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"Application of advanced technologies to achieve operational microspace systems"

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Any intelligent fool can make

things bigger, more complex, and more violent.

It takes a touch of genius – and a lot of courage -

to move in the opposite direction.

Albert Einstein

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1. Introduction

The notion of what a small satellite is has been the subject of academic and technical discussions since they regained importance in the 1980s. The expression “small satellite” is not always clear nor unequivocal. Much of the confusion comes from the definition of what a small satellite is. How heavy can a satellite be and still fit within the classification of small satellite? Or should the definition be instead based on its dimensions?

Nowadays, ranking of satellites in terms of their mass allows talking about different classes of satellites: we can recognize, as the most widespread and accepted convention stated, three classes of satellite:

- Large satellites;
- Medium-sized satellites;
- Small satellites.

The boundaries of each of these categories can vary according to the references, or the feelings of the people; one of the possible satellite classification by mass criteria is provided in [1, 2], and has been reported below.

1. Large satellites: mass greater than 1000 kg;
2. Medium-sized satellites: mass comprised between 500 and 1000 kg;
3. Small satellites: mass lower than 500 kg.

Such limits are not universally accepted, since the notion of a “small”, “medium” and “large” depends also on the applications, the common understandings and often the background of the readers.

Anyway, from this moment on, we will use the definition mentioned above.

At the beginning of the space age, all satellites were small satellites. The limited capability of launchers available at that time limited the mass of the satellites that could be injected into orbit. The first artificial satellite was the famous Sputnik-1, a USSR satellite weighing about 84 kg. It operated during 1440 orbits in the months between its in-orbit release and orbital decay. In the same year, 1957, the USSR also launched Sputnik-2, six times larger than the Sputnik-1, with a mass of approximately 500 kg. The next year, 1958, the U.S.A. succeeded in launching their first satellite, the Explorer-1, a cylindrical satellite weighing slightly less than 14 kg.

During that same year, Explorer-2 (14.5 kg), Vanguard-1 (1.5 kg), Explorer-3 (14 kg), Explorer-4 (25.5 kg), Pioneer-1 (85 kg), Pioneer-3 (5.9 kg) were all successfully injected into orbit, while other satellites failed to accomplish their missions either because of a rocket failure or failures in reaching their desired orbits. These include Luna E-1 No.1,Luna E-1 No.2,Luna E-1 No.3,Luna E-1 No.4, each weighing 360kg, Explorer-5, weighing 17 kg, Pioneer-0 and Pioneer-2, with a mass, respectively, of 38 and 40 kg.

With the sole exception of Sputnik-3, a huge satellite of 1,327 kg, the satellite launched in the first years of space activities, all the early satellites were characterized by mass within the category of small satellite.

Also the year 1959 was dominated by launches of small satellites, the larger of which was the Luna-2, a Soviet mission of 392 kg impacting the Moon.

1960 was characterized by the development of Vostok-L, a Soviet two-stage carrier rocket having the capacity to launch 4,550 kilograms into Low Earth Orbit. Its first flight took place the 15th of May , the first of four launches it accomplished that year. Meanwhile 1960 saw the U.S. successfully launch its first large satellite, the Discoverer 17, sent in orbit on the maiden flight of the Thor-Agena version B launcher capable of injecting about 1250 kg into a circular, 200km orbit.

As the time passed, launch vehicles became more and more capable, and the mass of satellites that could be injected in orbit increased dramatically. As the launchers became larger, the satellite designers tended to exploit their increased capability leading to construction of ever heavier satellites, all but abandoning the development of small satellites.

1.1. Renewed interest in Small Satellites

The 1980s began the period of renewed interest in smaller satellites which has continued and accelerated to today, with a commensurate increase in their development, launch and on-orbit operation. Their renaissance stemmed in large part from the new technologies, particularly in microprocessors and solid state radios and memories that allowed satellite developers and engineers to develop small satellites with capabilities previously possible only from much larger platforms. Their increased utility lead to a steadily increasing number of launches of small satellites, whose on-orbit population grew significantly, as illustrated in the figure below, showing the increase in the number of satellites in this category; the mass distribution for small satellites as launched is plotted below, and illustrates that there are no clear mass boundaries. A linear tendency line has been added to underline the small satellites’ trend during the period 1980-1999.

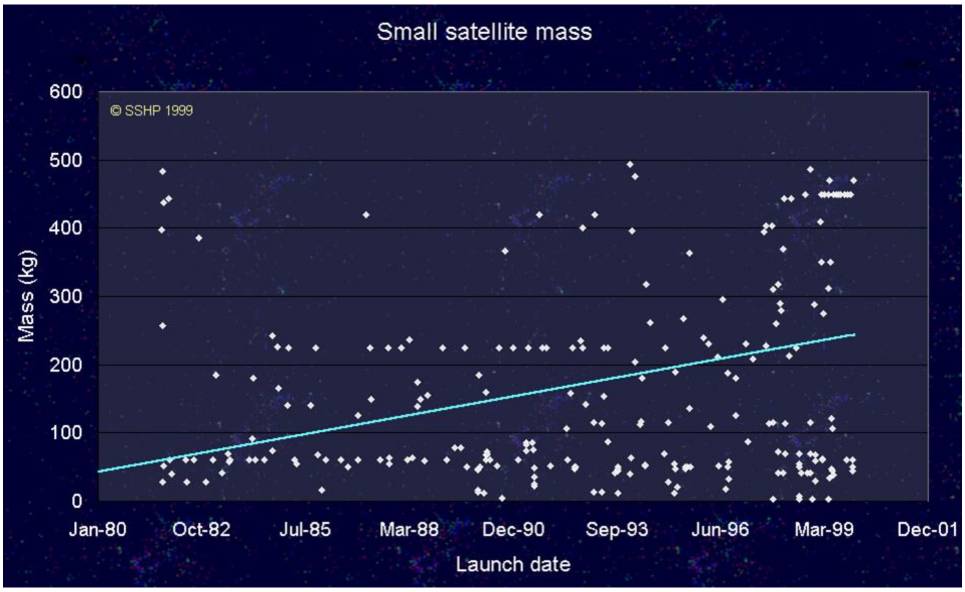


Figure 1 - 1 – Progress of small satellite in 1980-2000 - Courtesy of SSTL

Furthermore, in 1992, in response to pressure to stop budget overruns and in the wake of several high-cost missions which were completely or totally lost, NASA administrator Dan Goldin coined and spread the famous expression *"faster – better – cheaper"*, promoting the use of small satellites and the design and management methods characteristic of small missions to many NASA programs. This new generation of small spacecraft immediately demonstrated the advantages that lead to their widespread application across the range of space activities. These advantages included the short time required for their transition from the mission conception to realization of the space platform, often measured in months, compared with a ten to twenty years characteristic of traditional space programs. The short development time and low cost facilitates on-orbit replacement in case a failure should occur.

The growth in small missions continued through the decade of 2000, as recent studies[3] have confirmed. These studies document that, in the period between 1999 and 2010, 274 small government satellites (commercial satellites are excluded) had been launched, i.e. an average of 23 units per year (figure 2). The value of the government satellites launched during that period considered is consistent with what had been observed in the preceding fifteen years, 1983-1998. A deeper analysis of launches during the period considered lead to the result shown in the figure.

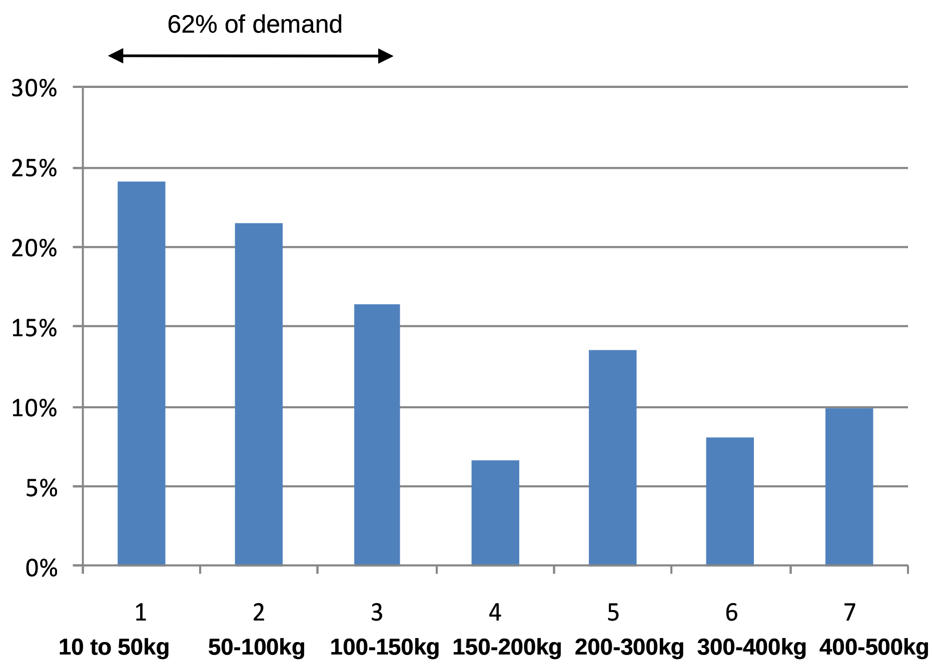


Figure 1 - 2 – % of smallsats launched in the years 1999-2010

The graph shows that, for the small satellites launched during the period investigated, a large percentage of satellites were in the range 10-150 Kg. In fact 62% of the satellites, constituting a total of 170 units, are in that range. These two studies neglected satellites with mass smaller than 10 kilograms.

The intense interest in this category of space platform drove the large increment in the number of small satellites launched and operated, as demonstrated in the studies previously shown. In reality the class of small satellites launched changed significantly. Besides the satellites considered in these earlier studies, about one hundred Cubesats (satellite weighing approximately 1 kg) have been launched, plus a similar number of satellites in the range of 1 to10 kilograms. This lead to a condition where satellites, which were extremely different, became lumped together. In fact they differ from each other not only in mass, but often drastically in the logic of their design and manufacturing, and in their mission objectives. There are small satellites designed with large expenditures of economical and managerial resources, using space-qualified components, aimed at carrying out highly demanding missions, and other small satellites designed and manufactured with commercially available components developed for terrestrial applications, without rigorous procedures and management oversight.

The identification of all these platforms, extremely different from one another, in the same category can be misleading and thus led to the definition of new categories to better distinguish among the different satellites. The categories are the following:

- Mini Satellite: 100-500 kg
- Micro Satellite: 10-100 kg
- Nano Satellite: 1-10 kg
- Pico Satellite: <1 kg

The main features of this class of satellites, the reason why they're becoming more and more popular, the advantages in using such platforms, but also limitations and problems arising during migration from a traditional, expensive and highly performing satellite to a small/micro-satellite, will be explored, investigated and described.

1.2. The traditional micro-satellite concept

As also indicated in the previously cited study, the category of microsatellite contains a large number of platforms which are being continuously developed and launched. The wide diffusion of these microsatellites owes mainly to three factors:

1. As launch cost depends strongly on weight, and the cost for each kilogram launched is extremely high (about $20.000), a lighter satellite promises to accomplish mission objectives at significant savings;

2. A microsatellite can be designed, built and launched in a shorter timeframe (24 months, typically) than a traditional satellite, which requires at least 5, and as many as 10 or more years to reach orbital operations. The small satellite will thus fly with current technologies on board whereas the larger satellite will be built with technologies already long obsolete.

3. A microsatellite, being lower cost than a traditional spacecraft, can be used as a "technology demonstrator", to test in orbit new technologies which newer flown before. Risk management dictates that traditional, expensive satellites use only components of known reliability, often meaning it is built with components that are old and already obsolete.

Small satellites achieve launch cost savings by sharing launches among multiple spacecraft. When a large satellite is launched, it must pay the entire cost of the launch vehicle and its operations. This total ownership of a launch is referred to as a "dedicated" launch.. Launch of a micro-satellite (or even a small one) need never pay the cost of a dedicated launch . Instead its launch is shared between two or more satellites. The main payload, a traditional medium or large satellite is the primary payload, paying most of the cost of the launch, and the other, smaller, satellites can be launched "piggy back", paying only their marginal per kilogram cost and not the entire launching expense. The great majority of micro-satellites have been launched as secondary payloads accompanying a larger spacecraft payload. Piggybacking gets the small satellite into orbit economically but inserted into approximately in the same orbit as the primary satellite, sometimes at the cost of some mission effectiveness.

These piggy-back launched micro-satellites have been typically developed using a low-cost approach that favored the use of commercial-off-the-shelf (COTS) components, despite that these devices are not designed to survive in the harsh space environment for the entire mission lifetime. Despite several drawbacks including the impossibility of assuring a certain performance level, or the increase of on orbit objects, has had a bright side, allowing many different organizations (academia, research centers and governmental institutions, as well as small companies) to access space, testing new technologies, fostering students working with real space platforms (even if really small ones, like Cubesat) and performing on-orbit experiments. The philosophy of small satellites (which includes also microsatellites) is well described in [6], where management of small, minimum cost, missions have been treated in detail.

The work I will present in these pages is a general study of the main characteristics of a micro-satellite, without focusing entirely or exclusively on the range of mass defined in the previous description, but also slightly encroaching on nano- and mini- satellites' territory, including those with mass up to approximately 150/180 Kg. Satellites with mass lower than 10 kilograms will be examined, but taken into account only if certain additional requirements will be met.

1.3. Novel concept

In contrast with the historical trend, in the last years a new, general tendency has arisen which should be highlighted: the increasing interest in augmenting performance of small satellites, seeking, at the same time, to improve the reliability of the system to be better compliant with space environmental conditions. A leading example of such a tendency is the recently launched ESA mission PROBA-2. This satellite demonstrates how technological innovations can be adapted to the space environment, providing high performance coupled with the capability to survive and correctly operate throughout the mission lifetime. This satellite demonstrates how such platforms can be taken in serious consideration for achieving demanding objectives and performing space operations.

This example might pave the way for the definition of a new paradigm for micro-satellites; this new concept recalls the features of the first pioneering satellite, which for the first time faced outer space, aimed at opening to mankind the frontier of space environment. Such platforms were developed and manufactured employing the most advanced technologies available in the field of engineering which have features in common with the space engineering. This tendency gave a strong push to the achievement of highly performing technology, necessary to make the satellite able to satisfy the unpredictable requirements that this new challenge presented.

The paradigm proposed in this thesis aims at exploiting innovative technologies, methodologies, procedures and behaviors for equipping micro-satellites with previously impossible capabilities, leaving the "low cost/COTS/piggy back" concept that permeated the small sat revolution of the former years. Such a paradigm is founded on three pillars: technology innovations, novel missions, the concept of responsive space.

Thanks to advancement of technology and miniaturization, the spacecraft bus and the payload will be equipped with devices able to guarantee high levels of performance, as already demonstrated by the ESA micro-satellite PROBA-2, which recently achieved a huge success. Such innovations do not simply include novel technologies, but also innovative procedures, methodologies and processes aimed at improving the microsatellite's capabilities and performance.

Thanks to these new capabilities, micro-satellites will be capable of satisfying the stringent requirements of substantive and valuable space missions. In fact, with the help of these "new", high performing platforms, novel kinds of missions can be proposed and performed: exploiting their reduced expense compared to traditional larger satellites, constellation missions are made economically affordable, and novel services can be proposed. But not only missions based on satellite constellation, also new single satellite missions, can be made feasible.

The third pillar composing the new proposed paradigm of high-tech micro-satellite is related to the space program timeline. In a world where technology runs faster and faster, and services and needs change constantly and even more and more rapidly, the time for designing, manufacturing and assembling a space platform needs to be shortened as much as possible: the concept of responsive space, with all its technical and technological implications, will be of outmost importance in completing the innovative paradigm proposed.

The success of this new, pioneering vision for the conception of microsatellites and their future implementation and usefulness speaks to the simultaneous existence of the abovementioned concepts. In fact, larger satellites have much higher capability than this category of platform, and the advantages of moving towards the implementation of (also) satellites of drastically reduced mass is not so immediate and evident.

In fact, if on the one hand the noteworthy miniaturization and the increased capacity in terms of mass memory storage, computational capability, and performance in general of electronic components goes in this direction, the unchanging geometrical limitations argue for the continuous further miniaturization of satellite components:

The power generated by solar cells is directly related to the area of solar panels; even improving the cells’ efficiency, the amount of power that can be generated cannot increase without limit. The resolution of images produced by electro-optical payload depends on the diameter of the objective optic; the satellite orbit can be lowered, the focal path can be split by means of mirrors and lenses, but of the maximum ground resolution cannot be achieved in an ever smaller payload. The possibility of operating the payload in eclipse condition, and, in any case, even if the payload is not in active condition when the satellite is not illuminated, the necessity of providing electrical energy to satellite vital subsystems depends on the presence on-board the satellite of batteries; and the more energy is necessary, the more massive the battery needs to be.

These examples demonstrate limitations to microsatellite capabilities that cannot be exceeded, even implementing the most advanced, performing and innovative technologies. Thus, larger satellites will always be capable of higher performance, and will continue representing the only way for accomplishing extremely demanding and leading-edge mission objectives; anyway, individuation of niche missions (and some of these will be also deeply investigated), that large satellites, for technical, managerial, or economical reasons cannot accomplish, would be the ideal motivation to attain such novel capabilities.

1.4. High-tech microsatellite

The first (and probably more important) aspect to consider in order to be capable to develop and design a really innovative and high performing platform, is the possibility to fully exploit technology innovation, developing a platform in an opposite way with respect to what traditionally happened for earlier microsatellites, based as they were on Commercial-off-the-shelf logic. This aspect is probably the most critical one due the following consideration:

Nowadays' stakeholders of space activities would like to see the results (in terms of produced services and missions accomplished) of the investment made in space project, so the tendency is that of mounting on board satellite reliable components, already flown in past missions, demonstrating their ability to survive in the space environment. For the newest components, long and expensive qualification processes are needed, and such procedures drastically reduced the injection of innovative technologies in the recently developed satellite. But these two mentioned behaviors if on the one hand are necessary to test and assure long exposure to space environment, on the other hand entail the use of old and obsolete technologies that significantly limit the on-orbit capabilities of space platform.

However, such long exposure might not be necessary for shorter duration missions, as the ones that can typically be imagined for most microsatellites. So, new highly advanced technology both for satellite bus and payload, would allow small platforms to use neither the costly, reliable but obsolete technologies used for larger satellite, nor the low cost, low performance and unreliable components commercially available and tested only for terrestrial application. Innovative technologies that can enhance the performance of such platforms will be proposed, and included in the architecture of possible innovative microsatellite, pushing upwards the capabilities and allowing more functionality, as has been proposed in other work and studies[4]. A shining example of such a concept is ESA's mission PROBA-2.

PROBA-2 is acronym standing for Project for On-Board Autonomy-2 and its successful eighteen months in orbit, with the satellite still fully operating in orbit, demonstrated that a microsatellite can provide the required performance for challenging missions; this microsatellite, weighing 120 kg, is the second mission in ESA's In-Orbit Demonstration (IOD) series.

The mission proved the efficiency of mounting new space technologies and equipment, testing also innovative technique and novel operational concepts in the actual space environment; spacecraft autonomy, ground segment autonomy and experimental attitude control algorithms, as well as previously untried Earth observing techniques or formation flying methods.

The main objective accomplished by Proba-2 is the demonstration that a small mission can couple new and advanced small payloads (delivering science data to users) with technology innovation capable to provide the necessary performance in order to successfully accomplish stringent payload requirements. It also proved that a mission and system design which uses embedded onboard autonomy and an automated ground segment can allow responsive missions with high flexibility and fast response time; that advanced development methods (such as code generation) are cost-efficient and robust; and that an attitude control system based only on an autonomous star tracker is sufficient for the pointing and stability needs of science and Earth observation missions.

Even if some of the seventeen technology demonstrators mounted on-board Proba-2 are 'guest' payloads, the great majority of them form part of the platform and system baseline design.

The PROBA-2 mission needs to be the starting point for the implementation of a novel concept of space exploitation. Low-cost, highly technological missions are the new frontier of the space mission: low-cost, high-technological satellite will be the answer for many missions that could never be performed by large satellite, while costly missions, requiring ten or dozen of years to be designed, manufactured and launched will continue to be the main way to accomplish extremely complex missions. The exploitation on board satellite of technology innovations will lead to improved technical performances.

Reliability will be strongly take into account, also in the frame of short mission, as these platform must guarantee, with a certain level of confidence, the ability to accomplish required mission.

1.5. Missions

The possibility of developing a microsatellite capable of provide high level of performances enable the significant opportunity of carrying out novel missions: new telecommunication Low Earth Orbit mission, or Earth Observation missions tailored over a certain area of the globe, with a high value of revisit time are specific examples of missions suitable for micro-satellites.

The mission is the starting point for this new microsatellite paradigm, a mission that such a platform, with its small mass and reduced volume, would be able to fully perform, accomplishing mission needs and being compliant with stringent requirements.

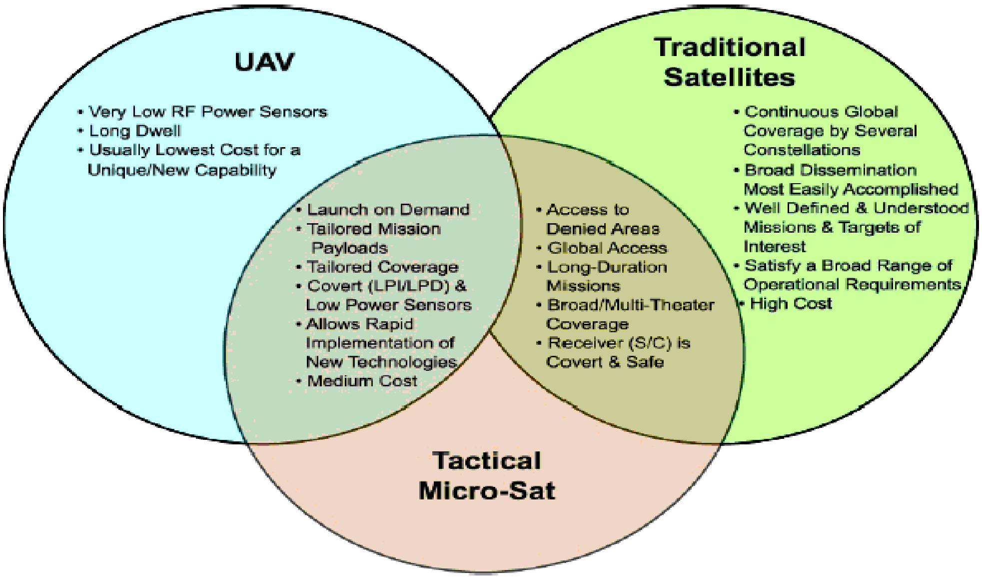


Figure 1 - 3 – Possible exploitation of microsatellites

A microsatellite can be the ideal platform for exploiting the benefits of a platform placed high over the Earth, with the advantages of low value of revisit time, large coverage (if required), duration of mission (compatibly with altitude) of several months, keeping, at the same time, some of the advantages of a different system, like the UAV (scaled for the case of space system), like the possibility of mounting a payload for a specific application, keeping low the cost of the system, allowing for rapid implementation of system and exploiting new technologies.

What must be developed, in order to be sufficiently successfully, is a system with a cost proportional to the results, the outcome of the entire project and the service produced: it must be the opposite of a large spacecraft, capable of performing traditional mission, as, for example, acquiring hundreds of image per day (Pleiades, 450 images/day), but a small platform, with a specific objective, as, for example, a localized area to observe. But these microsatellites need also to show valuable platform: a significant added value this platform could assure is *filling the gap* that traditional satellites have left uncovered: for example, satellites for Earth Observation are traditionally located over Sun-Synchronous Orbits, with Local Time Ascending Node (LTAN) equal, or at least close, to noon, because of the advantageous illumination condition in which observation in visible range of spectrum can be made. But this means that observation at different local time, like at 8:00 or 9:00 am cannot be obtained; while satellites with optical payload, especially if it works in the visible spectrum, will never launched on SSO with LTAN 8:00 or 9:00 am, it will not be possible to have a certain number of observation per day (as, for example 3 or 4 observations per day).

While the observations obtained at local time of noon are the best possible according to illumination condition (and this is the reason why traditional satellites are placed over these orbits) could be useful to have observation even in non-perfect illumination condition, in order to improve the number of images per day of a certain area; in addition, it could happen that meteorological conditions prevent the satellite from acquiring image of an area of particular interest, due to rain or clouds. Micro-satellite, or constellation of micro-satellites, could be used to fill this gap: launching platforms also on SSO with different LTAN can provide a variety of opportunity for acquiring images of a certain area, increasing the probability of obtaining an image.

A constellation of micro-satellites (or also a single satellite, but flying in a desirable orbit, tailored to the area to cover) could be also used to improve revisit time: in fact, traditional satellites designed and manufactured for acquiring significant quantity of images of the Earth (like Pleiades) are launched to orbits that do not benefit particular area with respect to others. These satellites are launched to have the possibility of making images of any zone of the Earth, but without any kind of preference for any area. Therefore, if the need of overflying a dedicated area with a certain repetitiveness arise, it would be difficult to satisfy such necessity implementing a satellite as performing as the traditional large satellite (say, for example, Pleiades); in such a situation, micro-satellite could, once again, come to aid, and one (or more, according to several mission requirements) micro satellite could be launched over proper orbits.

The mission such a platform faces needs to be tailored to microsatellite possibility: single-payload unique-purpose platform would help the satellite to be fully compliant with the mission requirements. Nice-to-have requirements need to be neglected, as they cause problems and large increments in mass/power (and consequently cost) budget: in fact, satellite systems largely suffer diseconomies of scale, and just a small increment in performances can cause a sharp rise in mass and cost of satellite.

Large satellite are capable of satisfying a large number of requirements, providing very high performance, but such platforms are expensive, and requires several years to be developed; on the contrary, microsatellite requires drastically less resources, can be developed in short timeframe, and have the capability to fully complete tailored objectives.

1.6. Responsive development & Air-launch capability

The possibility of rapidly implementing space capabilities is a technological and technical issue; its implementation depends on the development of components and equipment that could be rapidly integrated and tested: its accomplishment can enable space mission that cannot be performed in a different way but through the manufacturing of a microsatellite: large satellite requires a lot of time for being designed, manufactured, integrated and so on. Such platforms also require a large expense to be compliant with the requirements. Only the microsatellite can be the answer for a response request.

But the responsive space capability means a different way of facing the design of a satellite: no more development of mission-specific hardware or software, but function-devoted design: in such a way, space system could be easily assembled combining the functions and functionalities of different modular subsystem. But also the subsystem should be modular.

A further point investigated is the rapidly deployment of space capabilities, through the implementation of high tech microsatellites that are responsively designed, integrated and tested in order to be ready for immediate launch. Concepts like the space Plug-and-Play should allow reaching this ambitious goal.

Once the rapid development of a space platform has been achieved, such systems need to be responsively placed in-orbit, otherwise the rapid development turns out to be useless. Air-launch could be the solution: Air-launch, which is a launch characterized by a first stage of flight provided by an airplane (or, more generally, an aerial vehicle) is the final element that completes this new, innovative paradigm of micro-satellite and their exploitation.

Air-launch provides independence from the launch schedule of launchers of large satellites, assuring a high level of responsiveness, which is of fundamental importance in certain scenarios, like management of disaster. The concept of air-launch is not an innovation in the survey of launching possibilities: in the middle of the 1960s, Antonio Ferri participated in writing a document named *"The possibility of air-breathing launchers for space investigation"* as part of a contract developed at the Polytechnic Institute of Brooklyn[7]. Several studies have been performed during the last years, and a solution commercially available is still operating. These studies conducted to the identification of several options for air-launching satellite: captive on top, captive on bottom, internally carried are possible configuration for air-launching system.

The integration of satellites and launchers capable of pursuing the last phases of the in orbit placement presents major challenges, mainly in terms of technical solution.

Nevertheless, existing commercial services, and the increasing number of recent studies aiming at employing such capabilities, even seeking the development of an air-launch system for satellite up to about 6 tons[5], are clear signals of the interest for this alternative mean of space access.

1.7. Conclusion

Many of the concepts illustrated in any of the proceeding sections could refer to the other aspects of the dissertation illustrating the integrated nature of this study where all the aspect are strictly related and interconnected, as the only coexistence of any of the above mentioned, and further explained, concepts can justify the effort of such a shift in the conception and paradigm of the micro satellites.

An example could be the missions that a particular technology, that is not presently implemented in the space activities, would enable if it would be finally implemented in future microsatellites (or space systems more in general); talking about this specific technology might not make sense without underline the possibility of enabling missions that this technology would make possible, and vice versa.

So, in all the following chapters and paragraphs, references to other paragraphs and chapter will be continuously made, referring to previously introduced, but also successive described, capabilities and technologies.

In particular, in the second chapter, an investigation of the existing and in-development technologies has been done, looking for opportunities of improving microsatellites' performances; the technologies exhibiting such capabilities have been chosen; finally, supposing to have such technologies currently available, I briefly design possible application in space activities for these technologies.

In the third chapter, I introduce the aspect of "*Responsive Space*"; several studies have been undertaken on this topic, and the main results of these studies will be presented, together with personal consideration over opportunities offered, technical and technological difficulties to face, and possible methodological solutions to implement.

In the fourth chapter, looking for an instrument for integrating the concept of responsiveness with the idea of high technological micro-satellites, I developed a concurrent design tool, based on the ESA CDF kernel, for designing space missions, aiming at shortening as much as possible the time necessary for preliminary design and mission assessment; following the logic and methodologies of the "*Concurrent Engineering*", this tool has been developed involving in the preliminary design all the actors engaged in the design of a space mission.

In the fifth chapter, combining the concepts of responsiveness and technological innovations, and utilizing the concurrent design tool developed, I performed three intensive case studies. In each case study, the necessary premises have been done in order to highlight the needs to face.

In the sixth chapter, the conclusions have been drawn, underlying how the concepts that permeate the entire thesis can open a new perspective in the exploitation of space using microsatellites.

Figure Index

References

[1] – Herbert J. Kramer, “*Observation of the Earth and Its Environment: Survey of Missions and Sensors*” Ed. Springer

[2] – Jacob Job Wijker, “*Spacecraft Structures*” Ed. Springer

[3] – R. Villain, C.Talbot, *“Lessons from the Past for the Future of the Smallsat Market”*, THE 2010 4S SYMPOSIUM Small Satellite Systems and Services, 31st May – 4thJune 2010, Funchal, Madeira, Portugal.

[4] – Fleeter, “*The logic of microspace*”, Ed. Space Technology Library

[5] – Stratolaunch System, www.stratolaunch.com

[6] – F. Bruhn, E. Lamoureux, G. Chosson, J. Bergman, K. Yoshida, T. George, R. Thorslund, J. Köhler, “*Bridging the Space Technology “Valley of Death”: Two spaceflights in 2009 to validate advanced MEMS/Microtechnology systems and subsystems*", CANEUS 2009, NASA Ames, 1-6 March 2009.

[7] – A. Ferri, “*The launch of space vehicles by air-breathing lifting stages*,” Vistas in Astronautics, v. 2, Pergamon Press, London, 1958

2. Technology

In satellite design, it is common to use space-qualified components that promise the advantages of reliability surviving the harsh space environment, while providing, during the entire prescribed lifetime, the promised level of performance for which they were selected in the design phase.

These guarantees are of particular importance in the case of expensive, long duration, high performance missions. Large scientific and commercial geostationary satelliltes, need to survive as long as possible in the space environment, providing the performance necessary to carry out their mission objectives. In these systems a hardware failure could be disastrous. In fact, for scientific interplanetary missions, a failure could lead to the impossibility of ever performing that particular mission, as celestial condition might not be favorable for a reflight in subsequent years. In the case of a geostationary telecommunications satellite, the service needs to be continuously provided, and an interruption can lead to significant economic loss for the companies providing broadcasting services as well as possible sanctions against the company responsible for the satellite operations.

The consequence of a system failure in certain types of space missions can be the partial or total loss of a mission representing an investment of decades of work, billions of dollars, and the loss of service already contracted to clients, important scientific results or operational capability. Thus all the potential precautions to ensure system survival for the entire lifetime must be taken. Since component failure can result in systems failure and hence mission failure, avoidance of component failure is typically a central element in risk reduction.

The steps typically undertaken to minimize the risk of component failure are:

- Testing the components in facilities that simulate the space environment, to test their ability to withstand radiation, thermal excursion and mechanical stress occurring during the satellite's lifetime;
- Using components that have already been used in previous space missions, and that have already demonstrated their ability to reliably operate in the space environment;
- Designing the system taking into account cold and hot redundancy, in order to make the system able to tolerate unavoidable (random) component failures;
- Oversizing some subsystems in order to be still capable of providing necessary performances even in presence of component degradation (as for example in the case of solar arrays which degrade over time in the space environment)

Adoption of these measures brings with it certain disadvantages:

Increase in system cost: components tests have significant cost, as well as implementing redundancy, that has not only a direct impact, as more components need to be purchased or manufactured, but also logic of data selection among components (if the components are in hot redundancy) or switches for turning on switched off devices (if cold redundancy is implemented) are needed. Additional elements add mass which adds to launch cost, add to power consumption further increasing mass of solar panels and batteries (power is usually the single most massive subsystem) and the added complexity can in fact lower overall system reliability.

Long designing/development time: Test programs to ensure space qualification extend development program duration

Low performances: The use of components that have already flown in previous mission inevitably implies that such components are old, and, in many cases, even obsolete. Their lower performance results in more complex (and hence lower reliability) circuit designs and great cost and mass than would systems built with contemporary components.

The approach we propose in this chapter, and in this thesis more generally, is to enlarge the “field of view”, opening the possibilities to use technologies that traditionally are not taken into account.

As discussed in the introduction, the huge growth in the number of microsatellites and the number of developers devoted to them, is due to the fact that commercial-off-the-shelf components, devoted for terrestrial use, have been extensively used in designing, manufacturing and assembly of these platforms. These components have proved suitable when the mission needs are not so stringent as in other, conventional missions. But this approach does not allow the satellite developer to provide neither evidence of the platform’s ability to survive in the space environment, nor information about the possible lifetime, as components have not been tested, and heritage from past mission does not exist, so any kind of attempt to identify possible satellite lifetime is likely to have itself very low reliability.

The question is *”What type of components to use?”*

There is no right answer to this question, and no correct behavior to take. In fact, the lack of clear guidelines has led developers to the most obvious response, i.e. using space-qualified components, despite that the decision entails, as briefly mentioned earlier, a significant augmentation in the system cost, long time to bring the project to completion and, last but not least, a low level of performance. On the other hand, the performance that commercial-off-the-shelf components can provide are guaranteed for terrestrial application, not in the severe space environment.

Between these two behaviors, we propose a third way, which is located in between the two options currently in use, a way that tries to collect as many advantages as possible from the two ways, keeping the disadvantages at a minimum, and, in any case, maintained within well-known and acceptable limits. The way proposed consists in the use of components that exhibit a certain level of reliability coupled with significant advantages with respect to *commercial-off-the-shelf* components used for traditional terrestrial application. What is required of these devices is that they are manufactured to be used in a more severe environment that can (even partially) resemble the space environment. The present chapter will suggest technology innovations that will enlarge the capability of a microsatellite, giving it capabilities that were previously not achievable.

There are several consideration to bear in mind when choosing a component: in fact, components that have been designed, manufactured and tested for working in demanding environment can truly be suitable for typical application of space activities; if such components are produced in mass production, being used in large quantities, the cost of using even a few of them can be extremely advantageous with respect to components that needs to be designed, manufactured and tested in few samples, being devoted to just a single space mission.

