Mass ¹⁴ [kg]	
		Hohmann Transfer	Spiral Transfer		
#5	378.6	454.1	454.7	26	33
#5.1	182	134.8	134.9	13.5	9.5

Table 9: ΔV budget: Satellite disposal and orbit transfers for orbit #5 and #5.1. Equations from 2.2.4 to 2.2.7 are used.

To dispose the satellite, for orbit #5.1 target perigee altitude is taken as 100 km and for orbit #5 50 km since it's apogee is much higher and would result in several more orbits over which apogee decreases before the spacecraft re-enters the atmosphere. Otherwise an adequate control of the deorbiting progress might not be possible.

Hohmann transfer and spiral transfer requires almost same Delta-velocities, the reason for this is that the difference between initial and target altitude is not enough to create a significant difference. For example, for a GEO satellite transfer method would be more significant. However, it would not be a good choice for a GEO satellite since the satellite has to make lots of spirals before reaching the target orbit. In the case of using both methods for orbit options #1 to #4, it is assumed that the altitude will decrease to 550 km after mission life time and will decay within 25 years. Specific impulse I_{sp} of thrusters is taken as 300 seconds to estimate the amount of propellant mass, that should be saved for the final maneuver. And the initial total mass of the satellite is taken as 250 kg.

Orbit trade-offs are summarized in table 10 and signs are used to rank mission issues among orbits.

¹⁴Columns of Propellant mass dedicated to Disposal and Hohmann Transfer respectively.

		Orbit Number				
Issues		1	2&3	4	5	5.1
Eclipses	Duration	o	++	o	+++	++
	Free Days	--	+	---	+++	++
Ground Station Access Time ¹⁵		+	o	o	+++	++
Sensitivity		+++	++	++	-	+
Thermal Stability Easiness		---	++	-	+++	++
Radiation Concern		Low	Low	Low	High	Low
Cost	Orbital Cost	+++	++	++	--	o
	Development Cost	+++	+++	+++	--	+++
Contradictions		None	None	None	Yes	None

Table 10: Issues vs Orbits

It can be concluded that orbit #5 is the best option with respect to eclipse times which is directly related to thermal issues and solves that issue too. However having a high radiation concern and high orbital cost, orbit #5 does not seem preferably. Orbit option #5.1 is a better option than #5 since it eliminates radiation concern and decreases the orbital cost. The main disadvantage of #5.1 is the amount of propellant that is needed to be saved for satellite disposal. Which about 10 kg and for a small satellite mission it is still a significant amount.

Orbit #1 is a good option with low cost, good access time and with the best sensitivity. However the experiment is very sensitive to thermal fluctuations and requires very high thermal stability. Thus, orbit #1 can not be the best orbit option.

Asking for a high thermal stability, orbits #2, #3, #5 and #5.1 are the best choices. However, option #5 already discarded from the list. And #4 does not offer a good thermal stability. In addition, it is getting less sunlight than the others which means a lower power production. Thus, option #4 is not a better option than options #2 and #3.

A more detailed orbital cost and sensitivity estimation should be made for the option #5.1 after having one or more bus options for the mission. It is good to state that there is no significant difference between orbits #2, #3 and #5.1, as a consequence of this a satellite would be easily adaptable to switch between the orbital selection from orbits #2 or #3 to #5.1. Until then orbit #2 and #3 stay as the best options.

¹⁵Despite of having different ranks for the visibility, all orbits are found adequate for meeting with the downlink requirement.

2.5 Nominal Mission Scenario

2.5.1 Overview

In the previous section, candidate orbits are compared with respect to several important aspects and a conclusion on which orbit offers the best interest for the mission was made. In this section two, mission scenarios are suggested. The first one depends on orbit #2&3 and the second one an orbit on option #1. Scenario 1 should be accepted as baseline design. The ground station assumed for both scenarios is the ground station at DLR Space Operations and Astronaut Training Center, Weilheim. Mission life time is conserved as 2 years.

2.5.2 Scenario option 1

To achieve the mission goals, the experiment should run undisturbed for a duration as long as possible. An open question about this issue is, whether the payload will take experiment measurements during eclipse or not. The answer to this question lies behind thermal stability. If no thermal instability would be expected then the experiment can go on, and in the worst case during eclipses time the experiment can be paused for a while. Scenario 1 as a baseline scenario accepts the issue when eclipses occurs, measurements are not to be taken.

Inclination [deg]	Altitude	Period [min]	Orbital Velocity [km/s]	RAAN [deg]	MLTAN [local time]
97	575	96.02	7.575	10.7/190.8	18:00/06:00

Table 11: Orbital parameters of scenario 1.

Eclipse Free [Days]/[Periods]	Raw Expected Sensitivity	Ground Access Time [min]	Required Downlink Rate for PL [Mbits/s]
264 / 3960	$7.03 \cdot 10^{-10}$	32.15	1.21

Table 12: Experiment and ground communication parameters of scenario 1.

Eclipses are not expected to happen about 264 days over a year, with other words: for 264 days no thermal instability is expected to occur. The experiment is planned to run all the time over a year. In the worst case, (probable during eclipse occurring periods) experiment to be paused for a certain amount of time until reaching back a thermal balance.

2.5.3 Scenario option 2

In the case thermal stability is assured for orbit #1 which is the one of the most important risks for the mission, orbit #1 can be preferable for the mission. However, the experiment is very sensitive to thermal disturbances and this is not much expected.

Inclination [deg]	Altitude	Period [min]	Orbital Velocity [km/s]	RAAN [deg]	MLTAN [local time]
52	575	96.02	7.575	0	-

Table 13: Orbital parameters of scenario 2.

Raw Expected Sensitivity	Ground Access Time [min]	Required Downlink Rate for PL [Mbits/s]
$6.59 \cdot 10^{-10}$	46.2	0.84

Table 14: Experiment and ground communication parameters of scenario 2.

3 Sub-Systems

3.1 Attitude Control System (ACS)

3.1.1 Introduction

This subsection tries to answer the question of which attitude control system is the best option for the BOOST mission, rather than explaining ACS techniques or designing a full ACS for the mission. Initial condition to control an attitude is a good determination of the attitude by sensors like a sun sensor or a gyroscope. This system is called attitude determination system and is not covered here. Two attitude control methods, which are 3-axis stabilization and spin stabilization are examined here after short descriptions and later a trade-off is made for the mission.

A satellite, when placed in its orbit, experiences various disturbing torques such as gravitational forces from other bodies (Earth and Moon), magnetic field interaction, solar radiation pressure etc. Due to disturbances the satellite slowly drifts in time and orientation also changes. The orientation should be fixed at a degree of accuracy for various reasons. The most common reasons are *antenna directivity* to ensure that the satellite has enough access duration with a ground station, *sun-pointing* to allow solar panels get the maximum sun light and *nadir-looking* to scan a certain area on the surface which is mostly used for earth observation missions.

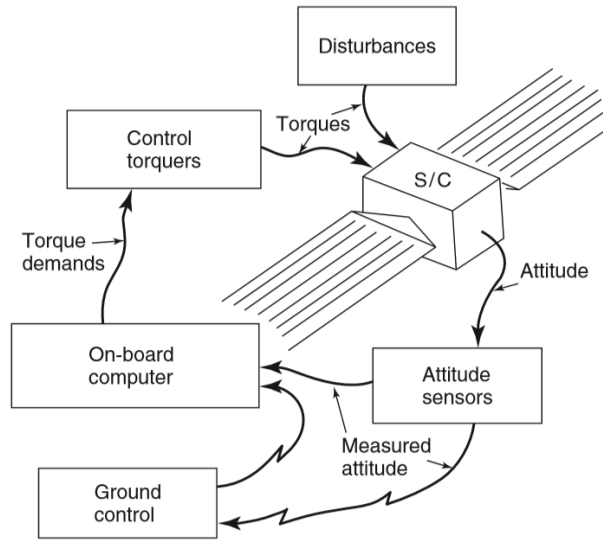


Figure 33: Block diagram [19] shows major elements of an attitude control system. Arrow indicates relationships between elements and information flow can also be noticed. The main body can get torques by internal (control thrusters) or external (disturbances) sources. Attitude is measured by sensors and this data flows to a ground station which can also send new attitude commands. The on-board computer has an attitude control software (or program) which takes decisions depending on inputs and/or desired attitude.

