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Conceptual Project of the Classic FireSat Mission for the Monitoring of Forest Fires in the Northeast of Brazil using the Space Mission Analysis and Design (SMAD) process

Erica Generoso Lima

Instituto Nacional de Pesquisas Espaciais - INPE, São José dos Campos/SP - Brazil
erica.lima@inpe.br

Mariana Conti Tarifa

Rodrigo Dantas Dias

Artur Robson Cutolo

Pedro Kukulka

Willer Gomes dos Santos

Instituto Tecnológico de Aeronáutica - ITA, São José dos Campos/SP - Brazil
mariana.tarifa@ga.ita.br; rodrigo.dias@ga.ita.br; pedrokuk@ita.br; willer@ita.br

Abstract. Catastrophic forest fires have been increasingly frequent. When the fire is detected early, it is easier to make fire control decisions and, in this way, the use of available resources is optimized and alerts the public in general. Given the importance of using remote sensing technology by satellites for the monitoring and preservation of natural resources, the current project aimed at the conceptual project of the Classic FireSat mission applied to the monitoring of forest fires in Northeast Brazil. The Space Mission Analysis and Design (SMAD) process was used to define the requirements and restrictions of the mission. Studies of the location and the ways of fire identification were carried out. In addition, the final orbit design was conceived using the STK software. A preliminary dimensioning of the propulsion, determination of attitude control, management of the onboard equipment, communications and energy subsystems was performed. There was a need to iterate the design process several times, as the phases of the project progressed. It was demonstrated that the proposed project is adequate and feasible for the purposes of a preliminary mission design, since the results are consistent with the literature and real applications, including components available on the market.

Keywords: conceptual project, space mission, analysis and design, orbit design, subsystems dimensioning

1. INTRODUCTION

In a global context, the occurrence of forest fires happens due to social, economic and natural factors. Catastrophic fires have been more and more frequent and recent disasters have attracted the attention of the media and strengthened the perception of them as bad events (Robinne *et al.*, 2018). In Brazil, the number of fires resulting from human action has increased considerably in recent years for several reasons, including the preparation of plantations, deforestation, clearing of pastures, land disputes and social protests (de Medeiros *et al.*, 2017).

The use of remote sensing technology by satellites is of great importance for the monitoring and preservation of natural resources, especially in estimating risks, such as the detection of forest fires in several phases, from previous conditions to post-fire impacts (Lasaponara *et al.*, 2016). When the fire is detected early, it is easier to make fire control decisions and, in this way, the use of available resources is optimized and alert to the public. Therefore, it is essential that these are properly monitored by forest fire managers and others stakeholders (Hua and Shao, 2017; Thomas *et al.*, 1995).

Among the advantages of using data from satellites, it can be mentioned the simultaneous acquisition of data in different bands of the electromagnetic spectrum, periodicity of data acquisition over the same region and wide-band imaging (Krug, 2009).

In view of this perspective, the Classic FireSat Space Mission was proposed, which aims to detect, identify, monitor and report fires in Brazilian territory in almost real-time and at low cost. The present project consisted of carrying out the conceptual design of the mission, making use of the Space Mission Analysis and Design (SMAD) process proposed by Wertz *et al.* (2011). First, requirements and restrictions were defined, followed by studies on the location and ways of identifying the fires. Then, the final orbit design was conceived using the STK software (AGI, 2020). Finally, the preliminary dimensioning of the subsystems was performed.

2. SPACE MISSION ANALYSIS AND DESIGN PROCESS

As cited by (Wertz *et al.*, 2011), the main goal of Space Mission Engineering is to establish mission parameters in order to meet the objectives of a space mission, taking into account the combined optimization of schedule, costs and risks. These three aspects are particularly important, as the first space missions, despite being successful in an engineering point of view, were too cost and time demanding. The New Space Mission Analysis and Design (SMAD) proposed by (Wertz *et al.*, 2011) differs from previous processes as it considers two aspects: requirements based on a timeline and the stakeholders, who are the interested groups in the project. After defining the conceptual phase in the SMAD process, the next steps are to design, build, launch, and operate the satellite, as well as to perform its de-orbiting.

