COMPREHENSIVENESS BALANCE FOR EFFICIENCY (CBfE) METHOD FOR A PLATFORM-BASED SATELLITE FAMILY

Otavio Luiz Bogossian

National Institute for Space Research (INPE), Brazil, otavio.bogossian@gsr.inpe.br

Geilson Loureiro, Roberto V. F. Lopes

INPE, geilson@lit.inpe.br, roberto.lopes@dss.inpe.br

Abstract: *This paper aims to propose a method to balance comprehensiveness and efficiency adjusting the main parameters of the space platform components during the conceptual design phase. The method helps the developers to assess the platform comprehensiveness in terms of possible missions and the platform efficiency, based on the additional mass necessary to comply with launch and space environment requirements like drag, radiation, torque, etc. The more over dimensioned, the less efficient is the satellite. The method will be exemplified with a real case, the Multi-Mission Platform (PMM) conceptual design. Conclusions are that the method promotes a great enhancement on the productivity of platform based solutions conception while increasing the quality of the conceptual phase results.*

Keywords: *platform, satellite, product-family*

1 Introduction

The family of products concept became relevant with the transformation of the mass production concept into mass customization aiming to comply with individual client needs (Pine, 1993). The segmentation market grid based on platform was introduced (Meyer and Lehnerd, 1997) as the way to leverage the family of products across different market niches. Meyer and Utterback (1993) attempt to map the evolution of a given product family based on platform by means of extensions and upgrades. The family is a set of similar products obtained from a common platform, given to each product the functionalities required by specific clients (Meyer and Lehnerd, 1997).

The space context has specific characteristics such as the product complexity and the very low production volume. It was remarked (Gonzalez-Zugasti *et al.*, 2000) that space products are designed to comply with a particular missions, contrasting with general applications in which they are designed for market niches. The space product referenced in this paper is a satellite. Each satellite is composed of a payload (components to implement the specific satellite mission) and a bus (i.e. house-keeping or services functions). The bus is usually divided into sub-systems, each one for a specific discipline like structure, thermal, power, communication, on board data handling, attitude control, etc. (Aerospatial and Sextant, 1995; Alary and Lambert, 2007; Buisson *et al.*, 1998; Bouzat, 2000; Galeazzi, 2000; INPE, 2011). Each sub-system is composed of several equipment (e.g. power sub-system: batteries, solar array, regulators, DC-DC converters, etc.).

Bogossian and Loureiro (2011a) grouped the product family based on design methods (modularity, platform based, configurational or scalable), generation of product variety to target market niches and based on technical aspects for improving the product process, stock reduction and component reutilization promotion.

The space products, designed according to Meyer and Lehnerd (1997) premises, are those composed of a common platform that includes usually the same components to all space products (satellites) and a set of mission specific components that characterize each particular product and mission. The platform for a satellite is usually composed of almost all components necessary to guarantee the satellite operation (structure, power, on board data handling, attitude control, communication for control purpose, propulsion, etc.). The mission specific components are those designed specifically for the mission including some for housekeeping and all the payload equipment such as scientific experiments, cameras (Earth or Sun observation), communication for the specific application, sensors, etc. (Aerospatial and Sextant, 1995; Alary and Lambert, 2007; Buisson *et al.*, 1998; Bouzat, 2000; INPE, 2011; Dechezelles and Huttin, 2000).

This paper aims to present a method for assessing space low Earth orbit platforms development at the conception phase, to guide the various decision making points. This paper has the following specific objectives:

- to introduce the platform knowledge for general and space applications (presented above);
- to present the general and space platform development process and the impact of the last one in the platform;
- to present the *Comprehensiveness Balance for Efficiency* (CBfE) method;

- to present an application case for the method.

In order to achieve the specific objectives, this paper is organized as follows: section 2 presents the development process; section 3 presents the CBfE method; section 4 shows method application case and section 5 draws some conclusions and sets up some further work.