Some of mission aspects are key factors for selection and specification of the ACS for the satellite. Pointing requirement for a module like the antenna, solar panels or an instrument are some of the constraints for an ACS. A spin stabilized satellite normally points one direction (which is actually along spinning axis) and it needs thruster fires to change its pointing direction. Thus, spin stabilization is not a good choice for earth pointing and pointing accuracy is lower than 3-axis stabilization. Orbit altitude is an important parameter for external disturbances. For example atmospheric drag and gravity gradient of the Earth decrease linearly as altitude increases. Mission life time defines how much propellant is needed to maintain stability and pointing accuracy by an attitude control system, if propellant requirement is not feasible or too expensive, another ACS should be chosen or the current one should be re-adapted. Slewing corresponds to reorientation of the spacecraft to re-point the antenna, solar panels or the payload, which is in a close relationship with propellant budget.

An attitude control system is in relation regarded to several other subsystems like thermal, power, structure and propulsion. A satellite may need special maneuvers for a payload reason or just thermal stability reasons. For example a payload may be required to accommodate with as low as possible temperature and the satellite ACS occasionally has to make a maneuver with respect to the sun to save payload compartment from direct sunlight. The relation to the power system is in both ways: The first one is solar array pointing requirement which should be met by ACS, this requirement basically is a margin in the angle of the incoming sun rays to the panels to make sure that the panels are able to produce a certain amount of power. The second one is related with ACS power load and the required power regulation for the ACS. Both should be met by the power system. The type and specification of the ACS also define attitude control thruster sizes, locations and minimum impulse amount, which are important parameters for accuracy and propellant budget. Several structural issues are directly related with ACS preference: center of mass tolerance, sensor mounting and design flexibility.

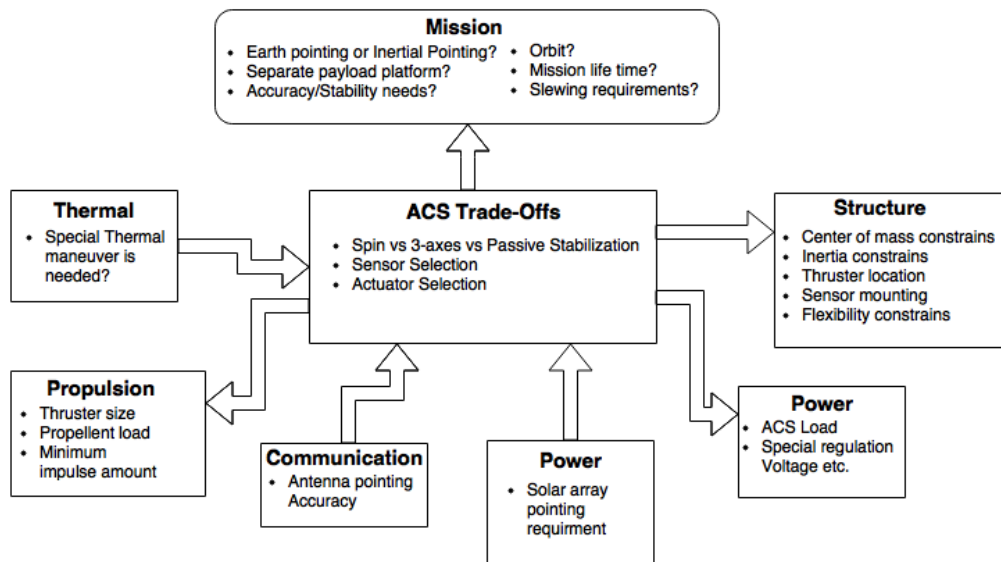


Figure 34: Main relationships between ACS, other subsystems and several mission aspects.

3.1.2 External Disturbances on the Satellite

There are four major disturbance sources for a satellite orbiting our Earth. Those are: solar radiation, atmospheric drag, magnetic field of the Earth and gravity gradient of the Earth. External torques that may act on the satellite should be estimated to decide which type of ACS would be the best option and specifications of the control system can also be estimated. The main question is which disturbance is dominant, which type of disturbance overcome others. Before starting to estimate external disturbances, we need to know the satellite shape, diameter, mass and inertia on the different axes and naturally some orbital parameters.

The first example design is a cylinder spin stabilized satellite which spins around the z-axis and travels along the y-axis. Radius is 0.65 meter, and height is 0.75 meter. The second one is a cube 3-axis stabilized and sun pointing satellite with 1 meter sides and four 1 meter solar panels on each side. Both satellite masses are assumed as 250 kg and total volume for both is 1 m^3 . Inertia for both satellite:

Spacecraft 1: $I_z = mr^2/2$ and $I_x = I_y = (m/12)3r^2 + h^2$.

Spacecraft 2: $I = ms^2/6$

Where s is the edge length of one side. However, we have four solar panels that influence two of the inertia. Symmetrically centered panels don't have any effect on inertia around z (yaw) and influence on other axes are estimated roughly.

Putting in the numbers we get: $I_x = I_y = 38.12[\text{kg} \cdot \text{m}^2]$, $I_z = 52.81[\text{kg} \cdot \text{m}^2]$ for SC1 and $I_z = 41.6[\text{kg} \cdot \text{m}^2]$, $I_x = I_y = 50[\text{kg} \cdot \text{m}^2]$ for SC2.

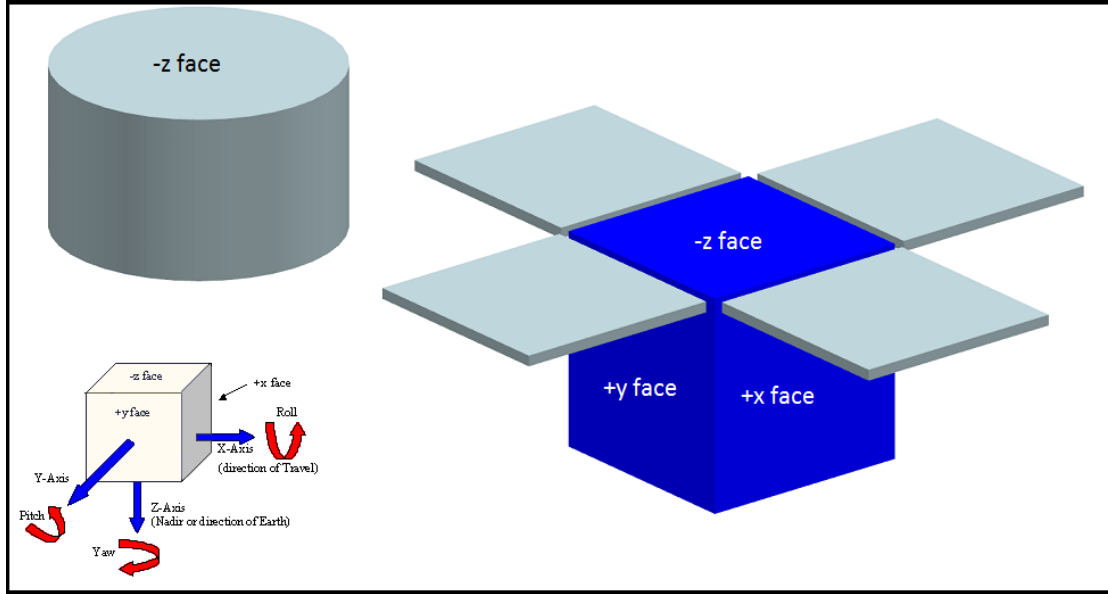


Figure 35: Two example bodies for external disturbances estimation. The cylinder represents the spacecraft one and the cube represents the spacecraft two.