The SMAD can be executed following the mission statement previously defined. There are four main process in the SMAD process: the definition of objectives and restrictions, the definition of alternative mission concepts, the evaluation of alternative mission concepts and the allocation of system functional and operational requirements (Wertz *et al.*, 2011). Following those steps, the Classic FireSat mission was conceived.

The main objectives of this mission are to identify, detect, monitor and report forest fires in a region of the Brazilian national territory. This process should take place in almost real-time and have a low development and operation cost. Among the secondary objectives, we can mention the acquisition of statistical data on the outbreak and growth of forest fires in Brazil, showing the public that there are ongoing actions to fight forest fires and obtaining as much data as possible to help in forest management throughout the Brazilian national territory.

To define the project's stakeholders, four distinct groups were classified in relation to their goals and objectives. The main customer, that is, who buys the space system, is the Brazilian Forest Service. The Government was chosen as the secondary customer, since this is the group that would finance the project. The operator, defined as the Mission's Operations Center, consists of the group that works on the daily operation of the satellite. The end-user, that is, the people who use the data provided by the satellite, was defined as the firefighters, individual rangers and forest fire management centers.

The life cycle schedule of the Classic FireSat mission followed as steps in a space project prepared by NASA (Wertz *et al.*, 2011). The life cycle estimate for the present project is presented in Table 1.

Table 1. Schedule adopted for the Classic FireSat mission.

Phase	Duration
Exploration of the Concept	6 months
Detailed Development	
Risk Reduction/Technology Development	8 months
Detailed Design and Development	12 months
Production	12 months
Launch	1 month
On-Orbit Check-out/Transfer to operational orbit	10 days
Operation and Support	
Operation	5 years
Disposal	0 to 5 years

As part of the SMAD process, it is necessary to establish minimum requirements and certain restrictions on the project. Such definitions are compiled in Table 2.

Table 2. Functional requirements for the Classic Firesat Mission.

Requirements and Restrictions	
Requirements	Mission Firesat Brazil
Performance	Identification of fire outbreaks through clear clouds, with up to 200 meters of resolution and accuracy of geolocation of 1km.
Coverage	Coverage of specific forests in the Brazilian territory at least five times a day.
Interpretation	Identify a fire outbreak within 5 hours, with less than 5 % false positives.
Response time	Data must be interpreted for the end-user within 30 minutes.
Mission lifetime	5 years
Schedule	Operational in up to 3 years.
Legislation and regulation	Obey orbital debris rule and civil program regulations.
Political	Meet public demand for action.
Development	Assembly constraints and mission management only on national soil.

3. IDENTIFICATION OF FOREST FIRES

As a first identification step, it was necessary to define the main area of the Brazilian territory to be monitored. As part of the INPE Burning and Forest Fire Monitoring Program (INPE, 2019), there is a detailed study of Fire Risk. This study takes into account aspects such as precipitation, effects of the type of vegetation and the natural cycle of its defoliation, the maximum temperature and minimum relative humidity of the daily air. In the data taken from the 2019 Yearbook, according to INPE (2019), it is possible to observe the fire risk in Brazil classified into 5 categories ranging from a minimum risk scale (green = 0) to critical (brown = 1). Figure 1 shows graphically the fire risk obtained in each of the four seasons of 2019.

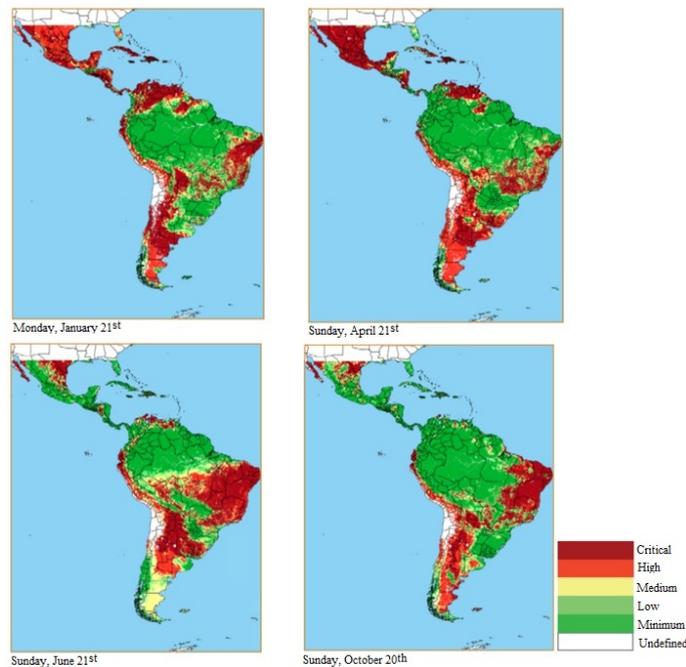


Figure 1. Fire risk in Brazilian territory throughout 2019. Source: Burning Program: Fire Risk, (INPE, 2019).