Another typology of components taken in consideration are components that could be particularly suitable for the space environment, being intrinsically resistant to the most critical characteristic of the space environment: an example could be the recently developed thin film CIGS (Copper Indium Gallium Selenide,) solar cell, that exhibits a sufficiently high value of efficiency (19.9%[18]) but that shows dramatic advantages: having a non-crystalline structure, they are intrinsically radiation tolerant. This aspect provides significant benefits for the development of the subsystem devoted to the generation of electrical power.

In the following sections, examples of technological innovations applied to the field of space engineering are detailed. Possible technology innovations are proposed and their potential applications are explained: preliminary guidelines for how these technological solutions can be assessed, together with hypothesis on how to implement the mentioned devices in a standard space system, and the possible advantages in using such components.

2.1.Inter-satellite link

The use of multiple spacecraft flying in coordinated formation can provide additional value to future space missions and services. The satellites flying in formation have to communicate each other in order to enable advanced functions like relative navigation and formation control, clock synchronization, science and spacecraft health data exchange. Such capabilities can be enabled via Inter-Satellite Link (ISL) technology.

Flying two or more spacecraft, maintaining with extreme precision the control of the formation, presents high requirements for ISLs, because inter-satellite sensors are needed both to keep the communications, and to permit formation acquisition and maintenance in a precise relative position using inter-satellite tracking. Direct inter-satellite ranging is one of the tracking strategies, and it can be used in support of the GPS system to keep the relative satellites attitude.

Therefore, the combination of inter-satellite communication and ranging into a single package, and the implementation of such a capability on microsatellites would allow the fulfillment of a wide range of applications related to distributed spacecrafts systems. Inter-satellite communication and ranging would also allow to diminish the interactions with the ground station, reducing the cost of the operations, as well as enhancing the system robustness and making easier real-time operations. This technology is of high interest, both for current and future missions, and both for micro- and large satellites, once its capabilities have been demonstrated.

In case a new technology needs to be introduced in a certain field, it is a wise to take advantage of the existing hardware and software; however, the space environment impose some challenges, ranging from limited on-board power and computing resources, bandwidth constraints up to intermittent communication links, or line of sight ranging. These limitations make the design of an inter-satellite link system a highly demanding task.

Having said that, key system drivers have been analyzed in order to provide hypothesis for the development of a fully functioning inter-satellite communication and ranging system, to mount on board microsatellites.

2.1.1. Radio Frequency or Optical Inter-satellite Links

The primary communication media for an ISL (Inter-Satellite Link) systems are the radio frequency (RF) and optical/laser link.

Optical links have the capability of providing very high speed communications, with a data rate on the order of Gbps[1], which is suitable for transferring image, or large amount of data. Optical sensors are also used for formation keeping once the involved spacecraft reach very short distances with respect to each other, since they guarantees very high position accuracy. The laser are characterized by narrow beam and high directivity, and this lead to an interference-free signal transfer and a very low possibility of interception. However, optical sensors tend to have a relatively small field of view. Hence, to obtain spherical coverage all around the spacecraft, either a large number of them are needed, or their field of regard has to be extended by scanning. A more challenging part of an optical sensor is that a highly accurate Acquisition, Tracking and Pointing (ATP) subsystem is required in order to establish and maintain the communication link. In the process of “scanning for target” and “maintaining link”, a direct line-of-sight is needed and a rapid relative motion between two spacecraft will make this process more complex if the sensor exhibit a tight field of regard. Although transmission of an image by laser link from one satellite to another has been demonstrated in SILEX[2] (Semiconductor Laser Inter-satellite Link Experiment) with a data rate of 50 Mbps, optical communication is still a very new technology for spacecraft.

RF sensors present their own problems, but they appear more manageable if compared to those of optical sensors. RF sensors can guarantee significantly lower level of achievable data rate, but data rates of less than 10Mbps can be considered more than adequate if in the distributed spacecraft mission the link message traffic only consists of navigation data, spacecraft health and status, and some science data, not including information such as in interferometric and optical mapping missions. An RF link can provide omni-directional coverage when considering multi-antenna combination, and although RF equipment can be easily subjected to co-channel interference, multipath, atmospheric and man-made noise, a careful system design and the use of technologies such as spread spectrum modulation can significantly reduce interference effects in most cases. Furthermore, long term experience with radio transmission for space-to-ground links makes RF-based inter-satellite communication more reliable and easier to implement in space.

On balance, once there is a need for a very high data rate or a very accurate positioning requirement, optical sensors can be the solution; in all the other situations, an RF sensor is preferable for a small satellite mission.

2.1.2. Frequency Allocation

The choice of the most appropriate band (or even bands) of frequency is one of the most important choices that the developers of an inter-satellite communication capability had to deal with; such bands can be chosen according to the spectrum regulations specified by the International Telecommunication Union (ITU), and had to fit technical characteristics and constraints (including availability of hardware), as well as mission requirements. Regulation from international and national actors needs to be borne in mind when identifying the spectrum or when making frequency allocations for distributed spacecraft inter-satellite communications. Basing on the indications provided by ITU, several frequency allocation options may be available for inter-satellite communications[3,4]: frequency can be allocated to Space Research Service (SRS); Inter-satellite Service (ISS); Radio navigation and radio navigation-satellite service (RNSS). Frequencies and bands suitable for the abovementioned services are listed in the following table

Band	Frequency Range	Service	Examples
S	2.025 – 2.110 GHz	SRS	PRISMA
	2.200 – 2.290 GHz	SRS	TPF
Ku	13.75 – 14.3 GHz	SRS	
	14.5 – 15.35 GHz	SRS	
Ka	22.55 – 23.55 GHz	ISS	Iridium
	25.25 – 27-5 GHz	ISS	GRACE
	32.3 – 33.4 GHz	ISS, RNSS	StarLight
W	59 – 64 GHz	ISS	
	65 – 71GHz	ISS	

Table 2 - 1 – Frequency allocation candidates for inter-satellite communications

Apart from spectrum regulations, the system developer has to consider also the availability of equipment, together with the characteristic of each of the frequency bands; in fact, technical parameters can seriously affect the selection of the frequency to use for ISL system. The most important technical to bear in mind are:

- Available bandwidth and data rates;

It is well known that the higher the frequency, the wider the bandwidth available. According to the Shannon theorem, the maximum theoretical data rate, given a fixed signal to noise ratio, is proportional to the bandwidth. So, in order to achieve high speed data it is necessary to choose high frequency bands for which wide band use is permitted. In the process of frequency selection, the starting point is the amount of data to transmit, that depends on the mission requirements; this value determines the required bandwidth, which will consequently influence the frequency allocation, as shown in the following table.

Bandwidth	Maximum Data Rate	Recommended Frequency Allocation
narrow	< 100 kbps	S
medium	100 kbps - 1 Mbps	S,Ku
	1 Mbps - 10 Mbps	Ku, Ka
wide	10 Mbps - 100 Mbps	Ku, Ka
	> 100 Mbps	Ku, Ka, W

Table 2 - 2 – Bandwidth vs. Frequency

- Number of channels and/or number of links in the local distributed spacecraft network and multiple access techniques;

When implementing such a capability for a distributed formation of spacecraft, it must be taken into account that the satellites will have to share communication links; some technical solutions exist for guaranteeing that all the transmissions correctly occur, and these solution are:

I. CDMA (Code Division Multiple Access); because it employs spread-spectrum technology and a special coding scheme to allow multiple users to be multiplexed over the same physical channel, the total data rate can be equivalent to several times the single access bandwidth, depending on the modulation and the number of links. Therefore, sufficient spectrum is needed to accommodate this increased data transmitted simultaneously by multiple users.
II. FDMA (Frequency Division Multiple Access) is the most simple division, assigning a sub-frequency band for each spacecraft, and the bandwidth for each channel contributes to the total spectrum. In addition, a certain isolation in needed between subsequent sub-frequency band in order to mitigate mutual interference. When the number of channels is high, lower frequency allocation bands (e.g. 2025-2110 MHz in S band has only 85MHz bandwidth available) may not provide sufficient when using FDMA technology.

III. TDMA (Time Division Multiple Access), consists in sharing a single carrier frequency with multiple users, using it at different time slots; this technique has the least influence to frequency allocation compared to CDMA and FDMA.

- Link performance, associated with required transmitter power, propagation, and antenna characteristics, which can vary greatly depending upon frequency;

The free-space loss L_{FS} is inversely proportional to the square of the carrier frequency f[5]:

$$L_{FS} = \left(\frac{\lambda}{4\pi d}\right)^2 = \left(\frac{c}{4\pi d f}\right)^2$$

where d is the distance between the transmitter and receiver, c is the speed of light, and λ is the wavelength. Thus, lower frequency has a smaller space loss.

In addition, the RF frequency also affects the satellite transmitter power, antenna size and beam width. In turn, these factors affect satellite size, mass, and complexity. Their relationship can be expressed by the following equations[5]:

$$G_t = \frac{\pi^2 D^2 \eta}{\lambda^2} = \frac{\pi^2 D^2 \eta f^2}{c^2}$$

$$\theta = \frac{21}{f_{GHz} D}$$

$$EIRP = P L_l G_t$$

where G_t is the transmitter antenna gain, θ is the half-power beam width in degrees (-3dB lower than the peak gain), D is the antenna diameter, η is the antenna efficiency, P is the transmitted power, L_l is the transmitter-to-antenna line loss, and *EIRP* is the transmitter effective isotropic radiated power

From the previous equations we derive that, with a fixed value for EIRP, moving towards lower carrier frequency means that the antenna's diameter has to increase to maintain a specific beam width. As the dimension cannot arbitrarily increase, further diminishing in carriers' value must be compensated with higher transmitter power. On the other hand, higher frequency implies an antenna beam narrower, which means that certain (high) frequency bands are not suitable for omni-directional antennas

The overall link performance can be expressed by the link budget using the received signal-to-noise power density ratio C/N_0 of the communication system. If we assume identical transmitter power, antenna gains, and implementation losses in both the lower and the higher frequency bands, the received signal will be much weaker at higher frequency band.

In general, higher frequency bands enable smaller antenna size, but require more transmitter power to compensate for all the attenuation effects, such as free-space loss, reduced effective C/N_0, and ultimately make the overall communication system more complex.

Formations of distributed microsatellites that usually are severely constrained by mass, power and cost limits are recommended to adopt inter-satellite links using lower frequency allocation (e.g. S band).

- Ionospheric errors, Multipath, and Doppler shift effects;

ISLs can be used both for the distribution of data among spacecraft and for navigation purposes. Performing pseudorange measurements is a common way to realize relative navigation. From work performed in the past years, regarding the feasibility of using C-band (2-4GHz) as future GNSS frequencies[6], we conclude that frequency selection has an effect on the pseudorange error budget, including ionospheric effects, carrier multipath and the signal acquisition process. Ionospheric effects are inversely proportional to the square of the carrier frequency, and carrier multipath is inversely proportional to the carrier frequency. Therefore, a higher frequency allocation helps to reduce the errors caused by ionospheric path delay, and to mitigate the carrier multipath as well. However, due to higher maximum Doppler shifts at higher frequency, the Doppler search region increases, which negatively influences signal acquisition. Assuming identical code length, signal acquisition takes a longer time at high frequency bands.

- The carrier frequencies for inter-satellite links must be assigned so as to avoid interference with other onboard communication systems like the TT&C subsystem.

If the inter-satellite sensor and the TT&C subsystem work in the same frequency band (e.g. S band), as for the FDMA separation technique, sufficient frequency separation between the inter-satellite link, and the TT&C uplink and downlink must be considered, in order to reduce the risk of disturbance.

2.1.3. Data type, Data rates and Bandwidth

The information that the satellite has to exchange can be classified into several data types or traffic, each of which classes of data has different levels of data rate and bandwidth requirements:

- Navigation data
- Payload data
- Spacecraft health & status

Navigation data can contain the measured absolute position, relative distance, velocity, attitude, and time information. The volume and transmission frequency of the navigation data is tightly coupled to the nature of the mission. In case of tightly cooperating spacecraft in close formations (separation distances < 1 km) with high positioning accuracy and tight control windows, the typical maneuver cycle for maintenance of the formation may be too short for a ground-controlled formation and thus many require a fully autonomous on-board control approach with the help of inter-satellite communication. In this scenario, the frequency of broadcast of navigation data over an ISL can be on the order of seconds or even continuously, which enables approximately real-time relative navigation corrections to command the drifting spacecraft back within its control window. The tight communication link requires significantly higher data rate and greater bandwidth requirements than those missions requiring kilometer level positioning accuracy.

Payload data can be exchanged or not among distributed satellite according to the mission objectives and requirements. If the science data are periodically downloaded to ground stations without any processing from the space segment, thus there will be probably any necessity for transmitting such data to each other. On the other hand, could be implemented collaborative missions which may require that the scientific data acquired by a satellite has to be exchanged to facilitate distributed space-based computing. In this second case, estimates of the data rates and bandwidth for payload data must be done; such estimates present the greatest challenge because payload data can be of many types, and the topology of onboard distributed computing can be significantly different for different missions. The mostly used distributed computing topology is the star topology[7], which facilitates the data collecting, comparison and processing within the master spacecraft, but is not as bandwidth efficient as a distributed topology.

Spacecraft health & status data is the less constraining data in terms of volume, needing in the meantime, a less frequent transmission with respect to navigation data. The amount of this information depends upon the complexity of the spacecraft's equipment, but, in general, hundreds of bit is sufficient for such data.

In order to design a proper ISL, navigation, payload and spacecraft health & status data must be taken into account together; mission requirements will determine the major contributors for the overall data rate and bandwidth. For autonomous formation flying like PRISMA, navigation data will probably be the source of the majority of data, while, for the scientific missions, will probably be the science data that drives the choice of the communication strategy.

Furthermore, the determination of the data rate is affected by power constraints and separation distance between the spacecrafts. Higher data rate or larger distance means higher power consumption, and therefore higher system costs. Fortunately, missions with large separation distances between spacecraft are characterized by positioning accuracy and data rate requirements lower than close formations. The following table shows two examples of this situation[9, 10].

Inter-satellite transceiver	**Spacecraft separation**	**3σ Ranging Accuracy**	**Data Rate**
FFRF on PRISMA	10m – 500m	1m (LOS*>45°) 20cm (6°<LOS<45°) 1cm (LOS<6°)	12 kbps
	500m – 30km	1m	4 kbps
IRAS on MMS**	250m – 640km	30m	4.096 kbps
	640km –1800km	9km	512 bps
	1800km – 3500km	35km	128 bps

* Line of Sight
** Intersatellite Ranging and Alarm System (IRAS) on Magnetospheric Multiscale Mission

Table 2 - 3 – Two option of ISL

The overall process that finally leads to determination of inter-satellite communication data rates and bandwidth is shown in Fig. 2-1. Multiple access technology should also be considered because they affect bandwidth in different ways as explained in frequency allocation paragraph.

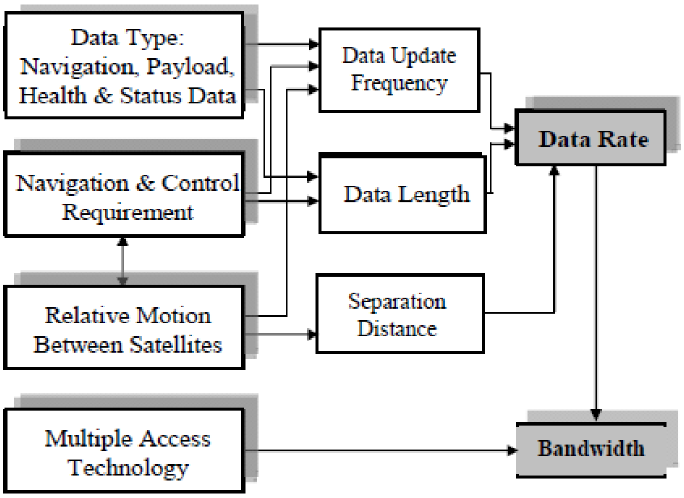


Figure 2 - 1 – Bandwidth Selection Process

2.1.4. Inter-Satellite Communication Network Architecture

The architectural complexity of the inter-satellite communication network is mainly dependent on its multiple access technologies and topology schemes: for this reason, without entering excessively in technical low-level detail, multiple access technologies and topology scheme will be presented.

2.1.4.1. Multiple Access Technology

Three basic techniques for sharing links in distributed spacecraft systems exist, and we already introduced them: these techniques are based on distributing various sources of signals in frequency (FDMA), in time (TDMA) and in coding (CDMA). The table 2-4 summarizes advantages and disadvantages, providing an attempt to identify a good candidate for microsatellites, bearing in mind the strong limitations on power, mass and cost that microsatellite impose. FDMA is not an economic choice, since it requires large frequency bandwidth to guarantee multiple channels assigned to multiple sub-frequencies. TDMA with its simple timing schedule is suitable for microsatellites, especially in the situation that the number of spacecraft is not large (<10). Some near future missions have the objective of testing formation flying technique using TDMA as their inter-satellite multiple access technology (PRISMA, DARWIN and MMS)[8,9,11]. The greatest challenge of TDMA technique is time synchronization with a maximal synchronization error that is equal to the propagation time of the signal (e.g. smaller than 100μs when the separation distance is less than 30km). CDMA is also an efficient way, especially when considering using GNSS-like signals to realize inter-satellite ranging. However, the near-far interference problem requires a power control scheme at the transmitters. The performance of CDMA decreases with increasing number of spacecraft because of mutual interference. The same constraint of limited number of spacecraft also exists in TDMA, but here the reason is that a large scalable number of spacecraft makes the cycle period of the TDMA sequence too long for data that is broadcasted with high update frequency (especially navigation data, which may be required for real-time data processing). Therefore, both CDMA and TDMA can be used for inter-satellite communications in small scale formations with small satellites.

MA	Advantages	Disadvantages
FDMA	• Multiple S/C transmissions can occur simultaneously • No complex timing	• The larger the number of S/C, the greater the frequency band allocation required for the mission • Multiple S/C cannot share the same transmitter without mutual interference • Frequency isolation between sub-frequency bands is needed to mitigate the mutual interference • Requires complex bandpass filters to separate channels • Increased cost due to frequency variation • Difficult filtering to separate large power signals from adjacent spacecraft • May require power control
TDMA	• Single frequency needed • Multiple S/C can use the same transmitters with high efficiency • Simple timing logic easily separates large numbers of S/C • Prevent interference from other S/C completely	• Inter-satellite communication can only occur at specific time slots • The overall throughput performance is reduced since each S/C must wait their turn to access the shared frequency • Time synchronization is needed • Signal transmission delay varies along with the different separation distances between S/C. Trade-off of the proper time slots needs to be made to avoid signal collision and guarantee efficient channel occupation as well • The greater the number of S/C, the longer the duty cycle of a TDMA sequence that transmits information once among all the S/C
CDMA	• Multiple S/C transmissions can occur simultaneously • GPS-like relative ranging method can operate simultaneously with communication • Relatively immune to transmitter distortion and interference	• Limited number of S/C due to mutual interference (noise generated by undesired S/C signal) • Near-far interference: different separation distances between S/C cause various signal power levels at the receiver • Closed-loop power control scheme may be needed to tightly control transmit power • Less bandwidth efficient than FDMA and TDMA • Complex signal processing needed

Table 2 - 4 – Comparison among MA Technology

As the number of spacecraft grows, hybrid versions involving mixes of these three multiple access techniques are possible. For example, the Crosslink transceiver (CLT) developed by the Applied Physics Laboratory (APL) in Johns Hopkins University considers to apply hybrid FDMA/CDMA for full duplex inter-satellite communications. This combination can support a scalable number of spacecraft and meanwhile achieve sufficient isolation between the transmit signal and receive signal[12].

2.1.4.2. Topology

The communication topology is tightly coupled to the nature of the mission and the number of spacecraft. Canonical topologies generally used for communications within distributed spacecraft systems include centralized (star), fully distributed (mesh), and hierarchical forms.

The centralized formation flying topology consists of a central node and several "supporting" satellites. The central node acts as the control point or data collector point for the formation, since it normally has a stronger data processing capability. Information is passed from the "supporting" spacecraft to the central via ISLs. However, the latter is a single point of failure due to its unique capabilities and its centralized role within the formation. The distributed formation flying topology consists of several spacecraft that have similar capabilities, such as in the MMS mission[9]. This topology of network allows for a more reliable system, since the failure or the temporary unavailability of a single node (whether it is) does not compromise the system working, but only, in case, its performances. A hierarchical topology is the combination of the previous two topologies and divides the formation into manageable subsets. Due to this, it works well for larger formations. A comparison of the different topologies is provided in Table 2-6[12].

Topology	Advantages	Disadvantages	Examples
Centralized (Star)	• Simple design • N-1 links for N nodes;	• Relies on the capability of the central resource, which is responsible for primary control and information dissemination • Potential faults within the central resource greatly influences the implementation of whole mission	DARWIN MAMIX TPF
Distributed (Mash)	• Support direct interaction among all distributed assets • Fault tolerate for each node • Real-time communication to each node	• Topology architecture is N(N-1)/2 for N nodes • Rapid growth in complexity as N increases • Resource limitations such as communication bandwidth and processing capability.	GRACE MMS
Hierarchical	• Robustness is supported • Control structure complexity depends on the functional relationships between S/C	• Needs multilevel approach; • Not necessary for small distributed missions.	

Table 2 - 5 Comparison among Topologies

2.1.5. Conclusions

At this point of the study, we can say that the “ideal” ISL system does not exist, since the choice must take into account mission-specific objectives and system priorities: in fact, a solution that is the best possible for a particular mission objective could not be applicable if a different mission is in phase of study, as the mission requirements could be different; it could happen that a lower data rate is required, but the data latency should be smaller, as the data needs to be exchanged faster among the satellite.

According to the choices regarding the multiple access technique, and the topology of the distributed satellite network, different level of performances can be reached; such performances will be obtained through the commitment of different mass, power and cost budget.

Once the user requirements has been identified, mission requirements must be extrapolated: the characteristics of the inter-satellite link, for example in terms of amount of data to transmit, must be fitted according to mission requirements; thus the technical solution are studied and implemented in order to be consistent with the mission requirements identified.

2.2.Inflatable Solar Panels enabled through use of Ultra-thin solar cells

A fundamental limitation of microsatellite systems is their limited physical apertures, for collection of solar energy (hence power production), for radiation of heat from the satellite (thermal control) and for focusing radio energy transmissions (antenna gain). Many designs and flight experiments have been carried out to test and demonstrate innovative methods to package augmented apertures within small spacecraft envelopes. Among the most promising of these methods are inflatable structures. The realization of large apertures aboard small space platforms has stimulated a number of innovative and high value missions[13,14]. Applications typically associated with inflatable system include:

- Mars Rover solar arrays
- Passive radar antenna arrays
- SAR (Synthetic Aperture Radar) arrays
- Solar concentrators
- Structural members such as booms and trusses
- Impact attenuation devices
- Human habitats
- Sun shields
- Satellite blanket solar arrays

Studies regarding some of these applications have been carried out, and many pre-flight technology demonstrations have been conducted. Among these applications, inflatable solar arrays are probably the most likely to be realized in the near term and surely would be the most fitting application for implementation on board an innovative microsatellite to enhance spacecraft performance and enable new missions for microsatellites. The capabilities that could be most significantly improved via this technology are the specific power and the system cost.

Inflatable structures offer several advantages: they are extremely lightweight and easier to deploy; their deployment systems are more reliable than traditional mechanisms; inflatables enable more flexible and denser packaging before launch.

In conjunction with advancements made in the space inflatable structures technology, the enhancements in the technology of flexible, ultra-thin film solar cells have made the inflatable solar array very attractive for the next generation of spacecraft, and in particular for platforms with very tight mass constraints, like the microsatellites. Combining ultra-thin film solar cells and inflatable structures, lightweight, low cost and small packing volume solar arrays could be further developed. In other words, spacecrafts could potentially have larger solar arrays than traditionally happen for nowadays space system, without the undesirable drawback of increasing in mass and difficulties in storage of rigid array systems once these technologies would be employed.

The enhancements in three different areas could directly contribute to the employment of inflatable solar array technology: I) the advancements in technologies of thin film solar cell (TFSC); II) the development of inflatable materials with correlated rigidization methods and III) the development of controlled deployment mechanisms. In general, space inflatable structures are manufactured with flexible materials (thin films or coated fabrics) that are inflated in orbit in the operative configuration.

Several environmental threats could damage the inflatable structure, being the impact with particles (in particular orbital debris) the most probable one; such threats are obviously dangerous, since damaging the inflatable structures could seriously affect the overall spacecraft performances. So, methods for solving, or at least reducing, the effect of such threats need to be taken into account: in orbit rigidization of inflatable structures could be one of the solutions to extend system lifetime. It's important to clarify, anyway, that methodologies for extending life of inflatable structures are needed only if the space system has long mission duration; for missions lasting some months, typically less than 12-18, the rigidization methods could not be necessary. Anyway, a brief overview of the most promising rigidization methods will be provided

2.2.1. Rigidizable Structures

Space rigidized structures are made of flexible materials that can be rigidized in space through external influence. Rigidizable structures can be manufactured into many different shapes such as toroids, spheres, tubes, etc., which can be designed into various types of structures. The rigidized components are designed for typical operational lifetimes of even to fifteen years without environmental concerns. Over the last forty years many rigidization methods have been investigated and many reliable rigidized structural components have been produced. Some of the most promising rigidization methods are[15]:

- Heat Cured Thermoset Composite Laminates (Thermal Heating)
- Thin-walled Aluminum/polyimide Laminates
- Thermoplastic Composite Laminates (Passive Cooling)
- UV Curable Composite Laminates
- Foam Inflation
- Inflation Gas Reaction Laminates

Of these rigidization methods 'Thermal Heating' and 'Thin-walled Aluminum' are the most promising for space and planetary surface applications respectively.

2.2.1.1. *Thermal Heating Method*

The composite laminate system, which consists of a thermoset matrix resin and a fiber reinforcement such as graphite, is cured or rigidized by heating. The thermoset resin hardens after being heated to a specified temperature and cure time. This rigidization method can be designed to cure from solar energy, or from the spacecraft power, or from a combination of both. The properties of the composite material are consistent with those used in today's spacecraft design.

2.2.1.2. *Thin-walled Aluminum Method*

In this approach of rigidization a laminate is fabricated from Kapton film and ductile aluminum, where the Kapton film is positioned on both sides of the aluminum. To rigidize the cross-section, the structure is inflated to eliminate the wrinkles in the laminate and to a point of just yielding the aluminum. After yielding the aluminum the Kapton/aluminum laminate is ‘rigidized’ and will maintain structural integrity. The inflation gas is then vented to space.

2.2.2. Deployment Technique

Together with rigidization methods, the possibility to successfully implement and use inflatable solar panel depends on the development of deployment techniques that are able to assure a proper and controlled deployment of the inflatable structures.

The reasons for controlled deployment are several:

1. Maintaining, during all the deployment, the system within acceptable boundaries;
2. Improving structural reliability, avoiding inflatable structure to twist other structures;
3. Minimizing shock and impulse during deployment, even the smaller ones, as they can affect the attitude of the spacecraft.

A certain number of techniques for safely deploying inflatable structures have been studied, and device for any of this technique developed; among these, two devices look to be particularly adequate for application of inflatable solar array: the roll-up device and the columnation device. Such devices and their functioning principles will be described in the following; for the sake of completion, we briefly cite also different devices, that will not be described more in detail:

- Internal compartmentalization
- Break cords
- Becket loops

2.2.2.1. Roll-up Devices

This first method for deploying inflatable structures is one of the most widespread approach utilized in tubes, i.e. the roll-up method: it consists in a rolled inflatable tube that is deployed through a mechanism that controls its rate of unrolling when the inflation gas in introduced. This method resemble the one used for the party favor, that unwinds when someone blow in it.

This method is characterized by two classes of devices devoted to control the rate of unrolling: the former are devices embedded in the tube, and the latter devices mounted at the end of the tube. The first category comprises internally as well as externally embedded mechanism; an example of internal mechanisms is the constant force spring, mounted longitudinally to the tube, while external mechanisms can be represented by longitudinally stuck Velcro strips. Introduction of inflating gas would unroll the spring, or separate the flexible material from the Velcro, allowing for the structure to inflate at a controlled rate. Each system provides a proper, evaluable resistance, which imposes the internal pressure in the inflatable tube; this, in turns, set the value of the of the beam stiffness during the deployment, that varies according to the completion of the inflation.

The second class of deployment control method is characterized by the use of a torque mechanism, placed at the end of the tube: such mechanism give stiffness to the “unrolling” beam, until complete deployment is accomplished.

The roll-up approach is really compact, reliable and does not envisage particular difficulties or limitations in packaging procedure.

2.2.2.2. Columnation Devices

The functioning principles of these devices are shown in the following figure

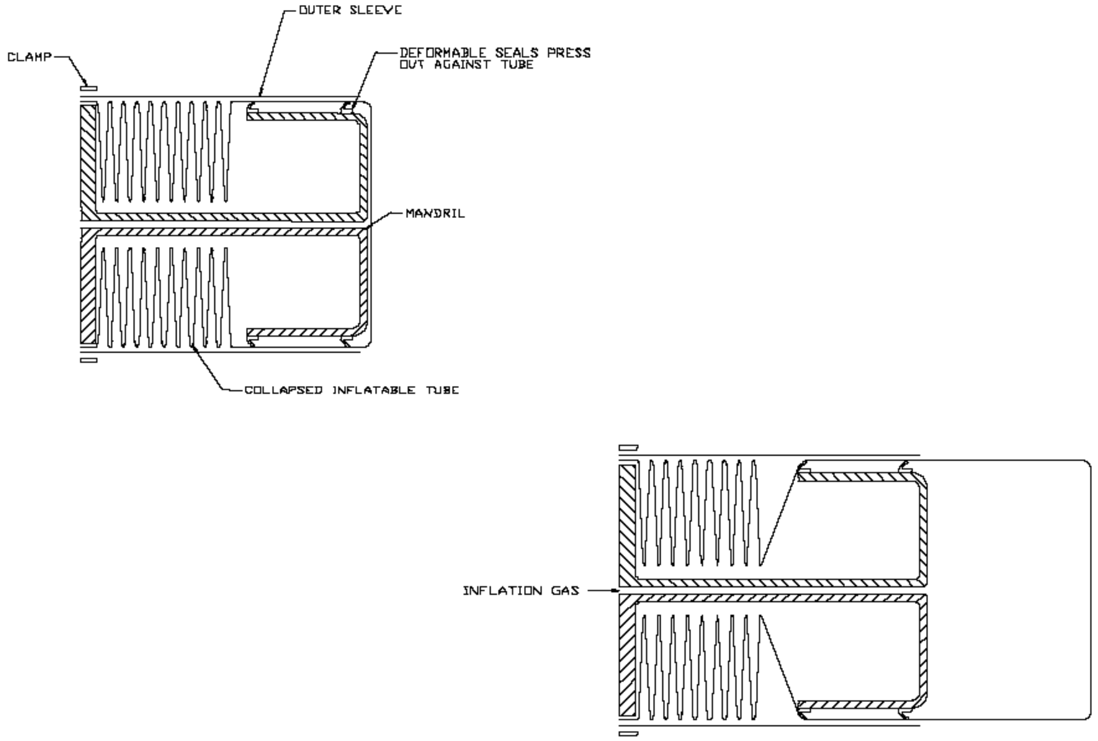


Figure 2 - 2 – Columnation Device

Such a method offers a deployment mechanism that axially extends the folded flexible material, in a way similar to what happen when a telescopic boom is deployed. The flexible tube is folded in a supporting structure, a mandrel, with one end of the tube clamped to this mandrel, while the other free to be pushed away by the inflating gas. Such gas is introduced through the mandrel into the free end of the tube; as this gas is injected into the tube, it tends to extract the folded flexible material, overcoming also the small friction that properly installed seals (as shown in the previous figure) create for avoiding unwanted deployment. Such method guarantees also a certain structural stiffness during deployment.

With respect to the previous method, this one introduces a supporting structure, which has its own mass and volume: several variations of this device envisage the use of inflatable, collapsible mandrel, which allow the overall system to fit into smaller volume.

2.2.3. Thin-Film Solar Cell

A Thin-Film Solar Cell (TFSC) is a solar photovoltaic which is manufactured through the deposition of one or more thin layers (thin film) of the photovoltaic material onto a substrate; the photovoltaic materials used are exceptionally thinner than both mono- and multi- crystalline solar cells. TFSCs are made up of layers with a thickness of around 10 nanometers, extremely small in comparison to the 200 to 300 nanometers layers found in crystalline silicon cells, or the 150 micrometer[16,17] of the triple junction Gallium Arsenide solar cells. They are also extremely lightweight, providing an excellent opportunity for the development of innovative systems for generating the necessary power on board small space platforms, while being also extremely flexible, as the successive figure incontrovertibly shows.

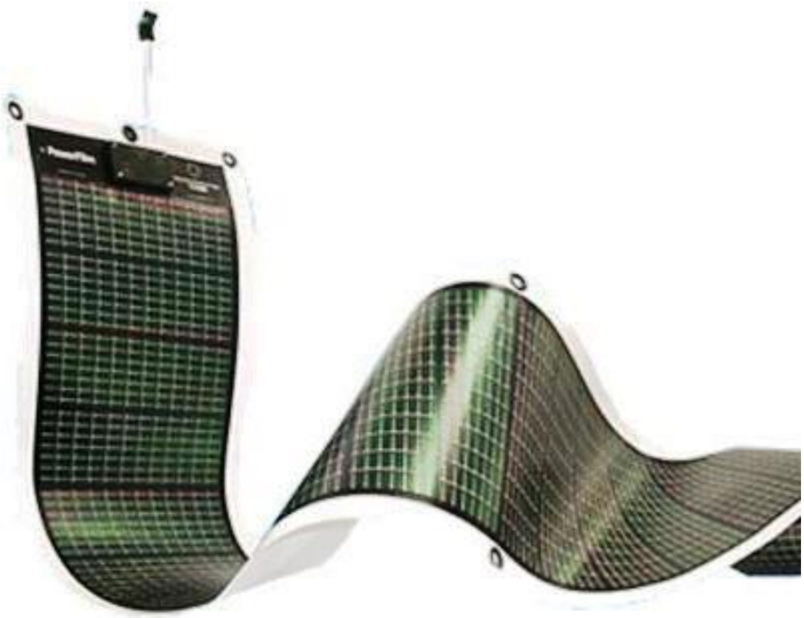


Figure 2 - 3 – Thin film flexible solar cell

The semiconductor junctions form of TFSC is of various types: in the case of amorphous silicon, it is in the form of p-i-n junction, which is a particular junction where between the positively (p) and negatively (n) doped region, there is an intrinsic (i, or undoped) layer sandwiched; in the case of copper indium gallium selenide (CIGS) and CdTe (Cadmium telluride) panels, it is in the form of hetero junction (that is the interface between two layers or regions of dissimilar crystalline semiconductors, that have unequal band gaps, and is the opposite of a homojunction). The advantages of using such thin film solar cells, in place of traditional crystalline cells, are numerous, the most important of which are:

- Versatility: thin film coating can be applied on almost any surface including plastic, metal or paper;
- Flexibility: While silicon- or gallium/arsenide- based solar cells are rigid and hence brittle, Thin Film Solar Cells can be deposited on elastic substrate material. The flexibility of thin films is based upon how they are installed and not in terms of how they will be used;
- Lightweight: such technology gives a tremendous advantage in term of mass of solar cell; a triple junction GaAs solar cell has a specific weight of about 0.85 Kg/m^2[16], while thin film solar cell have a specific weight of about 0.15 Kg/m^2 (such values are referred to active layers and the relative substrate, not to the only active materials)
- Increased specific power: according to the technology analyzed, the increments of specific power that characterize the TFSC with respect to traditional solar cell are significant, up to one order of magnitude.