Gravity gradient of the Earth: Produces a constant torque for earth looking satellite (always same side points the Earth) and cyclic for other oriented spacecraft.

$$T_g = \frac{3\mu}{2 \cdot R^3} |I_z - I_y| \cdot \sin(\Theta) \quad (3.1.1)$$

Where T_g is the maximum gravity torque, μ is the Earth's gravity constant ($3.986 \cdot 10^{14} \text{m}^3/\text{s}^2$); R is the orbit radius in meters, Θ is the maximum deviation of the Z-axis from local vertical. Mostly a satellite has more than one altitude control (AC) modes, some modes may require additional slew rates for pointing. For spacecraft one (SC1), Θ angle is taken as 1 and 2 degrees since it is a spin stabilized spacecraft and has less motion flexibility. For spacecraft two (SC2), Θ angle is taken as 1 and 10 degrees. Because a 3-axes stabilized satellite has no rotational maneuver limit.

Solar radiation: Is an energy flow from the Sun that carries all wavelengths of electromagnetic radiation. Lots of photon particles have been carrying by the flow and those particles hit the spacecraft sun facing surface. As a result a small torque appears. It is a constant torque for a Sun oriented satellite and for other types of orientations it acts cyclically. The magnitude of the torque depends on the Sun facing surface area, the surface reflectivity and the spacecraft center of mass. A surface is either transparent, absorbent or a reflector. However, a body is usually made of a mix of them. Solar panels act more like an absorber while the main body acts like a reflector.

$$T_{sp} = F \cdot (c_{ps} - c_g) \quad (3.1.2a)$$

$$F = \frac{F_s}{c} A_s (1 + q) \cdot \cos(l) \quad (3.1.2b)$$

Where T_{sp} is solar radiation pressure F_s solar constant (1367 W/m^2), and c is the speed of light. A_s is the surface area, c_{ps} is the location of solar pressure, c_g is the center of gravity, q is the reflector factor (ranging from 0 for perfect absorption to 1 for perfect reflection), and l is angle of incidence of the Sun. A_s is 2.6 m^2 (about half of the surface area) for SC1 and 5 m^2 for SC2 (Sun oriented). q is 0.6 for SC1, 0.5 for SC2 since we expect more solar panel surface absorbs more. Incidence angle is taken as 0° for both spacecraft. Center of mass difference is taken as 0.1 meter for SC2 and 0.2 meter for SC1. A cubic well balanced body directly facing the Sun has a lower center of mass difference than other cylindrical bodies. Because SC1 is not a sun oriented satellite where sun waves come with various different degrees.

Magnetic field: Earth has a strong magnetic field that protects the planet from outer disturbances like solar winds. The magnetic field lines flow from the south pole (which is magnetic north pole) to the north pole (which is magnetic south pole). Magnetic fields create a cyclic torque on a satellite depending on the satellite magnetic dipole, altitude and orbit inclination.

$$T_m = D \cdot B \quad (3.1.3a)$$

$$B = 2 \cdot M/R^3 \quad (3.1.3b)$$

T_m , the magnetic torque on satellite can be calculated according to: D is residual dipole of satellite¹⁶ in $\text{A} \cdot \text{m}^2$ and B is Earth's magnetic field in tesla. B can be estimated by the equation 3.1.3b

¹⁶Residual magnetic dipoles can range anywhere from 0.1 to $> 20 \text{ A} \cdot \text{m}^2$ depending on the spacecraft's size and whether any on-board compensation is provided. On a small-sized, uncompensated vehicle, $1 \text{ A} \cdot \text{m}^2$ is typical [17].

for a polar orbit. M is magnetic moment constant of earth, $7.986 \cdot 10^{-15} \text{ T} \cdot \text{m}^3$ and R is radius from earth center.

Aerodynamic: The surface of the satellite that is pointed along the direction of velocity gets hit by atmospheric atoms and molecules. The density of the atmosphere is strongly linked with altitude. It is very weak above 750 km and roughly at that altitude the aerodynamic disturbance on the satellite is overcome by other disturbances (see figure 41). However, for a very low orbit body (lower than 400 km), it is the most dominant disturbance. Atmospheric drag is constant for earth pointing satellite and varies for other oriented satellite.

$$T_a = F \cdot (c_{pa} - c_g) = F \cdot L \quad (3.1.4a)$$

$$F = 0.5(p \cdot C_d \cdot A \cdot V^2) \quad (3.1.4b)$$

Where F is the atmospheric drag force on body, C_d is drag coefficient (usually between 2 and 2.5), p is atmospheric density, A is surface area, V is the satellite velocity, c_{pa} is center of aerodynamic pressure and c_g is the center of gravity. In our example, the surface area (which is related with the direction of motion and the orientation of the spacecraft) is taken as 3 m^2 for SC1 and 2 m^2 for SC2. The offset from center of mass ($c_{pa} - c_g$) is assumed as 0.2 cm for SC1 and 0.1 cm for SC2.

Spacecraft	Gravity Gradient[N·m]	Solar Radiation[N·m]	Magnetic Field[N·m]	Aerodynamic[N·m]
#1	$\Theta 1^\circ: 9.03 \cdot 10^{-7}$ $\Theta 2^\circ: 1.8 \cdot 10^{-6}$	$3.79 \cdot 10^{-6}$	$4.73 \cdot 10^{-5}$	$8.36 \cdot 10^{-6}$
#2	$\Theta 1^\circ: 5.21 \cdot 10^{-7}$ $\Theta 10^\circ: 5.11 \cdot 10^{-6}$	$3.41 \cdot 10^{-6}$	$4.73 \cdot 10^{-5}$	$2.78 \cdot 10^{-6}$

Table 15: Estimated disturbance torques for both example spacecraft

The table tells us magnetic field has the most influence on both bodies, atmospheric drag is the second most influential for the second bodies and solar radiation is the second stronger influence for the first body. An additional point that requires attention is the direction of the disturbance torque, for a better estimate all disturbances should be added together with taking into account their directions.

3.1.3 Spin Stabilization

The main principle of spin stabilization of a spacecraft is making the satellite body spin on an axis perpendicular to the orbital plane to overcome any other disturbance momentum. Commonly used spin rate varies between 30 and 100 rpm [18]. Since disturbances decrease with altitude, it is more feasible to spinning with a lower rate for a GEO satellite than a LEO satellite. Lots of early satellite maintained their attitude by spin stabilization and the shape of the spacecraft is usually chosen as cylindrical to decrease the amount of aerodynamic friction while satellite spins. Inertial momentum which is created by a spinning disk connected to a motor should be at least 10% larger than the inertial moment by any transverse axis to maintain stability [18]. And the main condition for inertial stabilization is $I_z > I_x = I_y$ where the body spins about z-axis.

There are two spin stabilization methods: the first one is single spin stabilization where the whole satellite spins and the second one is dual-spin stabilization which is developed to eliminate the main disadvantage of a single spin system, low communication efficiency.

Single spinning is a simple and cheap attitude control system when compared to other methods, however most of orbital time the satellite antenna looks to outer space rather than to the Earth. This drawback does not cause only a crucial decrease for ground station access time but also results in a lower power efficiency since most of time antenna beams propagate to undesired directions. In a dual-spin stabilization the communication module is placed in a part of the satellite called *platform* and the rotating part is called the *rotor*. A dual-spin stabilization representation can be seen at figure 36.