2 Development process for satellite family based on platforms

The development process in this context is an important premise, considering it defines how well we know the family of products to be produced.
In the space context, the most common approach to develop a mission is the independent development (Gonzalez-Zugasti et al., 2000), in which the main product (satellite), will be developed for a specific mission requirements. The number of recurring products is very limited, usually of one or two units. There are some exceptions like the GPS (24 satellites and an additional number of spare satellites) and Galileu (27 satellites and 3 spares) constellations (Forrest, 2004), where the satellites are placed in different orbits and phases.
The satellite platform concept was adopted by some space programs to exploit common aspects of the products (satellite), from one mission to another. In general, the space agencies do not have a complete view of the satellite family to be generated before the platform has being designed. However, they aim to increase the reutilization of the common part (platform) as much as possible when future missions are defined.
Boas and Crawley (2006) presented a very particular example of simultaneous development applied to a family of fighter planes (Joint Strike Fighter program) to define the platform for the fighters. They presented a second example, a sequential development process, in contrast with the previous one. The Boeing 777 platform was designed at the same time of the first product. According to the authors, this second approach has the inconvenience of making the first plane, a strong reference for the platform design that could cause problems for the future planes. This approach is easier to adopt due to long development process and difficulties to define the different members of the family. Boas and Crawley (2007) compared the parallel development, when the platform is developed for a known set of family members and the sequential development, when each member is developed after the other based on the platform developed for the first member.
Bogossian and Loureiro (2011a) concluded that the sequential approach is often applied to the development of multi-mission satellite platforms demonstrated by missions like Jason 1 using CNES PROTEUS platform (Aerospatial and Sextant, 1995; Dechezelles and Huttin, 2000; Grivel *et al.*, 2000), Demeter mission using CNES Myriade Product Line platform and called previously Ligne de Produits Micro-satellite (Buisson *et al.*, 1998; Bouzat, 2000; Cussac *et al.*, 2004) and SkyMed/COSMO mission using ASI/Alenia PRIMA platform (Galeazzi, 2000). During or after the platform design, the space agencies define a certain number of space missions based on platform flexibility and constraints (Galeazzi, 2000; INPE, 2001; Dechezelles and Huttin, 2000; Boas and Crawley, 2006) like covered orbits, pointing accuracy, launchers, lifetime, mass and power limit for payloads, etc.

3 The Method

Comprehensiveness Balance for Efficiency (CBfE) in this context aims to balance the mission attributes and the platform efficiency based on the additional mass necessary to cover launch and space environment requirements like drag, radiation, torque, etc. The more over dimensioned, the less efficient is the satellite.

3.1 The measure of penalties

Mosffatto (1999) has remarked that the platform concept has several benefits but also some drawbacks, one of them is the open architecture necessary to define new products will produce heavier products.
Gonzalez-Zugasti and Otto (2000) mentioned that the main benefits of a platform adoption are the development, manufacturing and operation costs by means of the reutilization and scale economy. As a drawback a lower performance or efficiency is obtained when compared with a specific development. He also remarks the need of product flexibility to comply with new requirements and also be economically feasible.
Boas and Crawley (2007) mentioned as one platform penalty is the performance reduction, without exploring this subject. Bogossian and Loureiro (2011a) concluded that a sequential development process without quantifying the associated penalties will increase the comprehensiveness in terms of missions and will tend to increase the platform inefficiency.

3.2 Principles

It was chosen as inefficiency measuring index, the equipment mass due to its importance in the design. The total satellite mass is limited by the launcher and the increasing of the platform mass reduce the available mass for the payload.

The platform equipment capacity is significantly affected by the platform comprehensiveness. The method will capture the required capacity for different cases (e.g. required angular momentum storage capacity of the reaction wheels for different cases) and will determine the corresponding mass. The equipment considered in this paper and their capacities are: fuel tank (kg of propellant); reaction wheels (angular momentum in Nms); torque rods (Am2); solar array generator (surface m2) and battery (Ah). The method also includes the shielding mass and the capacity of the electronic components to cope with the radiation environment (krad). For the launchers, the structural mass was considered. The method will capture the inefficiency through the difference between the worst and best cases. This difference is reduced by the number of possible configurations of each equipment (e.g. two different tanks sizes defining each one according with the mission requirements).
Bogossian and Loureiro (2011b) presented a preliminary version of this method without defining how to consider the scalability, use of heritage and use of several launchers in the method.