The probably most important difference, from the point of view of the design of a space platform, deserves a particular underlining: such materials are intrinsically radiation tolerant: having a non-crystalline structure, the high-energy particle that impact the solar array does not affect in any way the performance of the solar cell, differently from what happen for traditional solar arrays. In fact, in crystalline solar cells, the particles hitting the cell damage the cell itself, creating gap in the doped region and deteriorating the crystalline structure, making, this way, more difficult the movements of electrons inside the cell.

The photovoltaic materials used to manufacture TFSC are different, and, for any of these, different deposition methods can be used. Thin-film solar cells are categorized according to the photovoltaic material:

- Amorphous silicon (a-Si) and other thin-film silicon (TF-Si)
- Cadmium Telluride (CdTe)
- Copper indium gallium selenide (CIS or CIGS)
- Dye-sensitized solar cell (DSC) and other organic solar cells

Among all these different technologies, the one who guarantees the best performance, at the date of writing, is the CIGS (Copper Indium Gallium Selenide). Using such technology, researcher recently reached 19.9 percent efficiency[18,19], setting a new world record for this type of cell. CIGS cells use extremely thin layers of semiconductor material applied to a low-cost backing such as glass, flexible metallic foils, high-temperature polymers or stainless steel sheets. Future improvement of such a value would only make this technology more and more tempting for future space mission.

Anyway, it must be noted that any of the different materials used for manufacturing flexible solar cells cannot guarantee the high efficiency in energy conversion that a traditional solar array, made with mono- and multi- crystalline solar cells, is able to provide. Recent studies demonstrated how triple junction solar cells reached the incredible value of efficiency of 43.5%[20]. This consideration does not compromise the utility of the TFSC, as such technologies, even if do not guarantee this value of efficiency, reveals themselves to be more efficient if considering specific power: triple junction (TJ) Gallium Arsenide (GaAs) solar cell specific power are around the 450 W/kg[21] (if no structural support are considered), while CIGS are around 3300 W/kg, under the same hypothesis[22]. Considering all the support structure, necessary in both cases for having a valuable power generator system, the specific power for TJ GaAs drops to about 100 W/Kg, while that of CIGS to about 250 W/Kg.

2.2.4. Inflatable Solar Array

The use of inflatable technology coupled with ultra-thin, flexible solar cells enables some breakthrough advantages with respect to traditional solar arrays; some of these advantages are the reduced stowage volume and mass, the increased specific power (as already said, 250 W/Kg against the about 100 W/Kg which can be obtained with the most advanced technologies for crystalline solar cell), and the reduced cost over the existing mechanically deployed solar array. The combination of these technologies is highly attractive for all missions limited by shortage of available power, both for large platforms that have to deal with launch vehicle envelope restrictions, and for small platforms that cannot exceed stringent mass and volume budgets.

Such combination can help reducing the envelope of the power-generating system, providing in a small volume (when in stowed configuration) the potential for providing high quantity of electrical energy.

These types of systems have been studied for many years, as evidenced by many papers[23,24,25] and satellite projects[26]. These studies and projects, however, had to deal with an important disadvantage that largely obstructed the success of such a technological solution: the very low level of energy conversion efficiency that flexible solar cell succeeded in achieving.

In the middle of the 1990's, the ratio of incoming energy converted in electricity, was very low, approximately on the order of 5-7%, achieving, only in case of successive advancements, the value of about 9%[27].

This means that, in order to have a flexible solar array that could, at least, produce the same quantity of electrical energy, flexible solar wing should be three, or even four times larger than the corresponding crystalline solar array.

Nowadays, instead, the energy conversion ratio of flexible solar cell has been dramatically increased, reaching the values of 20%, which is, about, the two-third of the conversion factor of the most efficient crystalline Triple Junction Gallium Arsenide solar cell for space application.

2.2.5. Hypothetical system

In the following, a hypothetical system for exploiting the advantages of flexible solar cell will be presented. The basic configuration of the system proposed consists in:

- Solar array blanket;
- Supporting structure;
- Deployment system;
- Inflation system.

The system, in its deployed condition, will appear as shown in the following figure.

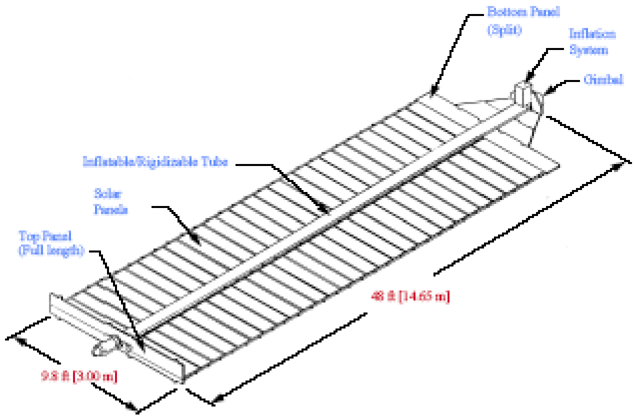


Figure 2 - 4 – Hypothetical system

The system is mainly composed of a deployment tube, which is in the middle of the wing; during its deployment, that occurs through inflation gas, it will unroll the solar blanket, that has been properly packaged in the stowed condition.

Let's see in more detail how the system is conceived.

2.2.5.1. Solar array blanket

The solar arrays are characterized by a blanket of flexible solar cells, split in the middle by the deployment tube. When the system is stowed, the blanket is bended like an accordion. The technology of solar cell we decided to use for the system does not affect the system itself but, for the sake of completion, the CIGS TFSC developed by NRL has been chosen as those used. If successive development will lead to lighter, more economic or more efficient technology, the same system can be used by simply replacing the solar blanket array. The blanket will be composed of a large number of TFSC placed side by side, with a packaging factor of approximately 0.82[19]. Even if a single cell could be created, differently from the case of crystalline solar cells, with the possibility to significantly improve the packaging factor, there are several drawbacks in doing so: the primary is technical, in fact a similar cell will have an high value of current, but a very low voltage, making difficult to manage this issue; even solving such a problem, a single shunt could have a dramatic impact, destroying the only cell, and thus the entire module.

Thin film module are typically made by series connected solar cell stripes, large about 1 cm

2.2.5.2. Supporting Structure

This subsystem comprises all the mechanical system, like stowage panels, launch ties, and tie release mechanisms, together with the inflatable beam. The inflatable beam has the task to provide a deployment mechanism, and, once the system reach the deployed condition, to provide support for the solar array, carrying out the role that in traditional solar wing are executed by truss masts.

2.2.5.3. Deployment System

The roll-up device and its embedded mechanism for controlling the unrolling are the primary system of the deployment system. When the inflation system starts releasing gas, it is piped to the base of the tube, which initiate inflating and unfurling. The torque mechanism gives the necessary resistance to unrolling action, allowing the tube to exhibit the necessary pressure, that is kept uniform. Such pressure assures the beam to have certain stiffness during the deployment; the torque mechanism is attached to the top panel, which deploys the solar array blanket.

2.2.5.4. Inflation system

Systems devoted to space inflatable structures are of different types: there are systems that uses compressed gas, systems characterized by a gas generator, or system which sublimate powder and liquid into gases. Each of these systems has its own pros and cons; the simplest system is, of course, those involving compressed gas. As this latter system, in the last years, became very reliable and with a small value of mass, it represents the best suitable option for an inflation system of a hypothetical flexible solar array for a microsatellite.

2.3.On Board Computer

The On Board Computer (OBC) is the nerve center of a space vehicle: it is the central intelligence of the system, the subsystem devoted to management and operation the satellite, providing the necessary commands to the different subsystems, executing functions.

In order to try to lower the cost of this subsystem, some options are available: the use of automotive electronics is a possibility[28].Such typology of electronics are subjected to screening processes and a large number of tests, well beyond those of typical industrial electronics: such devices are, in fact, field-tested in extremely severe environments of temperature, vibration, shock, duration, and electrical fault, as these systems are related to humans and their safety. These devices are typically not tested for vacuum or radiation properties, though the vacuum testing is generally performed at the board level, being any way most valuable at this higher level.

When computing systems, traditionally used for a terrestrial application, have to be implemented into space system, the incorporation of EDAC (Error Detection And Correction) on its memory, together with other state-bearing systems, is of outmost importance. The most frequent effect of the radiation typical of the space environment on a computing system is SEU (Single-Event Upset), where a single bit in a memory device, within the microprocessor or elsewhere, is changed in state. Even in a LEO, relatively benign orbit this can happen several times per orbit, and, for this reason, is worthy of attention; fortunately, simple algorithms are extremely common for detecting and correcting these errors, so the system is able to operate. For most critical registers, triple-voting logic is used to guard against similar errors. Since there will likely be some critical registers which cannot be SEU-protected, the other major architectural difference between space and terrestrial systems is the incorporation of more extensive watchdog and self-resetting capability.

One particular type of self-governance is a sensitive over-current monitor, which power-cycles the computer not only in case a short circuit occurs during ground test, but in the operational case where radiation has caused a Single-Event Latchup (SEL). An SEL is an ion channel to ground within a semiconductor, and shows itself as a sudden jump in current draw. SEL protection circuits must be designed to detect this increase in any single component, without confusing normal power transients from SEL events. This may call for multiple SEL protection circuits covering small portions of the overall circuit, or a detection system that anticipates known current increases and trips only when the increase was not planned.

In terms of watchdogs, a typical space computing system will have a software watchdog to ensure that all critical tasks are acting correctly, a microprocessor watchdog to ensure that the device itself is acting correctly, a subsystem watchdog (housed outside the computer itself, typically in the power system) to ensure that the computer as a whole is acting correctly, and a mission watchdog (serviceable only by ground contact) to ensure that the ground controllers agree that the spacecraft is performing as required.

The abovementioned capabilities are those traditionally required to a space-specific OBC system and they are rarely incorporated into terrestrial data management systems and their devices. However, systems using space-qualified components incorporating these space-specific features are exceptionally expensive, but will satisfy the requirements for quality and space-reliability of the system and its internal devices. Terrestrial industrial systems have a very agreeable price, but will face very significant SEU problems if used in the mission-critical role of flight computer. The solution could be found in the middle, where terrestrial (eventually mass-produced) systems can be assembled with architecture typical of space application.

It is evident from the above discussions that the primary advantage of industrial electronics stems from the fact that they are mass-produced. In fact, the effects of mass production are widespread and almost entirely positive, and as such deserve particular attention.

A mass-produced device takes advantage of economies of production scale to reduce cost, as described above. It also tends to encourage improvement of production process, as the deservedly praised "6-Sigma" efforts at many companies demonstrate[30]. In short, a semiconductor device rolling off a modern production line has a vanishingly small probability of being faulty, or conversely, such a device has an exceptionally high probability of performing exactly as its specifications describe. This in turn brings up another advantage of mass production: availability. Such devices are produced, warehoused, and available for immediate distribution. Lead times can be 24 hours or less. Compare this with a space-qualified device which may be produced specifically for the given contract, with a lead time of 6 months or more. There are certainly exceptions in both directions but the general rule is *mass-produced parts arrive faster*.

Mass-produced components are also able to demonstrate their reliability.

Let's make an example. To achieve an acceptably high ensemble reliability (say, >95%) in a typical spacecraft, the components much be much higher reliability. A spacecraft with 1000 electronic components must have better than e^[(ln 0.95)/1000] = 99.995% reliability in each component, before even beginning to count their interrelations (which is, as noted above for new designs, where the true failure probability occurs). In order to have even a 90% Confidence Level – 90-95% is typical – that a given component has 99.995% reliability, 46,052 devices must have been tested with zero failures in an environment similar to that operative, and for a duration similar to that prescribed. This is not particularly impressive for in industrial component that has a few million devices in the field, but is far beyond the entire production run over the entire history of most space-specific components.

2.3.1. System Description

The difference between industrial and space-devoted components is that for the latter, we are relying on a small sample population put through rigorous and expensive tests for limited duration, and, using the result of these tests, we affirm that this device is – essentially – reliable, and will be capable of satisfying mission requirements; on the contrary a typical industrial component, for which there is a plentiful sample population, are put through a range and extent of real-world tests for extended durations, and from the results, with a mathematical formulas, reliability are deduced. So, due to the low volumes involved, space-specific components must use qualitative arguments of analogy (particularly, that semiconductor fabrication tends to produce consistent devices) in order to present acceptable reliability figures, whereas terrestrial components offer quantitative data from which fully quantitative reliability figures can be derived.

At this point a caveat is needed: all of these statements about the quantitative comparison and reliability demonstrated hold true only in the environment where the sample population has been tested; very few industrial components are used or tested in vacuum, or in the harsh radiative environment typical of space missions. The severe space environment, extremely more aggressive that typical terrestrial ones, is the reason why the satellite developers tend to rely exclusively on space-qualified equipment.

In essence, the satellite developers have not to look for reliable industrial components – the reliability of these systems has already been proven – but whether the components which environment they use to face during tests and normal operations is comparable with space environment, and, to the extent these environments differ, what is the impact of such differences on the behavior of the devices.

So, the mission developers would now be asking the question: Of the blanket category “industrial electronics”, how many are actually used in environments comparable to space one? A logic gate from an industrial electronics manufacturer that is tested to industrial standards but which has never been used in more than a room-temperature application loses almost all of the strength of the previous section’s arguments. It would be monotonous and without any sense to call each potential manufacturer and get a list of customers, and from there contact the customers to see how the particular device is being used. More efficient is to work backwards from a well-known-comparable end use, and see what components are used in that application.

Components traditionally used for aeronautical application could be good candidate to satisfy some of the typical environment that space devices have to deal with. But in this paragraph we will focus on a different application: an impressively high-fidelity end use is automobile electronic systems. Cars, trucks, Sport Utility Vehicles, and every other variation of wheeled vehicle in modern times mount huge quantities of electronics devices: microprocessors, sensors, electromechanical actuators, passive components, active components, analog, digital, and so on. They must survive unattended for upwards of ten years, and in the unlikely event of a failure, must be replaceable for as long. They are likely to experience such electrical input anomalies as jump-starting, incorrect wiring, or other extremes or lacks of voltage as may occur in the normal course of mistake or misuse of an automobile. They operate between the 100°C engine and the –30°C typical of certain cold region. Drivers will not tolerate a failure in any of these systems, and accordingly the systems must fail with exceeding rarity. They has been tested to operate when offroading in a 4x4,with vibration and shock in all axes, for much longer than the few minutes of a satellite’s launch.

The above statements are just examples, but the reality is really impressive: automotive electronics are expected to survive in thermal, mechanical, and electrical conditions that match or exceed those experienced in a normal orbital mission, and to survive in such conditions for years on and without error or anomaly. The only factors missing are vacuum and radiation, as indicated earlier. Specifically, the difference between an automotive electronics test regime (to give confidence in a component to survive in automotive usage) and a MILSTD-883B test regime (to give confidence in a component to survive in aerospace usage) are as follows.

Test Condition	**MIL-STD-883 requirement IAW[3] MIL-PFR-38535**	**NASA GSFC 311-INST requirement**	**Automotive requirement**
Visual Inspection / physical size compliance	Inspection at 7-10X magnification for no visible defects	Inspection at 7-10X magnification for no visible defects	Inspection for no visible defects

Soldering heat resistance	Steam age 8 hours; bake 100°C 1 hour; solder test by dip in solder pot 245°C ±5C, 5 seconds	Steam age 8 hours; bake 100°C 1 hour; solder test by dip in solder pot 245°C ±5C, 5 seconds	Moisture 30°C 70%RH 168 hours: bake 125°C 24 hours; solder test by infrared reflow 235°C ±5C, 10 seconds
Thermal shock	-55° to 125°C, 15 cycles, 10 second transfer time, 2 minutes dwell	Not required	-65° to 150°C, 500 cycles, 10 minutes cycles
Particle Impact Noise Detection	Not required	3 1,000g shock, 3 second 20g 40-250 Hz vibration, repeated 4 times	Not required
High Temperature baking	IAW device specification	IAW device specification	150°C, 2000 hours
Span of life	125°C, 1000 hours, all Vcc's at max rated voltage	Max junction temperature, 1000 hours, all Vcc's at max rate voltage, performing operational test	150°C, 2000 hours Vcc5V= 6.5V Vcc3.3V= 4.2V, performing operational test
Lead Integrity	0.229kg, 90° bend, 3 times	0.229kg, 90° bend, 3 times	0.229kg, 90° bend, 3 times
Salt atmosphere	35°C, 95%RH, 24 hours, 20,000 to 50,000mg/m^2 NaCl deposited	Not required	Required but details unavailable
Temperature / humidity	130°C, 85%RH, 100 hours	Not required	85°C, 85%RH, 2000 hours, Vcc5V= 5.5V Vcc3.3V= 3.6V, performing operational test
Solvent resistance	1 minute in: alcohol solvent at 25°C; organic solvent at suitable temperature: ethyl inorganic solvent at 70°C	1 minute in: alcohol solvent at 25°C; organic solvent at suitable temperature: ethyl inorganic solvent at 70°C	Not required
Autoclave (PCT)	121°C, 100%RH, 2 atm, 96 hours, unbiased	Not required	130°C, 85%RH, 2000 hours, Vcc5V= 5.5V Vcc3.3V= 3.6V
Thermal cycling	-65° to 150°C, 100 cycles, 10 minute dwell	-65° to 150°C, 100 cycles, 10 minute dwell	-65° to 150°C, 1000 cycles, 1 hour/cycle
Static electric breakdown (Human body model)	C=100pF, R=1.5k, 3 samples; voltage breakdown range required	Not required	C=100pF, R=1.5k, 5 samples; voltage breakdown level recorded
Static electric breakdown (Machine model)	Not required	Not required	C=200pF, R=200, 5 samples; voltage breakdown level recorded

Table 2 - 6 – Test condition

The case has now been made for the high – indeed extremely high – reliability of automotive electronics in all realms save two: vacuum and radiation. These will be addressed in turn.

The problem of vacuum is dual: material degradation, and thermal runaway. Material degradation stems from the concern that plastic integrated circuit packages will outgas in vacuum; this problem must be accounted for, as particles can interfere with optics and other sensitive equipment. Degradation can be overcome by one of two simple approaches: using modern plastic IC packages, that don't outgas in vacuum at a notable rate, or, alternatively, coating the circuit board in a urethane derivative that decidedly will not outgas. Thermal runaway is an issue present whether the unit is operated in space or on Earth: it is solved ensuring a path (conductive or radiative in the orbital case) to dump heat out, in order to keep the temperature within the manufacturer's specifications. The IC itself works similarly whether its case temperature is being maintained by conduction and radiation alone, or by convection as well: an automotive fuel injection microprocessor placed near the manifold whose case hovers at 80°C, and a satellite flight computer microprocessor in sunlight whose case hovers at 80°C, will behave entirely the same. It is simply needed a path to expel heat.

The second difference is more difficult to deal with: the radiation is a difficult issue to face, for the nature itself of the electronic systems that a spacecraft carries on board; the counteractions are limited, and none of them is a panacea solving all the problems.

It is worthwhile here to explore the effects of radiation on electronic devices, as possible solution can be found only if the roots of the problems are clear:

- *Bulk charging*– It occurs because ionic radiation, particularly electrons, can be absorbed into metallic structures, giving those structures a (negative) charge. If the structure is suitably shaped such that a charge can build up on a piece of metal near another, uncharged, metallic surface, then this charge can suddenly dissipate, through a spark, which can cause damage to nearby electric circuits. This is a spacecraft system issue, and generally not a bothersome one, unless the vehicle is oddly shaped or poorly grounded. This is not an issue for spacecraft subsystems such as flight computers.
- *Single-Event Upsets (SEUs)*– They occur when a state-holding electronic is physically hit by radiation and changes state, from 0 to 1 or 1 to 0. Any of the radiation forms (photonic or ionic) can cause an SEU, but electrons and photons are the usual culprits. Obviously, this is extremely relevant to a spacecraft computing system, but SEUs are generally easily handled, by detecting and correcting a register state change, or by detecting and recovering from a functional failure resulting from an SEU.
- *Single-Event Latchups (SELs)*– They are short-circuits within a semiconductor caused by a radiation impact. They are temporary conditions which end when power is removed from the part. Power can then be re-applied safely and the short-circuit will not recur. It must be noted that SELs are seen far more often in the laboratory than in space, and not very often in the laboratory. However, it is still a good and responsible idea to design short-circuit protection into semiconductor circuits, both to advertise that a product has latch-up protection, and to guard against other more routine short-circuit conditions such as metal accidentally touching the board during testing.
- *Total Ionizing Dose (TID)*– It represent the total amount of radiation that a semiconductor device is able to tolerate before its definitive death. It cannot be easily worked around or dealt with. The most problematic aspect is that it can only be delayed, but never entirely eliminated. TID is in practice brought about primarily by protons, which cause the unfixable physical damage to the semiconductor.

Bulk charging is not a driving subsystem issue, and SEUs and SELs can be handled architecturally. TID is then the issue to worry about. It is at this point that the judgment of the mission manager comes completely to the foreground.

In conclusion, vacuum is handled by standard material coating and thermal design techniques; radiation is handled by cognizant architecture of the electronic subsystem, plus a dose of judgment on the part of the satellite designers: a six-month mission on a 500 km orbit need not worry at all about TID; a 15-year mission above 1200km altitude must worry about it; and in between there are plenty of situations to evaluate, and many trade-off to accomplish between lead time and component TID tolerance.

Since it would be wise to apply the principle of “Produce, don’t redesign” to this system as surely as it does to semiconductor components, this computer is created to support up to two generic Input/Output cards for mission-specific needs such as bus interface, line drivers, or mass memory. The FPGA routes the needed signals - parallel or serial ports for device interface, address/data lines for memory – to each of the two I/O card connectors, so that a single OBC design can be produced identically for a multitude of different missions and applications. The SH7055 also natively supports the Controller Area Network (CAN) bus, for peripherals or other computers which can share this bus for vehicle control and data exchange.

A hypothesis on how could appear the On Board Computer using the abovementioned methodologies is illustrated in the following figure:

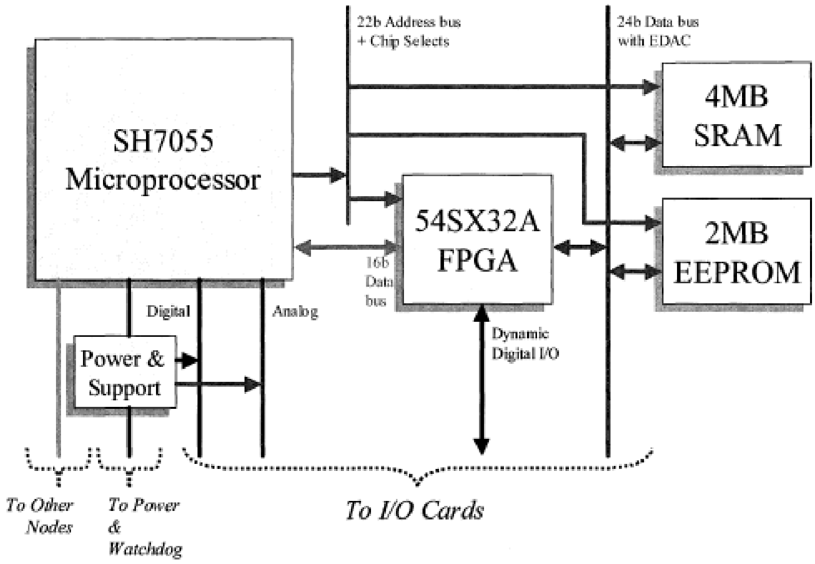


Figure 2 - 5 - OBC

A typical industrial component will typical survive 2 to 10 krad[29], with the denser products (microprocessors, gate arrays) at the low end, and large-feature products (operational amplifiers, low speed discreet logic gates) at the high end. With a very few identifiable exceptions, of which a flight computer is not one, no spacecraft integrated circuit is going to see unencumbered, fully exposed radiation of any sort. Any given incoming ray of radiation will have to pass through several millimeters if not centimeters of material, generally metal, and this shielding will tend to absorb, deflect, or otherwise impede incoming radiation before it ever has a chance to impact the 1cm squares of Silicon which are the integrated circuits inside.

Generally, a conservative hypothesis is to suppose that any of the electronic components has approximately 1 cm spherical protection; this consideration, coupled with the decreased amount of incoming radiation that a 1mm protection offers, is enough to suppose that, for the purpose of the missions we will propose in this thesis, terrestrial components, properly shielded through specific “metallic coating”, or through architectural choices, for example, placing the most critical components in a specific position, bordered by less critical components, acting like shields.

But radiation hitting shielding can generate Bremsstrahlung radiation, which goes on to hit the electronics inside the shielding. This is a real issue, and must be considered: the effect of the Bremsstrahlung radiation is typically generating high-energy photons (usually X-rays), in the place of protons. Anyway, shielding is a decidedly advantageous trade-off, since the former generally are the source of functionally tolerable SEUs, but the latter are the cause for the irreversible TID.

2.3.2. Conclusion

At the end of this discussion, we can now state that there do exist many space-appropriate components developed for and devoted to terrestrial applications, proven to operate in an environment that is as harsh as the orbital environment, but with some notable exceptions. One example of this typology of components are the automotive components, that can be used to manufacture the on board computer of a space system; one of the reason for using terrestrial components is the mass-production: since these components are mass-produced, their costs can be kept lower than costs of space-specific components, that are generally mission-specific developed and produced, leading to a significantly higher expense.

Differently from space-specific components, terrestrial components are not tested for any of the environment present in space: for the environment where the terrestrial components has not been tested, counteractions to contrast the effect must be undertaken.

For automotive components, vacuum and radiation are the environment in which the components have not proven their ability to operate, thus solutions needs to be undertaken. In the previous, such solution has been described.

Of course, all the discussions made above are valid, and the choice of using terrestrial components represent a good compromise, only in the case of specific missions: a 10-year mission, at an orbital height of several thousands of kilometers, crossing the inner Van Allen radiation belts, will likely not complete its mission if terrestrial components are used. In such a case, even rad-tolerant equipment must be neglected, and the use of rad-hard, space-specific components is unavoidable.

But, as said in the conclusion of the first chapter, even if the central concepts of the study are inevitably introduced and discussed in different paragraphs, the study must be considered as an integrated effort, where the contemporaneous presence of the principal actors are necessary to justify the implementation of such capabilities. And this integrated study envisage the implementation of innovative technologies into a microsatellite, limiting the range of application (and, consequently, the variety of problems) that such a platform entails.

Automotive is not the only field of engineering in which looking for potential industrial components; in the following paragraphs, other equipment, generally devoted to terrestrial application, will be presented, and innovative solutions proposed for improving the capability of microsatellite.

2.4.Optical link for downloading data

Spacecraft recently launched must accommodate a large increase in the capability of earth observation (EO) sensors flown. State-of-the-art payloads like high resolution optical or infrared cameras or SAR systems produce data at a rate of gigabits per second. Such a rate of data acquisition is significantly higher than the down-linking data-rate, so this aspect became the bottleneck in EO-systems since many years. Furthermore, the data acquired by the sensor can only be downloaded when the satellite is in the proximity of a cooperating ground station. Accordingly to number of cooperating ground stations, their placement and the satellite orbit, the visibility occurs for a limited period of time per day, and each downlink session lasting no more than around 9 minutes for a LEO (Low Earth Orbit) satellite. These conditions significantly limit the effective operational time fraction of such sensors to a really short duty cycle per day. In addition, due to limitations in the available frequency bands and in technological feasibility, the RF downlink technology is currently reaching its limits.

A solution to this bottleneck could be provided by the use of using optical free space high speed links: this innovative technology could multiply the downlink data rate by an order of magnitude, providing the opportunity of even faster links in the near future. Another advantage of optical communication system is the reduced budget required in terms of mass, size and power consumption with respect to traditional Radio Frequency system, being a really fascinating solution also for micro-satellite; furthermore, the size of the corresponding optical ground station, is extremely compact, enabling portable mobile station with only some decimeters of telescope diameter requested. This represents another essential benefit if compared to RF ground stations, which are typically characterized by antenna diameters of 5 meters and more.

Such system has, however, has certain definite drawbacks: the most evident and troublesome of them is the blockage of signals due to cloud coverage: in fact, even with a minimum fraction of cloud coverage, the optical signal is blocked.

In addition, a proper wavelength for optical system has to be chosen, paying also attention to the atmosphere absorption window and the interaction, in general, between molecule present in the atmosphere and the signal wavelength; in fact, even with the complete absence of cloud, molecule present in the atmosphere can absorb, reflect or scatter particular wavelength, reducing (up to the complete elimination) the signal crossing the atmosphere. The following graph shows the percentage of transmitted signal for wavelength from the visible up to the Far Infrared. Even if a not complete absorption occurs, it must be borne in mind that also a noteworthy percentage of signal cannot cross the atmosphere.

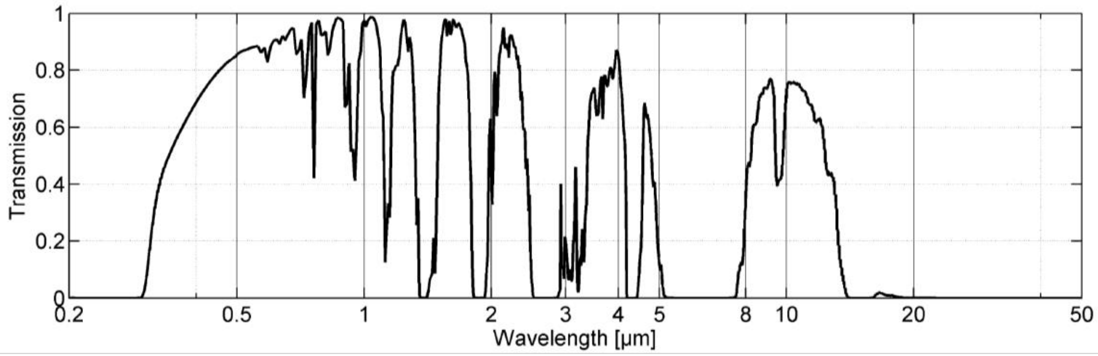


Figure 2 - 6 – Transmission factor vs. Wavelength

This latter limitation can be avoided by carefully analyzing the characteristic of the wavelengths, not only its transmission percentage, but also index-of-refraction that, at particular angle, and according to the wavelength chosen, can affect the transmission.

Despite these obstacles, the feasibility of direct optical LEO-downlink has been already demonstrated, exhibiting very good performance with measured bit error rates down to 10^{-6} with a transportable and inexpensive optical ground station[31].

2.4.1. Cloud Blockage

Once absorption, refraction and scattering by the atmosphere have been addressed, the major issue facing such a system will be blockage by clouds. Reliable optical downlinks should be limited to geographical sites characterized by a low fraction of cloud coverage, as the optical signal is blocked by thick water clouds. Therefore, these stations should preferably be situated on mountain tops (like the classical astronomical observation site) or in countries with low cloud probability, like the Mediterranean or sub tropic latitudes. A certain number of facilities located at equatorial latitude can assure a good probability of orbiting above a clear sky allowing the possibility to download.

Of course, this practice might not be acceptable for Earth Observation applications like those requiring an as fast as possible downlink of data acquired. Moreover, for non-EO applications like communications or broadcast, a nearly hundred percent availability is required for the satellite link. Therefore, an optimal "ground" station should be positioned above the clouds. Aircraft or aerostatic High Altitude Optical Stations (HAOSs) can provide a suitable solution for the problem, with the later having the advantage of stationarity together with lesser vibrations and position uncertainty. The final "last mile" to the ground can then be bridged by standard RF point-to-point links as used today for terrestrial applications. With a buffering strategy onboard the HAOS even optical downlinks from the HAOS to a terrestrial optical ground station could be used: it is possible, in fact, storing the data during cloud blockage, while re-transmit them when possible.

The implementation of high altitude receiving station will provide also an extended visibility time of the LEO satellite when compared with RF and optical ground stations: in fact, the link can already start at negative elevation angles as long as the line-of-sight stays above the maximum cloud altitude of about 13km.

2.4.2. Functional Hypothesis

In the following, a hypothetical mission scenario, with a payload producing an enormous amount of data, will be presented; such scenario will be investigated and the maximum payload duty cycle will be evaluated, considering different options for the data downloading.

2.4.2.1. *Earth Observation Scenario*

The system used in this case study consists of a satellite with a mean orbit height of 500 km, equipped with a payload (a high resolution camera, for example) which produces a large quantity of data at data rate of 6.7 Gbit/s. This high data rate suggests that the down link capacity of actual communication systems is a limiting factor for the operational time of the payload. For simplicity in calculations, there is only one receiving ground station for the data downlink, and I've decided to locate this ground station at Fucino, Italy.

2.4.2.2. *State of the Art RF Downlink*

Currently used RF downlinks have an effective user data rate of up to 262Mbit/s (e.g. TerraSAR-X). Using this RF downlink, it is possible to start data transmission at an elevation angle of 5° of the satellite. This results in a mean daily contact time of 2668s, which turns into a maximum transferable data volume of about 83 GByte per day. In this case, to underline how RF communication link is a significant bottleneck both for future, and also present application, a ground station availability for downlink of 100% of time will be used for calculation; in such a condition, the transferable data volume per year is approximately 32 TByte, and the camera can be used during the 0.12 % of the operational time of the satellite.

2.4.2.3. *Solving the data-rate download limitation: Optical Ground Station*

The innovative technology of the optical communication can be implement to partially solve the strong limitation of satellite downlink capability that affect the operational time of high data volume payload. We will imagine settling a ground station for optical communication system at Fucino, in order to better compare this technological innovation with conventional RF option.

In case of optical ground station, due to atmospheric effects, the availability of a profitable optical down link is limited to higher elevation angle w.r.t RF downlink, equal to 10° and more. This condition reduces the satellite-GS Field-of-View, resulting in a mean shorter mean daily contact time of 1650s. Nevertheless, using an optical downlink with a data rate of 5Gbit/s, the resulting transferable data volume per day is 1009 GByte per day when neglecting cloud blockage. The down link station at Fucino is a rather favorable place for an optical ground station because it has a mean availability (limited by cloud cover) between 40% to 50%[32], depending on the season, as the next figure witnesses.

