Spacecraft preliminary design for ice monitoring in the poles: GlazeSAT - Master Thesis - Technical Univerisity of Madrid - Czech Technical University

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Abstract

The purpose of the thesis is to apply the preliminary design of spacecraft projects to an educational example of an Earth Observation satellite in LEO orbit which is going to monitor the evolution of the ice at the poles.

The first phase of the design involves the main requirements and the concept of the mission. The final phase of the project involves the subsystems sizing and the mission analysis design.

Aspects explored are the main components of the subsystems, trying to determine the values of the most important variables of the elements in order to comply with the requirements. These subsystems are the payload, structure, attitude orbit and control subsystem, propulsion, data handling, communication, electrical power and thermal control subsystems. Sizing have been performed with some previous data estimated by several missions, which these properties have changed during the design, when the components were determined. Some comprise solutions have been taken, and only a few iterations in the design have been made in order to obtain a solution close to the optimal one. Integration issues had a predominant priority in the sizing and concept configuration, which had been developed in more time as it was estimated. All of this considerations and design solutions will have future consequences to further design phases, and the design is still susceptible to have prospective changes.

Summary

Global warming is a fact that can not be neglected nowadays, which must be treated in order to assure the planet and humans survival. Every day, new effects of the climate change are being discovered, which will provided catastrophic consequences. All the effects should be recorded in order to elaborate a great strategy to solve the problem. However the development of the industry and the globalization has connected the planet a huge way, making every part co-dependant of many different parts. This co-dependence with different geographic parts of the planets makes tougher the implementation of a global strategy to reduce the main sources of contamination, due to the strategy agreement among every part. Nevertheless, small changes can be implemented in a global way, reducing part of the effects and polluting emissions to the atmosphere. This small reduction of contamination can be performed with the exhaust control and monitoring of different variables of the planet. In other words, in order to perform significant affordable changes for reducing the impact in the global warming, precise and active measurements of the effects of the global warming must be recorded for perceiving the little changes. There are a lot of variables to measure which can be related easily to the global warming. Sea level, temperature of the seas, temperature of the air, forestation measurements, plastic production, fossil fuel consumption, meat industry investment, relocation of the main fresh water reservoirs, air pollution measurements, flora and fauna diversity in different zones, etc...

Nevertheless, this project is going to focus on ice edges measurements at the Earth poles. There is a background of several space missions with similar purposes, for instance, CryoSAT 1 & 2, ICESat 1 & 2, RadarSat 1 & 2, CloudSat, KompSat-5 or TerraSAR-X. Annual variations can be related to the global warming due to the extreme variations during a year. This variations can be measured as peaks, which can provide information of different effects in different parts of the world. Also if the peaks are observed of being reduced, some conclusion could be extracted. Also, the global warming has a growing rate every year, which is estimated as an exponential if there is no active human change during the following decades. This growing rate can be related also with the ice-edge reduction every year of the ice edges at the poles, and also with the dates record of the same events. Furthermore, iceberg detachment can be measured, extracting different useful variables. For instance, typical size of the icebergs, distance travelled until melting, detachment frequency, location of the hot and cold zones, etc...

All of these variables are going to be measured with a payload, situated in space segment, payload selected is going to be a SAR (Synthetic Aperture Radar), which will provide images during day and night, no matter the climate conditions. Satellite is going to be designed (in phase B, preliminary design) for this mission, which is going to have to more sizing requirements, like 7 years of life time, and full map of the recorded zone every

10 days. These user requirements are going to impose different requirements for the payload and the rest of subsystems of the spacecraft. The main ruled applied for the sizing is to have the minimum total mass with the best performances, the cost is not a parameter taken into account for this sizing analysis.

Once the requirements are defined, orbit altitude can be selected by the hand of a trade off. An excessive altitude will result in a huge payload mass, providing a big satellite of medium size. On the other hand, a reduced orbit altitude will provide a minimum mass for the payload, but giving an enormous mass of required propellant in order to maintain the orbit during the 7 years of life time. Trade off analysis studies the optimal preliminary altitude of the orbit. This altitude has been calculated with several models; a pre-sizing of the payload, which the payload mass can be computed as a regression of similar missions, and also can be related to the spacecraft mass with another regression. The other model used is an analytic model for the atmosphere density calculation (as and exponential), which will be used for the altitude decay of the orbit. With the dry mass of the satellite estimated with the regressions, and a nominal altitude of the satellite, the decay can be analysed, providing the required propellant mass for the lifetime. Iterating in this process, an optimal altitude can be found, which has resulted as a value of 470km.