A spin stabilized satellite is inherently stiff due to the rotational body which means it is less sensitive to any other physical disturbance and also stiffness is a wanted feature for spacecraft structure design. Due to the cylindrical shape of the spacecraft, solar panels are mounted on the body (side surfaces of cylinder). The main disadvantage of this necessity is a low power production by the solar panels: less than half of panels can work simultaneously, actually since sunlight varies over round surface, it becomes very low at far edges and actual total power production becomes around a third. On the other hand, it is easy to handle thermal issues on a spinning body. A more or less constant rate rotating body gets the same amount of solar fluxes to each side. Thus, the heat flows (with a close margin) from outer space to inner structure are well distributed over all sides, also the same is valid for heat outflows.

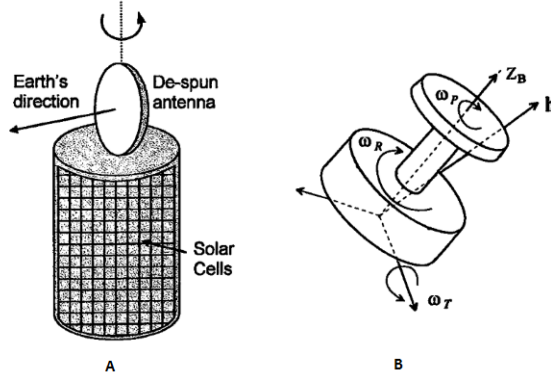


Figure 36: A: Outer look of a dual-spin stabilized satellite [18]. B: An inner design schema of dual-spin system: w_T denotes momentum in the transverse plane and P and R denote platform and rotor, respectively. When there is no external disturbance angular momentum, h , equals to zero.

Regarding pointing, it is a quite limited system where the satellite rotates about an initially fixed axis. Any re-pointing maneuver requires significant amounts of propellant and pointing accuracy is between 0.3° and 1° and proportional to spin rate.

The satellite bus for the BOOST mission is going to carry a sensitive payload, where any disturbance like undesired moments or any other noises (sound or vibration) may cause serious problems for the measurement or even the payload itself. One of the mission requirement states a 3 rpm as maximum tolerable spin rate, which obviously is a low rate to maintain stability by a spin stabilization control. The second concern is related with possible vibration due to moving parts and structure of the satellite. Even if the satellite would pass spinning noise tests before launch, it would not be certain that noise may pass tolerance level due to any internal or external reason.

To estimate the required spin rate for attitude stabilization, first the total momentum that is needed to keep the body spinning should be estimated. In theory, momentum storage in spinning body can be treated same as a momentum wheel storage. Angular momentum (h) should be known to estimate the spin rate:

$$h = \frac{T_d \cdot P}{4 \cdot \Theta_a} \quad (3.1.5)$$

Where T_d is disturbance torque, P is period in seconds and Θ_a is allowable motion.

$$w_s = \frac{h}{I} \quad (3.1.6)$$

Where w_s is spin rate and I is internal inertia of spinning axis. The inputs of above two formulas are: T_d as $4.73 \cdot 10^{-5}$, P as 5761 seconds, Θ_a 1 radian and I as I_z which is $52.81 \text{ kg} \cdot \text{m}^2$. We get angular momentum as $39.15 \text{ N} \cdot \text{m} \cdot \text{s}$ and spin rate as 0.741 rad/s which equals to 7.07 rpm from the equations 3.1.5 and 3.1.6. This spin rate does not have any margin that would guarantee the stabilization. So at the flight model, we can expect to use a higher rate then we found by our estimation. Just remember, only one type of disturbance (magnetic field) is taken into account, however other disturbances may increase the total disturbance.

3.1.4 3-axis Stabilization

Today 3-axis stabilization is a more common method than other stabilization methods. It is a very accurate stabilization method which is really needed especially for missions that require a very high pointing accuracy like Hubble space telescope for observation of the universe.

The control torques about the 3-axis, i.e. yaw, pitch and roll, come from a combination of reaction wheels or momentum wheels. A momentum wheel and a reaction wheel generally have the same design however the control principle of the two systems are different. A reaction wheel at off mode rotates very slowly unlike former reaction wheels to prevent jamming and when a rotation is needed it starts to speed up at the opposite direction of required rotation, then slows down. A momentum wheel rotates constant at high rates to give a high angular momentum to spacecraft to eliminate any possible disturbances.

Three reaction wheels can provide all degrees of freedom (which is totally six-two per axis) for a vehicle, however a fourth one is added for redundancy. Four basic reaction wheel designs represented in figure 37.

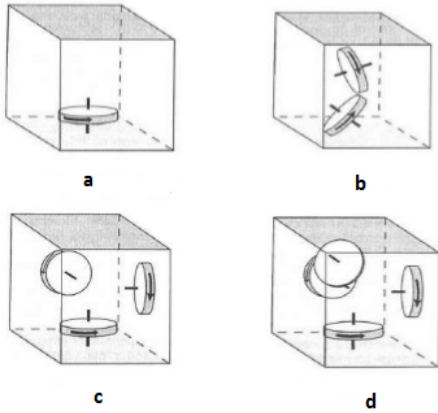


Figure 37: Represent the most common 3-axis stabilization designs with reaction wheel(s): single, dual, three and four reaction wheels design respectively.

There is another four wheels design called *pyramid reaction wheel configuration* where the reaction wheel's central axes points to one point with a 45° from central axis to the normal where normal line passes over the point. Optimum reaction wheel configuration should be set for a satellite to maximize the efficiency. Satellite sensors provides necessary inputs for the attitude control system. When the satellite sensors detect that the satellite is moving away from the proper orientation, the reaction wheels speed up or slow down to return the satellite to its correct position. A reaction wheel also works as a momentum storage device and stores disturbance momentum up to some levels. A reaction wheel has a specific momentum storage capacity and this quantity is also a matter that requires attention upon selection of the wheel.

Momentum storage in a reaction wheel can be estimated as below [17]:

$$h = T_d \cdot P \cdot (0.707) \cdot \frac{3}{4} \quad (3.1.7)$$

Where T_d is the disturbance torque, P is the orbital period in seconds, 0.707 is the rms average of sinusoid function and $3/4$ is the correction factor for magnetic disturbances since T_g is at the maximum rate over pole regions and lower at the equator regions. We get a momentum amount 0.144 N·m·s from the equation. A reaction wheel with about 10 times margin would be a good choice for our satellite with 1.4 N·m·s. On the other hand, the total disturbance probable would be higher than our estimated T_d since other disturbances may act (at some point) in the same direction with the magnetic torque.



Figure 38: An example of reaction wheel as a suggestion for baseline design from Blue Canyon technologies [22]. A higher model with 8 N·m·s momentum is also available if necessarily.

A 3-axis stabilized satellite in theory has no constraints for pointing, the satellite can point anywhere with a very high accuracy, 0.001° to 1° depending on the sensor and the actuator (e.g. reaction wheel or momentum wheel) selection. In orbit various external and internal disturbances would be stored in the reaction wheel and momentum storage capacity would eventually become saturated. The reaction wheel should be dumped by an actuator to allow it operates normal again. Those actuators can be a magnetic torquer and/or attitude control thrusters. To save propellant and make the design cheaper and simple magnetic torquers are very useful. Moreover a polar orbit mission like the BOOST mission offers more magnetic torque than lower inclined orbits since the magnetic field gets stronger at the poles.

A 3-axis stabilized system maintains its stability at zero momentum, which means any disturbance should be eliminated by applying a negative one of equal quantity. The main advantage of this system is high power productivity from solar panels and it also allows to use larger solar panels unlike a spin stabilized satellite. A zero momentum system allows satellite antenna connectivity easiness however a certain amount of steering is necessary if the satellite is not an Earth pointing satellite. An example of reaction wheel is given in figure 38 with its specification.