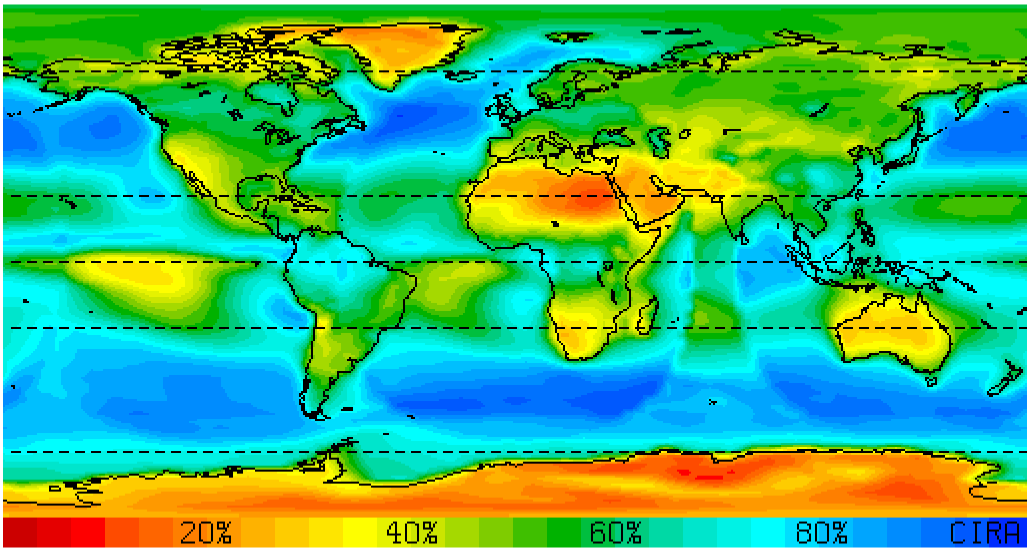


Figure 2 - 7 – Cloud Probability

But even in worst case, the transferable (40% availability over one year) data volume per year is at least 147 TByte (the most favorable situation of 50% of cloud blockage raises up this value to 184 TByte) and the camera could be used during 0.56 % of the operational time (0.70% if 50% availability over one year is considered). Such an improvement, by more than a factor of 4, can lead to a new condition, where the limiting factor is the available data storage on the satellite for cases when no data downlink is possible due to bad weather conditions.

It must be noted, anyway, that these values have been obtained considering just a single ground station, furthermore with a non-optimal placement: if a small network of four ground station, spaced by several hundred km to ensure uncorrelated cloud cover statistics, and distributed over the national territory (ground station diversity), the combined availability can be boosted to a mean value comprised between 81% and 94%, leading to an even further increase in payload operational lifetime (between 1,13% and 1,32%). This values are obtained considering ground stations located over territories that are characterized by cloud probability coverage between 50-60%, which limits the ground station mean availability to 50 – 40%. The payload operational lifetime can be further improved if the ground stations placement is driven by choice of profitable location from a climatic point of view: in fact, using just two ground stations in advantageous areas, like astronomical observatory sites or deserts, ground stations availability approaches 99%.

Furthermore, the use of a network of ground station, apart from improving the probability to have favorable condition for downlink, lead also to improvement of contact time, providing additional time slot for data download, improving, in turns, the payload operational time. In fact, with two ground stations placed in optimal location (cloud probability around 10%), spaced to ensure uncorrelated cloud cover statistics, the probability of establishing at least one contact is around 99%, allowing up to 1,39% of payload operational lifetime. At the same time, there is a probability of around 80% to establish a connection with both the ground stations, doubling the downlinking capability during the passes when the link with both the GS is possible.

2.4.2.4. Improving download data-rate: Optical and RF Ground Station

Even if optical downlink improve the volume of data a satellite can send to Earth, the possibility of cloud coverage (even really, as could happen with a network of ground station) cannot be acceptable for certain kind of application: we should think about the TV broadcasting, or a telecommunication service that needs to be operative 24/7. For overcoming the cases where an optical downlink is not possible due to cloud cover, we should imagine of combining in a same ground station both the equipment for optical and RF downlink. The proposed approach adds noteworthy complexity both in terms of ground support equipment and satellite download devices, that should be able to up/down load information in both the bandwidth, improving also the mass of the space segment. In addition, the use of a “double” GS impact the data downlink management: in fact, downloading of data has to be schedule with a certain advance to occur via RF or optical channel.

Such improved complexity, however, comes with a significant extension of the downlink availability, and, as a consequence, sensor usage. Using the value from above, this combined downlink capability will improve the volume of data downloadable between 166 and 203 TByte per year, improving the sensor usage between 0.63% and 0.76% yearly operational time. Such a value has been obtained considering that the optical communication system is the primary solution, used between 40-50% of the passages, when favourable weather condition are encountered; in case the cloud prevents the usage of such a system, RF backup solution would be used for send payload data to Earth.

2.4.2.5. Improving download data-rate: High Altitude Optical Station

The advantages that an optical ground station can offer in terms of data volume downloadable are clear and considerable; but the uncertainty related to weather condition can seriously prevent the use of such an innovative technology. In the previous paragraph, a solution of a “double” ground station has been proposed. In this one, another solution will be investigated: the use of a High Altitude Optical Station (HAOS).

Such an HAOS can be assumed to be placed above the ground station of Fucino, at an altitude of 20 kilometers; such station has an optical receiver, to be able to receive the data downloaded from the satellite. The downlink from this stratospheric station to the ground will occur by means of a point-to-point RF communication, through the use of a steered antenna; the RF technology has been chosen in order to have a 100% availability of such a “last mile” connection.

Since the HAOS is located at an altitude such that no cloud would obstruct the link between the satellites, such a stratospheric station can be considered available any time the satellite is in its visibility. Furthermore, being the station at a significant altitude, a link can be established even with the satellite at a negative elevation angle; at an altitude of 20 km, this angle is -2.7°.

This condition lead to an extended mean daily contact, that reaches the value of 4273s, and in a transferable data volume of 2670GByte. This daily downlinking capability lead to a 974TByte per year downloadable from the satellite to the HAOS, expanding the payload operational time up to 3.69% of the time

The following table summarizes the result of the hypothesis made, considering, as technological alternative to a 262 Mbps RF-downlink (the most efficient RF-downlink) an optical channel with a data rate of 5Gbps (a moderate data rate for such a technology).

	RF Downlink	GS Optical Downlink	HAOS
Downlink data rate [Mbps]	262	5000	5000
Payload data volume produced [Gbps]	6.7	6.7	6.7
Mean daily contact time [s]	2668.216	1615.193	4272.820
GS availability [%]	100	40	100
Data downloaded daily [GByte]	87.384	403.798	2670.512
Data downloaded yearly [TByte]	31.895	147.386	974.737
payload operation [%]	0.12	0.56	3.69

Table 2 - 7 – Comparison among different systems for downloading data

2.4.2.6. Wavelength Selection

Due to the station keeping altitude of HAOS, the effects of the atmosphere on optical beams are much smaller if compared with those typical of scenarios where optical station are "grounded". But, as already said in the beginning, it is important to correctly evaluate the interaction between the optical beams and the atmosphere, especially at low elevation angles, where longer atmospheric layer needs to be crossed; for such a reason, the wavelength has to be carefully selected because of the large variance of the absorption coefficient over the wavelength.

Three absorption effects needs to be evaluated. The first one is the freespace loss, which is determined by the beam divergence angle, and is not dependant on obstacles that can reflect of cause diffraction, while the two remaining attenuation effects characterize the propagation in the clear atmosphere: absorption and scattering. These effects lead to specific transmission windows, which are then suitable for optical communications in the atmosphere. But it is also important that laser sources and detectors are available in the regions with minimal atmospheric attenuation: in fact, it must be borne in mind the transmission window previously shown.

Having a look at the possible choice, three wavelength regions for Free Space Optical (FSO) systems has been identified[33,34]: 785nm, 1064nm and 1550nm. The 785nm technology presents some disadvantages with respect the two other frequencies: the presence of strong background light and the higher Rayleigh-scattering compared to 1064 and 1550nm. For 1064nm and 1550nm technology one of the clear advantages is the availability of high power optical fibre amplifiers to boost the transmission signal.

The physical properties of any of the usable wavelength must be investigated more in detail, as it encounters several problems that cannot be solved or discussed from an engineering point of view. But, for the sake of completeness, it must be noted that optical system working around 1550nm, with on/offkeying and direct detection, is widely used in terrestrial fibreoptical transmission, and technological experiments involving such a system in stratospheric situation have been executed[35].

2.4.2.7. Mobile Optical Ground Station

Finally, another opportunity provided by such a technological solution is that of having a portable ground station, small and lightweight, that can, anyway, assure a significant data rate capability. The possibility of having such a portable ground station could potentially allow deploying a global network of small ground station capable of downloading information from satellite equipped with such a communication system.

The possibility of having at disposal such a ground station has been demonstrated by the DLR, that developed and manufactured a transportable GS, shown in the next figure.

Figure 2 - 8 – Optical Ground Station (1)

The illustrated system has an aluminum mirror of 60cm; the structure is realized looking for low weight and for such a reason composite materials have been extensively used. Such a GS has a data-rate capability of 1,25 Gbit/s.

In the next figure, the portable ground station mounted and tested.

Figure 2 - 9 – Optical Ground Station (2)

2.4.3. Conclusion

We have calculated the practical advantage of optical downlinks from earth observation (EO) satellites over conventional RF-downlinks. The usability of the EO-sensor could be boosted by nearly a factor of thirty with a future system based on HAOS (High Altitude Optical Station) that in addition offers the capability of downloading data from the satellite for longer contact time and without any dependencies from weather condition.

Alternatively, a more practical system can enhance the down linkable data by approximately a factor of ten with simple direct downlinks via just a small network of four optical ground stations: in fact, the use of one optical ground station enhances the payload's operational time, with respect to reliance solely on conventional radio downlinks, from 0.12% to 0.56%; this value is additional improved with four ground stations, characterized by cloud blockage used for calculation o that approximately double the ground station availability, from 40% to 81%, in the worst case. The implementation of such a system for products and services requiring 24/7 availability can be problematic in this latter case, due to the constraint that meteorological conditions may impose on the downlink capability.

In both the cases, wavelength selection must considered carefully as many technical factors can affect the performance of such a system. The high data rate performance (from 1 up to 10 Gbps, with current technology) is offered by low-power on-orbit transmitters with very small optical apertures, in the range of few centimeters, and having in accordance a very low mass. This allows possible future users to use portable ground station(s) that can be deployed according to evolving requirements, evolving gradually the capacity and capability of a system based on this technology.

Figure Index

REFERENCES

[1]- Pasquale Maurizio De Carlo, Leonardi Roberto, et al. Intersatellite link for Earth Observation Satellites constellation, http://www.corista.eu/Docs/intersatellite_link.pdf

[2] - Gilles Planche, Vincent Chorvalli, SILEX in-orbit performances, Proceedings of the 5th International Conference on Space Optics, Toulouse, France, 30 March - 2 April, 2004.

[3] - Bernard L. Edwards, *Distributed Spacecraft Crosslink Study Part 1: Spectrum requirements and allocation survey report and recommendations*, Goddard Space Flight Center, May 2002.

[4] - National Telecommunications and Information Administration, *Manual of Regulations & Procedures for Federal Radio Frequency Management*, U. S. Government Printing Office, Washington, D.C., May, 2003.

[5] - James R. Wertz, Wiley J. Larson, *Space Mission Analysis and Design*, Third edition, pp 535-585.

[6] - Avila-Rodriguez J.A., Wallner S., Hein G.W., Eissfeller B., *A vision on new frequencies, signals and Concepts for future GNSS systems*, Proceedings of the International Technical Meeting of the Institute of Navigation (ION GNSS), Fort Worth, Texas, 25-28 September, 2007.

[7] - Abdul-Halim Jallad, Vladimirova T., *Distributed Computing for Formation Flying Missions*, Proceedings of the IEEE International Conference on Computer Systems and Applications, Dubai/Sharjah, UAE, 08 - 11 March, 2006.

[8] - Lestarquit L., Harr J., Grelier T., et al. *Autonomous Formation Flying RF Sensor development for the PRISMA Mission*, 19th International Technical Meeting for the Satellite Division (ION GNSS), Fort Worth, TX, 26-29 September 2006.

[9] - Heckler G., Winternitz L., Bamford W., *MMS-IRAS TRL-6 Testing*, 21st International Technical Meeting of the Satellite Division (ION GNSS), Savannah, GA, 16-19 September 2008.

[10] - Harr J., Delpech M., Grelier T., Seguela D., *The FFIORD Experiment: CNES' RF Metrology Validation and Formation Flying Demonstration on PRISMA*, 3rd International Symposium on Formation Flying Missions and Technologies, Noordwijk, the Netherlands, 23-25 April 2008.

[11] - Bourga C., Mehlen C., et al. A *Formation Flying RF Subsystem for DARWIN and SMART-2*, International Symposium Formation Flying: Missions & Technologies, Toulouse, France, 29-31 October 2002.

[12] - Stadter P. A., Heins R. J., Chacos A. A., et al. Enabling Distributed Spacecraft Systems with the Crosslink Transceiver, AIAA Space Conference and Exposition, Albuquerque, NM, 28-30 August 2001.

[13] - C. Jenkins, *"Gossamer Spacecraft: Membrane and Inflatable Structures Technology for Space Applications"*, Progress in Astronautics and Aeronautics

[14] – F. Svelto, *"Technologies for Human Space Exploration: ASI PROGRAMS"*

[15] - E. Oñate and B. Kröplin *"Recent Advances in the Rigidization of Gossamer Structures"*, Textile Composites and Inflatable Structures, 259–283. © 2005 Springer.

[16] - "Triple Junction Solar Cell with 30.0% Efficiency and Next Generation Cell Concepts", 9th European Space Power Conference, St. Raphaël, France, 6-10 June 2011

[17] – Data Sheet Solar Cell type 3G30; http://azurspace.de/index.php?mm=97

[18] – http://www.nrel.gov/news/press/2008/574.html

[19] – Repins, I.; Contreras, M. A.; Egaas, B.; DeHart, C.; Scharf, J.; Perkins, C. L.; To, B.; Noufi, R. "19.9%-Efficient ZnO/CdS/CuInGaSe^2 Solar Cell with 81.2% Fill Factor." Prog. Photovolt: Res. Appl. 2008; 16, 235-239.

[20] – Solar Junction Breaks Concentrated Solar World Record with 43.5% Efficiency. Optics.org (2011-04-19) Retrieved on 2012-06-05.

[21] – D. M. Wilt, M. A. Smith, W. Maurer, D. Scheiman, P. P. Jenkins, *"GaAs Photovoltaics on Polycrystalline Ge Substrates"*, 4th World Conference on Photovoltaic Energy Conversion, Conference Record, 2006 IEEE

[22] – Otte, K.; Makhova, L.; Braun, A.; Konovalov, I. "Flexible Cu(In,Ga)Se2 Thin-Film Solar Cells for Space Application." Thin Solid Films. 2006; 515-516, 613-622.

[23] – X. Mathew, J. P. Enriquez, A. Romeo A. N. Tiwari, *"CdTe/CdS solar cells on flexible substrates"*, Solar Energy 77 (2004) 831–838

[24] – A.N. Tiwari, A. Romeo, D. Baetzner, H. Zogg, *"Flexible CdTe Solar Cells on Polymer Films"*, Progress in Photovoltaics: Research and Application 2001; 9:211-215

[25] – B.M. Basol, V.K. Kapur, C.R. Leidholm, A. Halani, *"Flexible and light weight copper indium diselenide solar cells"*, Conference Record of Photovoltaic Specialists Conference, 1996

[26] – M. A. Sturza, F. Ghazvinian, *"Teledesic Satellite System Overview"*

[27] – N.P. Kim, J.A. Nielsen, M.A. Verzwyvelt, *"Lightweight CuInSe2 space solar cells fabricated by chemical thinning"* Conference Record of Photovoltaic Specialists Conference, 1996

[28] McDermott, S.A., Jacobovits, A. ; Yashiro, H.. *"Automotive electronics in space: combining the advantages of high reliability components with high production volume"* - Aerospace Conference Proceedings, 2002. IEEE

[29] Pyzdek, Thomas, The Complete Guide to Six Sigma, Quality Publishing, 1999.

[30] –*"Space Radiation Effects on Electronic components in Low Earth Orbit"*, NASA Preferred Reliability Practices, Practice No. Pd-Ed-1258

[31] N. Perlot, M. Knapek, D. Giggenbach, J. Horwath, M. Brechtelsbauer, Y. Takayama, T. Jono, "Results of the Optical Downlink Experiment KIODO from OICETS Satellite to Optical Ground Station Oberpfaffenhofen (OGS-OP)", Proc. of SPIE 6457A, 2007

[32] - http://xjubier.free.fr/en/site_pages/SolarEclipseWeather.html (data acquired by the International Satellite Cloud Climatology Project, established in 1982 as a part of the World Climate Research Program)

[33] – G. Soni, “*Free Space Optics System: Performance And Link Availability*”, International Journal of Computing and Corporate Research, Volume 1 Issue 3 Manuscript 4 November 2011

[34] – Kim, B. Mcarthur, and E. Korevaar, “Comparison of laser beam propagation at 785 and 1550 nm in fog and haze for opt. wireless communications,” Proc. of SPIE, vol. 4214, pp. 26-37, 2001

[35] J. Horwath, M. Knapek, B. Wilkerson, B. Epple, M. Brechtelsbauer, D. Giggenbach, N. Perlot, "Backhaul communications for High Altitude Platforms, Results of STROPEX, the Stratospheric Optical Payload Experiment", 15th IST Mobile & Wireless Communications Summit, June 2006, Myconos, Greece

3. Rapidly Deployable Platform

3.1. Introduction

The capabilities of small spacecrafts have increased significantly over the past decade. Indeed, it is generally accepted that small spacecrafts will continue to offer reduced cost solutions to conduct space missions, offering a large fraction of larger satellites capabilities, at only a small fraction of their cost: this is due to the fact that large (and consequently expensive) satellites are assumed to perform for a long period, with a lifetime ranging from at least 5 up to 15 years (typical lifetime of geostationary satellite), while the micro-satellites have a lifetime significantly shorter, ranging from 6 months up to, typically, 24 months.

The reduced lifetime allows microsatellite designers and developers to consider less the effect of the degradation of subsystem performances due to the severe space environment than the developers of satellites with a longer lifetime: the Total Ionizing Dose in the electronics, the degradation of solar cells efficiency, and the reduction of component performances due to the space environment can be treated with significantly less criticality.

On the contrary, bearing in mind that satellites cannot be maintained once launched, in order to keep the satellites operating for a long period of time it is necessary to implement several tricks to guarantee the performances of the satellite along the entire lifetime. The most widespread actions undertook are: the component redundancy, to avoid loss of functionalities if one or more components fail; subsystem or component oversizing, to compensate the performances degradation due to incoming radiation (this is typical during the solar array design, which area is calculated to provide a certain power at End-of-Life, and the area calculated is increased to keep into account performance degradation). But, since satellites are really complex systems, the increasing in mass of certain subsystems impacts also any other of them. The typical example is the AOCS (Attitude and Orbit Control System), which performances are get worse by an increasing of spacecraft's mass. So, if the satellite mass increases due to component redundancy and subsystem oversizing, in order to keep the AOCS performances (pointing control and accuracy, for example) we need to upgrade its components, which will probably lead to a heavier and more power-demanding subsystem, which in turns will affect other subsystems, such as the EPS (Electric Power System) and TCS (Thermal Control System), for example. The changes in these latter subsystems will probably impact the design of other spacecraft subsystems, and so on, leading to a series of design iteration necessary to reach a complete design.

This lead to the situation in which a satellite performing a particular mission for a short lifetime can be extremely smaller than a satellite performing the same mission, with the same performances, but with a largely longer lifetime.

The high complexity of medium and large satellite is also cause of another aspect seriously limiting the potentiality of space capabilities and their implementation: the lead time. The lead time is the time passing through the conception of the service that a space system can provide, and the time the space platform is placed into orbit, ready to perform the prescribed service. For large and medium platform, it could take several years (from 5 up to 10 for extremely complex and large spacecraft) for designing, manufacturing, testing and launching the satellite. This situation precludes such space platform the possibility to be launched shortly for responding a sudden arisen need, or for replacing in the short a failed system. Also for such applications, the microsatellite can provide opportunity otherwise unthinkable: a microsatellite can be developed in timeframe shorter than those common for large platforms, typically 18-24 month; but, even if the lead time typical of microsatellite are shorter, they cannot anyway respond to short-term needs.

In this chapter, aspects, methodologies and procedures for implementing the capability of responsively deploy a space system will be investigated, embracing both the space segment and the launch segment. No consideration will be done on ground segment, even if its importance is undeniable in the vision of responsively provide a service to a proper user.

A new vision will be provided for microsatellite, conceived as the only (presently) candidate for arranging a capability of deploy a space system, from the conception of the mission requirements, in few months rather than years.

3.2. Responsive Space concept

The ORS Office is an office of the U.S. Department of Defense (DoD), being ORS an acronyms standing for Operational Responsive Space; the Operational Responsive Space Office has the task, in synthesis, to "*allow a Joint Force Commander to rapidly implement and reconstitute a space capability*". For this reason, the Operational Responsive Space has been introduced in the space field with a clear military sense.

Going through the way paved by the U.S. DoD, and modifying some of the paradigms introduced, in this chapter a novel definition will be coined, and the concept of *Responsive Space* will be redefined. All the characteristic of this concept will be explained, the advantages enumerated and investigated, and the incredible opportunity that such a capability will be able to provide to the space users will be listed and explained.

The reasons and the opportunities given by the implementation of innovative technologies into microsatellite platforms have been described in the previous chapter. The possibility of a responsive capability will enable several additional opportunities. The main reason for seeking such a capability is related to the large amount of time needed for space programs to be completed. Traditional space programs need many years to achieve their completion and such a long period of time can have harmful effects:

- Mission operational needs can change in the meantime, making the effort done during the years and the economic resources invested useless, as well as the service intended of no interest.
- In such a long period of time, novel techniques or innovative technologies could have made simpler the realization of certain operations.
- Economic difficulties, changes in the socio-political scenarios or even simpler reason, like the rise of different sudden necessities, can lead to the cancellation of a program that is requiring too long to be completed.

Furthermore, as the great majority of the space programs exceed economic budget, as well as the scheduled timeline, the implementation of methodologies, procedures and technological solutions for accelerating the development of space systems will be of outmost importance both for having space capabilities that could be rapidly implemented for the required service, and also as a support to space programs to respect development time constraints.

As an example of *Responsive Space* application, we can cite the necessity of having images, with a frequency of about 10 images per day, over a particular area, for a period of time of several months, and the first image needs to be obtained after just a couple of months from the time of the mission kick-off. This situation cannot be satisfied if traditional procedures for the deployment of a space capability are used.

In order to make this possibility feasible, new techniques, new procedures and new conducts must be studied, developed and implemented. The responsive space capabilities could also give the possibility to rapidly reconstruct space assets that could suddenly be unavailable, due to a failure.

The studies performed over the concept of responsive space derives from the identification of several needs presently not covered: the possibility of rapidly developing a space capability for responding to suddenly arisen needs; the necessity of having the capability of (even partially) reconstitute a space asset that unexpectedly faced a failure; the opportunity of using the most innovative technologies in exploitation of space services through the use of space system (microsatellite in particular) rapidly developed.

All these necessities converge into a single aspect: Responsive Space. The implementation of the related capabilities can allow the engineers to satisfy the following needs:

1. Reconstitute lost capabilities, subsequently to a system failure, for example;
2. Augment existing capabilities, if additional resources are demanded to a particular space system;
3. Fill unanticipated gap in capabilities, if a new request for space service arise;
4. Exploit new technical/operational innovations;
5. Respond to unforeseen events, like a natural or man-made disaster.

The responsive space is a capability that goes well beyond the (however difficult) possibility to rapidly assembly space platform.

A comprehensive study needs to be done in order to underline all the difficulties, highlight all the technical aspect that is to be investigated to allow the implementation of a responsive capability. In this chapter, consideration about the operation, the existence and availability of ground station (and ground equipment) will not be done, as the work has been focused on the space segment (the satellite) and the launch segment (the launch vehicle).

From the point of view of the space segment, the fulfillment of one of the abovementioned need requires the possibility of rapidly design, manufacture, integrate (and eventually test) a space platform. The realization of such a possibility is to be achieved through procedures different from those traditionally implemented when a satellite has to be designed. In the latter, in fact, a one-of-a-kind design is performed, while a "responsive" satellite and its subsystems are to be designed differently: to be rapidly assembled, a variety of "pieces" must be available, and their collection and assembly must be lead to the final space system. This implies that the objectives of the design must not be the final product, but functionalities; in other word, in a similar way to what happen in software design, where many reusable functions are written, and then, once the necessity arise, assembled to have the desired code, a collection of subsystem modules must be available. Each module will be capable of providing one or more functions and, according to the mission needs, they will be assembled and integrated to satisfy such needs.

From the point of view of the launch segment, the fast response in integrating a space system must be coupled with an innovative method for launching satellites: the traditional surface launches, and the less widespread sea-launches cannot be compliant with stringent requirements in terms of time-to-launch that certain activities needs. A new method, capable of responding to severe time constraints allowing a rapid integration of the launcher vehicle with the payload, and a fast access to space, is of critical importance for the exploitation of responsive capabilities in satellite development.

3.2.1. Reason for *Responsive Space*

In the last years, and in particular in the last decade, a growing attention and interest raised towards such concepts, named *Responsive Space*, and many attempts have been undertaken for the purpose of significantly reducing time and cost needed to implement a certain operative capability. Such an interest is supported by several arguments:

- Market research underlines the strong necessity of such a reduction, together with a reduction of risk related to the development of a typical space system. One of the main reasons for such an effort is the present impossibility of employing a new technology in a space project. In fact, typically, the launch of a satellite is the conclusion of a project which design phase freezes the technology to mount on board satellite several years before, at least 3 and up to 5 years for bigger projects). Furthermore, the complexity, the long time spent to build a satellite, and the high cost related to the development, launch and in-orbit operations, suggest to the companies that manufacture satellites and their equipment to rely on components which have successfully flown on past missions. The more missions a component accomplished, the more reliable it is and the more confident are the companies that a particular component will perform the required mission. The drawback of this way of proceeding is the difficulty of introducing in a space mission newer components, keeping the level of performance of a particular system the same over the years.
- The capability of rapidly fulfilling specific operative needs through the use of space is an important asset to acquire. This capability is important for many civilian application, such as the monitoring and the management of natural disaster.
- In addition, it is generally known that a trend that most of the space project have in common is the slippage in terms of planned costs and schedules. Initiatives that have as main objective the implementation of methodologies, procedures and processes that contrast these trends are well-accepted.

With the aim of investigating responsive capabilities, both in Europe and U.S, a certain number of efforts have been undertaken. In particular, a deeper investigation has been conducted with the aim of producing standards and encouraging modularity of components with the intention of simplifying the development of the space platform. Standards should promote interchangeability and interoperability, while modularity tries to reduce the complexity through integration by reducing the coupling between system components.

As a consequence, in order to get the abovementioned capability, it’s necessary to overtake technological, but also cultural, barriers typical of space projects, like the distrust of new technology. If these capabilities are reached, a systematic and significant reduction of design phase, manufacturing and integration of a satellite will be possible.

To support the Responsiveness, a particular attention must be put on the development of a “low cost” launch capability able to quickly place in orbit a satellite. In fact, traditional satellite orbit injections require expensive and time-demanding launch schedule. If no alternative launch strategy is adopted, the advantage of designing, manufacturing and assembling a satellite in a short period of time would be wasted.

The process of standardize the interface between components helps the rapid integration of the satellite equipment, but the rapid creation of a complex satellite needs to face several technological challenges: in particular, the improvement of on-board avionics and software represents the major issues. Nevertheless, avionics and software are the components with the largest improvement margin, driven by solution found for terrestrial equipment, like the Plug-and-Play (PnP) technology. This technology, largely implemented in the field of electronics, in particular with USB devices that can be plugged into personal computer, lead to development of an impressive number of low cost equipment requiring just a few second after its connection to PC to configure it. The advantages suggest transferring this technology also into the space field.

The development of the PnP standards in the field of space activities could fasten the design and the manufacturing phase of a satellite, and, together with a well-thought modularity of the equipment, provide the capability of *Responsive Space*.

To make the long story short, *Responsive Space* is motivated by the growing necessity of rapidly deploying space platform, shortening the lead time from several to some months (from one to three); such an effort will hardly be accomplished if large platform and their related procedures will be used as subject of the study; microsatellite, for their nature, are the most promising space system to be used as subject: they are small enough to make subsystem development and integration compliant with time constraint, and also large enough to accommodate payload and equipment capable to provide a real, valuable service (as it will be demonstrated in the fifth chapter, where three case studies have been performed). Technologies, methodologies and procedures will be conceived and expresses, trying to create an as comprehensive as possible paradigm for "responsive platform development"

3.3. Responsive Space – Space Segment

3.3.1. State of the Art

A complex system requires more time and additional effort for design, manufacturing and integration than a simpler system, of course; thus, the objectives of developing a space system in the timeframe of several months instead of many years can appear as unreasonable; nevertheless, combining powerful development tool, that will be discussed in a successive paragraph, together with clear methodologies of components development and manufacture, and a fair attention to the aspect of modularity (which allows for re-usability) and system re-configurability, such objective can be achieved.

Anyway, the possibilities of time reduction for space programs are not so evident. Part of the problem comes from the use of old, but well-defined and approved approach; it is well known, in fact, and we extensively made mention of it, the issue known as *legacy:* the tendency of composing space system with "space qualified" components. These are components that have already flown, and, as they demonstrated their ability to survive in space environment, they give a certain confidence about their capabilities; anyway, such a confidence is often apparent. Components that already flown, in fact, are often integrated in a new assembly that never flew; furthermore, it is extremely rare that devices could be integrated without the development of tailored hardware and software.

This problem is well established, preventing the opportunity of introducing modern and more powerful infrastructure. According to the previous consideration, the avionics in the satellites are extremely obsolete, and really less performing than those of terrestrial system made for application of similar complexity. All these consideration found an appropriate answer in the methodological approach proposed in the entire thesis: it is important, in fact, to consider a new approach to the design and development of such systems.

The term *Responsive Space* suggests the capability of rapidly accomplishing a specific objective, suddenly arose, through the use of space systems. Initially, this could be identified as an on-demand launch capability, able to be immediately deploying a space system. But a more careful analysis advises that such a view is extremely restrictive, since does not envisage some of the most significant challenges to face, so as integration and testing phase, for instance.

Even if an immediate launch capability should be available, the payload and the bus should be already integrated (excluding, this way, a priori the possibility to introduce novel payload); or the components should be off-the-shelf, ready to be rapidly assembled. This latter solution involve technological limitation, represented by the huge amount of electronics into a satellite, which is often source of error both in terms of software's lines of code and interface.

Moreover, the difficulty that could arise in functional verification would be of tricky resolution. There are at least four different factors that limit the development and implementation of complex system:

- **Thought limited**: the time needed for conceiving, planning, designing, and settling is strictly related to the complexity of the project
- **Process limited**: the time needed for manufacturing components is limited by the fabrication processes
- **Coordination limited**: the common process of communication, approval, authorizations and so on can strongly impact the development time.
- **Geographic limited**: the process of component procurement envisages the possibility to wait components that are manufactured in different geographical places. Furthermore, given that not all the components can be assembled in the same place, it is generally necessary to use several plants for completing the system.

There are several examples of systems in which one class of limitations overlaps the others: for example, in the case of FPGA (Field Programmable Gate Array), delay due to process are converted in delay due to the thought limitations; procuring components and storing them into a warehouse can convert the process and geographic limitation into a different process limitation, consisting in the selection and integration of components in a single plants.

Analyzing all these aspects, it is extremely clear how the concept of *Responsive Space* not only rely on one single, isolate element, like that of rapidly launching a satellite in space, but also on a design methodology, which faces all the issues that are time-consuming, and responsible for project delay.

A lot of studies have been conducted having as central theme the concept of *Responsive Space,* witnessed by the conferences on such a topic, which reached in 2012 the ninth edition. Some of them[1,2,3] identified the Plug and Play (PnP) technologies, especially in the field of avionics, as outstanding opportunities to be used to help achieving responsive space capabilities such as the rapid reconstitution and augmentation of existing space assets.

Re-configurability and modularity of the space system help also in case a late intervention on the satellite needs to be performed, for substituting and/or adding particular component. During a traditional space program, anyone would advise against doing such an operation, as this intervention could prevent the correct functionality of the satellite.

From the investigation undertaken on traditional satellite design and development procedures, several problems and criticalities arose, as well as many opportunities and principles that need to be followed and exploited for being able to implement a responsive capability, at least in the area where these competences can be applied.

3.3.2. Innovative Architecture

In this paragraph will be presented methodologies, technology and technical solutions aiming at shortening as much as possible the time to get the satellite final design.

This new approach must be opened to the use of terrestrial components, adequately selected, tailored to space environment, as extensively shown in the previous chapter, with the principal aim of augmenting system performances, but with the additional step further of inserting such innovative components in a re-organized satellite architecture: after proposing in the previous chapter possible innovative technologies, in this chapter a deeper look towards a re-organized and more effective satellite architecture will be done.

Space systems tend to centralize the intelligence, and, even if processor is appearing in some satellite subsystem, the logic is still that of a centralized approach. On the contrary, many terrestrial complex systems have distributed intelligence, which reduce the necessity for cabling: Internet is the brightest example of distributed intelligence. In addition, centralized systems embrace the increased complexity of longer flight codes, as well as more cabling, and the increased risk related to the presence of a single point failure that could damage the entire mission.

In the area of software, it is important to change the development philosophy, switching from the present tendency of develop mission specific software to a different logic, that of populating a database of module of software, each devoted to a particular functionality: in such a way, it will be possible to develop a mission specific software by simply selecting the modules necessary for the mission under investigation. This will allow for reusability of module of software.

Critical element is to develop a process of component manufacturing and assembly able to embrace many of the key principles behind a modern plug-and-play (PnP) approach. At the beginning we will focus on avionics, which has been identified as the area most readily to be transformed into a PnP system; the following four elements were those retained as the most crucial:

- Plug-and-Play Network;
- Innovative spacecraft bus;
- Adaptive computational capability;
- Configurable Radio (via software).