Based on the observed data, it was decided to monitor the Northeast region of Brazil, where critical risks remain constant in part of the territory throughout the year. In addition, it is a region made up of several biomes such as the *caatinga* (predominantly), *cerrado*, Atlantic forest and the Amazon forest. Therefore, it is an important region to be monitored.

To define the form of identification of the fires, several data from real space missions for monitoring forest fires were compiled, with some examples being as follows: Amazônia-I (Rudorff *et al.*, 2003), Uniform-1 (Yamaura *et al.*, 2014), Terra and Aqua satellites from NASA. The extensive use of infrared sensors was observed, which is an application used in the practice of fire fighting for many years (San-Miguel-Ayanz and Ravail, 2005). Fire detection takes place based on the medium infrared (3–5 μm) and thermal infrared (8–14 μm) channels of remote sensing images (Hua and Shao, 2017).

Global fire monitoring over the past thirty years has been based on sensors such as the AVHRR (Advanced Very High-Resolution Radiometer), present on the NOAA (National Oceanic and Atmospheric Administration) satellite, ATSR (Track Scanning Radiometer) present on ERS satellites (European Remote Sensing), MODIS (Moderate-resolution imaging spectroradiometer) onboard Earth and Aqua terrestrial observation satellites, the OLI sensor (Operational Land Imager) present at Landsat-8, among others (Hua and Shao, 2017).

The MODIS sensor is characterized by its wide spatial and spectral coverage, in addition to better continuity in acquisitions, in thinner spectral regions, when compared to other existing sensor systems (Latorre *et al.*, 2003). This sensor was designed to monitor the atmosphere, ocean, and land on a global scale daily, with spectral and spatial resolution bands specified to meet these purposes. However, sensors like MODIS are very robust sensors and do not fit the scope of the Classic FireSat mission.

Thus, it was proposed to use a compact infrared camera, the CIRC Compact Infrared Camera, developed by the Japanese Space Agency (JAXA) to detect fires. The CIRC is a light, compact and low energy consumption infrared sensor, whose specifications meet the requirements of the Classic FireSat mission. The main characteristics of CIRC are shown in Table 3.

Table 3. Main characteristics of the CIRC, obtained from (Katayama *et al.*, 2009).

Compact Infrared Camera (CIRC)	
Item	Value
Dimension, Mass	150 mm x 100 mm x 200 mm, 3 kg
Spectral Range, Pixels	8-15, 640 x 480 μm
Resolution, Field of vision	< 200 m, 128 x 96 km

4. ORBIT DEFINITION

This section aims to present the definition of orbit from the fundamental parameters of the orbital elements, such as the choice of altitude and inclination to meet the necessary restrictions of the project presented.

4.1 Project Assumptions

In order to define the orbit for the Classic FireSat mission in the Northeast Region of Brazil, the following premises were adopted: the orbit must be circular, the altitude of the orbit must be greater than 300 km and less than 1000 km. The drag coefficient is 3.13 (Santos and Albuquerque, 2020). The ballistic coefficient is 108.9 kg/m^2 .

4.2 Orbit Design

Given the assumptions described in the last section, in order to obtain more accurate values, an orbit simulation was performed using the Systems Tool Kit (STK) software (AGI, 2020) to obtain a better visualization of the passage of the satellite over the Northeast region of Brazil.

Since the payload operates with an optical sensor, luminosity is an important issue to be covered, it was chosen a sun-synchronous orbit. Thus, the choice of the satellite's altitude at 850 km was made, with an inclination of 98.82° and with the Local Time of Ascending Node of 01:14 HMS, shown in Figure 2. This choice was due to the fact that the satellite has a quantity of 6 accesses in the delimited region, given that the use of these characteristics promotes a better cost/benefit ratio for this mission.



Figure 2. Orbit defined for the Classic FireSat mission to monitor the Northeast region.