After these analysis, orbit can be selected, in order to comply with the observation zone requirements. This analysis has been performed in chapter 3, with a Sun-Synchronous orbit selection for obtaining the best performances. Initial RAAN of the orbit has been selected to has the minimum eclipses among the lifetime. During this chapter, main perturbations have been studied, and which of them will need corrections has been discussed. Also, a preliminary study of the re-entry, eclipses, angle with the Sun and coverage with ground stations is studied. These analysis will provided a lot of useful data for the subsystems sizing, for instance, the Solar panels sizing, communications antennas, propulsion subsystem sizing, etc...

At the same time the orbit was studied and selected, different segments were analysed, in order to study the compatibility with the orbit. For example, ground stations are selected in order to be able to receive the payload data, and telemetry housekeeping of the satellite. This stations are Kiruna(Sweden) and Troll (Antarctica). This stations will provide useful information for the communication subsystem sizing. Another segment studied is the launching segment. Before selecting any launcher, a particular calculation has been implemented in order to check the main performances of the launchers. Only four launchers were selected for this analysis, Soyuz, Falcon 9, Dnepr and PSLV-G. In the calculation, the first needed datum is the required velocity for the orbit, which is composed by the orbit velocity, the losses (calculated by Townsend and Schilling models) and the rotation velocity of the Earth. Then, with the propellant masses and the gross masses of the launchers, and with the rocket equation, maximum payload mass can me computed with a Newton Rapshon method. This method was quite accurate for Soyuz ST-A and Falcon 9 capabilities, because the researched data of the launcher characteristics was quite specific. However, this computation had no relation with Dnepr and PSLV-G user manuals, and the results were not accurate at all. After this analysis, Falcon 9 is selected as the main launcher, for performing the operation at Vandenberg space complex. Secondary launcher selected is Soyuz ST-A, which will perform at Baikonur.

User requirements are transmitted to the payload sizing as necessary resolution, Swath, look angle, and equivalent noise parameter. Usually, SAR has two performance modes (actually, up to five modes, but in this analysis only two are studied), Stripmap mode, which has a great resolution with a low Swath, and ScanSAR with has a wide Swath with worse resolution. First, frequency of the carrier signal is selected to comply with the required resolution with an acceptable antenna size, frequency is selected inside of X-band, due to bands Ku and Ka are reserved for military purposes. Band width and Azimuth length of the antenna are selected in order to comply with both resolutions (range and azimuth). Later, with all the efficiencies, geometry, backscattering coefficients, the rest of the characteristics of the SAR can be determined. With this analysis, frequency, look angle, dimensions, resolutions, gain, band width, average power, peak power and mass are given as a final specification. 4 different ScanSAR modes are calculated later with the previous data, with the aim to reach a resolution and an overlap between each subswaths.

Structure subsystem has been designed in order to withstand the most restrictive forces during the lifetime of the mission. Prismatic shape (squared base) of the satellite has been selected, according to similar missions and integration requirements of the different subsystems. Skin frame configuration has been selected in order to withstand the forces without two many stress concentrations given by the loads. These loads are giving during the launch, so the satellite must be able to withstand them with a constrain of having the frequency of the first vibrational modes higher than the vibrations frequency of the launch. With a simple configuration of stiffeners and coating, buckling, yield strength, and frequency analysis are analysed in order to obtain the minimum thickness of the stiffeners and coatings in longitudinal and lateral cases. This analysis has been performed as a function of the height and base-side length of the structure. Final dimensions of $1 \times 1 \times 1.3$m have been selected in order to reduce the dry mass of the satellite. Also, no bigger dimensions were required because in this analysis, only one payload was studied for this spacecraft. Aluminium alloys are utilized for the manufacturing; 6061-T6 for the coating and 7075-T6 is selected for the coating, with thicknesses of 1mm for the coating and 5mm for the stiffeners giving a total mass of 94kg for the structure. The final part of this section is the analysis of the integration of the different subsystems, the mechanisms analysis, and the final response to the sizing loads.

Attitude and Orbit Control subsystem starts analysing the pointing requirements of the payload in order to have a great image quality, assuring the continuity of the recording and allowing a maximum error in range direction. With the required swath and resolution, and the pulse description, maximum values of 0.2405 for yaw and pitch, and 0.106 for roll angle are allowed as maximum pointing errors. Also, pointing manoeuvres, detumbling and desaturation manoeuvres are sizing parameters for the AOCS. Three axis stabilization is required for this pointing error, due to the antennas must point to Nadir, and the Solar panel must point the Sun perpendicularly. With these specifications, Star trackers are selected to determine the attitude during nominal modes. Sun Sensors and magnetometers will determine the attitude during detumbling, orbit control, slew and safe modes and IMU will perform instead of Sun sensors during eclipse. After selecting the main sensors for the attitude and control subsystem, perturbations in attitude must be analysed. Gravity gradient, aerodynamic, magnetic and solar pressure torques are the main disturbances, which only the gravity gradient and aerodynamic have a secular nature for three axis stabilization. With the secular torques, reaction wheels can be selected in order to unload them in a reasonable time, around 46 days for the selected reaction wheels. Magnetorquers are not a suitable option in order to desaturate the wheels, because the manoeuvre would last around 14h, creating magnetic fields which will disturb the measurements. Thrusters are selected in order to perform these manoeuvres, with a mean separation of 0.75 (1 metre for 4 pairs of thrusters, and 0.6 metres for two pair of thrusters). A total number of 12 monopropellant thrusters are selected, requiring a total mass of propellant for desaturation performances of 6809 grams for the 7 years of lifetime. After the actuators selection, it can be checked that the reaction wheels and the thrusters comply with the pointing manoeuvres requirements. Last part of the analysis is the elaboration of the pointing budget with all the sensors and actuators. This error is computed with the procedure described during the AOCS section, complying with a margin of 500%.