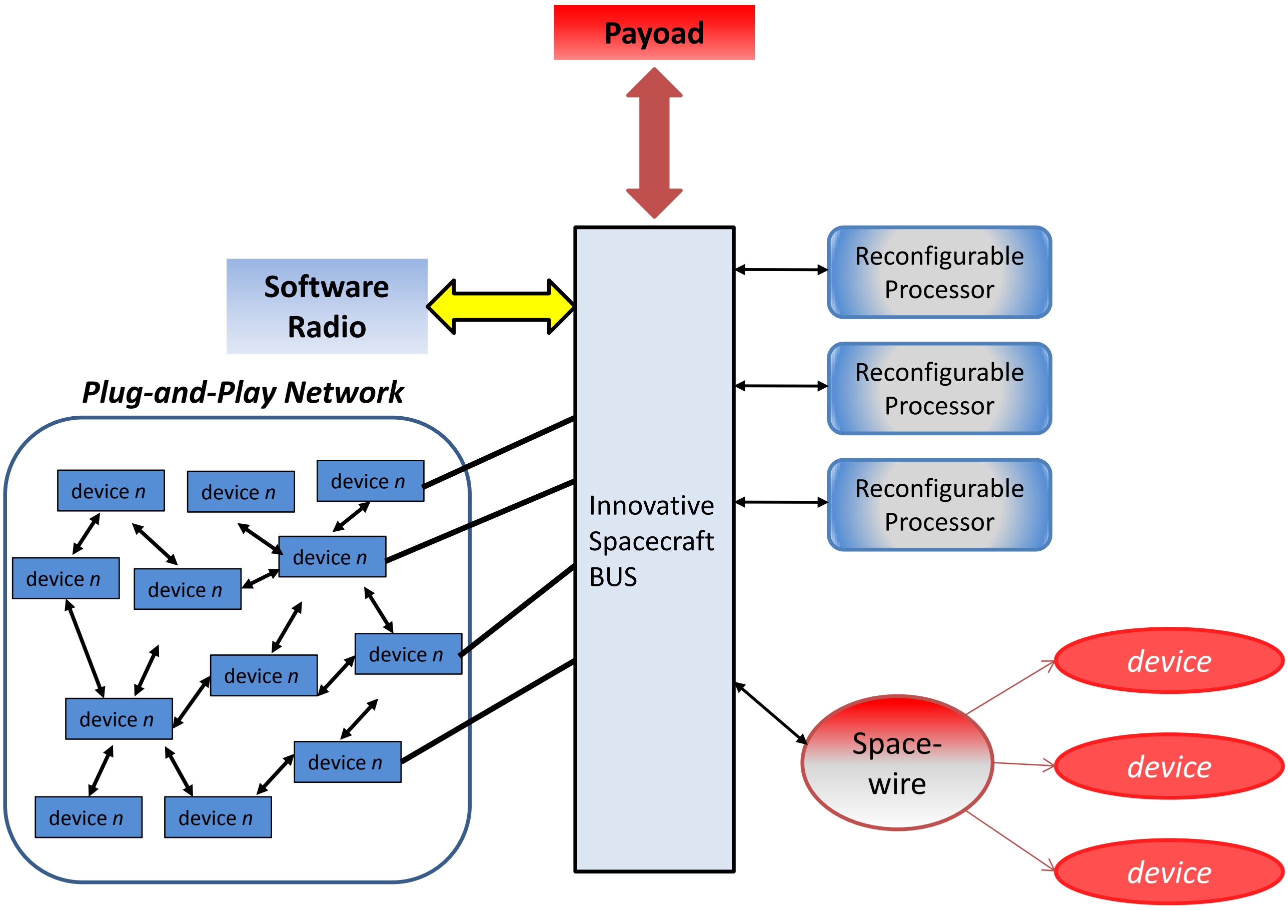


Figure 3 - 1 – Conception of a PnP Space system

The component of a PnP system must be designed to be re-configurable: for example, a traditional transceiver should be substituted by a device able to modify the waveform in order to be able to support any mission requirements; or reconfigurable adaptive processor should be adopted for the purpose of being able to tailor the computational capabilities the satellite is able to provide with those required by mission objective. The previous figure shows the relationship and the interconnections among the abovementioned components; this scheme is in contraposition with the next one, which shows the architecture of a traditional space system.

Anyway, the most innovative concept that can be extrapolated by the previous figure is that of innovative BUS, which is characterized by programmable cabling, which allows the use of cabling constituted a-priori, which can be configured during the integration.

Networks of this typology have functionalities similar to those of traditional PnP network, having the automatic recognition and configuration of new devices as the most evident and powerful application, because it allows a rapid system integration.

3.3.2.1. Plug And Play

Usually, the architecture of the avionics on board satellite is characterized by a Command and Data Handling System exploiting a common bus (the PSI Bus is the most widespread one) allowing peripherals to communicate, as shown in the next figure.

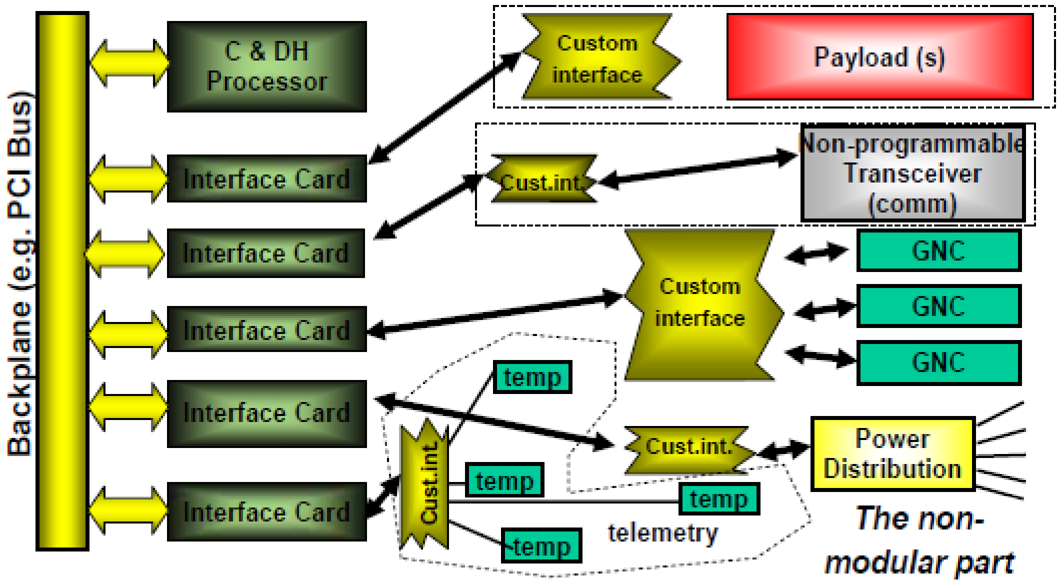


Figure 3 - 2 – Traditional non-modular system

The use of such a standard bus allows the different peripherals to communicate each other; nevertheless, even if only standard interfaces (e.g., RS–422) are used, it's often necessary to customize both satellite components hardware and software. It is clear, therefore, that such an attempt to standardize do not correctly face the main problem related to assembling complex system; in fact, such a solution has lead, in the previous years, to the necessity of software writing, harness building and tailoring of electric interface; but such operations, as witnessed by previous space program, could take several months.

The concept proposed in this paragraph for allowing responsive space focus on the standardization of interface as one of the principal aspects, probably the most important one, even if not the only one. For this reason, responsive network must adopt some technological innovations:

- A computational capabilities that can be modularized to mission needs (Malleable Signal Processor)
- Plug-and-Play network
- Radio with modular bandwidth (via software)
- *'Switched fabric'* approach
- Innovative BUS.

The use of Plug-and-Play Network suggests the idea of comparing the satellite to a self-organized network: in fact, the logic of PnP equips the satellite with a sensors and actuators connection mechanism that is self-organized, through the use of plug-and-play protocol. The implementation of Software Definable Radio (SDR) gives the satellite flexibility in operation, and compatibility with past and present communication systems. Malleable Signal Processor provides the satellite with the necessary computational capability.

In the following, detailed descriptions of each of the element identified as revolutionary for responsiveness will be provided.

3.3.2.1.1. Reconfigurable Computational Capabilities

In order to be capable of rapidly implementing the correct computing capability, and in order to be compliant with computational requirements that could be modified up to few days previous launch, the development of reconfigurable processor could allow space system designer to tailor such computational capability according to mission requirements. The computational architecture is, in turn, reconfigurable, and its capability could exceed the elaboration capacity of any other satellite. The processing unit so defined will be connected by a communication bus based on Spacewire standard. Such high computational capability will provide the satellite with the capability of running algorithms of on orbit image processing, target detection, mission re-planning, etc, giving the space system also feature that traditional micro-satellite does not have.

3.3.2.1.2. Innovative Bus Architecture

Such an innovative BUS has been conceived for providing a powerful instrument for implementing a completely reconfigurable system for satellite application; such a system embeds hundreds of relays throughout packaging and interconnect structures, each accessible and mutable through ordinary software commands. Connections that might be beneficially altered over the course of a mission can be routed through this new kind of BUS: examples of modifiable connections include power distribution, internal communications buses, special discrete interlock control signals, and off-line spare avionics units (for cold sparing).

Such a typology of BUS can provide significant advantages for the implementation of responsive space capability: it can be used in the construction of universal prefabricated wiring harnesses, which can be pulled off-the-shelf and configured rapidly for a new mission: during test and integration, this BUS can be altered to rectify interface errors once they have been discovered; on orbit, it can be reconfigured as necessary, due to changes in mission objectives, or faults that could emerge over a mission lifetime.

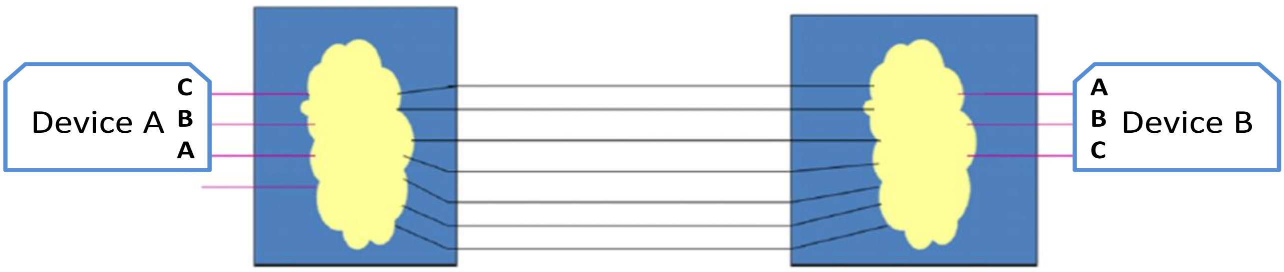


Figure 3 - 3 – Concept of an adaptive connection system

In the previous figure, two devices with three terminals each are schematized: in a traditional satellite bus, the connection among the terminals are to be defined, frozen and physically realized during the design and development phases, precluding the possibility to be later changed. On the other hand, instead, adaptive wiring systems do not face these limitations, and have many useful properties, including the potential of self-healing/diagnostics and soft-definable probe signals. Algorithms used in FPGA and switch routing are exploited to guide the formation of switchable wire paths in the adaptive wiring manifold. A reconfigurable switch fabric enables dynamic routing of signals in many different applications. Power routing, digital and analog signals and high frequency transmissions can be routed for space systems.

As previously discussed, adaptive wiring manifolds offer a number of benefits in developing new systems. Since the adaptive wiring substrates (or panels) may be pre-built and inventoried until use, it is possible to retrieve them as needed and configure them on demand. Rather than wait for custom-defined wiring harnesses to be developed and delivered, a process that could take several months in case of development of complex system like a space one could be, the adaptive versions can be configured very quickly. Unlike custom wiring harnesses, whose wiring pattern is permanently locked in, adaptive panels can be altered as needed to accommodate late-point changes.

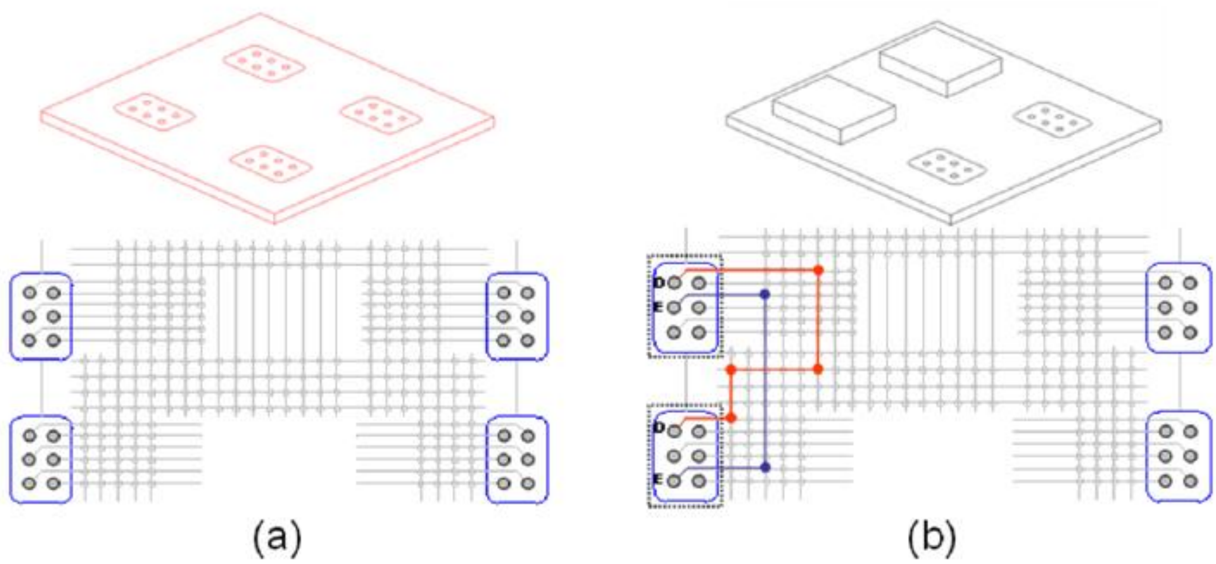


Figure 3 - 4 – Adaptive Wiring Manifold

Previous figure depicts a notional implementation concept to provide some intuitions about how an adaptive wiring system might actually be implemented. In this case, the substrate takes on the aspect of a physical panel featuring four sockets where components or “modules” can be mounted (previous Figure, a). To implement the amorphous “cloud” of wiring resources as shown in the precedent figure, we depict here a deliberate configuration consisting of a matrix of wires in rows and columns, with circles shown at the intersection points. The circles represent electrical switches that, when closed, connect together the associated row and column. At this moment, we are not concerned over the specific medium for switches. They could be, for example, metallic relays, solid state switches, microelectromechanical systems (MEMS) devices, or combinations of these and other switch types. Using such fabrics, to create a connection it's necessary to close, a number of switches, as shown in the previous Figure (b), which shows how a two-net list problem (involving two placed modules) might be solved (through a total of six switch closures).

Adaptive Wiring Manifold (AWM) combines MEMS (Micro Electro Mechanical System) technology and Field Programmable Interface Device (FPID) to redirect connections among components, both for on board anomalies mitigations and for responsive pre-launch access. The sensors' inputs are connected to the satellite intelligence through the AWM. This architecture is used for dynamically connect terminal on board the satellite, reaching a final result that can be schematized, going back over to the previous image, as follows.

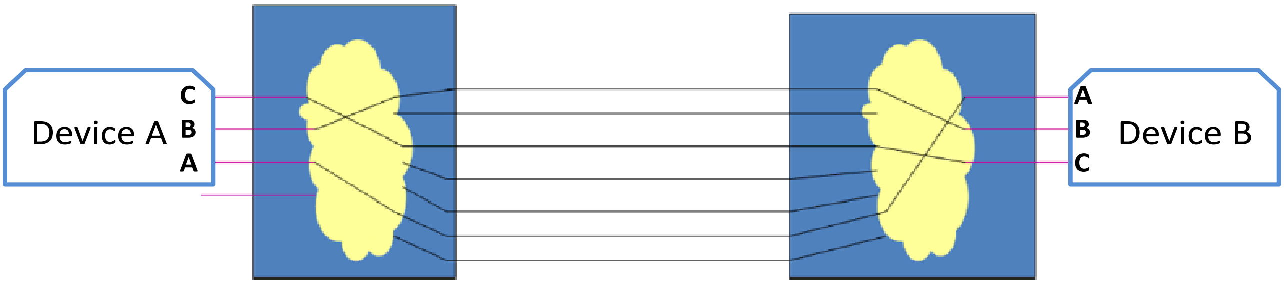


Figure 3 - 5 – Example of adaptive connection system (1)

Adaptive wiring systems furthermore, have two powerful benefits that are impossible in any other wiring technology. The first advantage of this architecture is the ability to adapt to faults occurring after the system is placed in operation. Since wiring patterns can be software-definable, defects can conceivably be bypassed by computing an alternate configuration. This concept is illustrated in the next figure. In this case, a faulty connection between B-B can be rectified by configuring other wiring resources that can achieve an equivalent connection without removing the system from the field (which is impossible for space system).

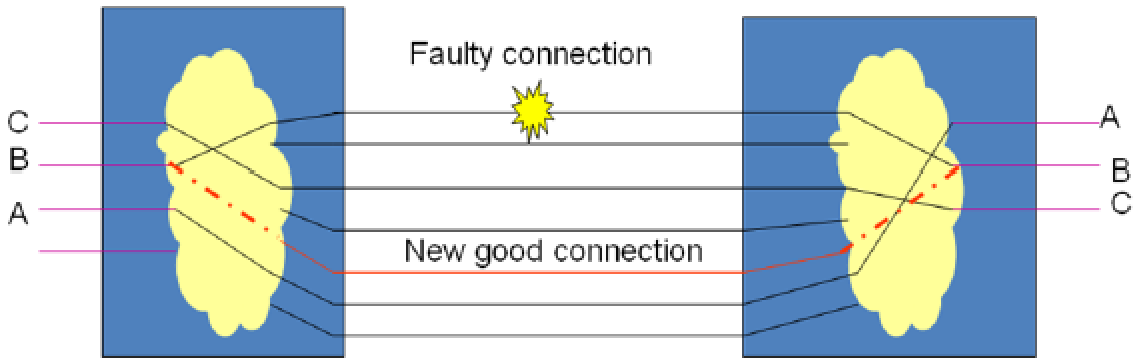


Figure 3 - 6 – Example of adaptive connection system (2)

The second unique advantage of adaptive wiring systems is the ability to form temporary connections for diagnostic and maintenance purposes. Temporary probes can be inserted at normally inaccessible buried nodes within a wiring system and eliminated when no longer needed. This concept is depicted in the next figure. In this case, we use the adaptive wiring system to set up a temporary connection to check a possible problem with terminal C on the right panel.

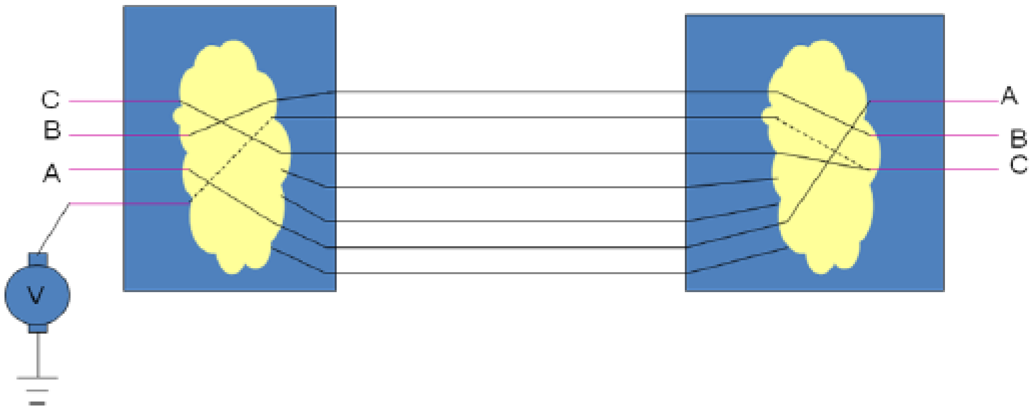


Figure 3 - 7 – Example of adaptive connection system (3)

3.3.2.1.3. Interface module

Software and cabling represent maybe the most crucial obstacles for the introduction of innovative technologies during the satellite development.

The implementation of a BUS like that conceived and presented can partially solve the problem, providing cabling with the possibility of dynamic reconfiguration; in such a way, the network can easily evolve, leaving the designer the possibility of adding and removing components without the necessity of modifying cabling and wiring.

Another important aspect is the possibility to add to the network as much components as possible: for this purpose, the use of interface module, providing a common interface for easily adding sensors and actuators to the network, is of outmost importance. In addition, the use of plug and play protocol provide the network with a self-organizing capacity: such capacity not only gives the networks an easy way to be augmented and modified, but also make them failure-tolerant: in fact, the distributed intelligence and the absence of a centralized handling system allow all the nodes of the network to perform tasks, so any nodes failure can be bypassed through network re-organization; in addition, the data can be routed through many different path (as already seen), making a single connection failure a problem of minimal importance.

The ports are distributed over the satellite , and the components can be simply plugged-in, and the self-configuring/self-organizing logic automatically add the components to the network; this automation allow a rapid integration, leading, at the same time, to a drastic reduction (up to the complete elimination) of human interpretation, that is usually cause for errors. The automation of processes like those of network configuration and organization can favor the development of a responsive space capability.

The interface modules are, as a consequence, elements of extreme importance for a PnP infrastructure, as they allow interfacing traditional components with the PnP networks; these modules have two main tasks:

1. Handle and supply the client device (the device connected to the interface module)

2. Allow the connection of the client device with the PnP network, enabling the client device to correctly operate in it.

In such interface modules are contained the electronic datasheets describing the functionalities performed and the commands accepted by the client device. The main characteristic that this kind of module needs to have are:

- Mass memory: a memory for storing information (primary the electronic datasheets)
- USB interface: An interface for the connection of USB devices;
- Digital Analog I/O: as for the USB, they must be able to manage this kind of communication protocol
- Power management: such modules need to have a system for receiving in input the power and supply the client device; they receive the power from the port where are connected, but they also have to manage the electricity supply of the client device
- Central processing unit: it would be a wise behavior to have a MCU (microcontroller unit) in any node of the network, so a central unit is required also to interface module.
- Clock management: as already said, for the correct synchronization a universal notion of time are required, and, thus, needs to be managed by the single node of the network

Electronic Data Sheets

The enabling mechanism for achieving really plug-and-play (PnP) capability in space systems pass through the development and implementation of Electronic Data Sheet (EDS). Every hardware device or software application used within a PnP system must have an associated self-describing electronic data sheet that fully explains the component (either device or application) to other components in the system. Similarly to what happens when an USB device is plugged into any personal computer, it is necessary that a device (or application) added to space PnP network self-describe itself, its attribute, functionalities, commands accepted and data transmitted, in order to keep the "system aware of its presence".

In the following figure, the comparison between a space Plug-and-Play component and terrestrial USB device are shown: the electronic data sheets replace what in consumer electronics the driver does. But, from the following figure another important aspect is highlighted: the interface module, used to connect a "normal" device to such a network, without the development of tailored hardware and cabling.

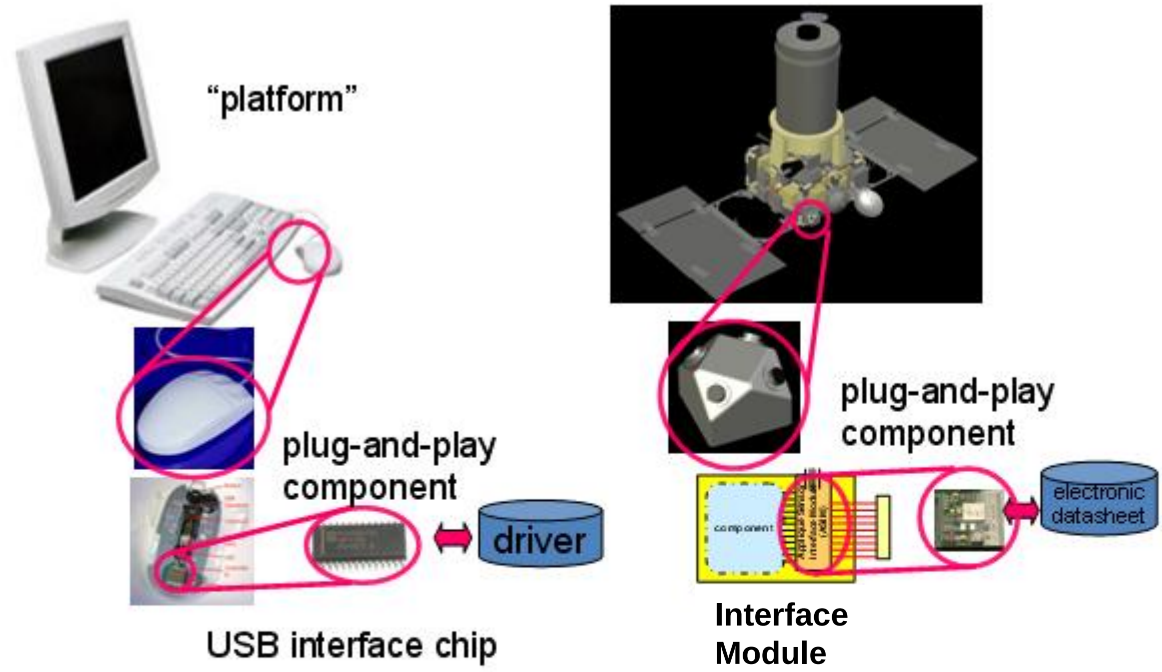


Figure 3 - 8 – Comparison between terrestrial and space PnP concept

The electronic data sheet contains descriptions of all component-specific commands accepted, variables produced, and data messages that can be delivered by the component. It fully describes the services or data provided by the component and represents the complete protocol for accessing these services or data.

The electronic data sheet uses the eXtensible Markup Language (XML) to provide a schema-controlled language for the data sheet.

A common set of terms shared by all PnP applications allows for the creation of electronic data sheet that can be understood and accessed by components throughout such a system. Descriptions of data products within data messages are constructed from a common data dictionary (CDD) of standard terms. Terms used in the CDD must be easily recognized by the system developers, unique for each variable type, and non-duplicating.

3.3.2.1.4. Reconfigurable Radio

The future space communication equipment should be able to provide:

- Compatibility with past and future communication system
- Identification and protection from ground based disturbance for safety of communication

The idea of a Software Reconfigurable Radio (SRR) embraces these concepts, making this system completely compatible with the features of system re-configurability. Such a device, in fact, needs to be equipped with a high bandwidth receiver able to shift the frequency; this allows the satellite to communicate in a large bandwidth, ranging UHF to Ku band.

Another capability that such devices need to have is the ability to detect, identify and geo-localize possible source of disturbance and its emitting frequency; in such a way, shifting the operating frequency and the waveform, it will be possible to bypass the disturb, keeping a high quality of service. This functionality can be useful either for bypassing communication jamming, but also to counteract possible on-board interferences that could occur.

3.3.2.1.5. Switch Fabric Approach

Switched fabric, also named as switching fabric, is a typology of network where nodes are connected each other through switches: such approaches are Internet-like, meaning that a node can be connected to hubs, and, in turns, hubs can be connected to each other. The presence of multiple physical links offers a faster data rate, as the traffic among nodes is spread across multiple connections. The introduction of very high-speed connection was one of the motivations for the development of switch fabric approaches.

In such a way, through the use of these building blocks, network of arbitrary size can be created; switch fabric systems rely on packet based communications. Intelligent routing approaches examine the structure of these packets to facilitate moving them from sources to destinations. High-bandwidth links and non-blocking crossbar hubs with large numbers of ports contribute to the construction of extremely high-performance networks. Examples of switch fabrics include Infiniband, RapidIO, Myrinet, and Spacewire.

The use of switched fabric approaches is crucial in the implementation of a responsive capability: it can allow for rapid integration of components into 'in-developing' network; it can allow for late substitution or addition of components into satellite architecture. The potentiality of such a philosophy needs to be coupled, however, with newer procedures of equipment developments that enable such a possibility.

3.3.2.2. Reconfigurability

In the architecture proposed in the previous paragraph, has been introduced the important concept of re-configurability, intended as the ability of exhibiting diverse behavior through the only execution of remote software command; this characteristic is extremely desirable for any space system, given the obvious impossibility to physically operate on the satellite.

From a *responsive space* point of view, instead, the concept of re-configurability assumes an increased importance: in fact, it can play a crucial role in accelerating the component assembly and integration. Moreover, once in orbit, the system functionality can be modified in an adaptive way, re-programming the components to respond to new mission needs, or varying the configuration to get around possible component failure. In other words, combining electronic components and reconfigurable system we have an “adaptive space system”, able to satisfy several different mission objectives. A space system designed in a traditional manner would have required a fabrication tailored to specific requirements, and should be capable of satisfying only one or few objectives.

Reconfigurable components, furthermore, could be stored in a warehouse, ready to be rapidly integrated and programmed. A classification of reconfigurable/reusable components is presented:

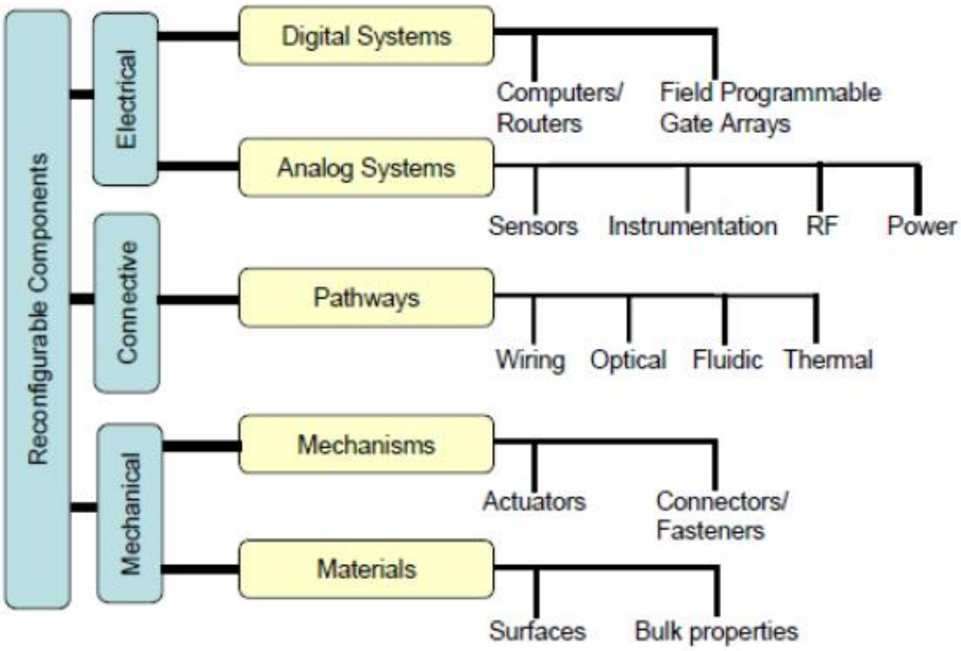


Figure 3 - 9 – Reconfigurable Components

- *Digital System* – Such class of system are the simplest reconfigurable system: example are the FPGAs (Field Programmable Gate Arrays) that are digital devices which functionalities can be programmed via software.
- *Analog System* – Such systems are generally linked to sensors and actuators; differently from the FPGA, an analog device is not able to implement any analogic function, but, using gain, impedence and filter characteristic that are programmable, it is possible to incorporate several function in a single circuit;

- *Paths* – Links (wired or wireless) are essential to connect components, that would otherwise be just a group of independent and not-communicating elements: cabling is, thereby, indispensable. Reconfigurable paths allow to handle links among components, providing the flexibility already mentioned (as the capability of getting around a failed component);
- *Mechanisms* – Example of reconfigurable path is the flexible adaptive optics in phase of development;
- *Materials* – It can be considered the possibility to alter the properties of particular materials, like the emissivity, or the thermal conductivity.

The reconfigurability represents one of the most innovative concept at the base of a responsive system; talking about reconfigurable system, it is not always clear *how much* the system is reconfigurable. The concept of an adaptive hierarchy may be useful in understanding the different levels of '*adaptiveness'*. In the following, description of levels of *'adaptiveness'* is provided:

- ***Fixed systems:*** This is the lowest level of adaptiveness, basically the non-existence of reconfigurability. A system can only operate in the way it is designed for.
- ***Programmable systems.*** A programmable system can be thought of as a system with a certain number of control button that can be "turned" on and off via software. Programmability at its most basic level is a static concept. Software is compiled, FPGAs are synthesized, etc. At this most basic level, systems are configurable. The button can be pressed, but not agilely and all at once.
- ***Context-Switchable systems.*** A context-switchable system is a more dynamic extension of basic programmability. Here, it is possible to press the control button more agilely, but usually in a binary way, like an "all-or-nothing" proposition. The closest analogy is the jukebox in which many different songs (analogous to "programs" in a reconfigurable system) are available. The songs can be alternated, switched back-and-forth, but, one the jukebox has been "closed", not otherwise changed.
- ***Self-configurable systems.*** This level of *'adaptiveness'* is the successive level in the scale: it is represented by the possibility of not just compiling programs, but also to "fly the compiler". A self-configurable system can, following certain well defined rules, build its own programs. This concept is not as exotic as it seems: boundary conditions, detectable by a system through dedicated sensors, can lead to the automatic generation of a section of code, which can be then compiled and executed. Remote systems take advantages of the ability to do their own compilation, saving time if distance and bandwidth are limited.
- ***Evolvable systems.*** When a system can compile or alter its own behavior, and the compilation can be iterative (tested and repeated), the processes are analogous to training. Evolvable systems are systems that formulate and reformulate their own configurations in order to fulfill a target objective. One simple familiar example is artificial neural networks, whose weight settings are tuned algorithmically in an attempt to converge on patterns of a given training set.
- ***Cognitive systems.*** This represents the highest level of *'adaptiveness'* presently conceivable: a cognitive system is a system that can partially manage its own configuration processes, in order to be compliant with high-level mission objectives and/or mission profiles, that could also change during the system lifetime.

The last two level of *'adaptiveness'* are of extremely interest in the development of responsive capability: a system capable to change its configuration during lifetime, according to variable objectives, could also be rapidly designed to respond to suddenly arisen needs.

3.3.2.3. PnP Networks

A network based on plug-and-play concept needs a series of technical characteristics that allow and facilitate its application in space systems; such characteristics are:

- ***Distribution***. Distributed network needs no centralized control: the tasks of network control, as well as the tasks the network must complete are executed by participants of the network. Putting the centralized control to one side help avoiding the problem that such architecture entails: the amount of software and wires located in one single place, and the number of errors and difficulty in testing related software, as well as the existence of a dramatic single point failure.
- ***Amorphous.*** The users should not be concerned over the shape of the network, in the same way of consumers that are not worry about the shape of the power grid when they plug a device into a wall socket.
- ***Time and space.*** Important feature for a distributed network is the exact knowledge of a precise notion of time and spatial location. For a GNC system, for example, the knowledge of sensors and actuators location is of outmost importance, as well as a unified notion of time is needed for the correct execution of all the operation in the scheduled order.
- ***Hot-swappable.*** The nodes of this network must be dynamic: this means that it should be possible to modify the network morphology, by adding or removing components, without compromising the correct behavior of the network.
- ***Fault tolerance.*** The concept of hot swapability itself entails a certain notion of robustness: in fact, providing the network with the capability of dynamic reconfiguration, failed components can be substituted by a network reorganization. Anyway, such a network should also embrace fault tolerance and other concept of robust design.
- ***Coherence and unity.*** The distributed network should appear as a single, unique system, and the application design not necessarily need to know the entire system, but must operate over the idea of a virtual service, which functions are performed by real devices.
- ***In-situ Reconfiguration.*** The ability of updating the software of any component of the network is critical for providing the possibility of errors correction, and system augmentation, during integration and/or normal operation.

3.3.2.4. PnP for Space Application

Several studies have been undertaken in the recent past with the aim of developing a system that would permit the rapid integration of a space system. Such studies envisage to adapt for use in space technologies developed for terrestrial application and that gathered an incredible diffusion due to their characteristic: the technologies related to plug and play. These technologies, if introduced in space systems, colud have a significant impact on the time necessary for designing, manufacturing and testing space platforms, in particular microsatellite.

In the field of consumable electronics, the diffusion of PnP protocol caused the industries involved to launch an incredible amount of new products on the market: any peripheral, that up to the moment of introduction of PnP protocol needed to be connected directly to the motherboard, while the computer was switched off, and once mechanical operation of removing part of the case has been executed, was realized with this technology, allowing a reduction of time necessary to mount, use, and un-mount these peripherals. Hard drive, network interface card, CD and DVD reader, and so on was developed with these standards, enabling fast and rapid substitution, and “on-line” plugging, with the PC switched on and without any type of installation. Furthermore, the installation of a new device in an old PC implies the installation of devices’ drivers, to allow the bus to communicate with this new devices. The USB standard enable the possibility to plug a device into the main bus, and after a few second, to have the device installed and perfectly working: this happens because USB components are self-installing and self-configuring. The diffusion of such technologies in terrestrial application was due to the presence of a standard interface and the existence of a common protocol of data transfer.

In order to provide the capability of a quicker integration of satellite components into the final configuration, a similar approach could drastically reduce the time necessary to integrate devices adequately developed: a PnP approach for space systems would not only facilitate the assembly of components properly manufactured, but would also allow having a network easily expandable and modifiable, and an higher level of robustness of the space system, as the network could better face component failure. In the next figure, and hypothesis of a satellite manufactured exploiting the capability of PnP technologies.