3.3 Cases and premises

In order to assess the mission comprehensiveness, it is necessary to limit the method coverage according to the present available information about the platform projects, reducing when possible, the number of cases. The multi-mission platforms considered to define the scope (Bouzat, 2000; Galeazzi, 2000; INPE, 2001; Rougeron, M., 2000) were Myriade (Centre Nationale d'Études Spatiales - CNES), Proteus (CNES), Plattaforma Riconfigurable Italiana Multi-Applicativa - PRIMA (Agenzia Spaziale Italiana - ASI/Alenia Aerospazio) and Plataforma Multi-Missão - PMM (INPE). A set of compatible parameters were obtained from the considered reference platforms. They are:

- Only circular orbits (low eccentricity);
- Three altitudes, 400 km, 700 km and 1500 km;
- Three low inclination orbits, 0°, 12° and 25°;
- Three SSO orbits altitudes/inclinations with two descending node crossing time each one, 10:00 am and 12:00 am;
- Two pointing target, Nadir and Solar;
- Satellite configurations with one or two Solar Array Generators (SAG) wings;
- Satellite with a parallelepiped shape.

Some combinations were excluded from the previous list. The 400 km is too low for solar pointing and SSO orbits are Nadir pointing exclusively.
For each considered orbit, an assessment of the environmental characteristic that could affect the platform equipment was taken into account. Equipment that do not depend on the environment were considered only with respect to the electronic component radiation hardness (transmitters, on board computer, platform sensors, etc.). Payload equipment are not part of the platform and were not considered, including the launcher interface.
The thermal control for low orbit platforms is normally passive, with heaters placed when necessary and, as a consequence, the solution should be defined in a case by case basis. This study will not consider the thermal dimensioning as part of the platform but a specific component to each mission as considered for Myriade platform (Bouzat, 2000).
Some equipment will be classified here as a Multiple Source Dimensioning (MSD) because they have their dimensioning based on a budget of several environment effects. Other equipment will be classified as Single Source Dimensioning (SSD) for a single effect. For the MSD case, the method will take into account only the dimensioning corresponding to the environment effect being considered.
For the SSD market available equipment, a minimum size is established according to the capacities available in the market. For the orbits that require less capacity than the minimum size, it will not be considered the inefficiency associated to this difference. Only when a higher capacity is necessary will be considered as inefficiency.

3.4 Environmental effect on components and equipment

For the platform equipment, *Table* 1 shows the capacity and the environment effect considered by the method. For the cases and conditions presented in section 3.1, using several simulation data (STK EE v 8.12) and specific platform characteristic, the method will determine the required capacity to cope with the environment. If necessary, the method will perform some interpolations for the specific application case. The method will transform capacity in equipment mass using a specific capacity (e.g. for reaction wheels kg/Nms) using a mean value of up to three equipment of the same type. The obtained mass represents the amount of equivalent mass necessary to cope with the specific environment. The method will determine the maximum and minimum mass values and will consider as inefficiency, the difference between the best and worst cases. If the design considered more than one equipment capacity, the inefficiency will be divided by the number of different configurations (scalability concept). If the equipment is considered heritage, the inefficiency will not be taken into account.

The method, implemented in spreadsheets, generate tables covering all standard cases that are applicable.

4 The PMM Application case

The PMM project (INPE, 2001) began in 2001 at INPE with the objective of providing the necessary mean of producing low Earth orbit satellites in a reduced time and cost. The satellite considered for the first mission at that time was the Remote Sensing Satellite (SSR). At the present, the first satellite is the Amazonia-1, a remote sensing satellite planned to be launched in 2013 and shown in Figure 1. The satellite dimensions are 2.35 x 0.95x 0.95m. The total mass is around 560 kg and the platform mass is around 300 kg. The satellite has always two wings (symmetrical configuration) with a total surface of 6.3 m2 with SADA (Solar Array Drive Assembly) to rotate the wings. The Platform lifetime is four years.