Beside of various advantages of a reaction wheels stabilization system, it is an expensive and power consuming system. The cost is high by a few reasons: reaction wheels are expensive parts (a single medium size costs about 100k\$), additional propellant should be added to the spacecraft to use for momentum dumping and larger solar panels to feed power makes the spacecraft larger, this volume difference is an additional cost especially for launch expense. A reaction wheel in action can also produce some noises, this noise should be minimized while in production of the reaction wheels, also noise can be decreased by a method called "passive vibration isolation"[25] and finally the position of the payload with respect to the reaction wheels may affect the total noise disturbance on the payload.

A zero momentum stabilized system reserves spacecraft orientation in space, as a result of this only one side of the spacecraft looks to the Sun. With respect to thermal stability of the spacecraft: the sun side can absorb solar radiation (usually by solar panels) and should reflect if there is a body exposed to the Sun or the sun faced surface should be highly insulated. Also heat can be transferred to the cold side of the vehicle to make that region warmer. A simple design can be acceptable as following: Sun side unwanted heat can be exhausted by heat pipes and far-cold side can be heated by heaters.

3.1.5 Conclusion

A single spin stabilized system is a basic and old technology, however still finds its place in today's space missions. Because it offers simplicity and decreases the cost of the mission. Some missions specifically need spinning and spin stabilization becomes the best option. Dual spinning is an advanced version of single spin system and still in use for GEO communication satellites. A spin rate estimation is made at section 3.1.3 and at least 7 rpm spin rate is found. This rate is higher than the BOOST payload system requirement which is maximum 5 rpm. In an ordinary dual spin stabilization system antenna and a sensor is used in de-spun (platform) part. A new design can be developed that has the payload in platform, which rotates very slow (like 1 rpm) or stationary and other spin (rotor) body can spin very fast (like 60rpm) to create enough inertial momentum to preserve stability. However, there are two concerns, firstly this kind of development can eliminate the low cost advantage of spin stabilization and secondly rotating parts can be risky for a very sensitive experiment like the BOOST experiment.

The second option in the trade-off was 3-axis stabilization, which offers less risk and looks more feasible for the BOOST mission. As a baseline suggestion: Magnetic torquers shall be used for momentum dumping, and if possible even for attitude controlling of the spacecraft. To decrease possible reaction wheel noise, wheels should be used as less as possible and three magnetic torquers on each axis should be used with as high as possible efficiency for the momentum dumping.

3.2 Power System

3.2.1 Introduction

In this section after a short introduction, a power solar panel and a battery estimation are made with size, capacity and mass parameters.

A satellite power system consists of three units: power source unit, power storage unit and power distribution and conditioning unit (PDC). For the satellite orbiting Earth, the most common method to obtain energy is usage of solar panels where electricity is generated by the photovoltaic effect using sunlight. For deep space missions such as Galileo and Cassini, nuclear power can be used where sunlight is weak and power gain by panels is quite low. These type of missions have been using a system called Radioisotope Thermoelectric Generators (RTG) which is an electrical generator that uses an array of thermocouples to convert the heat released by the decay of a suitable radioactive material into electricity. Produced power by solar panels can be used directly or can be saved to a power storage unit which is basically consisting of space batteries. A PDC is responsible for a few points as given:

- Aiming at as low as possible power lost on cables by using suitable metals and making cables as short as possible.
- Avoiding of overloading and overheating.
- Good regulating and distribution of current and voltage for the satellite.

The BOOST mission currently has two mission scenarios where the first one is based on a 97° SSO orbit and the second one is based on a 57° inclination. The main (the first) scenario has advantages over the secondary scenario with respect to the power system. The maximum eclipse duration for the first scenario is 21.8 minutes and for the second one 35.5 minutes. The maximum eclipse duration is a key parameter for estimating the power system. It simply defines how long the satellite will not be able to produce power by solar panels and the power system needs to be fed by batteries. The period time, 96.1 minutes for the both scenarios, is also an important parameter by defining how long the solar panels can produce energy by subtracting the eclipse time from it. Thus, daylight period for the first scenario is 74.3 minutes and 60.6 minutes for the second scenario.

3.2.2 Power Budget Estimation

The power system estimation is made for two scenarios and depends on two different power requirements. SY-IN-0020 states a 95 W of power required for the payload. However this amount may go up to about 200 W according to current discussions about the payload design. Thus, there are two different estimations by using these power requirements. For the bus, the power requirement is estimated based on historical (small size) satellite power budgets [17], payload power requirement and the satellite estimated size. A 200 W power requirement for the bus is assumed as nominal and used in this section¹⁷. To sum up, the total power requirement is assumed to be 300 W and 400 W for the two scenarios, respectively.

Parameters for the BOOST mission are listed in table 16. In principle the bus voltage should be selected depending on the total power requirement and the satellite electrical parts. For a lower than 2kW power budget, 28 volts is the most common bus voltage of today's space missions [35] and for this the BOOST power estimation it is taken as 28 volts. The mission life time is taken as three years instead of two years to provide a longer time for the experiment in case of it is preferred to have more time.

Bus Voltage [volts]	Peak Load [W]	Time in eclipse [hours]	Time daylight [hours]	Mission lifetime [years]
28	Peak Load 1: 300 Peak Load 2: 400	Scenario 1: 0.36 Scenario 2: 0.59	Scenario 1: 1.23 Scenario 2: 1.01	3

Table 16: Power system parameters

Today, solar cells are produced mainly by three different materials: silicon, gallium-arsenide and multi-junction cells where different materials are used for each layer. Silicon is an old technology and used commonly at early space age, gallium-arsenide offers a higher power density and became common after silicon cells. Multi-junction is a technology that offers even more power density. In this power budget estimation, triple junction solar cells are used which offer a higher performance and become more popular today. As an example solar cell, "ZTJ Space Solar Cell" from SolAero Technologies is taken. This cell has a high power efficiency with 29.5 % and has a light cell design with a 84 mg/cm^2 mass to area rate. Moreover, it is fully space qualified and has already a flight heritage [33].

A solar cell slowly degrades in time due to some factors such as temperature of the environment and radiation. Also some factors decrease the cell's performance such as cell shadowing and design & assembly degradation. Radiation degradation on solar cells for BOOST is expected to be around 1% per year [35]. Shadowing of the cells is a negative factor for missions where the satellite is not sun pointed or sun orientation of the panels is very bad. The BOOST mission is a sun oriented mission where the orientation of the satellite is only limited by the antenna connectivity. Thus, the shadowing cell factor is a negligible issue. The degradation of solar cells due to temperature is about 2% [17]. To simply all degradations they can be merged as inherent degradation term, I_d , which is assumed to be around 0.8 [17].

¹⁷However, when the payload power requirement goes higher the bus power requirement may increase some due to power losses and additional electronics.

The first step of a solar array estimation is making an estimation of the total power requirement for the solar arrays, P_{sa} . And it can be done as follows:

$$P_{sa} = \frac{\frac{P_e \cdot T_e}{X_e} + \frac{P_d \cdot T_d}{X_d}}{T_d} \quad (3.2.1)$$

where P_e and P_d are the spacecraft power requirements during eclipse and daylight, respectively, and T_e and T_d are the length of these periods. P_e and P_d are assumed to be equal to the peak load (P_l) since there is not any differentiable mission requirement between day and night. X_e and X_d represent path efficiency from solar panels to batteries which is around 0.6 and 0.8 for a peak power tracking system [17].

Power efficiency of solar cells decay in time, this means at the end of the mission total power availability, P_{EOL} , is going to be lower than the initial power availability, beginning of life (P_{BOL}) which can be estimated as below:

$$P_{BOL} = P_0 \cdot I_d \cdot \cos(\theta) \quad (3.2.2)$$

where θ represents the sun incidence angle between the vector normal to the surface of the array and the sun line, I_d represents inherent degradation and P_0 represents power density of a cell. θ angle is taken as 23.44° , the worst case sun angle. P_0 is estimated by multiplying the solar cell efficiency with the solar illumination intensity, 1367 W/m^2 .