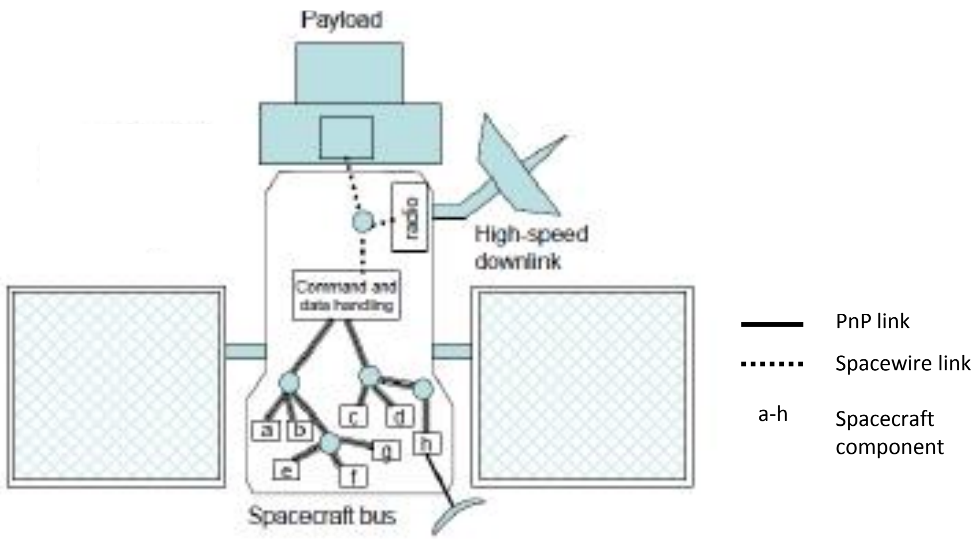


Figure 3 - 10 – Example of a PnP satellite

Together with the PnP approach, that facilitate the integration of the satellite, also a switched fabric approach is necessary: the network must be conceived as an Internet-like network, meaning that it would have an amorphous topology.

The example of plug-and-play technology is a clear example of how the components and the architecture need to be intended; anyway, it is important to underline how this technology, how has been conceived, is not perfectly fitting all the requirements that a space project has to deal with. The PnP technology, in fact, does not embrace important attributes for space application, such as:

- **Environment/Fault tolerance:** reliability is an important concept, even for low-cost microsatellite; the electronics needs to be fault tolerant, and capable of surviving the harsh space environment.
- **Synchronization:** a unified notion of time is necessary for correctly execute all the commands in the proper way
- **High Power Delivery:** traditional USB technology for terrestrial application does not support level of power and voltage typical of space systems. Several attempt has been done, like those of PoE (Power over Ethernet), trying to couple power supply and data transmission, that witnesses how such argument is kept in mind; unfortunately, PoE technology does not support the voltage of 28V, typical of many satellite.
- **Driverless PnP:** the terrestrial version of the PnP protocol envisages the use of driver: in most of the cases, the drivers are stored in a small memory present in the device, or, alternatively, looked for on Internet. In a space system, the development of drivers is really troublesome, due to the necessity for any component of supporting a wide variety of configuration. In an attempt of extending PnP principles for space activities, the plug-and-play for space application needs to be driverless: in such a way, the information necessary for access the service of the device are settled in a universal way, allowing the operative system to control and use the device without to know how the device works, but communicating via a standard interface.

Such attributes are crucial for the implementation of a capability of rapidly developing, integrating, and testing a space system. As done with the technology in the previous chapter, terrestrial standard has been considered, in order to detect their intrinsic limitations. Once these limitations has been identified, a software infrastructure has been conceived, that must collect all the suggestions and warning on how to implement such a capability.

An architecture for a PnP system needs to have:

1. Capability of autonomously identify components;
2. Flexibility, in order to satisfy the requirements of a responsive mission;
3. Robustness.

In order to provide to space PnP network with these characteristics, a set of standards that have been considered with the aim of identifying their limitations, and finding a way for solving such limitations. The standards analyzed has been chosen finding among those having characteristic as similar as possible to what we’re looking for: USB, Spacewire, Ethernet and other standards.

Once these standards have been analyzed, a software infrastructure responding to responsive needs has been devised, equipping such infrastructures with the features of automatic component identification, independent exchange of tasks among devices, robustness and flexibility.

The first standard analyzed is the USB: such standard support data transmission up to 1.5 Mbps (USB1.0), 12 Mbps (USB 1.1), 480 Mbps (USB2.0) or 4.8Gbps (the most innovative USB3.0). Investigating the differences among these versions, the most suitable for space application could be chosen: the main difference (apart from the speed in data transmission) is the level of power that the BUS can guarantee to USB peripherals connected: USB 1.0, 1.1 and 2.0 can provide a maximum current of 500mA, with a voltage of 5 Volts; USB 3.0 is able to provide a maximum current of 900mA, keeping the same level of 5V.

As none of the different technology is able to support high level of power, this aspect will not be a discrimination in the choice of the appropriate standard to investigate. With respect of the four pins used for USB (up to 2.0 version), a “space USB” standard should have additional pins, in particular it will need 2 pins to provide power supply, and 2 additional pins for the synchronization, that conventional USB does not properly support.

Being conceived as an extension of the traditional USB standard, also the “space USB” network can be composed of three elements:

1. Host
2. Device
3. Hub

The host is the master device, the only authorized to start and terminate a communication over the bus; the device is any “Space-USB” peripherals connected to bus which is not the host (hub included); the hub is a device that allow multiple peripherals to be connected to a single port.

The communication standard “Space-USB” so conceived allows having the necessary support for the power supply and synchronization of satellite components.

The second standard taken in consideration has been the Spacewire: it is a standard devoted to spacecraft communication, which is in part based on the IEEE 1355 standard, coordinated by the European Space Agency[4] in collaboration with NASA, JAXA, and Russian Space Agency and defined in the ECSS. Such standard has been extensively used in development of space system as it allows speeds connection between 2 and 400 Mbps; such a standard guarantees a bidirectional, full-duplex connection, allowing the devices to communicate in both direction, and simultaneously (example of full-duplex connection is the one between telephones). Such connection is generally used for communication with the payload as it assures, apart from high speed in data transmission, also routing, flow control and error detection in hardware, with minimum necessity for software to perform such operations. The disadvantages of such a connection mainly rely on the fact that this is a point-to-point serial connection, which allows direct connection and communication.

Technical solution for adapting this standard (born for space application, so more suitable for a Space-PnP architecture) is that of using the concept of switched fabric approach, in order to have adjustable connection among devices. Such connection can guarantee a higher level of power with respect to USB, being capable of manage up to 40 Ampere. Last consideration over such typology of connection is that the spacewire network are peer-to-peer, meaning that there is no client-server logic, but any device can start and complete a transition.

3.3.2.5. Space-PnP Network

The PnP for space applications necessitate the implementation of standard of data exchange, and, most of all, a software architecture that implements software for commanding the plugged devices and data gathering, exchanging, processing and analysis. This architecture must be capable of harmonize modularity, standardization, and common interfaces in order to treat any single component of the system as a black box.

What is different between such a proposed approach and the "terrestrial version" of the PnP is that the black boxes must contain the information necessary for their discovery, being capable to autonomously aggregate the network and the other devices.

These information need to be contained inside the components, and must regard:

- Commands: the list of commands that the components can execute;
- Variables: all the variables that components is able to produce once commands were executed;
- Messages: that the components can send in response of particular commands.

Such a logic allows the satellite developers to rapidly assemble a satellite, simply collecting and integrating the correct components, that are self-discovering and self-organizing. The components can autonomously configure themselves as not simply a collection of components, but as a system in which the components co-operate, as happen in any satellite. The difference is that, in this architecture, no specific hardware, cabling or software are necessary; the components are capable to co-operate by simply connecting themselves. Such black box can be different one another and a small synthesis of the elements constituting this network is:

- ***Normal Device***: it consists in any device (reaction wheel, temperature sensor, etc.) that can be integrated into a PnP network through the use of a device, an interface module, that handle the normal device, providing commands and supplying power. These interface modules, by means of specific files, called electronic datasheets, will communicate with the bus, receiving the commands that the device accepts, and providing data and information that the device produces.
- ***Command and Data Handling computer Module***: this device handles the network and all the devices plugged; it looks after device discovery and service handling, management of information, transmission of command and data among PnP device.
- ***Connection Module***: it works as an hub in Ethernet networks, allowing to expand them by providing additional port where connecting devices.
- ***Bridge Module***: such a component could be useful if more than a single standard is used in the architecture. For example, bridge module could be used if Spacewire devices should be connected to PnP device.

3.3.2.6. Satellite software Architecture

The desired modularity for space systems needs to be coupled with the capability of developing software application before than mission requirements, the satellite and its components are known. In order to facilitate the simultaneous and independent development of hardware and software application, software architecture needs to be developed: such architecture will allow the integration last minute of hardware and software, supporting the self-configuration and self-discovery. When discovered new connected components, it will assign a rule and a priority to any of them; in such a way, no flight code needs to be developed, as the system is capable of discover all the components, recognize their rules inside the network, and the software architecture has the task of commanding each of them in the proper way. In addition, the software infrastructure can encompass functionality of fault tolerance to implement after the losses of device, losses of software applications and components.

This architecture rely on the electronic data sheets, files containing the basic information of any device, like commands accepted, variables produced and so on; in this electronic data sheets are contained all the information that allow the devices to be self-discovered, to organize and configure itself, allowing, this way, the architecture to manage the PnP network.

Complex application are based on the exchange among devices of data contained in the electronic data sheets; so, these applications necessitate the information that, at system level, the devices exchange, so must be developed exploiting the information contained in the electronic data sheets to effectively be plug and play. The satellite architecture so conceived exploits the concept of data-centric architecture, since the attention is concentrated over the data provided by devices or used in the process. In such a context, a process can be a software application, a PnP devices, or any satellite user or device that produces data on board satellite. For letting the processes communicate each other, they need to have a common dictionary, they need "to speak the same languages"; for this reason, the architecture must be accompanied with a common data dictionary.

The software architecture will act this way: once a new component is plugged in, its electronic data sheet will be register in a section of the main architecture, in order to maintain a database of all the electronic data sheets of the components of the network, in order to have a dynamic database that is updated any time a new component is plugged. Keeping this database updated, also the functionalities that any component is able to provide (i.e. the functionalities of the space system) are kept updated. In this software architecture, computational capabilities will be assigned to different functionalities: there will be capabilities devoted to discovery of new components, that will elaborate the electronic data sheets, keeping updated the dynamic database; there will be computational capabilities devoted to the data transfer from place where are stored and components that requires such data; additional capabilities will be devoted to the co-existence of different type of standard, like the SpaceWire and this introduced new PnP standard.

The software infrastructure on the basis of PnP standard support the automation of designing, integration and configuration processes without which the rapid development of a space platform would not be possible.

3.3.2.7. Conclusion

As a conclusion, elements that support the concept of responsive space have been identified:

1. Modular intelligent components: the incoming devices must incorporate software-definable components, so that the properties can be adapted to mission specific objective (i.e. software definable radio)
2. Minimization of custom-made cabling: cabling is low-level technology, but entails several problems and can cause serious delays in development of complex systems. Nonetheless, they are necessary as, even in presence of wireless approach, they are necessary to transfer the power. The AWM has been designed and conceived for the purpose of supporting the rapid integration
3. Reduction (or complete elimination) of mission-specific software: the writing of the flight code significantly affect the development of a complex system, and can cause delay in time and overrun of the prescribed costs. Until traditional logic of development of tailored code for specific mission objective will be pursued, very few opportunity of code re-use will be available: in addition, the software is strongly affected by phenomena of diseconomies of scale, causing the non-linear cost (and computational resources) increment if lines of code increase. The use of peer-to-peer network allows maintaining the line of code to sustainable level. By means of appliqué sensor, we can obtain self-organized network, where each sensors, actuators, and any other generic component, are equipped with its own intelligence, and are able to recognize the network and establish their role. In such a way, can be created network of arbitrary complexity without the necessity of tailored system

These cited elements are the main factor that would allow the rapid construction of space system, exploiting the potentiality offered by a modular plug and play approach; they are, thereby, the focal ingredients for acquiring a *responsive* capability.

The proposed PnP architectures devoted to space application exploit self-discovery mechanisms similar to those found on personal computer: such mechanisms simplify the integration of modular components, allowing a faster system development (in the first place). The concept of providing the components that will constitute the satellite with a proper intelligence underlie the technology of the plug-and-play devoted to space application: such concept, apart from assuring a quicker system integration, offers a reduction of the uncertainties, while improving robustness (for the existence of multiple path and a distributed intelligence) and flexibility (the system can be commanded via software to modify the configuration in consequence, for example, of new, different mission objectives. In the implementation of such a capability, a high importance is placed in the development of interface modules: such modules can to be thought as advanced adapter, that handle the client device and simplify the necessity of common devices to uniform themselves to network standards.

3.4. Responsive Space – Launch Segment

3.4.1. Innovative Launching Capability: Air Launch

The concept of Responsive Space, and the noteworthy advantages related to the implementation of such a capability, is based on several aspect, that need to exist simultaneously.

In the previous paragraphs, we cite the importance of having the capability of rapidly manufacture and assemble a satellite; but such a capability represents just one side of coin. In fact, also having the capability to fully develop a space platform, traditional launching methods are not compatible with the timeline typical of a *Responsive Space* project: they need to be booked largely in advance, in the order of couple of years.

It is evident how such timeframes are not compliant with responsive necessity, where mission requirements can impose months (or even weeks, according to the service to provide) as time limit for the deployment of space service. Furthermore, even in presence of the capability to schedule a launch with very restrict advance, exist several factors that can undermine the respect of the launch date: the Shuttle mission STS-101 of May 2000 illustrates the problem with surface launchers. Its ISS logistic mission was delayed by total of 25 days due to a combination of waiting for suitable launch weather, waiting for other launch vehicles that shared the same range resources, and waiting for the ISS orbital plane to pass overhead the launch site.

In order to be able to satisfy the requirements related to responsive launch, new methodologies of in-orbit placements needs to be investigated.

However, also another important aspect suggests the research of alternative methods for launching satellites. Even if the attention for small (and also micro) platforms raised considerably in the last decade, and even though, approximately in the same period, many initiative arose aimed at developing launchers smaller than those existing (the European VEGA and the U.S. Falcon 1 are valuable examples), launching small satellite (and even more micro-satellite) can be problematic: in fact, the launchers developed have launching capability significantly higher than those required by micro-satellite. This means that the microsatellite developed will never be the primary payload of the currently existing launching system, but will inevitably be “relegated to be piggy back payload”; but this means, even disregarding the issue related to the launch date, that the microsatellite developers cannot have any influence on the decision of the releasing orbit: the microsatellite will be released approximately over the same orbit of the primary payload. Such a situation imposes dramatic limitations to the capability of the microsatellite that has been designed, as however happen for any other satellites, maximizing the performance for a specific orbit. Such limitations can reveals to be dramatic for any mission based on microsatellite. In fact, the really limited mass of such platforms allows to be able to perform significant mission only if several aspects are present, and one of these is the possibility to place over the designed orbit, that rarely will be the same that a satellite with a lifetime of many years should fly.

Therefore, an alternative methods to launch microsatellites, is extremely important for giving such platforms the possibility to demonstrate their potentialities.

There are many potential launches opportunity for small satellite launch vehicles (SLVs) made possible by miniaturization technology in the past 50 years. Today's mantra of ''*Faster, Better and Cheaper*'' has persuaded mission planners to rethink how big a satellite needs to be to accomplish a task. Cost and flexibility are two of the most salient factors influencing this way of thinking. Marketing survey claims that 95% of performance of large satellites can be reached with small satellites at 5% of the cost or 70% performance at 1% of the cost [5].

In recent years, small launch vehicles have become the leading option of launching satellites to LEO due to their low cost and high operational effectiveness [6].

One of the possible ways of launching small and micro satellites with considerable benefits would be having an air launched vehicle. Such an air-launched concept consist in the possibility of using an airplane to carry at a certain operative height and at a certain initial velocity a launcher vehicle, that will fly its payload into final orbit. In practice, the launching system is composed by an airplane, which acts as the first stage of the system, and a rocket, that will be released to certain conditions (height, velocity).

This conception comes from a simple observation: the velocity that an object in a 200 km circular low earth orbit (LEO) must have is about 7.8 km/s; launching from a certain altitude (and, possibly, with a certain initial velocity) is better than launching from the surface, with a null initial velocity, as it happens for traditional surface launches. Of course, this simple consideration must be coupled with deep studies, as the situation is not so simple and immediate.

In order to provide the altitude and velocity for orbiting a space system, it must be accelerated, and taken to the proper orbit: in other words, is necessary to spent energy. But the energy necessary to carry the satellite into orbit from the Earth surface is higher than that necessary if the launch occurs from certain height and velocity condition, due to the losses that arise during the launch: gravity loss arises because part of the launch vehicle's energy is wasted in holding it against the pull of Earth's gravity; drag loss is another loss and is caused by the friction between the launch vehicle and the atmosphere.

The possibility of air-launching offers several advantages; some of these intrinsic benefits include:

- Providing mobility and deployment advantages;
- Providing flexibility in the choice of launching location;
- Offering the potential for aircraft-like operations that provide responsive launch on demand or launch on schedule;
- The propellant mass of air-launching vehicle could be significantly reduced as compared to conventional ground launched vehicles;
- Air launching also increases propulsion system's efficiency by using a large area ratio nozzle;
- Launch at altitude can reduce gravity and drag losses during initial flight;
- This launch method can take advantage of the initial speed of mother plane.

Air launching provides mobility and deployment advantages over surface launching; air launch systems can fly over or around weather condition that impedes surface launches; they can chase orbits and achieve any launch azimuth without out-of-plane orbital maneuvers that consume large amounts of on-orbit propellant; air launch systems can operate free of national range scheduling constraints, have minimum launch site requirements, and they may have reduced range safety concerns. Air launching significantly reduces the acoustic energy from the engine since there is no reflection from the ground and air density is lower: the strength of the thermal protection system (TPS) and structures near the base of a surface launch vehicle are sized by acoustic energy at launch. Finally, some air launch methods can improve mass inserted into orbit over a similarly sized surface launch vehicle. In addition, air launching systems offer the potential for aircraft-like operations that provide responsive launch on demand or launch on schedule.

However, besides all the advantages that this typology of launching methods offers, such a system is affected by the limitations in terms of mass and dimensions that intrinsically are related to the possibility for the aircraft of carrying (in the way proposed) a launch vehicle.

The air launched vehicle can be categorized into five launch method categories:

1. Captive on top
2. Captive on bottom
3. Internally carried
4. Towed
5. Aerial refueled

In the following, a description of each of these solutions will be provided, coupled with examples of existing past and present studies aiming at developing these air-launch methods.

3.4.1.1. Captive on top

Advantages of this method include the capability to carry a large volume rocket on top of the carrier aircraft. Disadvantages include extensive modifications (which means high cost) to the carrier aircraft; further, placing a launch vehicle on top of the carrier aircraft affect the lift produced by the fuselage and causes a large amount of drag that in turn limits launch altitude. In addition, the rocket must be provided with active control that flies the rocket at release from the carrier aircraft, and wings large enough to support it at separation from the carrier aircraft. No examples of captive-on-top launch vehicles have been actually operated, but several studies has been performed based on this option; the first one appeared in the 1966[7], year in which the development of Russian aerospaceplane Spiral 50-50 started.

Figure 3 - 11 – Captive on Top concept of the Russian aerospaceplane Spiral

The feasibility of such option for air-launch has been demonstrated and guaranteed by the lift difference of the two winged vehicle. More recent example of studies investigating such launching method are the Boeing concept studies (fig. a), the Dassault Aviation project “Vehra” (fig. b) and the Telemaque project (fig. c).

(a)

(b) (c)

Figure 3 - 12 Captive on Top concept: Boeing concept studies (a), Dassault Aviation project “Vehra” (b), and Telemaque project (c)

3.4.1.2. Captive on bottom

The advantages of the captive on bottom launch method include proven and easy separation from carrier aircraft. Disadvantages include limits to rocket size due to under the carrier aircraft clearance limitations and the high cost of carrier modifications. A new carrier aircraft can eliminate clearance limitations. Such *captive on bottom* method is, presently, the only operating method for air-launching a satellite. This services is provided by the Orbital Science company (fig 3-13): the three-stage Pegasus rocket boosts small satellites weighing up to 450 kg into low-Earth orbit[8]; the Pegasus rocket is carried aloft by an L-1011 carrier aircraft to the operative height of 12,000 m, where it is released, and, after a free fall of five seconds, its first stage rocket motor is ignited. Other options for captive on bottom method envisage the use of a different typology of aerial vehicle: at least two different solutions can be implemented as first stage of launching system. A first alternative to airliner could be represented by fighter aircraft, like Rafale, Eurofighter, Tornado, or Mig-31. Such aircraft guarantee an easier integration of the launch vehicle, as they has been extensively designed and used to carry under the fuselage a heavy cargo. Such option has pros and cons with respect to the “airliner” solution. On the one hand, it has, in fact, the ability to maximize the gain due to airborne launch thanks to the dynamic capabilities of the aircraft through the optimization of the kinematics at separation: according to the fighter selected and its flight profile, in fact, at release can be achieved flight conditions more profitable, in terms of flight path angle, velocity and altitude.

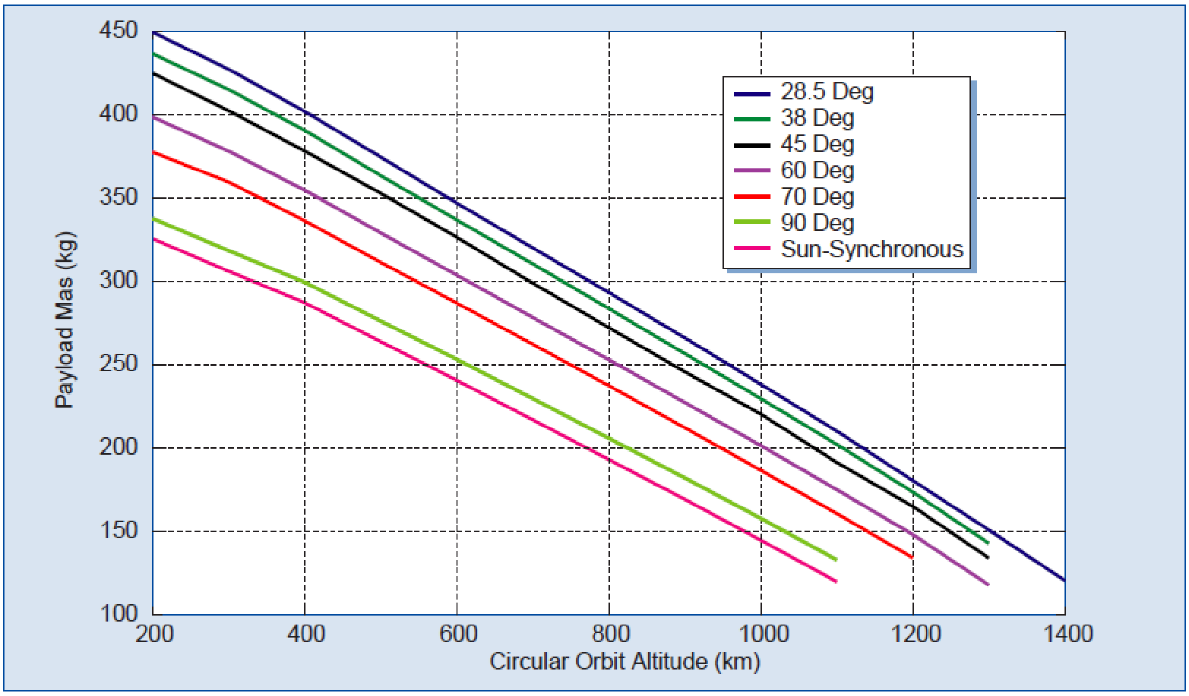


Figure 3 - 13 – Picture and launching performances vs. circular orbital altitude of the Orbital Science launching system

Such capabilities also offer the possibility to imagine a simplified sequence of jettisoning and a launcher without important aerodynamic support (big wings like those necessary for the Pegasus rocket, or extraction parachute like for internally carried method, explained afterwards). Furthermore, no (significant) modifications are necessary for the carrier aircraft.

Drawbacks are the limited mass of the launcher depending of the aircraft capability, and the affordable volume due to the generally small size of the plane and any constraint such as the train trap deployment, aerobrakes, ground clearance, etc. The limitations to the fighter aircraft translate in limited capability of in-orbit injection with respect to the airliner solutions, but also with respect to others typology of air-launching systems. In fact, while the Pegasus is able to insert up to 450 kg in very Low Earth Orbit with small value of inclination, studies over this solution suggest an upper limit of approximately 120 kg for the most capacitive aircraft available. Examples of this solution are provided: the French Space Agency (CNES) studied the possibility to mount a launcher for micro-satellite (MLA: Micro-Lanceurs Aéroportés, Micro Airborne Launcher) under the fuselage of a Rafale aircraft[9].

Figure 3 - 14 – Captive on Bottom: CNES MLA study

Other studies have been also performed by other agencies and industries using other fighter aircrafts, like the Tornado[10].

The second alternative for captive on bottom air-launch is represented by the implementation of an original concept, consisting of a dedicated aerial vehicle (eventually, unmanned), to integrate with the launch vehicle that has the task to inject the satellite into orbit. This is the solution adopted by the Virgin Galactic for the development of the SpaceShipOne (shown in the next figure) and its successor, the SpaceShipTwo.

Figure 3 - 15 – Captive on Bottom: Virgin Galactic's SpaceShipOne

In the picture, the launch vehicle is a spaceplane devoted to space tourisms. In an air-launching system for satellite, it will be substituted by a common launch vehicle. Such idea has been also the basis for the conception of both *Stratolaunch* air-launching system and the Dedalus project (CNES), depicted in the next figure.

Figure 3 - 16 – Captive on Bottom: CNES project Dedalus

The main advantage is the architecture (2-bodies) of the UAV allowing to easily accommodate a Launch Vehicle (LV). Furthermore, no limitations related to previous usage of the aerial vehicle arise, as the entire system needs to be developed from the beginning. And this aspect is also most clear disadvantages, that requires high amount of resources, both time and money.

3.4.1.3. Internally Carried

This typology of air-launching methods envisages the use of a cargo aircraft that allows to transport the launch vehicle for in-orbit injection into its cargo bay. A significant advantage of this typology of launching method includes little or no modifications to the carrier aircraft (lowering both development and operations cost). Most propellant boil-off concerns are eliminated since the launch vehicle is not subject to either radiation heating from the sun or convective heating from the airstream. Maintenance crews have access to the launch vehicle until just before the launch, which reduces the safety concerns of carrying a launch vehicle with a manned carrier aircraft. The launch vehicle is in a benign environment inside the carrier aircraft and maintenance and safety problems can be detected and resolved. Also, internal carriage eliminates weather induced launch failures (such as the Shuttle Challenger) by launching into a known and benign environment (the stratosphere). Such an air-launching method has been proven in the past: on 24 October 1974 a C-5A Galaxy dropped a 78,000 lb LGM-30A Minuteman I missile using drogue chutes to extract the missile and its 8,000 lb launch sled. Parachute airdrop of the 195,000 lb and 92 inch diameter LGM-118 Peacekeeper MX missile was also considered. In January 1997, the second stage of a Minuteman I was successfully parachute airdropped from a C-130. A disadvantage of internal air launch is that the launch vehicle must be sized to fit inside the carrier aircraft.

Also recent examples exist: in the figures 3-17 and 3-18 two different examples are depicted. The first one are taken from a study involving a Boeing C-17 GlobeMaster III carrying a Small Launch Vehicle capable of injecting a 450 kg satellite into 190 km, 28.5° inclined orbit[11]. Such study succeed in both extracting a perfect mock up of the rocket, with the same mass and dimension of the final rocket, and firing (in ground tests) the engines.

(a) (b) (c)

Figure 3 - 17 – Internally carried: AirLaunch Quickreach project

The second figure refers to a recent study, using a C130J to lift the 10m length, 12ton weight launcher, constituted by 3 solid stages and 1 upper stage with storable bi-propellant, housing equipment for platform service provision to the operative payload. The performances are still in evaluation, but the system should be capable of injecting 150 kg in equatorial, 300 km height orbit.

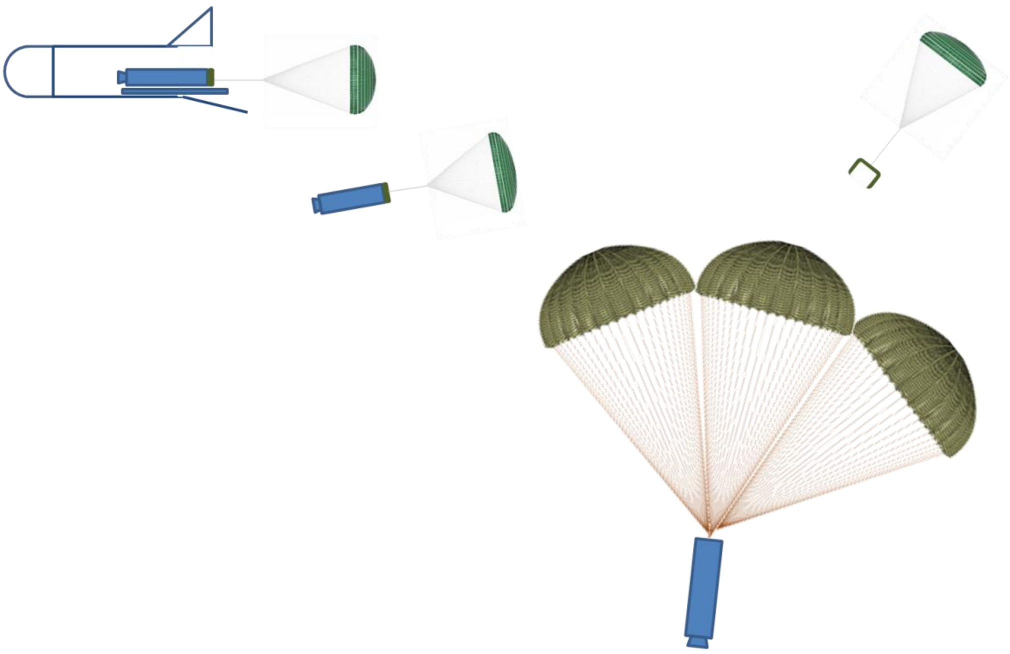


Figure 3 - 18 – Internally carried: AVIO SALTO project

Both the examples show a cargo airplane carrying expandable multi-stages rocket launcher, which are lifted up to approximately 10,000-15,000 meters altitude, then released by the cargo bay through deployment of extraction parachute, stabilized in vertical position by means of the same parachute (or a set of stabilizing ones), and then ignited. The first seconds of the engine ignition are needed to slow down, and finally stop, the rocket descent. From that point on, the launch will occur like a ground-based traditional launch, with the difference that it occurs from a significantly higher altitude. The limitation of such a method is represented, as already said, by the internal envelope of the cargo airplane which imposes constraints over the dimensions, shape and weigh of the expandable rocket, that, in turns, affect the mass of the satellite that can be injected in orbit.

3.4.1.4. Aero-Towed

Another possibility for air-launching system is that provided by aero-towed concept: it consists in towing a large glider that would carry the launch vehicle, as shown in the following figure.

Figure 3 - 19 Aero-Towed Concepts

The main advantages of aero-towed concept are the easy separation, the low-costs modifications of the aircraft and the relatively unconstrained size of the LV. Unfortunately, the drawbacks are also numerous: the most critical ones are related to complexity of conception (wings and gears required, sophisticated flight control during aero-towed phase, abort mission management), but also from technical point of view: the aero-towed vehicle would be towed off of a runway using the thrust of its own jet engines and the excess thrust from a airplane, probably a airliner, acting as a tow aircraft; even if the basic concept is feasible and so complex from a technical point of view, its implementation is obstructed by the limitation in thrust that the tow aircraft can offer to the vehicle to tow.

3.4.1.5. Aerial refueled

The last method for air-launching investigated is the possibility of having a combined jet and rocket powered aircraft, capable of taking off, be refueled and then ; it would use its turbofan engines for take-off, rendezvous, and refueling with an aerial tanker. Once refueled, the launch vehicle is able to perform the procedure for injecting the satellite into orbit. This refueling concept would reduce the size of the launching vehicle’s wings and landing gear up to about 1/2 of the same vehicle that had to carry all its oxidizer at take-off.

Even if reduced size of the carrier aircraft's wing and landing gear. Can be achieved, it must be noted that aerial refueling does not reduce the size of the jet engines – they must be sized to maintain level flight for a fully fueled carrier aircraft.

3.4.2. Final Considerations

The last two methods of air-launching (aero-towed and aerial refueled) are the most complex of those presented, as, differently from the others, envisages the use of a launch vehicle that needs to have the capability to take off and flight, which means they must be provided with wings, causing an increment of launch vehicle mass, reducing the capacity of in-orbit injection. Furthermore, such vehicles needs to face additional structural and aerodynamics problems related to these additional capabilities, which drastically increase their complexity, causing the impossibility to solve all the issues with feasible solutions from a technical but often also economical point of view[13]. In addition, as these vehicles needs to fly, they must have wings and a shape similar to that of the Space Shuttle, which implies additional problems for the operation of satellite release. As last analysis, it is evident how spaceplane like the Space Shuttle demonstrate their disadvantageous behavior if they has to act simply as a launch vehicle.

Thus, the solution that envisages mounting a rocket, similar to those conventionally used for surface launch, seems to be economically affordable, and technically and technologically easier. Anyway, the typology of air-launch (internally carried, captive on top or bottom) and the typology of aerial vehicle used influences also the design of the launch vehicle: the launch vehicle of a captive on bottom method using airliner needs to have wings to nose up as the launch vehicle is released in horizontal direction; furthermore, loads that arise during the phase the rocket noses up needs to be taken into account, differently from an internally carried launching method, where the LV is vertically ignited.

Figure Index

REFERENCE

[1] – T. Kuwahara, K. Yoshida, Y. Sakamoto, Y. Tomioka, K.Fukuda, *Satellite system integration based on space plug and play avionics*, 2011 IEEE/SICE International Symposium on System Integration, SII 2011 , art. no. 6147568, pp. 896-901

[2] – K. B. Center, D. C. Fronterhouse, M. Martin, *The Software Strategy for SPA Plug and Play Spacecraft*, 2010 IEEE,IEEEAC paper # 1348.

[3] – N.C. Anderson, G.G. Robinson & D.R. Newman, "*Standardization to Optimize Integration and Testing*", AIAA Responsive Space Conference 2003, AIAA RS3-2005-4005

[4] – spacewire.esa.int

[5] – Wei Sun, Proceedings of the IAA Symposium on Small Satellites for Earth Observation, Berlin, 2001.

[6] - S.J.Isakowitz,"*International Reference Guide to Space Launch Systems*", 3rd ed, American Institute of Aeronautics and Astronautics, 1999.