Table 1 - Environment Effect for Equipment and Components

ENVIRONMENT	AFFECTED EQUIPMENT	CAPACITY	UNIT
Aerodynamic drag	Tank	kg propellant	kg
Aerodynamic drag torque	Reaction Wheel	Angular momentum	Nms
Magnetic field	Torque rods	Magnetic dipole	Am^2
Sunlight	Solar wings	Surface	m^2
Eclipse	Battery	Ah	Ah
Electronics components	All electronics	TID	krad
Launchers (quasi static and stiffness requirements)	Structure	Stiffness and strength	kg shielding

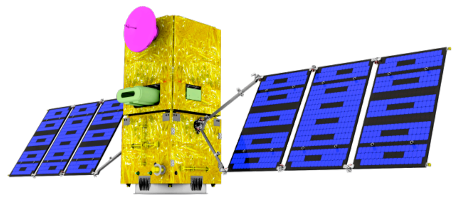

Figure 1 - Amazonia-1. The first PMM satellite

For each environment effect will be presented a table with the method result for each reference case of the application (PMM). The last two yellow columns of these tables shown the interpolation applicable due to lower orbit limits (600 and 1200 km) than the standard limits. These tables left clear that PMM has the possibility to point to the Sun and Nadir and always have two wings to avoid large torques.

4.1 Drag

The atmospheric drag affects the amount of propellant necessary to keep the orbit under an error limit. The fuel tank is considered a MSD and some of the input parameters are: Isp, number of wings and satellite surfaces. *Table 2* shows the mass tank based on the fuel necessary for the different PMM cases. It is possible to observe that only the two wings case (W=2) is applicable. *Figure 2* shows the curve used in one of the cases to estimate the PMM tank mass. Only one tank size was included in the platform design that represent only one configuration.

Table 2 - Tank Mass

TANQ MASS								
				ORBIT ALTTITUDE (km)			PMM	
ORBIT	INC.	POINT.	WINGS	400 (kg)	700 (kg)	1500 (kg)	600 (kg)	1200 (kg)
EQU	0	NADIR	1	NA	NA	NA	NA	NA
			2	27,25	0,70	0,00	3,47	0,09
LOW INC	12	NADIR	1	NA	NA	NA	NA	NA
			2	27,25	0,70	0,00	3,47	0,09
LOW INC	25	NADIR	1	NA	NA	NA	NA	NA
			2	27,25	0,70	0,00	3,47	0,09
EQU	0	SOLAR	1		NA	NA	NA	NA
			2		0,88	0,01	1,39	0,04
LOW INC	12	SOLAR	1		NA	NA	NA	NA
			2		0,88	0,01	1,37	0,04
LOW INC	25	SOLAR	1		NA	NA	NA	NA
			2		0,86	0,01	1,36	0,04
SSO 10H	97/98	NADIR	1	NA	NA	NA	NA	NA
	100		2	27,25	0,70	0,00	3,47	0,09
SSO 12H	97/98	NADIR	1	NA	NA	NA	NA	NA
	100		2	27,25	0,70	0,00	3,47	0,09
Max:	3,5	Min:	0,04	Unit:	3,43	Inef:	3,43	
# TANQS CONFIG.:			1					

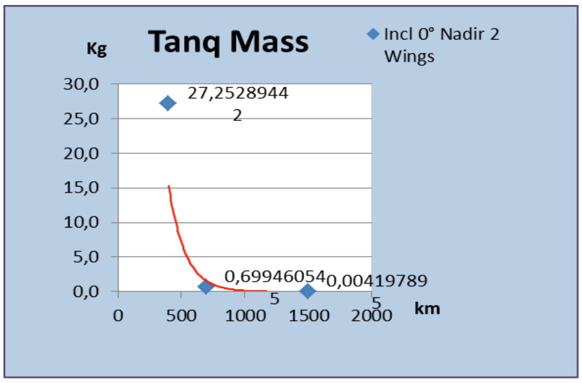

Figure 2 – Tank mass interpolation

4.2 Aerodynamic torque

The atmospheric drag affects the reaction wheels dimensioning in terms of angular momentum to be store for a number of orbits. It is considered as MSD and some inputs are the center of mass, center of pressure, satellite surface, number of orbits to be stored, etc.