Typical degradation rates for solar cells are between 2-4% per year. Life degradation, L_d , occurs because of thermal cycling in and out of eclipses, micro meteoroid strikes, remnants from thrusters, and material out gassing along the duration of the mission. L_d of solar cells can be estimated as below:

$$L_d = (1 - \text{degradation/year})^{\text{mission-life-time}} \quad (3.2.3)$$

And power that has to be available at the end of the mission can be estimated as below:

$$P_{EOL} = P_{BOL} \cdot L_d \quad (3.2.4)$$

Finally, the total size of solar array(s) that is required for the mission can be estimated as below:

$$A_{sa} = \frac{P_{sa}}{P_{EOL}} \quad (3.2.5)$$

Power Budget [Watts]	Power of the Array [Watts]	Power Density of the Cells [W/m ²]	Power of the Cell: BOL [Watts]	Life Degradation	Power of the Cell: EOL [Watts]	Solar Array Size [m ²]	Solar Array Mass [kg]
300 W	S1: 521 S2: 670	403	295	0.941	277.6	S1: 1.87 S2: 2.41	S1: 1.57 S2: 2.02
400 W	S1: 695 S2: 889					S1: 2.5 S2: 3.2	S1: 2.1 S2: 2.68

Table 17: Solar array estimation results, where S1 corresponds to scenario one and S2 corresponds to scenario two. The total power of the arrays is obtained by the equation 3.2.1. The power density of the cell is estimated by multiplying the efficiency of the cell (29.5%) with the solar constant, 1367 Watts/m^2 . The power of the cell BOL is calculated by the equation 3.2.2.

As the total solar array size 1.87 m² for the scenario one and 2.41 m² for the scenario two are foreseen for a 300 W of power budget. And for a 400 W budget, they are estimated to be as 2.5 m² and 3.2 m², respectively.

The power storage unit which is suggested in this section is based on Lithium Ion batteries, which have a better performance compared to Nickel-Cadmium¹⁸ and Nickel-Metal Hybrid batteries with higher energy density rates. VES 16 model of space batteries from Saft Groupe S.A. are used for the power storage estimation. This model has lots of flight heritage for LEO space missions [34].

The number of battery cells, N , that are needed to satisfy the bus voltage, V_{bus} , can be calculated from equation 3.2.1.

$$N_{cell} = \frac{V_{bus}}{V_{cell}} \quad (3.2.6)$$

where V_{bus} is the bus voltage and V_{cell} is the average cell voltage. The equation expresses the number of battery cells needed in series, which equals to 7.7 cells. Rounding up the value to 8 cells, the average voltage, V_{avg} , becomes 28.8 V. Then the total charge capacity is also determined, in order to know the set of batteries needed in parallel to fulfill the current requirement. The total charge capacity is calculated from the following equation,

$$C_{bat}[Wh] = \frac{P_l \cdot T_e}{\eta \cdot DOD} \quad (3.2.7)$$

where P_l is the peak load, T_e is the eclipse time, η is the transmission efficiency of the cell which is assumed to be around 0.9 [17] and DOD is the depth of discharge. The battery capacity in Ah can be estimated by dividing C_{bat} Wh by the average voltage. And the total mass of the battery cells can be estimated as below:

$$M_{bat} = \frac{C_{bat}[Wh]}{Bat_{den}} \quad (3.2.8)$$

where Bat_{den} represents the specific energy density of the battery cell.

DOD rate simply describes how deeply the battery is discharged. By other terms, it means the portion of the battery capacity that can be in use. If the DOD rate is 100%, all capacity can be used, discharged, however if it is 50%, only half capacity of the battery can be used. DOD rate decays by time but mostly by usage, when satellite enters an eclipse battery starts to be discharged and needs to be recharged again. This charge and discharge process is defined as a cycle.

To estimate the DOD rate, number of charge/discharge cycle can be calculated as below:

$$\#cycle = (\#missionyears) \cdot \left(\frac{\#Eclipse}{days}\right) \cdot \left(\frac{\#Orbits}{day}\right) \quad (3.2.9)$$

By plugging mission life time as three years, eclipse occurred days as 138 and 342, the number of cycles are estimated to be 6624 and 16416 respectively. A LEO mission has more charge/recharge cycles than a GEO mission, this results in a lower DOD rate. DOD rate for the selected battery can

¹⁸not in use anymore.

be read roughly from figure 42 (Appendix) as 85% and as 50% for the end of lifetime (EOL) of the mission.

Power Budget [Watts]	Battery Energy Density [Wh/kg]	Battery Cell Voltage [volts]	Battery Cell Capacity [Ah]	Battery Cell Mass [kg]	Total Battery Capacity [Wh]	Total Battery Capacity [Ah]	Total Battery Mass [kg]
300W	155	3.6	4.5	0.155	S1: 169.3	S1: 5.88	S1: 1.09
					S2: 471.9	S2: 16.3	S2: 3.34
400W					S1: 225.8	S1: 7.83	S1: 1.45
					S2: 629.2	S2: 21.8	S2: 4.46

Table 18: Results of the battery estimation with a margin of 20%. The margin is included for the reasons such as "spare" batteries.

The battery capacity requirement for scenario one is much lower than the power budget due to having very short eclipse duration. Only 169.3 Wh is enough for a 300 W budget and 225.8 Wh is suitable for the 400 W budget. On the other hand, the battery capacity requirement is higher than the budget for scenario two with 471.9 Wh and 629.2 Wh, respectively for the budgets. The total mass of batteries is estimated to be 1.09 kg for the lowest capacity and 4.46 kg for the highest capacity, those weights can be accepted as a light weight with respect to the satellite mass.

As a conclusion for this section, for the most possible current option which is a 400 W budget with scenario one: a solar panel size of 2.5 m² and a battery mass of 1.45 kg are estimated to be suitable for the mission. A trade off should be made between usage of body mounted solar panels and deployable panels. A deployable panel with a degree of freedom by steering according to sun angle can offer a higher efficiency than a body mounted panel and it is more efficient than orienting a body mounted satellite to the sun. However, if the size of the satellite's one side is equal or more than the required solar panel size, it is more logical to use a body mounted panel.

4 Conclusions and Future Work

This thesis study showed that the mission can be accomplished by a solar synchronized dusk or dawn orbit with an inclination of 97° . The optimum altitude is estimated to be around 600 km. The satellite can decay during a time that is in agreement with the 25 years rule. Ground visibilities are found to be sufficient for both ground stations. Regarding the satellite downlink size requirement, contact durations are found to be enough to meet the requirement.

The feasibility of two attitude control methods are examined. Due to a high spin rate requirement spin stabilization method is found infeasible. Thus, 3-axis stabilization method is suggested. Depending on the estimated external disturbances, a reaction wheel model is suggested which is expected to be sufficient. ACS suggestion is finalized with an ACS system that has four reaction wheels and one magnetic torquers on each axis.

The total power requirement of the satellite is estimated to be about 400 Watts. And depending on this, the total size of the solar arrays is estimated to be about 2.5 m^2 . A total battery capacity of 225 Wh is found sufficient (which already includes the spare batteries).

To sum up, the BOOST mission is examined with respect to several aspects and it is found that the mission is a feasible mission. The expectations from the mission are also found as achievable.

In this thesis several issues are examined. However, there are still a few issues that need to be further examined. As a future work, a detailed thermal analysis of the payload and for the satellite are required. Firstly, the payload should be analyzed without the satellite (standalone) and secondly after selecting the satellite bus, a secondary thermal analysis should be made. Another issue that requires some attention are the specific orbital parameters. The final form of the orbit should be defined after selecting the satellite bus.