Thus, after defining the orbital parameters, the CIRC sensor chosen for the mission was inserted, operating with a vertical angle of 12° and 9° for the horizontal angle in order to represent the spectral coverage (Katayama *et al.*, 2009).

5. PROJECT OF THE SUBSYSTEMS

5.1 Propulsion Subsystem

The choice of the propulsive system is determined by the mission objectives. There are three categories that differentiate space propulsion technologies: escape propulsion (from Earth's surface to orbit), in space propulsion (in orbit) and deep space propulsion (from orbit to orbit) (Salgado *et al.*, 2018). Each category requires different levels of thrust and specific thrust. For the case of the small space vehicle propulsion subsystem, the main types used and their characteristics can be seen in Table 4.

Monopropellant systems, which have hydrazine as their main propellant, constitute the most widespread type of propulsive system currently. The configuration of this system is not very complex, since the hydrazine is decomposed when it finds the catalytic bed and releases energy that was absorbed by the decomposition of the products (London, 1996).

Table 4. Main types of propulsive systems for small spacecraft and its main propellant features. Adapted from. (NASA, 2020).

Propulsion Type	Thrust	Specific Impulse(s)
Monopropellant (Hydrazine)	0.5 - 30.7 N	200 - 235
Cold gas	10 mN - 10 N	40 - 70
Alternative Propulsion	0.1 - 27 N	190 - 250
Electric Propulsion	10 - 20 μ m	500 - 5000

Initially, three parameters were taken into account: the satellite dry mass, which does not take into account the amount of propellant to be used, the *Total ΔV* , which includes the main functions of a propulsive system should provide for near-Earth missions (orbital transfer, attitude control, and orbit maintenance) and *Larger ΔV* which is mainly used for thruster design and selection. The satellite Uniform I, launched in 2014 for detecting fires in the Southeast of Asia, has a total mass of 50 kg (Fukuhara *et al.* (2017)). It is also estimated that 65% of a satellite initial mass is composed only by its propellant (Wertz *et al.* (2011)). Taking these into account, it was adopted 20 kg as initial dry mass. The Total ΔV considered that the satellite had an initial altitude of 400 km and final of 850 km, with 300 revolutions per year and 50 km for de-orbit. The largest ΔV was considered for small satellites applications.

Thus, the parameters considered for the propulsive system dimensioning are summarized as follows:

- Dry mass: 20 kg, Total ΔV : 360 m/s, Largest ΔV : 5 m/s;

After performing a detailed analysis to determine the best propulsive system to meet the project's requirements, aiming at lower energy consumption and less mass, the single propellant propulsive system was chosen, opting for the choice of the propellant AF-M315E, as it has good performance and occupies a smaller volume than the others, in addition to being considered a green fuel (NASA, 2020). Figure 3 shows the final result of the analysis, with the comparison between the three propulsion systems studied.

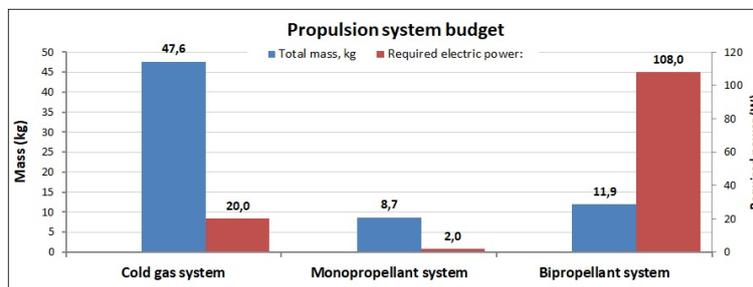


Figure 3. Ratio of mass and power required of the chosen single propellant system.

Thus, the tank offered by the company *Rafael Space Systems* (Rafael, 2014) has a spherical shape with a length of 0.23 meters, a volumetric capacity of 6 liters and a mass of 1.3 kg, and the propeller selected from the company *Micro Aerospace Solution* (Solutions, 2015) has a thrust capacity of 0.05 N, meeting the proposed project requirements.