Table 3 shows the necessary mass for the different cases. It is possible to observe that four wheels are considered on board (Inefficiency is four times the unit mass).

4.3 Magnetic field

The Earth magnetic field will be used to dimension the torque rods to unload the reaction wheels. They are considered SSD and a minimum value is established. *Table 4* shows the torque rod mass for the reference cases and the estimation for PMM orbits. The minimum capacity adopted for this case was 12 Am^2.

Table 3 - Reaction wheels mass

REACTION WHEEL MASS								
				ORBIT ALTITUDE (km)			PMM	
ORBIT	INCL.	POINT.	WINGS	400	700	1500	600	1200
				(kg)	(kg)	(kg)	(kg)	(kg)
EQU	0	NADIR	1	NA	NA	NA	NA	NA
			2	0,00	0,00	0,00	0,000	0,000
LOW INC	12	NADIR	1	NA	NA	NA	NA	NA
			2	0,00	0,00	0,00	0,000	0,000
LOW INC	25	NADIR	1	NA	NA	NA	NA	NA
			2	0,00	0,00	0,00	0,000	0,000
EQU	0	SOLAR	1		NA	NA	NA	NA
			2		0,07	0,00	0,066	0,002
LOW INC	12	SOLAR	1		NA	NA	NA	NA
			2		0,07	0,00	0,066	0,002
LOW INC	25	SOLAR	1		NA	NA	NA	NA
			2		0,07	0,00	0,065	0,002
SSO 10H	97/98 100	NADIR	1	NA	NA	NA	NA	NA
			2	0,00	0,00	0,00	0,000	0,000
SSO 12H	97/98 100	NADIR	1	NA	NA	NA	NA	NA
			2	0,00	0,00	0,00	0,000	0,000
Max:	0,1	Min:	0,00	Unit:	0,07	Inef:	0,26	
	# Wheel config.:		1					

Table 4 - Torque rod mass

TORQUE ROD MASS								
				ORBIT ALT. (km)			PMM	
ORBIT	INC.	POINT.	WINGS	400	700	1500	600	1200
				(kg)	(kg)	(kg)	(kg)	(kg)
EQU	0	NADIR	1	NA	NA	NA	NA	NA
			2	0,000	0,000	0,000	0,41	0,41
LOW INC	12	NADIR	1	NA	NA	NA	NA	NA
			2	0,001	0,000	0,000	0,41	0,41
LOW INC	25	NADIR	1	NA	NA	NA	NA	NA
			2	0,001	0,000	0,000	0,41	0,41
EQU	0	SOLAR	1		NA	NA	NA	NA
			2		0,049	0,000	0,41	0,41
LOW INC	12	SOLAR	1		NA	NA	NA	NA
			2		0,049	0,000	0,41	0,41
LOW INC	25	SOLAR	1		NA	NA	NA	NA
			2		0,059	0,000	0,41	0,41
SSO 10H	97/98 100	NADIR	1	NA	NA	NA	NA	NA
			2	0,001	0,000	0,000	0,41	0,41
SSO 12H	97/98 100	NADIR	1	NA	NA	NA	NA	NA
			2	0,001	0,000	0,000	0,41	0,41
Max:	0,41	Min:	0,41	Unit:	0,00	Inef:	0,00	
Min. Config.: 12 Am2 (0,41 kg				# Torque Rods Config.:			1	

4.4 Sun

For each case it was determined the solar panel surface necessary to provide the minimum power established for the platform. The solar wings are considered a SSD but they are not a commercial product, therefore is designed specifically for the platform and no minimum value was established. For each solar panel, the standard surface adopted by the project will be considered in pairs, one for each wing. *Table 5* shows the necessary surface and *Table 6* presents the even number of panels necessary to cover the required surface. A fixed mass was considered per wing (yoke, hold-down, etc.) and a variable mass with the specific capacity associated. Some inputs are taken into account as the cell and equipment efficiencies. The mass reduction due to the exclusion of the SADA mechanism for Sun pointing is not considered as inefficiency.