Additionally, launcher options for the mission should be investigated. The ground segment can be detailed further which may cover communication equipment requirements and a work distribution or definition for the ground facility staff. The data handling concept for the payload (or even for the satellite) can be investigated. And finally, a detailed cost analysis for the mission is required.

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5 Appendix

	25	26	27	28	29	30	31	32
		ATMOSPHERIC DENSITY			ΔV TO MAINTAIN ALTITUDE			
Alt (km)	Atm. Scale Ht. (km)	Minimum (kg/m ³)	Mean (kg/m ³)	Maximum (kg/m ³)	Solar Min 50 kg/ m ² (m/s)/yr	Solar Max 50 kg/ m ² (m/s)/yr	Solar Min 200 kg/ m ² (m/s)/yr	Solar Max 200 kg/ m ² (m/s)/yr
0	8.4	1.2	1.2	1.2	2.37×10^{13}	2.37×10^{13}	5.92×10^{12}	5.92×10^{12}
100	5.9	4.61×10^{-7}	4.79×10^{-7}	5.10×10^{-7}	8.95×10^6	9.90×10^6	2.24×10^6	2.47×10^6
150	25.5	1.65×10^{-9}	1.81×10^{-9}	2.04×10^{-9}	3.17×10^4	3.94×10^4	7.93×10^3	9.85×10^3
200	37.5	1.78×10^{-10}	2.53×10^{-10}	3.52×10^{-10}	3.40×10^3	6.72×10^3	8.51×10^2	1.68×10^3
250	44.8	3.35×10^{-11}	6.24×10^{-11}	1.06×10^{-10}	6.36×10^2	2.02×10^3	1.59×10^2	5.04×10^2
300	50.3	8.19×10^{-12}	1.95×10^{-11}	3.96×10^{-11}	1.54×10^2	7.47×10^2	3.86×10^1	1.87×10^2
350	54.8	2.34×10^{-12}	6.98×10^{-12}	1.66×10^{-11}	4.37×10^1	3.11×10^2	1.09×10^1	7.78×10^1
400	58.2	7.32×10^{-13}	2.72×10^{-12}	7.55×10^{-12}	1.36×10^1	1.40×10^2	3.40×10^0	3.50×10^1
450	61.3	2.47×10^{-13}	1.13×10^{-12}	3.61×10^{-12}	4.55×10^0	6.66×10^1	1.14×10^0	1.66×10^1
500	64.5	8.98×10^{-14}	4.89×10^{-13}	1.80×10^{-12}	1.64×10^0	3.29×10^1	4.11×10^{-1}	8.23×10^0
550	68.7	3.63×10^{-14}	2.21×10^{-13}	9.25×10^{-13}	6.59×10^{-1}	1.68×10^1	1.65×10^{-1}	4.20×10^0
600	74.8	1.68×10^{-14}	1.04×10^{-13}	4.89×10^{-13}	3.03×10^{-1}	8.81×10^0	7.58×10^{-2}	2.20×10^0
650	84.4	9.14×10^{-15}	5.15×10^{-14}	2.64×10^{-13}	1.64×10^{-1}	4.73×10^0	4.09×10^{-2}	1.18×10^0
700	99.3	5.74×10^{-15}	2.72×10^{-14}	1.47×10^{-13}	1.02×10^{-1}	2.61×10^0	2.55×10^{-2}	6.52×10^{-1}
750	121	3.99×10^{-15}	1.55×10^{-14}	8.37×10^{-14}	7.04×10^{-2}	1.48×10^0	1.76×10^{-2}	3.69×10^{-1}
800	151	2.96×10^{-15}	9.63×10^{-15}	4.39×10^{-14}	5.19×10^{-2}	8.63×10^{-1}	1.30×10^{-2}	2.16×10^{-1}
850	188	2.28×10^{-15}	6.47×10^{-15}	3.00×10^{-14}	3.97×10^{-2}	5.23×10^{-1}	9.94×10^{-3}	1.31×10^{-1}
900	226	1.80×10^{-15}	4.66×10^{-15}	1.91×10^{-14}	3.11×10^{-2}	3.30×10^{-1}	7.78×10^{-3}	8.25×10^{-2}
950	263	1.44×10^{-15}	3.54×10^{-15}	1.27×10^{-14}	2.48×10^{-2}	2.18×10^{-1}	6.19×10^{-3}	5.45×10^{-2}
1,000	296	1.17×10^{-15}	2.79×10^{-15}	8.84×10^{-15}	1.99×10^{-2}	1.51×10^{-1}	4.98×10^{-3}	3.77×10^{-2}
1,250	408	4.67×10^{-16}	1.11×10^{-15}	2.59×10^{-15}	7.69×10^{-3}	4.27×10^{-2}	1.92×10^{-3}	1.07×10^{-2}
1,500	516	2.30×10^{-16}	5.21×10^{-16}	1.22×10^{-15}	3.68×10^{-3}	1.95×10^{-2}	9.20×10^{-4}	4.88×10^{-3}
2,000	829	—	—	—	—	—	—	—
2,500	1,220	—	—	—	—	—	—	—
3,000	1,590	—	—	—	—	—	—	—
3,500	1,900	—	—	—	—	—	—	—
4,000	2,180	—	—	—	—	—	—	—
4,500	2,430	—	—	—	—	—	—	—
5,000	2,690	—	—	—	—	—	—	—
6,000	3,200	—	—	—	—	—	—	—
7,000	3,750	—	—	—	—	—	—	—

Figure 39: Atmospheric parameters for satellite[17]