[7] - http://www.buran.ru/htm/molniya.htm

[8] - http://www.orbital.com/NewsInfo/Publications/Pegasus_fact.pdf

[9] – C. Talbot, E. Louaas, P. G. Gotor, A. R. Merino, L. Froebel, *"ALDEBARAN", A launch vehicle System Demonstrator"*, AIAA/7th Responsive Space® Conference, Los Angeles, April 27–30 2009

[10] – Rosati, Gatti, *"Air Platform and Missile integration challenge for Launch on Demand"*, Workshop on High Tech Small Sat Mission, Centre for High Defence Studies. Rome May 5-6 2011

[11] – M. Sarigul-Klijn, N. Sarigul-Klijn, *"Trade Studies for Air Launching a Small Launch Vehicle from a Cargo Aircraft"*, AIAA Aerospace Sciences Meeting and Exhibit, January 10 – 13, 2005

4. Concurrent Design Tool

The concept of technological innovation does not simply refers to innovative components, or novel devices able to guarantee improved performance with respect to components traditionally adopted in the field of space engineering; technology innovation refers also to innovative methodologies for the deployment of a space program, methodologies for more accurately and rapidly be able to design a space platform, or processes of manufacturing and assembly.

In the next paragraph, a profound attention will be done on the methodology of Concurrent Engineering.

4.1. Concurrent Engineering

Concurrent Engineering, which is also known as Integrated Product Development (IPD) or Simultaneous Engineering, was defined by the Institute for Defense Analysis (IDA) in a 1988 report as

> *"A systematic approach to the integrated, concurrent design of products and their related processes, including manufacture and support. This approach is intended to cause the developers, from the outset, to consider all elements of the product life cycle from conception through disposal, including quality, cost, schedule, and user requirements".*

Concurrent Engineering is not a method for rapidly fixing design's problems, and it's not a way to improve Engineering performance; on the other hand, it's a strategy that addresses important resources. The major objective this strategy aims to achieve is to improve product development performance. Concurrent Engineering is a long-term strategy that involves major organizational and cultural change in the organization of the development processes.

Traditional *serial development* envisages the product first completely defined by the design engineering department, after which the manufacturing process is defined by the manufacturing engineering department, etc. This approach is usually very slow, costly and low-quality, due to the fact that a problem discovered during the manufacturing process, or during a successive phase, entails that the design (and all the processes and phase prior the one who face the problem) need to be performed again, with a significant waste of time and money. This situation leads to a lot of engineering changes, production problems, and in delays product development.

An evolution of the serial development is the *centralize design*: in such a centralized organizational structure the decision-making authority is concentrated in a single executive team, with information flowing from the people of this team to various business and engineering units. Such methods is more effective than the serial development, as problems, errors and discrepancies arising between two areas, being engineering, or manufacturing, or even managerial, are discovered earlier, since the development proceed in parallel. The problem with this organizational structure is the responsibility the single executive team has towards all the other areas involved in the project, together with the required expertise in any of these fields, in order to deal with any of the different teams. In addition, immediate and direct communication between two areas is not so appreciate, as the information needs to pass through the “central” team.

An innovation with respect to these organizational structures is the Concurrent Engineering, which brings together multidisciplinary teams, in which product developers from different areas of engineering, manufacturing and also product assurance work together and in parallel from the beginning of a project with the intention of getting things right as quickly as possible, and as early as possible. The cross-functional team required for performing *Concurrent Engineering* design contains expertise of different functions and areas of competences, such as systems engineering, mechanical engineering, electrical engineering, manufacturing, reliability and maintainability, as well as product development and quality assurance, cost engineering and program management.

The concurrent team can be either composed by only engineers (manufacturing, design, production, etc.) or also representatives of different areas like purchasing, marketing, quality assurance; in some cases, also costumers are present in the team.

The approach typical of Concurrent Engineering methodology helps to have, from the beginning of the process of product development, a clear understanding of what the product requires in terms of mission performance, environmental conditions during the operation, budget and scheduling: this is achieved through a clear definition of any relationship among the different functional areas. The teams acting together at the beginning of the workflow can be informed and agreed on any decision taken from any of the other group, either of product, process, cost or quality. Such interactions can help to choose among technological solutions, evaluating with a system view the impact that such choices have on different areas; working in close cooperation can help making the right choice from a system point of view, evaluating all the consequences and the impacts that the choice has not only on the area mainly involved in the decision, but also on the others. Such cooperation in early design is also helpful in easily reconciling discrepancies. With this approach, more proper choice can be made, evaluating really fast what are the impact on design feature, part manufacturability, reliability issues, cost and time constraints, and so on.

This approach is of outmost importance in helping making the correct design from the beginning of the product development: this, as a consequence, will reduce successive difficulties during the later program phases. The number of changes necessary in later design phase will be strongly decreased, as well as redesign successive to the impossibility of accomplishing demanding requirements coming either form technical or managerial groups: all the groups concerned in the program development, in fact, will be involved in the initial design. As major activities will be executed together, the time needed to achieve the final design will be drastically shortened. The reductions in design cycle time that result from Concurrent Engineering invariably reduce total product cost. In conclusion, *Concurrent Engineering* provides benefits such as reduced product development time, reduced design rework, reduced product development cost and improved communications

Concurrent Engineering addresses three main areas: people, process, and technology: it involves important changes in the business organization, as it requires the integration of people, business processes and technology, being dependent from interaction among these actors rather than hierarchical organization. Collaboration is of primary importance, as well as cross-functional work. Sharing information is the key for success, and the team leader has the primary role of supporting and encouraging such exchanges.

Many examples exist that demonstrated the reduction in the time necessary to get to the final design, and the improved efficiency in early discover of problems. One example is the General Electric's Aircraft Engines Division's, which design team achieve reductions in design and procurement cycle time.

Another example concerns Boeing's Ballistic Systems Division where Concurrent Engineering was used in 1988 to develop a mobile launcher for the MX missile and was able to reduce design time by 40% and cost by 10% in building the prototype.

To be able to correctly implement *Concurrent Engineering* approach, all the necessary information must be available to any subject who needed them, at the right time: a lot of information concerning products, parts, processes, procedures and any other data useful for design need to be exchanged.

4.1.1. Concurrent Engineering for space application

At the end of 1998, also the European Space Agency embraced the methodologies of *Concurrent Engineering*, by setting an appropriate facility, developing proper processes and integrated design models, and using dedicated software architecture and a necessary infrastructure. The outcome of the efforts were the development and implementation of the Concurrent Design Facility, a mission design environment (hereafter referred to as CDF) in which the conceptual design of space missions could be performed in a more effective way.

Concurrent Design allows the parallel design of several subsystems, managing their mutual interactions, which are then assembled to form an engineering system. The use of this methodology is particularly useful in any field of engineering related to the design of really complex system, like the aerospace engineering, where the design of a space system is certainly a strong effort. ESA has adopted this methodology or the early stage of the design of space systems, following the example of other space agencies, like NASA.

This approach is of great success, as demonstrates the duration of the pre-Phase A analysis, that are currently between 3 and 6 weeks, against a period of 6–9months,typical of about 10 years ago[1]. The implementation of Concurrent Engineering methodologies generated a series of advantages: saving of time and better cost prediction, improved compliance with mission requirements and reduction of design errors as well as major design changes in later design phase; however, the most important aspect of such methodologies dwells in the decrease of time to produce feasibility study. The elements on which the ESA CDF has been based are:

- Process
- IDM: Integrated Design Model
- Multidisciplinary team
- Facility and Infrastructure

Among these, the key element is the process; the next figure depicted the conceptual model of the process, highlighting how, in a space system, not only the engineering areas are extremely interconnected, but also aspect as Risk, Cost, Programmatics are involved in the study.

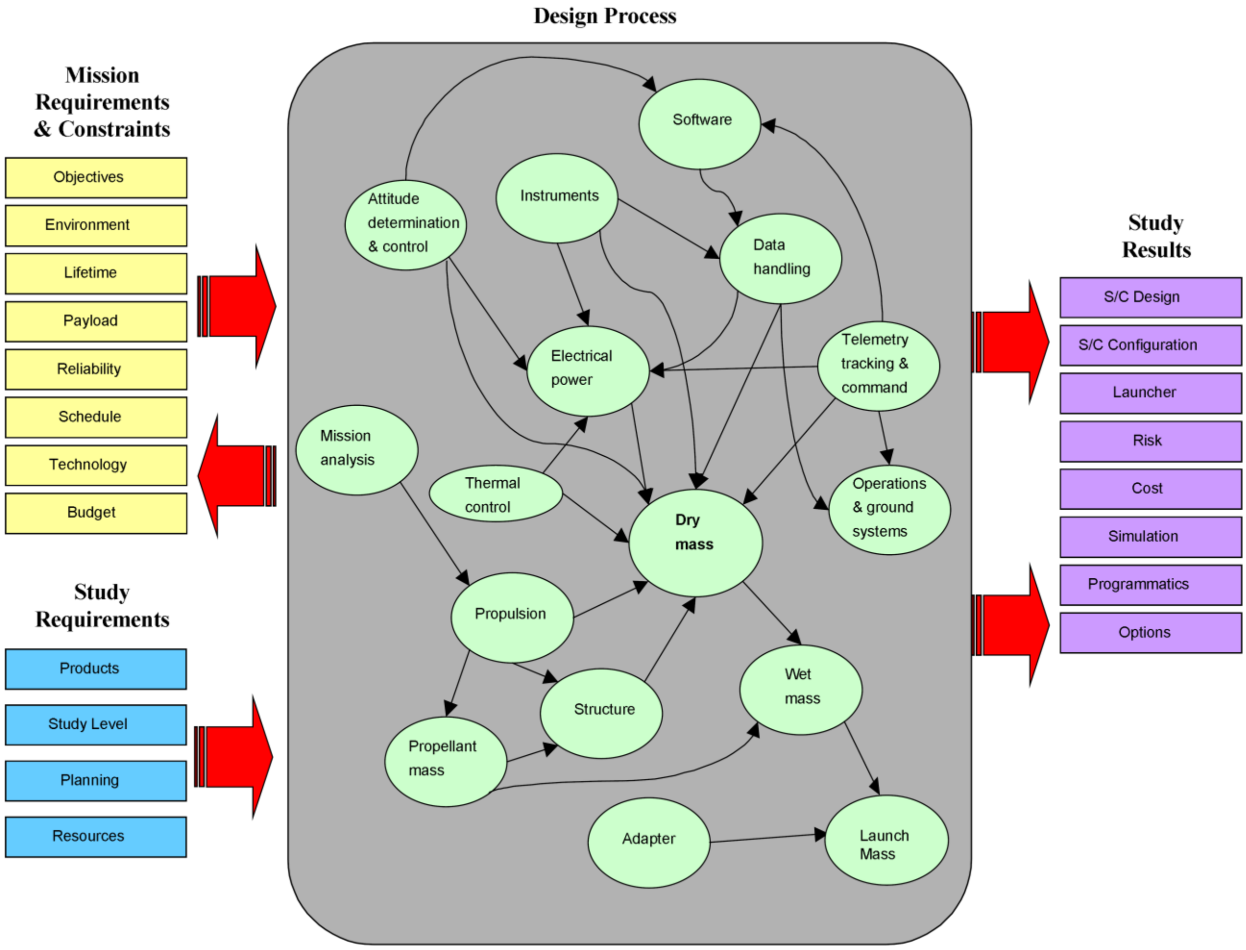


Figure 4 - 1 – Concurrent Desing Process

The strong correlation among all these aspect witnesses how any choice made has an impact on any other components, and how the consequence of any change will propagate through the system. This also underlines the importance of making the right choice, as also the errors made propagate and have impact on the entire system.

4.1.2. Concurrent Design Tool *SAPIENZA*

In this framework, and following the intention of ESA, which promotes the use of this approach in the frame of the European Space Industry, also the University of Rome "Sapienza" developed a proper Concurrent Engineering environment, established following the process I've defined in the next pages.

The development of such a functional Concurrent environment passes through the implementation of a process, that has the main task of clearly define the relationship among the different areas involved in the design; it is in fact of outmost importance, being the reason for the definition, in 1988, of the Concurrent Engineering, to underline the relationship among areas involved in the system design: not only the relationship among engineering fields dedicated to design of propulsion, structures and so on, but also the interaction of these technical issues with areas of design like the those related to space system operations, for example.

Once the process has been defined, an IDM, Integrated Design Model, is necessary to physically carry out the design process: the IDM is the collection of tools and software used to size the space system, i.e. the software environment used to perform the design of the space system, to calculate the components' properties that the various engineering teams have to evaluate and share, and to enable the "physical" exchange of data among the different engineering area involved in the design process.

A space mission can be described in terms of space, launch and ground, segment: any of these segments has its own activities, as well as procedures and processes. In this chapter, the procedure proposed is only focused on the preliminary design of the space segment. The preliminary design is characterized by actions useful for defining the mission requirements, from which requirements and constraints for any of the area devoted to space system design are extrapolated. Such requirements and constraint must be taken into account when satellite subsystems are to be designed, as the former are the features that the subsystem must be able to perform, while the latter are the limitation to deal with.

The space system is traditionally divided into several subsystems, and a specialized unit has been assigned to any of these subsystems. Obviously the design of each subsystem is strictly correlated with the others: an information flux is necessary. In the preliminary design phases, each subsystem can be characterized by a series of 'design parameter', which describe its most relevant physical and technical properties. Design parameters are the information exchanged during concurrent design sessions among engineering units.

4.1.2.1. The Process

The work I've done has been that of clearly underline the relationships between the areas of competence, by identifying the 'design parameters' that the different subsystems exchange one another. This is the first operation that must be done in order to correctly implement an architecture that fully exploit the capability of the methodology of concurrent engineering: in fact, the exchange of data among the team responsible for the design is of primary importance for respecting the principles of commonality of trade-offs and openness of any action undertaken. A general description of the design and of the design variables is provided along with their mutual interactions in the dimensioning and sizing schemes provided for each subsystem and implemented according to Unified Modeling Language (UML) formalism. The parameters that two or more subsystems exchange each other depend also on the typology of mission, in fact some variables can be very important for a particular mission, while do not used if a different mission is chosen: the propellant mass for altitude recovery can be important as they drive the satellite lifetime in low orbit, while can be neglected for a satellite in higher orbit that do not use thrusters, but rely on different systems (like reaction wheels, or gyroscopes) for attitude control.

In this paragraph, the most relevant UML diagrams of the design procedure are shown: the structure and organization are representative of the procedure, while the data contained depends, as already said, on the mission considered. The main class diagrams are presented, as well as a collaboration diagram.

The set of class diagram gives a complete description of the static view of the entire system, while collaboration diagram describes the sequence of the activities and the involved subsystem of the data handling engineering unit. The three class diagrams here exposed are examples of the work done to identify the relationship among the different subsystem. The first two diagrams do not directly refer to a well-defined engineering area, as could be the last one, but to two areas of competence of extreme importance in the early phase of the space system design.

4.1.2.1.1. Class Diagram

The first class diagram represents the mission analysis engineering unit, the part of the system that makes the first analysis and translates the system and mission requirements in specific requirements for each subsystem. Therefore, in this diagram there are not input labels because Missions generates only outputs and basically is not receiving inputs from other units, but only user requirement that has been already converted to mission requirements.

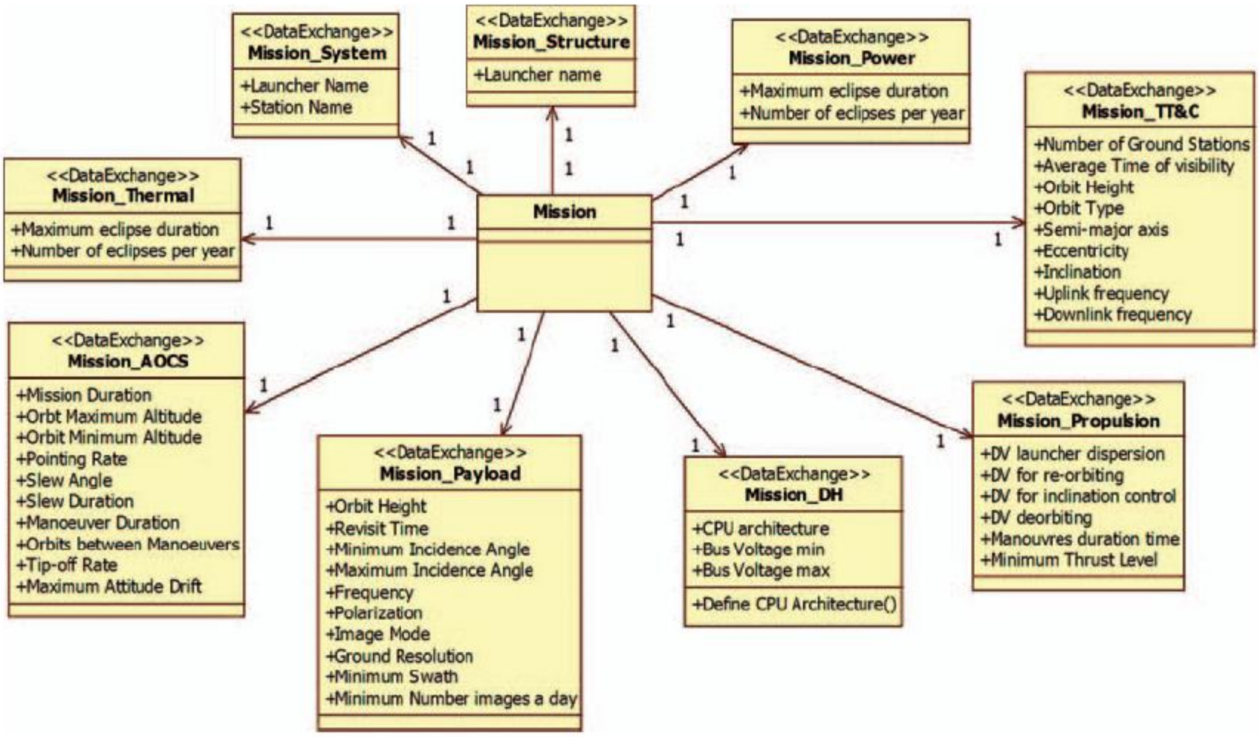


Figure 4 - 2 – Mission Class Diagram

The second diagram describes the activities of system designer: the principal feature of this diagram is the presence of interfaces with all other subsystems; in fact, any subsystem, at least, receives from the system designer the mass and power budgets. Given these budgets, each subsystem has to develop the design respecting these constraints, and at the end of each CD step, gives the results to system designer in terms of budgets and eventually mass and power margin. Besides these general features, there are a plethora of more specific data that characterize the interaction with other subsystems: just for making a few example, the system designer gives to the payload engineering unit information on pointing accuracy and on the range of power that can be furnished, while it passes to structure basic mass properties; it gives to power engineering unit receives from system the requirements on bus properties while AOCS gets some pointing requirements. The inputs to system are essentially data regarding budgets of mass and power, except for mission that gives some information on the launcher and the chosen ground stations. Additional specific properties are given by other subsystems, also in accordance with the mission in phase of study.

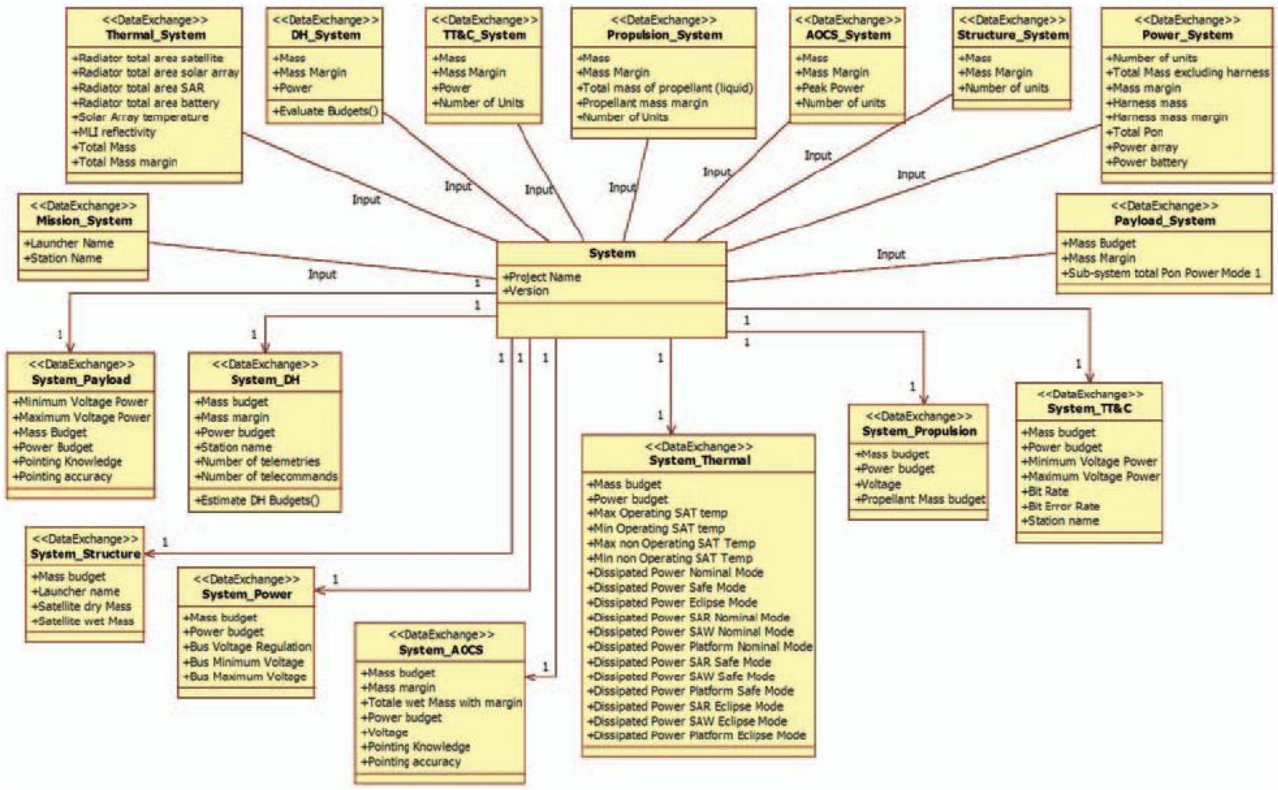


Figure 4 - 3 – System Class Diagram

The third class diagram is instead related to a precise engineering area: the structural one. This unit receives inputs from all other subsystems; the inputs received from the other subsystems are components dimensions, mass, suggested and denied positions that they can or cannot occupy when the configuration of the satellite is made. Obviously, structure not only gives data to the system designer, but also to other subsystems: to AOCS, it gives information about satellite dimensions and its inertia properties; to Thermal Control System, it gives satellite dimensions; and to Power it gives the requirements and the constraints for the implementation of solar wings and/or body mounted solar panel. Structure is composed by several elements that are designed using the appropriate functions and is described in the corresponding component class (successive figure).

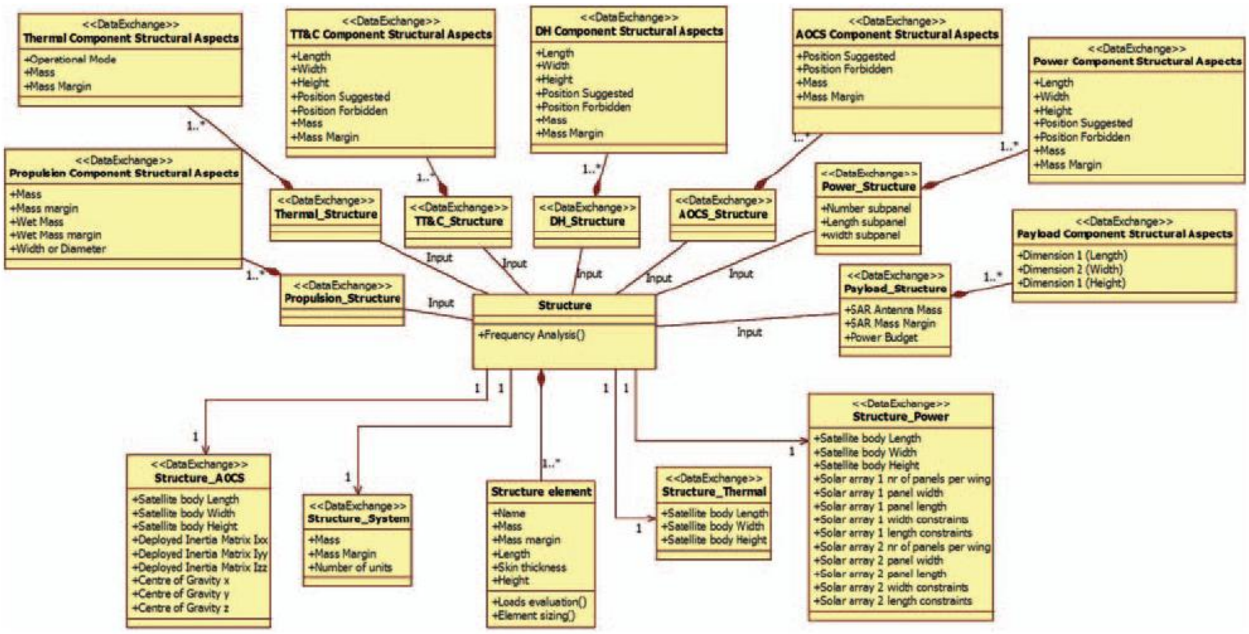


Figure 4 - 4 – Structure Class Diagram

In the two successive figures, class diagram of the Data Handling subsystem has been reported, together with its collaboration diagram. From the first class diagram, it is immediately evident (in the top-right box) how such subsystem receives budgets like mass, power and the so-called telemetry budget (number of telecommand and telemetries)

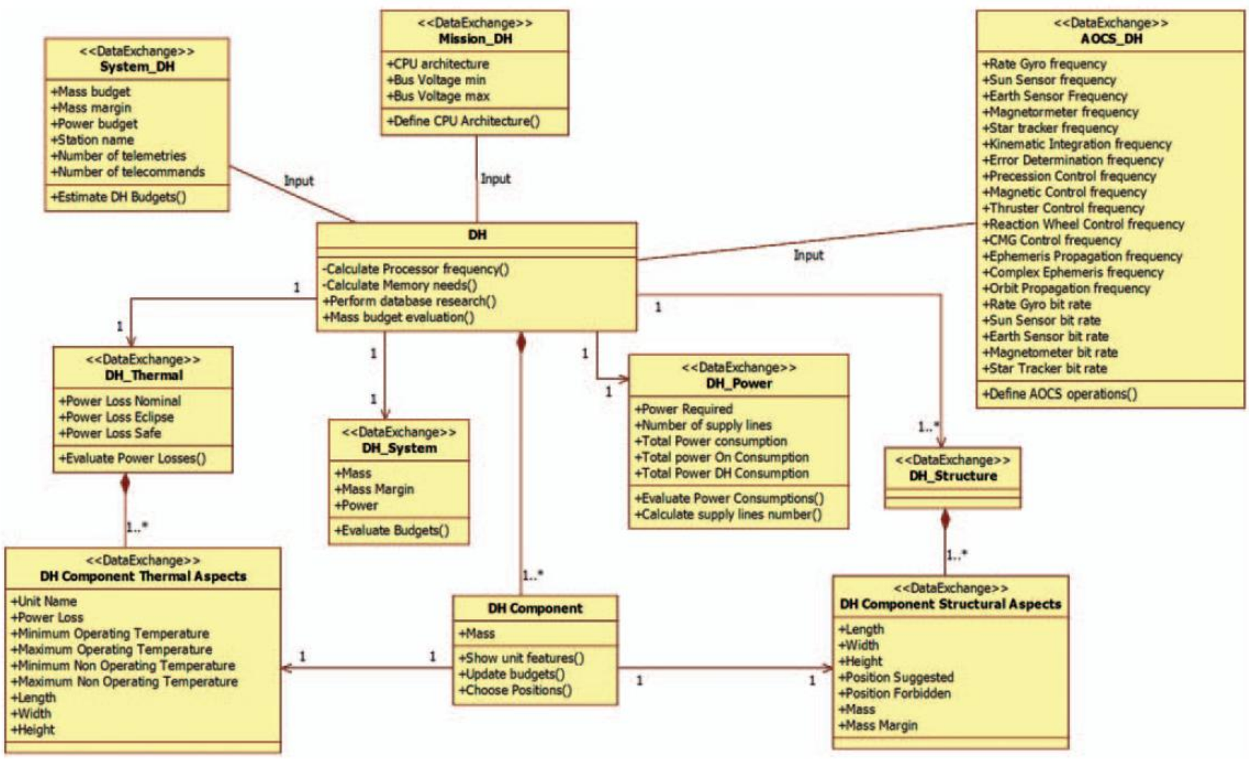


Figure 4 - 5 – Data_Handling Class Diagram

From the mission design team it obtains CPU architecture and bus characteristics. From the other engineering units it receives additional information, while, in the output area, information about dimension, mass, power consumed, losses and so on are exchanged with the other subsystems, as any other subsystem does.

4.1.2.1.2. Collaboration Diagram

The successive diagram is an example of collaboration diagram: it shows the actions performed when the concurrent design process is in active phase, when the design parameters are exchanged among subsystems, and engineering units make their design choices. Each arrow indicates a flux of information between objects. The numbers give the sequence of the actions that has been set. Each action is represented by a function whose complexity is unpredictable, depending on the design parameter it has to calculate. As an example, the function 'Calculate Processor Frequency()' can be very complex considering many variables: Update rate of the sensors, historical information, availability of existing central process unit (CPU), etc. 'Define_CPU_Architecture()', instead, could be a simple choice made by an expert designer.

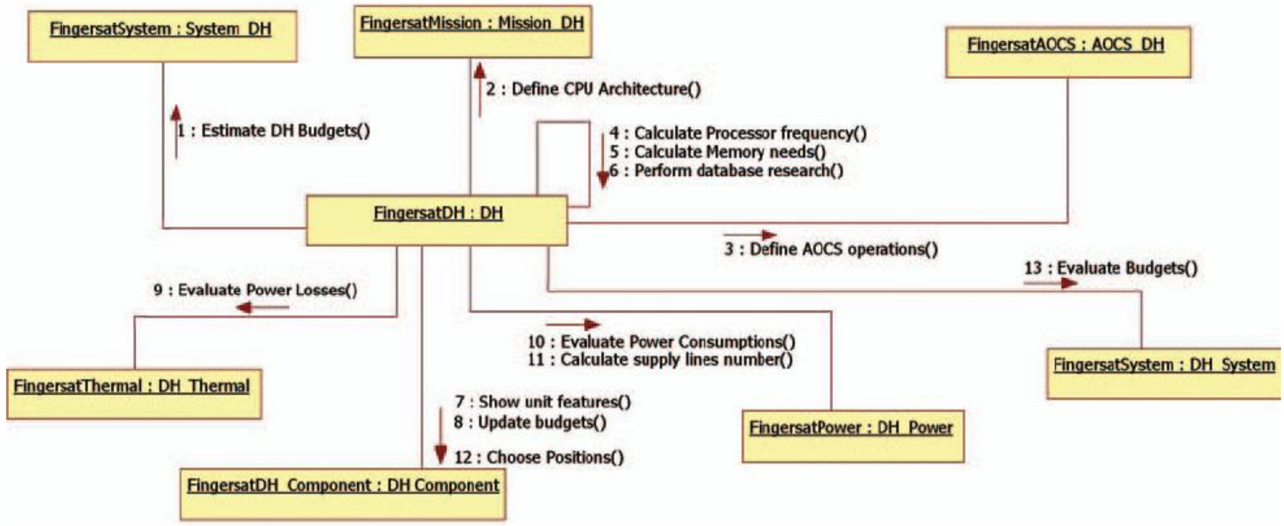


Figure 4 - 6 – Data_Handling Collaboration Diagram

These types of class diagram (the static view showing the parameters that the different engineering areas exchange each other) and collaboration diagram (the dynamic view, showing the fluxes of information and the operations to perform when design iteration occurs) has been produced for any engineering areas devoted to the design of spacecraft subsystems.

4.1.2.1.3. Concurrent Design process and design iteration

The process of Concurrent Engineering is, as already extensively said, a process where interactions among all the actors involved in the design is the core elements, and is the reason for all the advantages that such a methodology entails.

Such a process of concurrent design is characterized by design iteration: several attempts are, in fact, necessary to ultimately get to the final, definitive design. In fact, if on the one hand exists a flux information that is continuously exchanged among all the design actors, on the other hand any areas involved in system design needs to have a set of information (regarding, for example, mass, power consumption, or any other data related to spacecraft performances) for implementing its design operations. The results and the outcome of all these design operations, performed by all the engineering teams involved, will be used as necessary input information in a successive design phase; this process is witnessed by the successive diagram, used to clarify the process of concurrent design.

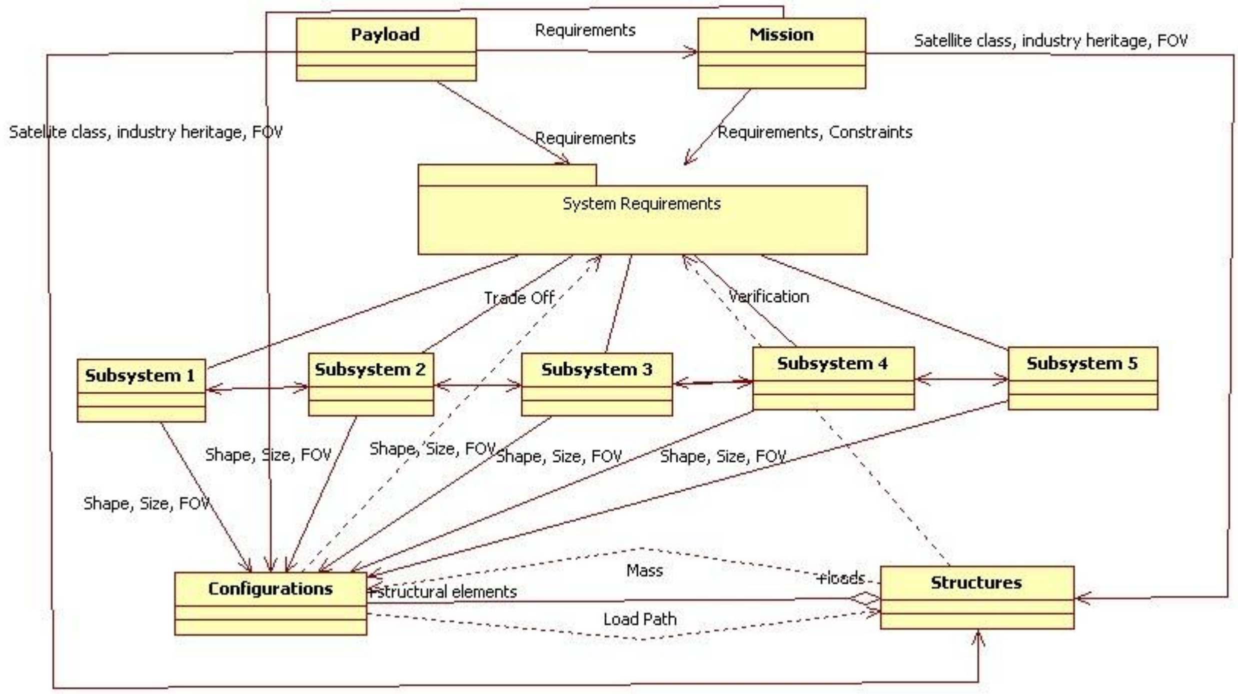


Figure 4 - 7 – Concurrent Design Process

The process starts with the definition of the payload, which is clearly chosen accordingly to the costumers' needs, and, in turns, by defining the mission requirements for giving the payload the ability to act in the most proper way. Once the mission has been defined, and more than a single option is possible, and in accordance with the requirements imposed by the payload (orbital height, revisit time, duty cycle, and so on) the requirements at system level are evaluated.