Table 5 - Required surface for SAG

SOLAR ARRAY GENERATOR - SAG (only panel surface)							
			ORBIT ALTTITUDE (km)			PMM	
ORBIT	INC	POINT.	400	700	1500	600	1200
			(m2)	(m2)	(m2)	(m2)	(m2)
EQU	0	NADIR	3,49	3,29	2,98	3,38	3,08
LOW INC	12	NADIR	3,78	3,57	3,19	3,66	3,32
LOW INC	25	NADIR	4,43	4,00	1,00	3,88	2,22
EQU	0	SOLAR		3,01	2,74	3,05	2,84
LOW INC	12	SOLAR		2,92	2,61	2,95	2,72
LOW INC	25	SOLAR		2,75	2,64	2,76	2,68
SSO 10H		NADIR	3,77	3,43	3,19	3,64	3,25
SSO 12H		NADIR	3,26	3,09	2,83	3,17	2,92

Table 6 - # panels and panel wing mass

ALWAYS TWO WINGS					1 ou 2 WINGS					
# Panels (*)		Fixed M	Total Mass		# Panels		Fixed Mass		Total Mass	
600	1200		600	1200	600	1200	600	1200	600	1200
(#)	(#)	(kg)	(kg)	(kg)	(#)	(#)	(kg)	(kg)	(kg)	(kg)
4	4	9,596	24,69	24,69	NA	NA	NA	NA	NA	NA
4	4	9,596	24,69	24,69	NA	NA	NA	NA	NA	NA
4	4	9,596	24,69	24,69	NA	NA	NA	NA	NA	NA
4	4	2,856	17,95	17,95	NA	NA	NA	NA	NA	NA
4	4	2,856	17,95	17,95	NA	NA	NA	NA	NA	NA
4	4	2,856	17,95	17,95	NA	NA	NA	NA	NA	NA
4	4	9,596	24,69	24,69	NA	NA	NA	NA	NA	NA
4	4	9,596	24,69	24,69	NA	NA	NA	NA	NA	NA
(*) Sempre par			Max:	24,69	Min:	17,95	Inef:	0,00		
			Min. Without Sada			24,69				

4.5 Eclipse

The eclipse duration was determined for each case and the battery capacity required, coping with the duration. The battery is considered a SSD, but considering that is a developed product based on off the shelf accumulators, it is possible to implement the necessary configuration adopting several strings, composing the necessary capacity not being necessary to establish a minimum value. Inputs considered are efficiencies, minimum bus voltage, mean DoD (Depth Of Discharge). *Table 7* shows the mass battery necessary for the reference cases and for PMM. *Figure 3* shows the interpolation for the PMM orbit limits.

Table 7 - Mass battery

BATTERY							
		ORBITA ALTITUDE (km)			PMM		
ORBIT	INC.	400	700	1500	600	1200	
		(kg)	(kg)	(kg)	(kg)	(kg)	
EQU	0	9,98	9,93	9,64	13,54	13,30	
LOW INC	12	9,97	9,76	9,64	13,41	13,18	
LOW INC	25	12,03	9,76	9,65	14,71	13,18	
SSO 10H		9,68	9,37	8,80	14,47	12,13	
SSO 12H		9,95	9,45	9,34	13,03	12,49	
Max:	14,7	Min:	12,13	Unit	2,58	Inef.	2,58
# Battery Config.			1				