5.2 Attitude Determination and Control Subsystem

Attitude control is the process of orienting an object in space in a predetermined direction. The attitude control determination subsystem is critical for on-orbit operations as its degradation and failure can occasionally be fatal to the mission (Wayer *et al.*, 2013). The development of this system has great importance, as it is used to dimension the actuators, consisting of reaction wheels and magnetic coils. The final dimensions of the satellite and the solar panels were used to obtain the inertial moments. Thus, with the parameters obtained in the simulation performed in the software STK, the initial values for the actuator selection were defined, shown in Table 5.

Table 5. ADCS model input data.

Symbol	Description	Value	Unit
-	Satellite Dimension	63x63x63	cm
-	Solar panel dimension	80x63x2.5	cm
W	Final weight	29.5	kg
R	Orbit radius	7221000	m
P	Orbital period	6029.44	s
$P_{sunlight}$	Day period	3901.99	s
λ_{max}	Maximum latitude of satellite orbit	90	°
V	Satellite Orbital Speed	7524.89	m/s
I_x	Moment of inertia about the X axis	3.144316	$kg.m^{-2}$
I_y	Moment of inertia about the Y axis	2.249916	$kg.m^{-2}$
I_z	Moment of inertia about the Z axis	3.442158	$kg.m^{-2}$
I_{xy}, I_{xz}, I_{yz}	Product of Inertia XY, XZ, YZ	0	$kg.m^{-2}$
A_s, A_r	Illuminated surface, Ram Area	2, 0.5	m^{-2}
CPS-CG	Distance between CPS and CG	0.001	-
q	Reflectance factor	0.6	-
D	Residual satellite dipole moment	1	$A.m^{-2}$
C_d	Drag coefficient	3.13	-

Considering the values of 5° for the angles of pitch and roll, it was possible to obtain a summary of the torque results of the main environmental disturbances, with a total of 2.70×10^{-5} Nm, as shown in Table 6.

Table 6. Disturbing torques suffered by the satellite.

Disturbing torques suffered by the satellite	
Torque Type	Value (Nm)
Solar radiation pressure	$0.0015 \cdot 10^{-5}$
Atmospheric drag	$0.0004 \cdot 10^{-5}$
Magnetic Field	$2.6658 \cdot 10^{-5}$
Gravity gradient	$0.0329 \cdot 10^{-5}$

In order to determine the dimensioning of the actuators, a satellite rotation speed of 2 km/s caused by the deployment phase was adopted. To cancel this unexpected rotation it was considered that the reaction wheel was responsible for the reduction of 0.9 km/s in the 3 axes, while the magnetic coil reduced about 1.1 km/s , which allows the satellite to be fully braked when reaching the desired orbit. Moreover, all torques considered during the different phases of the mission are presented in Table 7. In addition, a 20% safety margin was applied. Table 8 shows the minimum capacity that the reaction wheel and magnetic coil must exert to meet the design requirements of the ADCS subsystem, in which the minimum torque of the reaction wheel is $3.49 \cdot 10^{-5}$ Nm, the angular momentum storage is $0.052 \cdot 10^{-5}$ Nms and the magnetic dipole of the magnetic coil is at least 1.54 Am^2 .

Table 7. Maximum torques for the parts of the system.

Maximum torques for each part of the system	
Item	Value
Reaction Wheel Torque for Disturbance Rejection	$3.22 \cdot 10^{-5}$ Nm
Orbit acquisition torque slew for reaction wheels	$0.13 \cdot 10^{-5}$ Nm
Storage of the acquisition moment of the orbit slew by the reaction wheel	$51.37 \cdot 10^{-3}$ Nms
Slew maneuvering torque for reaction wheels	$0.24 \cdot 10^{-5}$ Nm
Storage of <i>slew</i> maneuver moment by reaction wheel	$3.23 \cdot 10^{-3}$ Nms
Gravity Gradient Moment Storage in the Reaction Wheel	$0.696 \cdot 10^{-3}$ Nms
Solar radiation moment storage in the reaction wheel	$0.075 \cdot 10^{-3}$ Nms
Atmospheric drag moment storage in the reaction wheel	$0.033 \cdot 10^{-3}$ Nms
Magnetic torques for disturbance rejection	1.54 Am^2
Magnetic torques for initial moment discharge	0.72 Am^2

Table 8. Actuators Project Summary.