For the propulsion subsystem, same process is pursued, main perturbations are analysed. Semi-major axis decay and inclination increasing are decided to be corrected, setting 3 months for the manoeuvre for the semi-major axis, and 1 year for the inclination, giving a total mass of 51 kg of Hydrazine required. Propellant budget is analysed, giving conservative values for the safety margin. Propellant breakdown consist, in the injection, maintenance, disposal, contingencies, and AOCS desaturation, giving a total mass of 69.5 kg or Hydrazine. Non extra thrusters are placed in the satellite, propulsion manoeuvres are going to be performed by two pair of thrusters placed in one of the lateral faces of the satellite. Duration time of the performance will consist of 11.25 minutes, 5 minutes and 40 seconds in each node of the orbit (ascending and descending). For the orbit determination, two GPS receivers will be placed inside the satellite, with a total of 4 antennas connected to both receivers.

Data handling subsystem will have a particular architecture. Two different on board computers are going to be placed, due to the data requirements contrast between the payload and the subsystems. A high performance computer and memory will be required for the payload due to data storage requirement per orbit, with a LEON 4 processor, and a memory flash of 320 Gbytes. However, for the central computer, only a LEON 3 will be required, with a RAM memory of 4kb and a Flash memory of 102Mb will be required. Central computer will be the one which will give the orders to the subsystems, payload, switchable elements and the communications subsystem.

For communications subsystem, the requirements are divided for the payload and the subsystems categories. For the payload, it is necessary to communicate with a bit rate of 1.25 Gbps; however, for the housekeeping telemetry only a bit rate of 510 kbps is needed. These constrains will result in two different antenna types for the communication. X-band antenna, parabolic will be selected and calculated for the payload data communication. Parabolic antenna will have a diameter of 0.12m, with a frequency of 12GHz, a Gain of 21.35 dB and a power of 8.39 W. Nevertheless, for the housekeeping telemetry, it will be transmitted in S-Band, with an isotropic antenna (G=1), which will require a frequency of 4GHz for the carrier signal and a power of 5.55W. In order to obtain this isotropic antenna without wasting to much power, a patch antenna of 3 different parts will be considered for the sizing, with dimensions of $30 \times 20 \times 5$mm each antenna. Placing the 3 antennas in perpendicular way, almost isotropic radiation diagram can be achieved.

Focusing in electrical power subsystem, demanded power profiles must be studied, in order to size the different components. Mean power during the orbit is estimated of 196.42W (225W during eclipse). Also, the system must be able to provide peaks of 1417W. Due to the life time of the mission and electrical power required for the mission performance, solar array as the main power source is selected. Also, batteries are chosen for the electrical supply during the eclipses times. First, the bus voltage is selected as a function of the power consumption in the satellite. Later, cells (with serial connection) of the batteries can be determined, in this cases, 8 cells are required. Finally the capacity of the batteries can be designated dividing the energy required for the discharge by the discharge voltage and the depth of discharge. Two batteries of 8 serial cells and 10 strings with a capacity of 15Ah each are selected for the mission. Also, during the peaks (pulse transmission by the SAR) batteries provides a current of 2.91 times the nominal one; manufacturer allows this peak current during microseconds. Also the efficiency, and the dissipated power can be calculated. For the panel solar sizing, the process involves the energy supply to the loads, and the required energy to charge the batteries during the illumination time. Counting the dissipations, losses, and the MPPT and DET controls, solar panel should be able to provide 473.8 W in the worst case (EOL, hot case, summer, maximum temperature). With all this input data, it is necessary to have 21 serial cells for the solar panel with 31 strings, giving a total area of 2.36 m^2. Solar panel is able to provide a maximum power of 969W in the best conditions. For this calculation, a mechanism is supposed, due to the deploy orientable panel configuration which is able to turn around one longitudinal axis. Panel is able to set perpendicular almost all the time of the orbit, maximum misalignment error is about 17.5. MPPT and DET controls are implemented in order to have redundancy. Also two charge/discharge regulators are placed inside of the vehicle in order to control the batteries safely.