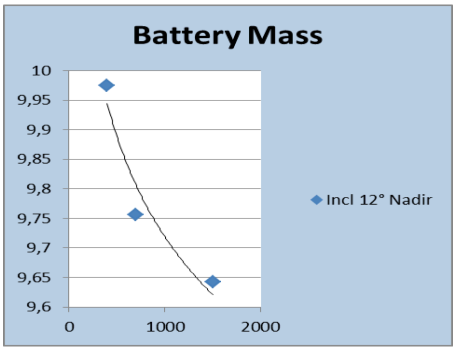

Figure 3 - Interpolation for one of PMM case

4.6 Space Radiation

The Total Ionized Dose (TID) impact will be determined for each orbit in terms of krads on silicon components. For equipment developed specifically for the platform, the electronic components could be considered shielded by the box and by the platform structure. Market provided equipment will be considered with its TID specification. An equipment will be considered as SSD, as a consequence, a minimum krads value was established. When the TID of the environment exceed the component specification, two aluminum plates must be added at the edge of the boards with a thickness necessary to reduce the radiation. The method requires as input the lifetime, component TIDs, number and size of the boxes. The simulation was performed using Spenvis [Spenvis, ESA]. *Table 8* shows the result of the method application for TID determination. The electronic components in the PMM project were dimensioned for the worst case but in this study it was necessary to reduce the TID definition to capture the inefficiency in terms of mass.

4.7 Structure

To capture the inefficiency from the structure design for several launchers, it was considered the longitudinal acceleration effect and the required rigidity to decouple the satellite first mode from the launcher. The over dimensioning was considered for the platform top, lateral e bottom panels. For the bottom panel, two configurations were considered for different launcher interface. *Table 9* shows the method results.

Table 8 - Shielding mass for TID limitation

TOTAL IONISED DOSE			
		PMM	
ORBIT	INC.	600	1200
		Mass	Mass
EQU	0	0,00	0,00
BAIXA II	12	0,00	0,00
BAIXA II	25	0,00	6,67
SSO 10H		0,00	2,72
SSO 12H		0,00	4,69
Max:	6,7	Min:	0,00
		Inef	6,67

Table 9 - Effect over the platform panels

LAUNCHER EFFECT			
PANEL / COFIG.	EFFECT	MASS PER PANEL (kg)	TOTAL MASS (kg)
TOP	Longitud. Accel.	1,60	1,60
LATERAL	Longitud. Accel.	0,89	3,55
BOTTOM I/F PANEL	Longitud. Accel.	0,00	0,00
BOTTOM I/F FRAME	Longitud. Accel.	3,37	3,37
BOTTOM I/F PANEL	Decoupl. 1st mode	0,00	0,00
BOTTOM I/F FRAME	Decoupl. 1st mode	0,99	0,99
TOTAL			5,15

4.8 PMM Final Results

Based on *Table 10* it is possible to conclude that 3.9% of the platform and 2.1% of the satellite corresponds to the direct masses that measure the inefficiency of the PMM adopted comprehensiveness. For the total mass that includes the indirect mass, these values are 6.1% and 3.2% respectively.

Table 10 - Final Results

TOTAL	PMM
COMPONENT	MASS (Kg)
TANQ	3,43
WHEELS	0,26
TORQUE RODS	0,00
SAG	0,00
BATTERY	2,58
STRUCTURE	5,15
SHIELDING (INDIR. MASS)	6,67
MASS	(Kg)
DIRECT INEFFICIENCY	11,4
TOTAL INEFFICIENCY	18,1
PLATFORM	295
SATELLITE	557,0
INDEX	%
DIRECT INEFFICIENCY	3,9%
TOTAL INEFFICIENCY	2,1%
PLATFORM	6,1%
SATELLITE	3,2%

5 Conclusions and Further Work

The platform previous knowledge and the development process were presented. *The Comprehensiveness Balance for Efficiency* (CBfE) method was described with its main elements. The method was illustrated by an application case of an existing platform, the PMM.
Table 10 shows the PMM Final Results giving an estimation of the inefficiency derived from the generality of the solution and provide to the platform designer, a tool to perform trade-off studies during the conception phase enhancing the productivity of the project.
The method should be improved on the adopted models for each environment, the analytical solutions for the structure, the interpolation to reduce the orbit inclination range if necessary and the environment not covered by the method (e.g. Sun pressure).

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