Final Actuator Design	
Torque Type	Value
Torque from Reaction Wheel	$3.49 \cdot 10^{-5}$ Nm
Momentum Storage by Reaction Wheel	$0.052 \cdot 10^{-5}$ Nms
Magnetic Torquer Dipole	1.54 Am^2

According to these values, it was selected the commercial actuators shown in Table 9.

Table 9. List of selected actuators.

Selected Actuators	
Item	Model
Reaction Wheels and momentum	Rockwell RSI01-5/28i
Magnetic Coil	Sputnix SW-MT-20

The choice of sensors followed the same methodology as the choice of actuators. Table 10 shows the list of chosen sensors that meet the design requirements. Two star sensors model Sinclair Inter ST-16 (Interplanetary, 2014), six solar sensors model SSBV-FSS (SSBV, 2014), a magnetometer model BAD-TFM65-VQS (Defense, 2014), a GPS with redundancy were selected. of the antenna model SGR-07 (Surrey, 2014) and three gyroscopes model SDI-QGES116 (Inertial, 2015).

Table 10. List of selected sensors.

Selected sensors	
Item	Model
Star Sensor	Sinclair Inter ST-16
Magnetic Coil	SSBV - FSS
Magnetometer	BAD-TFM65-VQS
GPS with Antenna	SGR-07
Gyroscope	SDI-QGES116

5.3 Onboard Management Subsystem

The onboard management subsystem has several functions, including internal data communication, management of satellite operation modes, fault detection and diagnosis, reception, decoding, processing and distribution of remote controls for subsystems and payload, the acquisition, formatting, storage and transmission of telemetry of subsystems and payload, on-board data processing, and data storage control (Wertz *et al.*, 2011).

For the dimensioning of the on-board system, it is necessary to define the processing requirements. All initial operations and launches are handled by the launch vehicle provider. It is also important to note that payload processing needs to be separate from general rover processing. Thus, the autonomy functions necessary for identification must be provided by the payload supplier.

In addition, the standard norms used to determine the sequence of commands applied in the identification of fires must be offered by the payload supplier, but implemented from the Command and Data Handling (C&DH) functional elements of the rover onboard processing. The storage required for this function is based on the possible identification of 500 fire events. Control requires thrusters that must have enough propellant for the de-orbit functions, following the mission lifetime of 5 years. Attitude control is provided by the reaction wheels and associated moment management is provided through torques and magnetometers to keep steering.

The star sensor, the gyroscope as well as the GPS receiver are part of the configuration of the basic attitude control and orbit determination system.

The fact that there is interoperation with the INPE-ITA ground station will guide the antenna sizing and frequency allocation that is assigned to the communication protocol, but the antenna pointing directly impacts on-board processing. In terms of C&DH the stored command string is provided by embedded processing, as well as basic utility functions, thermal and energy control.

The total Source Lines of Codes (SLOC) count for the processing of the rover used in the Classic FireSat mission is 4100 SLOC. The RAD6000 @20MHz processor was chosen for this project, as it has the highest percentage of At-

titude Control Determination SLOC among the other processors presented in (Wertz *et al.*, 2011). From the choice of processor, the onboard computer found in the market was defined, whose main characteristics are observed in Table 11. Lockheed Martin's SMEX-Lite onboard computer (Martin, 2020) proved suitable for the project due to its small mass and characteristics consistent with mission requirements.

Table 11. Key features of the SMEX-Lite onboard computer, retrieved from the Lockheed Martin catalog (Martin, 2020).

Parameter	SMEX-Lite
Performance	up to 27 MIPS
Frequency	up to 25 MHz
Power (High)	9 W @ 25 MHz
Power (Low)	2.5 W @ 3.1 MHz
Memory	4 MByte SRAM
Mass	< 0.6 kg
Interface	PCI, 1553, QHSS

5.4 Communications Subsystem

One of the most critical subsystems of the satellite is the communication subsystem, also known as Telemetry, Tracking, and Command (TT&C), since the use of this subsystem must guarantee the sending capacity and receiving the data in a reliable and compatible manner (Alminde *et al.*, 2002).

For this project, two reviews of the link satellite communications were considered: the uplink analysis, which consists of the transmission of data sent to the satellite, and the downlink analysis, which consists of the transmission of ground health data housekeeping of the satellite and data collected by the payload to the ground station.

The ground station used as a parameter to insert some of the initial data is defined as INPE-ITA, located in the city of São José dos Campos/SP.