Last subsystem analysed is the thermal control subsystem, which is going to be able to set the different devices in the operational range of temperatures. First study of the problem has been made by the hand of the energy balance equation, in which several terms have been described. The most important terms are: dissipated power, Solar incidence, Albedo incidence, Planet radiation incidence, intern conductive terms, intern radiative terms and the outer space radiation. Before computing any term, components description and selection has to be made. Most of the components of the satellite have a passive nature, which does not consume any electrical power during their performance. These components are the surface coatings, MLIs and the radiators. The outside of the satellite is painted in white (SAR antenna, Communications antennas, back side of the Solar panel), the interior part of the vehicle is painted in black, in order to homogenize the temperatures. Then, the exterior part of the vehicle is covered by MLIs, in order to insulate the interior part of the outer space and the Sun incidence, and finally two radiators are located in the lateral faces of the structure, in order to emit the dissipated power to the outer space as infrared radiative energy. Radiators are needed for the hot case; however active components, in this case heaters are required for the cold cases (at the end of eclipses), which will provide an extra thermal power to the system in order to heat the critical components (batteries). With all this data, the minimum radiator surface needed is calculated and also the minimum power required for the heaters. Two radiators of 0.45 m^2 each have been placed, and two heaters with a maximum power consumption of 20W each have been placed inside of the vehicle. Verification of the temperature ranges of the components along the orbit should be checked, the software proposed was ESATAN-TMS, however due to problem with licenses and installation the last week of the development, it was not possible to fulfil the verification in this analysis. This verification will be a part of the future work of the project.

Finally, last part studied of the project is the brief cost analysis. This estimation have been done with a parametric model extracted of [?], which relates the mass of each subsystem to the price. Two different costs are analysed, the Research, Development, Testing and Evaluation cost, RDT&E, and the cost of the Theoretical First Unit, TFU. For every cost, an inflation factor is applied to correlated the results to the fiscal year 2018. For RDT&E costs, inheritance factors are applied to the different subsystems, because already developed technology (commercial) is used for this missions, reducing the cost of each subsystem. For production costs, the learning curve is applied to the cost of TFU, however, for this mission only one satellite is required, so the production costs are equal to the TFU costs. Finally the cost of the segment space is about 143.9M$, with an uncertainty around 30M$ due to preliminary phase of the analysis. Launching costs are estimated around 21.5 and 36.7M$, assuming dual launch in both launchers, sharing the fairing with another mission which will require the same orbit.

Degree of maturity is going to be explained in the following paragraph. This analysis was a preliminary design of the space segment of the mission, for example launching and ground segment were analysed in a briefly way, in order to be able to obtain some data for the rest of subsystems. Orbit configuration has the most maturity level of the project, however issues related to eclipses and coverage should be analysed deeply in further analysis. Configuration is not frozen yet, because the thermal verification should be done, and also a deeper analysis in integration and compatibility analysis should be done in order to assure the correct performance. SAR development still counts with a relevant uncertainty level, which could be subject to change during future phases of the mission. Also, Management Plan should be done in order to organize the rest phases of the work, collaboration among several departments and a deep calendar should be programmed in order to comply with the different dates of the program.

There are also some required changes and future work, orbit characteristics and launcher selection should be analysed together in future analysis. Satellite final mass and volume have resulted smaller than was expected, so it would be possible to launch the mission with an smaller launcher in order to reduce the costs. If the launcher is not changed, dual launch should be studied in order to share the main costs with another mission, but however, this option it is susceptible to have changes in the nominal insertion orbit, which will require a further study. Also, SAR should be analysed in a deeper way, in order to reduce the ambiguities of some variables. Moreover, ScanSAR was not analysed as deep as Stripmap mode. Additionally, performance should be expanded to more operation modes as TopSAR. And even more important, different components of the SAR must be detailed as the rest of the subsystems.

Structure is also susceptible to future changes due to the integration issues. Also, individual study of the mechanisms should be performed, in order to reduce the criticality of the mission. NASTRAN should be use in order to check the stress concentration and the vibrational modes of structure during the launching. For AOCS and Propulsion subssytems, perturbations should be studied extensively, placing in time and with a control algorithm the different manoeuvres. Mathematical systems and communications protocols among the sensors and computers should be established and detailed. CHD requires a deep study of the connections and interfaces by the hand of the software development. For communications subsystems has the same future work as the CHD apart from the manufacture of the antennas and the proper testing and evaluation. EPS also presents and initial design, which will require an analysis of the power consumption profiles of the different loads. Also electronics configurations and mechanisms should be described more detailed in further analysis. Finally, for the thermal control subsystem, verification should be made with a proper software, and after testing should be done before the launching in clean rooms in order to test the hot and cold cases of the electronics.

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