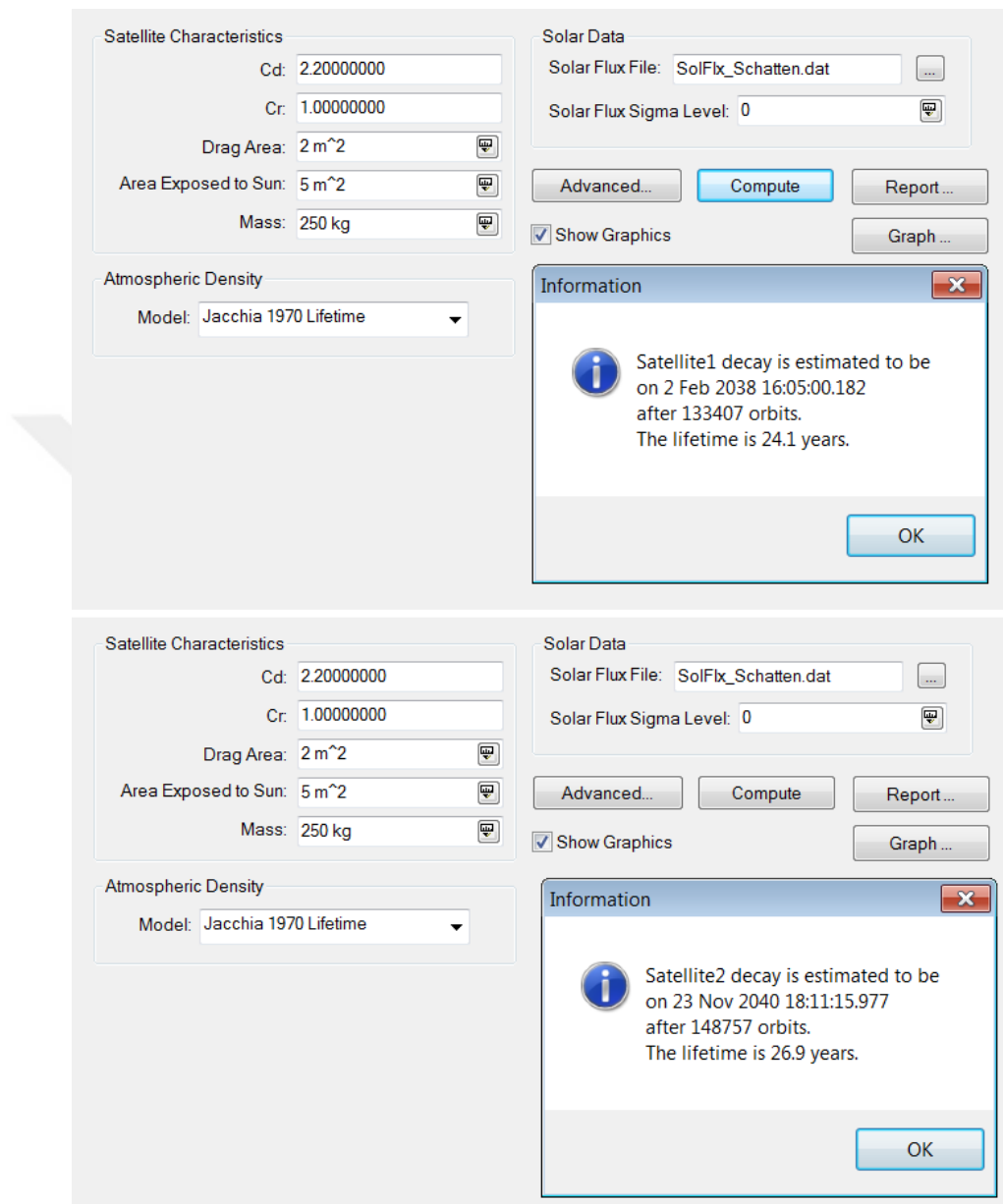


Figure 40: Satellite life time estimations for orbit option 1 and 2 by STK. Mission starting year is taken as Jan of 2014 instead of Jan of 2025 to have a more realistic result(Hint:11 years solar cycle is subtracted).

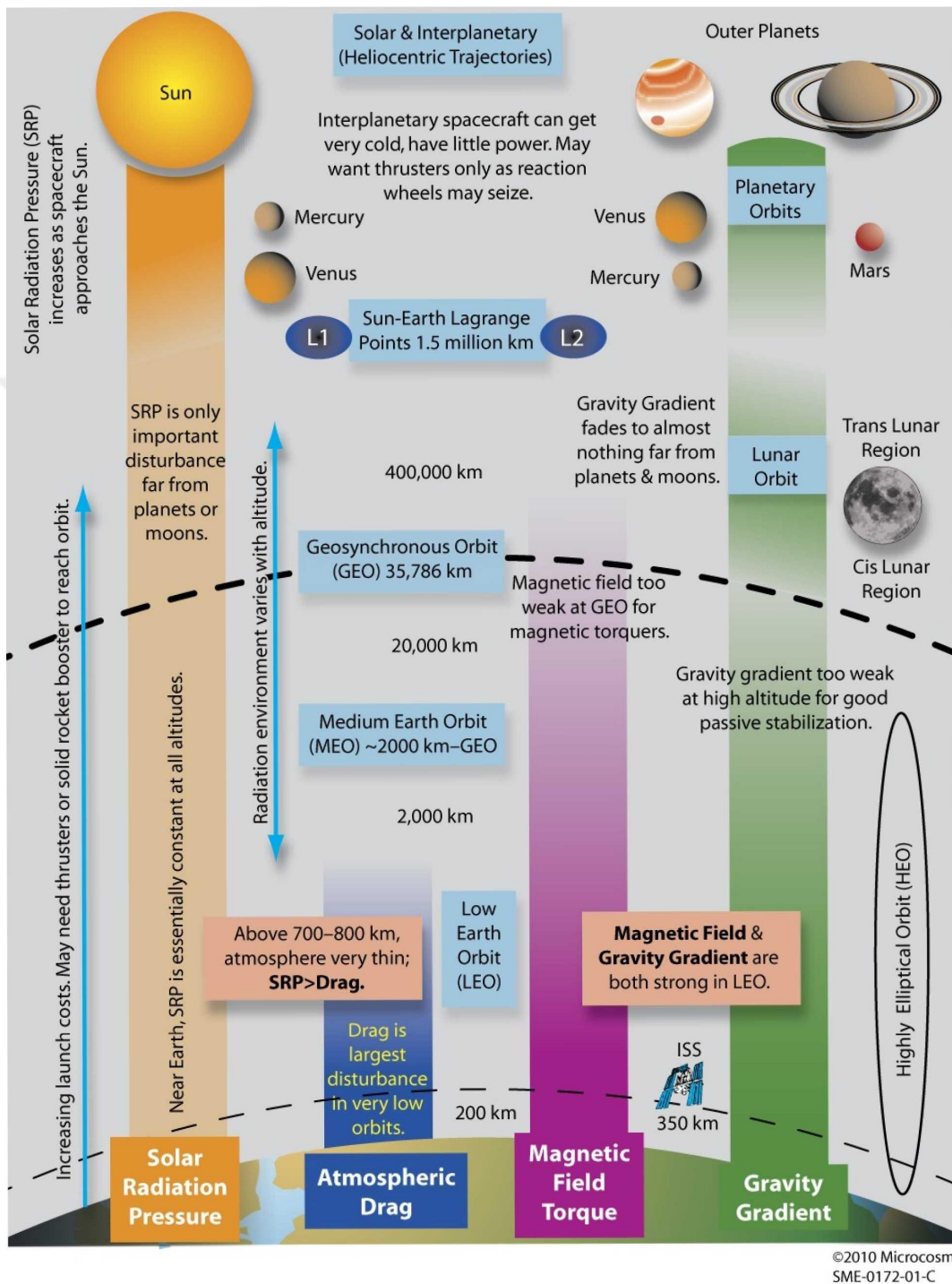


Figure 41: Overview of external disturbance in space [17]



	VES 140	VES 180	VL 48E	VL51ES	VES 16	VL 6P
Guaranteed capacity (Ah)	39	50	48	51	4.5	6.6
Mean voltage at C/1.5	3.6	3.6	3.6	3.6	3.6	3.6
End of charge voltage (V)	4.1	4.1	4.1	4.1	4.1	4.1
Energy (Wh)	140	180	170	180	16	22
Specific energy (Wh/kg)	126	165	150	170	155	65
Height (mm)	250	250	250	222	60	143
Diameter (mm)	53	53	54	54	33	38.2
Weight (kg)	1.13	1.11	1.13	1.08	0.155	0.34
Power capability current pulses A						225
Main application	GEO, MEO	GEO, MEO	GEO, LEO	GEO, MEO, LEO	LEO, GEO	Launcher

Figure 42: Space batteries from Saft Groupe S.A.[34]

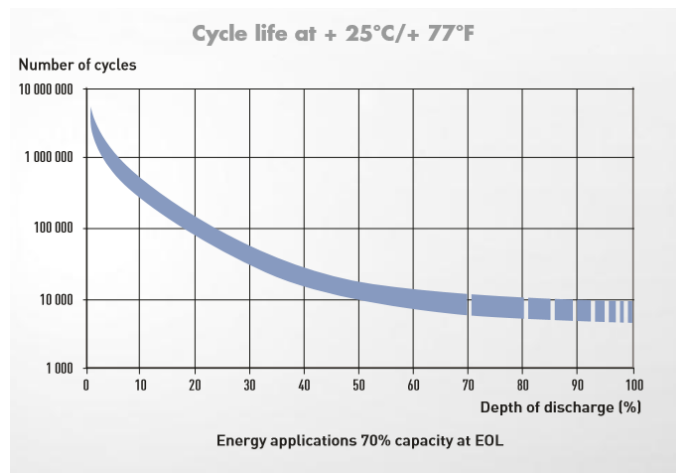


Figure 43: Number of cycles vs depth of charge[34]