Therefore, was used the spreadsheet SMAD Design (Santos and Albuquerque, 2020) for the analysis, which resulted in a margin of 21.02 dB for the Signal-to-Noise Ratio in the uplink and a margin of 16.24 dB of the Signal-to-Noise Ratio for the downlink, thus, attributing the use of the UHF frequency to the uplink and the S-Band frequency to the downlink.

5.5 Energy Subsystem

The energy subsystem has as its main function to generate, store, regulate and distribute electrical energy for the components and other subsystems of the satellite. The main power generator for low-orbit satellites is based on solar panels, which convert solar energy into electrical energy needed to run the equipment. A solar panel contains a set of solar cells (or photovoltaic cells) whose basic working principle includes a PN junction in a semiconductor, in which the main light-receiving part is the N material and the lower part is the P material (Sampaio and González, 2017).

There are several advantages of using energy from solar sources, which include low operating and maintenance costs, environmental factors, it is also a clean and renewable energy which is widely available in the environment.

The triple junction GaAs (Gallium-Arsenic) is one of the most used for space applications. When compared to other types of cells (silicon, indium phosphide, and others) it has greater efficiency and less degradation over time, in addition to a good working regime in environments with high radiation (Micha *et al.*, 2018).

Table 12 shows the parameters used to determine the area of the solar panels and also the number of batteries needed to supply the satellite's energy demand.

The dimensions of the panels designed can be considered oversized for the beginning of the mission, however, the longer the mission lifetime, the greater the power difference between EOL (End of life) and BOL (Begin of life).

Satellites that use photovoltaic cells to generate energy for their components demand a system that performs the storage of this energy, mainly to be used during periods of the eclipse and certain excess demands that may occur. Such storage can be done via batteries, super capacitors, fuel cells, and others. For this application, batteries were used.

A battery consists of individual cells connected in series. The number of cells is determined by the working voltage of the components that are connected to them. When connected in series, there is an increase in the output voltage, and when connected in parallel, the current increases. Primary batteries are those that convert chemical energy into electrical energy but do not do the reverse process. Secondaries batteries are capable of being fed back thousands of times.

The secondary battery selected was Li-Ion (lithium-ion), since it has lower volume and mass when compared to Ni-Cd (nickel-cadmium) and Ni-H₂ (nickel-hydrogen) (Wertz *et al.*, 2011).

The bus voltage chosen was 28 V, since most sensors and actuators work in this voltage range. The average discharge depth (D_{OD}) for Li-Ion batteries is 30 %. Therefore, considering two batteries, the minimum capacity is 2.85 Ah.

Table 12. Main parameters for the energy system.

Parameter	Value
Average power of day and eclipse	64 W
Day period	3901.99 s
Eclipse period	2127.45 s
Panel/charge/battery efficiency	60 %
Panel/loads efficiency	80 %
Cell conversion efficiency	33.8 %
Solar constant	1368 W/m ²
Inherent degradation	77 %
Incidence of the sun (worst case)	23.5°
Degradation by cell year	0.5 %
Discharge depth	30 %

A load cell used in space applications was selected so that it could meet the energy demands of the systems. In this case, seven cells connected in parallel must be used in order to meet the electrical voltage demanded by the bus. The technical characteristics are presented in Table 13.

Table 13. Main parameters of the battery chosen for the project.

Parameter	Value
Mass	222 g
Voltage	3 to 4.1 V
Nominal Capacity	6 Ah
Energy Density	263 Wh/L
Specific Energy	105 Wh/kg

6. CONCLUSION

Given the importance of using satellite remote sensing technology for monitoring and preserving natural resources, especially in estimating risks, such as the detection of forest fires, this paper presented the conceptual design of the Classic FireSat mission for monitoring fires in the Northeast of Brazil. The stipulated requirements and restrictions influenced the choice of the form of identification, the orbit, the subsystems, and their components. There was a need to iterate this process several times as the design phases progress. Therefore, it was demonstrated that the proposed project is suitable and viable for a preliminary study since the results are in agreement with the literature and real applications, including components available on the market. It is reiterated that this project consists of a preliminary study, therefore, it does not encompass all the details and complexity of a real conceptual project. However, it has a consistent theoretical basis that allows an initial estimate for projects that aim to meet the same objectives.

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