From the requirements at system level, which mainly summarize the performance of the system in general, and the service the satellite has to provide, the system engineering team evaluates the subsystems requirements, that are more technical requirements, that are transferred to the various designing teams, that have task to perform all the operations necessary to get the subsystem design. Once all the designing teams performed their design, the resulting information are exchanged among the teams, and used as input in successive design iteration. The design iterations are necessary as the data at disposal of the various engineering team, that comes from the system engineering team and are used as inputs in the first design iteration, are data of first attempt: in fact, for the engineering team is not possible to evaluate, for example, the exact power necessary to entire spacecraft (which components have not been already defined), or the precise amount of data that the payload will produce, or other information that would be available once the design process will be completed; in addition, the engineering team has to provide to any designing areas budgets:

mass and power budgets to not exceed are necessary as the system could not have a total mass as large as possible; link budget has to be provided to communication teams, otherwise it cannot define the communication system; a target power needs to be identified, otherwise the team devoted to power generation cannot start its designing process.

All these preliminary information will be used by the various engineering team as inputs for performing the necessary operations in order to have a design of any subsystem, characterized by design parameters; these parameters, and the subsystems designed, represent the satellite system and its functionalities. In these values will be probably present inconsistencies: hardly all the devices defined by the designing team will necessitate exactly the power that the engineering unit devoted to such task will have designed. For such a reason, such parameters will be shared among all the designing team, that will now perform a second iteration, carrying out again design operations that will lead to a refined subsystems design, as it has been obtained using as input data not the budget or the first attempt value defined by the engineering team, but the data coming out from the design operation of the other engineering team.

Since the inputs has been changed, probably the design of the subsystems needs to be modified, producing design parameters different from the previous ones. All these parameters will once more verified, evaluating the compliance with mission and system requirements, and the presence or not of discrepancies among the various areas; in case discrepancies exist, the design parameters resulting from the last iteration will be used as input for a successive design iteration, following the spiral model depicted in the next figure, over and over again until a consistent design is reached. Such process assures that all the interaction has been correctly evaluated, identifying a solution capable of satisfying all the mission and system requirements.

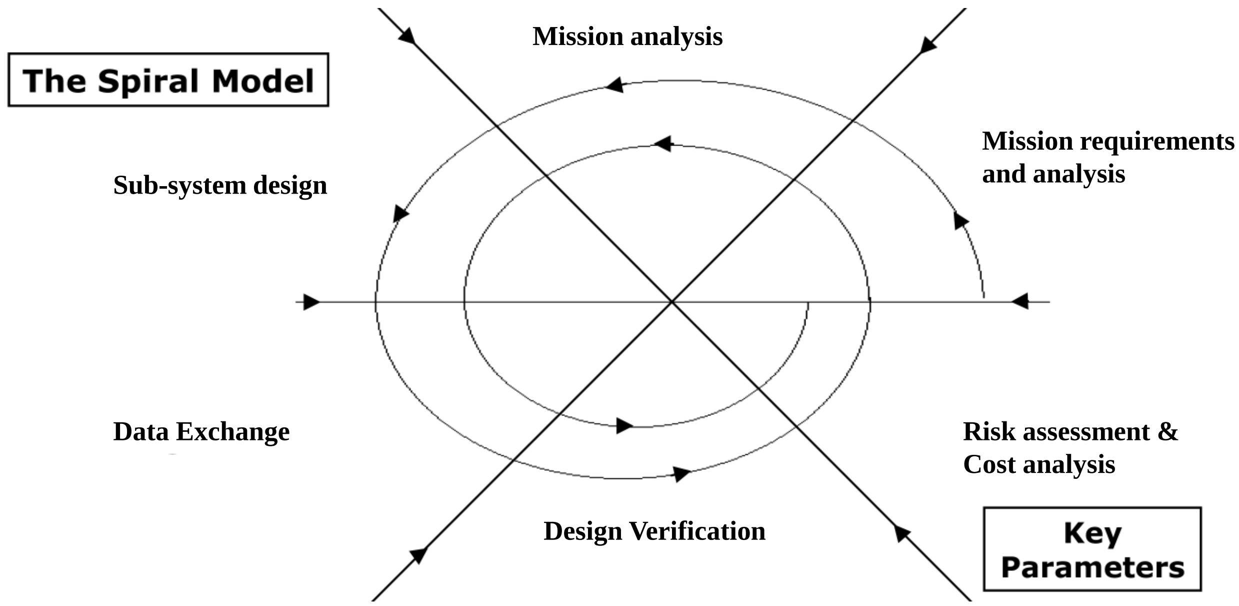


Figure 4 - 8 – Spiral Model, suggesting the recursive process of design iterations

The methodologies of the *Concurrent Engineering* became largely widespread, and numerous space agencies and industries developed its proper software for concurrent design during the last decade: what all these software for concurrent design have in common, from a functional point of view, is the fact that all these software has been developed for furnishing support to preliminary design phases: according the ESA definitions of a space project phases [1], such software have been developed for phase-A studies, aiming at assessing the technical feasibility of the space mission investigated.

Several attempts have been done in the last two or three years, with the aim to expand software develop according to this methodology also for later design phase[2].

In this Ph.D. research, the software has been developed for accomplishing pre-feasibility studies, classified, according to ESA nomenclature, as pre-Phase-A; the reasons for choosing to implement such early design phase software are simple.

Firstly, using such software for phase-B studies implies a level of detail for all the design aspects that goes largely out of the scope of studies performed in the framework of university projects. Secondly, the implementation of a concurrent methodologies for later design phase necessitate a knowledge of satellite subsystems and its components noteworthy deeper than that required for feasibility or pre-feasibility studies. Thirdly, the implementation of software for concurrent design for later design phases (A, B, etc.) is a step that must necessarily follows the implementation of concurrent software for earlier phase, being software for phase-A or -B design its natural successor.

4.1.2.2. The Integrated Design Model

As already said, the process represents the first step for settling a concurrent design capability: once the process has been established, the relationships among the engineering and managerial units underlined, and the rules and logic for designing any area of space systems has been identified, the successive phase I've performed was that of implementing the software environment of the concurrent design: a collection of proprietary tools, one for any area involved in the design, supported by specialized commercial software for more detailed and accurate analysis, and for evaluating parameters hardly obtainable with simple tools; the collection of these proprietary tools has been named *Concurrent Design Tool.*

The Concurrent Design Tool is composed by two type of software:

- Excel workbook, devoted to system and subsystem design;
- Commercial software (STK, Autodesk Inventor Professional, MatLab, StarUML, etc.), used as supporting tool during the process of satellite preliminary design.

The workbooks are the main elements of the CDT: the workbooks defined in the CDT are one for any of the area involved in the system design, plus one supporting *Data_Exchange* workbook; with the exception of the *Data_Exchange* workbook, that will be described in the following paragraph, all the workbooks are characterized by three types of worksheets:

1. Inputs/Outputs sheets;
2. Calculation Sheets;
3. Database sheets.

The Inputs/Outputs worksheets are used to "physically" transfer the information from a designing area to the others; the Calculation Sheets are those worksheets containing all the equations, formulas and design options that the teams involved in satellite design have to undertake in order to complete the design of the related subsystem. The Database sheets are instead the worksheets that collect the components necessary to subsystem design. As an example, the workbook devoted to Propulsion subsystem design contains database of spacecraft thrusters, mono/bi-propellant technical characteristic (specific impulse, thrust, etc.), valves, and any other device necessary to preliminary design.

It must be borne in mind that, being the process, and consequently the IDM, conceived for a preliminary design, the equations, formulas and sizing criteria are tailored to such needs. No complex analysis are to be performed, or particularly time-demanding calculation are necessary; simple rule of thumbs, basic trade-off, and the expertise of the people involved in the design will be the way for performing the subsystem design. To make an example, let's try to size a structural components of the space system: initially, a preliminary estimate of the devices mounted on this structural element must be done; then, using the information coming from mission specialists of the launcher that would be used for in-orbit injection, axial and lateral loads factors that the structural element will face are loaded from the Database sheets; using these data, loads that the structural elements have to withstand are evaluated, and a proper thickness of these elements (once the engineering team devoted to structure subsystem would have chosen the material used for manufacturing) will be calculated using simple design criteria, for example a Von Mises criterion could be implemented, calculating the von Mises stress or equivalent tensile stress.

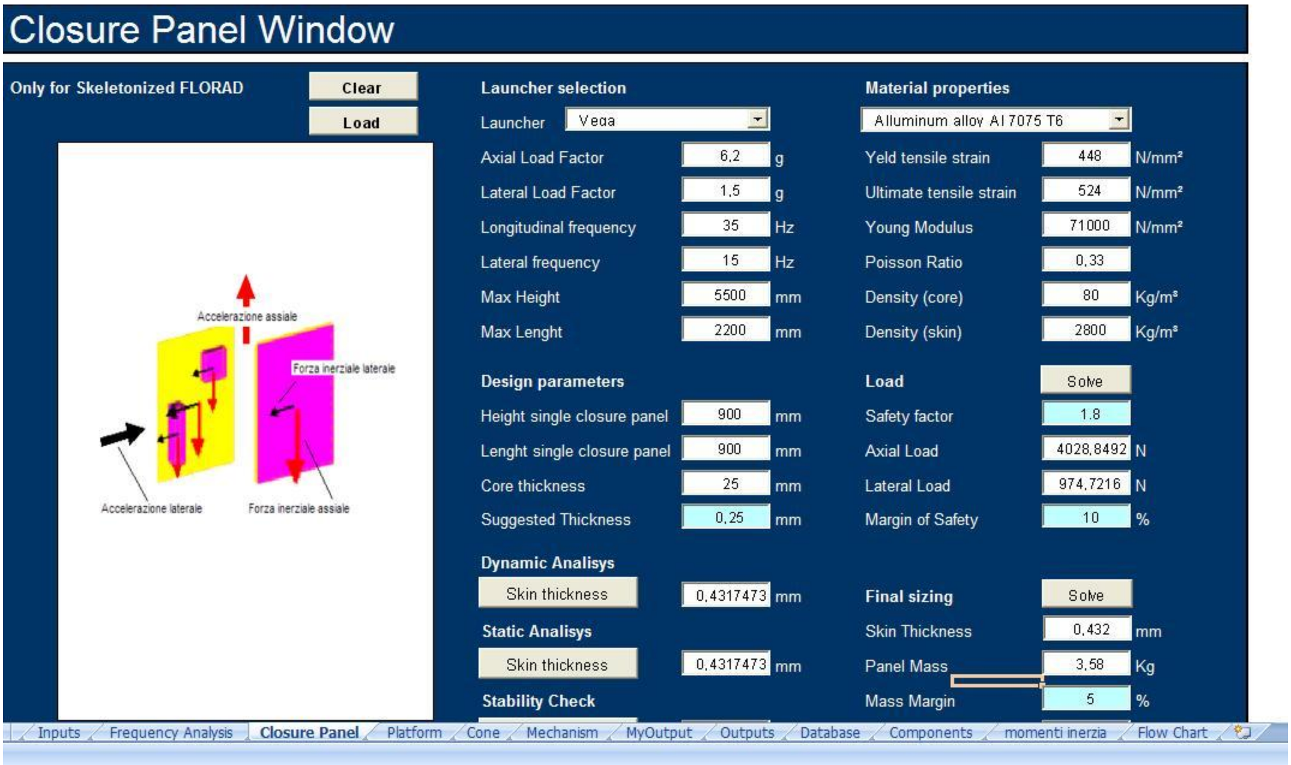


Figure 4 - 9 – Example of the IDM (Integrated Design Model)

This procedure is in opposition with the absolutely more refined method of performing Finite Element Analysis for evaluating with a better precision the satellite structural elements; but, in a preliminary design, the level of details that a FEA can provide is excessive, and often not justified, mainly for two reasons: the first is related to the fact that the design is still in evaluation, and an accurate analysis performed this way became useless if changes occur, and the changes will certainly occur, especially in the former design iterations; secondly, the preliminary design has to be rapid, and with the aim of evaluating the feasibility or not of space mission; in addition, accurate analysis are absolutely out of scope, since major mission changes can still occur in this phase.

The commercial software are used, also in the preliminary design, in order to execute complex operations, that generally are out of concept of such a tool, more devoted to preliminary design, that are anyway necessary in order to have a correct understanding of certain design parameters. A good example can be provided by Satellite Tool Kit (STK), used for orbit propagation: the use of such software provide important information about coverage area, revisit time, mean daily contact time with the ground station, eclipse durations, and so on. This information, useful for the process of subsystem design, from the Electrical Power Subsystem to the Telemetry, Tracking & Command subsystem, to the Thermal Control Subsystem, that could be hardly obtained in different way; as so Autodesk Inventor, a CAD 3D software with which the satellite configuration could be realized, providing the possibility to visualize a 3D model of the satellite, inspecting if the subsystems designed fit into the envelope.

In the following, a deeper analysis of the workbooks, their working principles and interaction will be provided, together with screenshots for better describe them.

4.1.2.2.1. Concurrent Design Tool working principles

The working principles of the CDT (Concurrent Design Tool) is the same used by the European Space Agency, and by all the other institution and industries having a proper tool for performing design in a concurrent environment: this has been largely promoted by ESA with the aim of facilitating the exchange of information among those entities. The core is represented by:

    - An Excel workbook, named *Data_Exchange* that is used to collect all the data coming from all the designing areas involved in system design;
    - Two worksheets in any of the Excel workbook devoted to designing of areas of competences; these couples of worksheets have the task to both download the data from the *Data_Exchange*, that are to be used as inputs for the design process, and to upload the data produced as results of the design process in the *Data_Exchange* workbook, in order to be successively downloaded by any other subsystem (in particular, the worksheet devoted to such operation).
- *Data_Exchange* workbook:

	A	B	C	D	E	F
89	Sub-system Power Margin M	AOC_E1_ss_PM_M1	%	internally linked	10,000	
90	Sub-system total Safe Power	AOC_E1_ss_Safe	W	internally linked	0,000	
91	Sub-system total Dissipated	AOC_E1_ss_Dis_Safe	W	internally linked	12,214	
92	Sub-system Duty Cycle Powe	AOC_E1_ss_Dc_M2	%	not shared	-	
93	Sub-system total Pon Power	AOC_E1_ss_Pon_M3	W	not shared	-	
94	Sub-system total Pstby Powe	AOC_E1_ss_Pstby_M3	W	not shared	-	
95	Sub-system Duty Cycle Powe	AOC_E1_ss_Dc_M3	%	not shared	-	
96	Sub-system total Pon Power	AOC_E1_ss_Pon_M4	W	not shared	-	
97	Sub-system total Pstby Powe	AOC_E1_ss_Pstby_M4	W	not shared	-	
98	Sub-system Duty Cycle Powe	AOC_E1_ss_Dc_M4	%	not shared	-	
99	Sub-system total Pon Power	AOC_E1_ss_Pon_M5	W	not shared	-	
100	Sub-system total Pstby Powe	AOC_E1_ss_Pstby_M5	W	not shared	-	
101	**Other Information**					
114	**Subsystem Equipment Details**					
115	Nr of Units	AOC_E1_nr_units		internally linked	17,000	
116	**Unit1**					
117	Unit1 Name	AOC_E1_unit1_name	-	internally linked	MicroWheel-10SP-S	
118	Number of units	AOC_E1_unit1_nr	number	internally linked	1,000	
119	mass (without margin)	AOC_E1_unit1_mass	kg	internally linked	1,100	
120	Mass Margin to be applied	AOC_E1_unit1_massmargin	%	internally linked	10,000	
121	Dimension 1 (Length)	AOC_E1_unit1_dim1	mm	internally linked	104,000	Length
122	Dimension 2 (Width or Diame	AOC_E1_unit1_dim2	mm	internally linked	93,000	Width or Diameter
123	Dimension 3 (Height)	AOC_E1_unit1_dim3	mm	internally linked	93,000	Height
124	Temperature					
125	Maximum Operational Tempe	AOC_E1_unit1_Top_max	C	internally linked	0,000	
126	Minimum Operational Temper	AOC_E1_unit1_Top_min	C	internally linked	0,000	
127	Maximum Non-operational Te	AOC_E1_unit1_Tnop_max	C	not shared	-	
128	Minimum Non-operational Ter	AOC_E1_unit1_Tnop_min	C	not shared	-	
129	Power Consumption					
130	Peak Power Consumption	AOC_E1_unit1_Ppeak	W	internally linked	0,000	
131	Mode 1 Power Consumption (	AOC_E1_unit1_PonM1	W	not shared	-	
132	Mode 1 Power Consumption (	AOC_E1_unit1_PstbyM1	W	not shared	-	

power / Propulsion / thermal / structures / system / instruments / mission / data_handling / comms / aocs / wiemer / MenuSheet

Figure 4 - 10 – Data_Exchange workbook - AOCS worksheet

The previous figure is taken from the *Data_Exchange* workbook, and in particular the AOCS worksheet: in such a workbook are collected all the design parameters of all the designing units involved; in the worksheet depicted, we can see a glimpse of the information that the engineering team devoted to Attitude and Orbit Control System has obtained after a certain number of design iteration. Entering more in detail, the first column described the design parameters, the second the name gave to the variable, useful for data migration among workbook, in the fifth column the value of the design parameters, and in the third the unit of measure; in the sixth column are provided additional (facultative) information, while the fourth column contains an indication regarding the design parameters: if it is internally linked, i.e. it is shared by the subsystem in exam (AOCS) with the other ones, or not (not shared).

The *Data_Exchange* workbook is composed by one worksheet for any of the designing areas (plus some supporting worksheets), and the format and functioning of any of them is always the same, as witnessed in the next figure by the Thermal worksheet of the *Data_Exchange* workbook

	A	B	C	D	E	F	G
88	Total_TCS_Power_nominal	the_E1_ss_power_tot_nomina	W	internally linked	0,000000		
89	**Safe mode**						
90	TCS_Power_sat_safe	the_E1_ss_power_sat_safe	W	internally linked	0,000000		
91	TCS_Power_sar_safe	the_E1_ss_power_sar_safe	W	internally linked	0,000000		
92	TCS_Power_solar_safe	the_E1_ss_power_solar_safe	W	not shared	-		
93	TCS_Power_battery_safe	the_E1_ss_power_battery_sa	W	internally linked	0,000000		
94	TCS_Power_propul_safe	the_E1_ss_power_propul_saf	W	not shared	-		
95	Total_TCS_Power_safe	the_E1_ss_power_tot_safe	W	internally linked	0,000000		
96	**Eclipse Mode**						
97	Total_TCS_Power_eclipse	the_E1_ss_power_tot_eclipse	W	not shared	-		
98	**Other information**						
99	Number of supply lines	the_E1_ss_power_Number_lines		internally linked	1,000000		
100	Suggested element position	the_E1_ss_suggested_element_pos		not shared			
101	Forbidden element position	the_E1_ss_forbidden_element_pos		not shared			
102				not shared			
103				not shared			
104				not shared			
105	**Thermal Equipment Details**						
106	**UNIT 1**						
107	Unit Name	the_E1_unit1_name		internally linked	MLI		
108	Number of Units	the_E1_unit1_nr		internally linked	1,000000		
109	Unit Mass (without margin)	the_E1_unit1_mass	kg	internally linked	0,505090		
110	Margin to be applied	the_E1_unit1_massmargin	%	internally linked	20,000000	5% OTS, 10% Modified, 20% Developed	
111	Dimension 1 (Length)	the_E1_unit1_dim1	m	not shared	-		
112	Dimension 2 (Width)	the_E1_unit1_dim2	m	not shared	-		
113	Dimension 3 (Height)	the_E1_unit1_dim3	m	not shared	-		
114	Thermal						
115	Maximum Operational Tempe	the_E1_unit1_Top_max	C	not shared	-		
116	Minimum Operational Temper	the_E1_unit1_Top_min	C	not shared	-		
117	Maximum Non-operational Te	the_E1_unit1_Tnop_max	C	not shared	-		
118	Minimum Non-operational Ter	the_E1_unit1_Tnop_min	C	not shared	-		

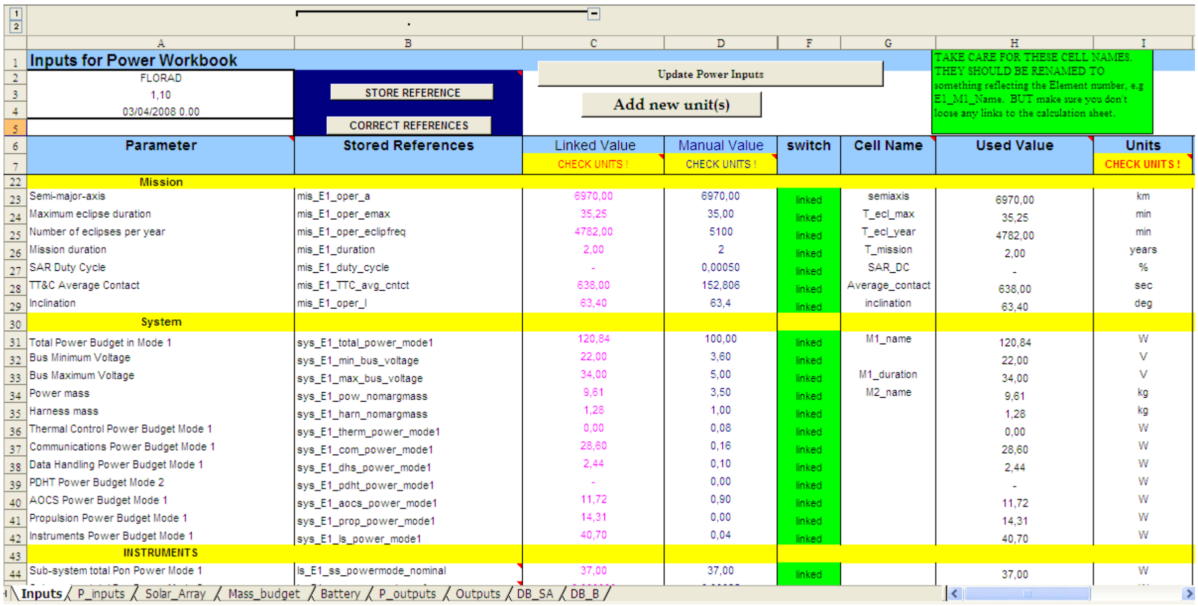
power / Propulsion / **thermal** / structures / system / instruments / mission / data_handling / comms / aocs / wiemer / MenuSheet / Constants / external_o

Last updated on: CUBESAT

Figure 4 - 11 – Data_Exchange workbook - Thermal worksheet

- *Inputs* worksheet (one for any workbook)

The first worksheet present in any of the workbook devoted to satellite subsystem design is the *Inputs* worksheet: such worksheet has the task to download data from the *Data_Exchange* workbook, making them available for the design operation that will be performed by the engineering team with the support of the same Excel workbook. In the next figure is illustrated one of the *Inputs* worksheet, in particular that of the *Power* workbook: the format is really similar to the worksheet collecting data of the *Data_Exchange* workbook, and the reason for such a similarity is right because the functions are similar.



Inputs for Power Workbook

FLORAD

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STORE REFERENCE

CORRECT REFERENCES

Update Power Inputs

Add new unit(s)

TAKE CARE FOR THESE CELL NAMES. THEY SHOULD BE RENAMED TO something reflecting the Element number, e.g E1_M1_Name. BUT make sure you don't loose any links to the calculation sheet.

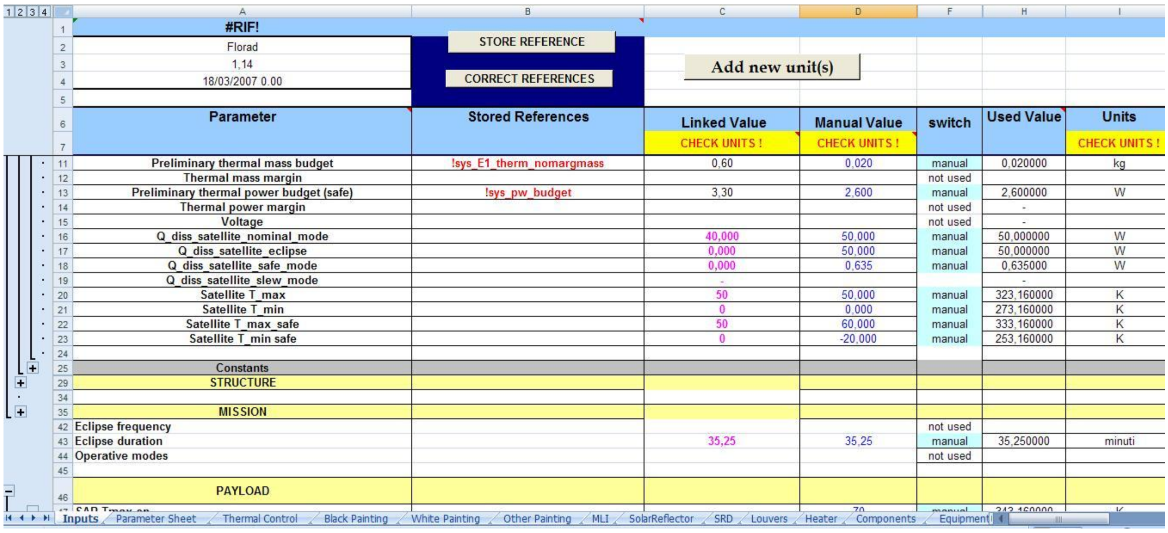
Parameter	Stored References	Linked Value	Manual Value	switch	Cell Name	Used Value	Units
		CHECK UNITS !	CHECK UNITS !				CHECK UNITS !
Mission							
Semi-major-axis	mis_E1_oper_a	6970,00	6970,00	linked	semiaxis	6970,00	km
Maximum eclipse duration	mis_E1_oper_emax	35,25	35,00	linked	T_ecl_max	35,25	min
Number of eclipses per year	mis_E1_oper_eclipfreq	4782,00	5100	linked	T_ecl_year	4782,00	min
Mission duration	mis_E1_duration	2,00	2	linked	T_mission	2,00	years
SAR Duty Cycle	mis_E1_duty_cycle	-	0,00050	linked	SAR_DC	-	%
TT&C Average Contact	mis_E1_TTC_avg_cntct	638,00	152,806	linked	Average_contact	638,00	sec
Inclination	mis_E1_oper_I	63,40	63,4	linked	inclination	63,40	deg
System							
Total Power Budget in Mode 1	sys_E1_total_power_mode1	120,84	100,00	linked	M1_name	120,84	W
Bus Minimum Voltage	sys_E1_min_bus_voltage	22,00	3,60	linked		22,00	V
Bus Maximum Voltage	sys_E1_max_bus_voltage	34,00	5,00	linked	M1_duration	34,00	V
Power mass	sys_E1_pow_nomargmass	9,61	3,50	linked	M2_name	9,61	kg
Harness mass	sys_E1_harn_nomargmass	1,28	1,00	linked		1,28	kg
Thermal Control Power Budget Mode 1	sys_E1_therm_power_mode1	0,00	0,08	linked		0,00	W
Communications Power Budget Mode 1	sys_E1_com_power_mode1	28,60	0,16	linked		28,60	W
Data Handling Power Budget Mode 1	sys_E1_dhs_power_mode1	2,44	0,10	linked		2,44	W
PDHT Power Budget Mode 2	sys_E1_pdht_power_mode1	-	0,00	linked		-	W
AOCS Power Budget Mode 1	sys_E1_aocs_power_mode1	11,72	0,90	linked		11,72	W
Propulsion Power Budget Mode 1	sys_E1_prop_power_mode1	14,31	0,00	linked		14,31	W
Instruments Power Budget Mode 1	sys_E1_is_power_mode1	40,70	0,04	linked		40,70	W
INSTRUMENTS							
Sub-system total Pon Power Mode 1	Is_E1_ss_powermode_nominal	37,00	37,00	linked		37,00	W

Inputs / P_inputs / Solar_Array / Mass_budget / Battery / P_outputs / Outputs / DB_SA / DB_B

Figure 4 - 12 – Inputs worksheets of the Power workbook

Figure

For the sake of completion, in order to demonstrate that the Inputs worksheets are similar, also the Inputs worksheets of Thermal workbook is depicted



#RIF!

Florad

1,14

18/03/2007 0.00

STORE REFERENCE

CORRECT REFERENCES

Add new unit(s)

Parameter	Stored References	Linked Value	Manual Value	switch	Used Value	Units
		CHECK UNITS !	CHECK UNITS !			CHECK UNITS !
Preliminary thermal mass budget	!sys_E1_therm_nomargmass	0,60	0,020	manual	0,020000	kg
Thermal mass margin				not used		
Preliminary thermal power budget (safe)	!sys_pw_budget	3,30	2,600	manual	2,600000	W
Thermal power margin				not used	-	
Voltage				not used	-	
Q_diss_satellite_nominal_mode		40,000	50,000	manual	50,000000	W
Q_diss_satellite_eclipse		0,000	50,000	manual	50,000000	W
Q_diss_satellite_safe_mode		0,000	0,635	manual	0,635000	W
Q_diss_satellite_slew_mode		-			-	
Satellite T_max		50	50,000	manual	323,160000	K
Satellite T_min		0	0,000	manual	273,160000	K
Satellite T_max_safe		50	60,000	manual	333,160000	K
Satellite T_min safe		0	-20,000	manual	253,160000	K
Constants						
STRUCTURE						
MISSION						
Eclipse frequency				not used		
Eclipse duration		35,25	35,25	manual	35,250000	minuti
Operative modes				not used		
PAYLOAD						

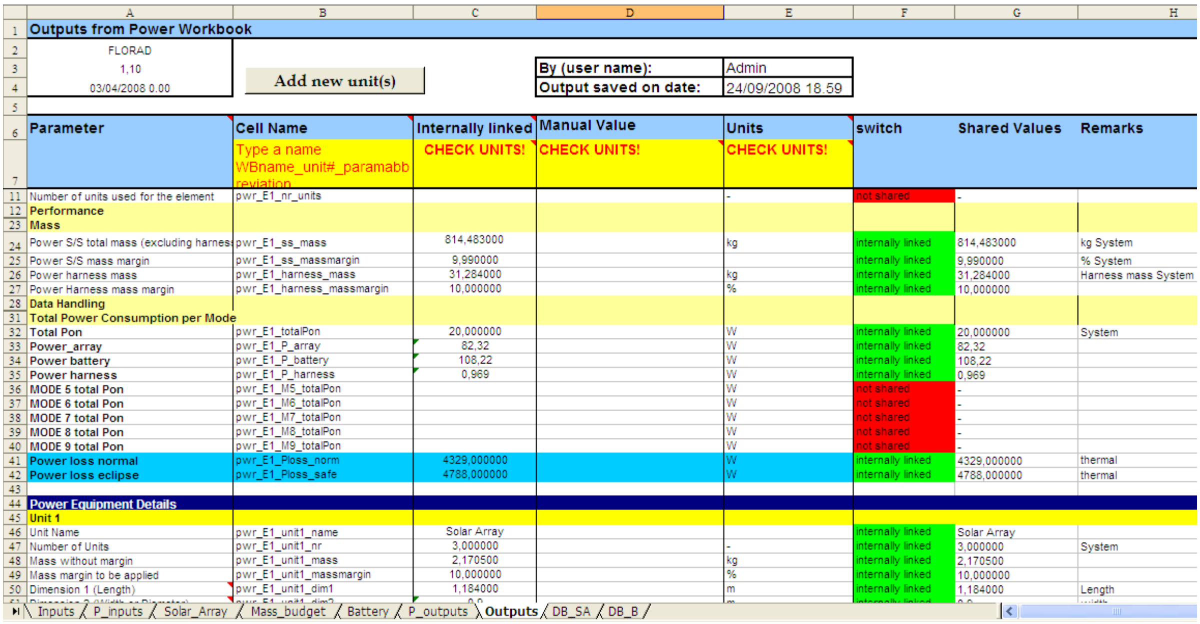
Inputs / Parameter Sheet / Thermal Control / Black Painting / White Painting / Other Painting / MLI / SolarReflector / SRD / Louvers / Heater / Components / Equipment

Figure 4 - 13 – Inputs worksheets of the Thermal workbook

In the first column the name of the design parameters, and in the second the name of the variables; the third column contains the Linked Values, which are the value downloaded from the *Data_Exchange* workbook (results of the design operations of some other designing area), while the fourth is a column where the people of the team can insert Manual Values, that could be used as inputs in alternative of values coming from *Data_Exchange* workbook: this could be done, for example, if the engineering team would like to evaluate the impact of the change of one or more parameters on its proper design, using values different from those downloaded. The fifth column contains a switch that can be varied from "linked" (as shown in the figure) or manual; according to this choice, the corresponding value will be copied in the seventh column, (Used Values) and used for design iteration. The eighth column contains the unit measure, while the sixth additional information regarding the name assigned to the cell (useful for correctly migrate the data among worksheet and workbook)

- *Outputs* worksheet (one for any workbook)

The second worksheet present in any of the workbook devoted to satellite subsystem design is the *Outputs* worksheet: such worksheet has the task to upload data into the *Data_Exchange* workbook, once all the design operation has been performed by the engineering team with the support of the Excel workbook. In the next figure is illustrated one of the *Outputs* worksheet, in particular that of the *Power* workbook: also this worksheet is extremely similar to the worksheet collecting data of the *Data_Exchange* workbook, and, also in this case, the similarity is due to the fact that functions performed by these worksheets are similar, more than in the previous case: in fact, the worksheets in the *Data_Exchange* workbook are copied by the *Outputs* worksheet of any of the designing workbook.



	A	B	C	D	E	F	G	H
1	Outputs from Power Workbook							
2	FLORAD							
3	1,10	Add new unit(s)		By (user name):	Admin			
4	03/04/2008 0.00			Output saved on date:	24/09/2008 18.59			
5								
6	Parameter	Cell Name	Internally linked	Manual Value	Units	switch	Shared Values	Remarks
7		Type a name WBname_unit#_paramabbreviation	CHECK UNITS!	CHECK UNITS!	CHECK UNITS!			
11	Number of units used for the element	pwr_E1_nr_units			-	not shared	-	
12	Performance							
23	Mass							
24	Power S/S total mass (excluding harnes	pwr_E1_ss_mass	814,483000		kg	internally linked	814,483000	kg System
25	Power S/S mass margin	pwr_E1_ss_massmargin	9,990000			internally linked	9,990000	% System
26	Power harness mass	pwr_E1_harness_mass	31,284000		kg	internally linked	31,284000	Harness mass System
27	Power Harness mass margin	pwr_E1_harness_massmargin	10,000000		%	internally linked	10,000000	
28	Data Handling							
31	Total Power Consumption per Mode							
32	Total Pon	pwr_E1_totalPon	20,000000		W	internally linked	20,000000	System
33	Power_array	pwr_E1_P_array	82,32		W	internally linked	82,32	
34	Power battery	pwr_E1_P_battery	108,22		W	internally linked	108,22	
35	Power harness	pwr_E1_P_harness	0,969		W	internally linked	0,969	
36	MODE 5 total Pon	pwr_E1_M5_totalPon			W	not shared	-	
37	MODE 6 total Pon	pwr_E1_M6_totalPon			W	not shared	-	
38	MODE 7 total Pon	pwr_E1_M7_totalPon			W	not shared	-	
39	MODE 8 total Pon	pwr_E1_M8_totalPon			W	not shared	-	
40	MODE 9 total Pon	pwr_E1_M9_totalPon			W	not shared	-	
41	Power loss normal	pwr_E1_Ploss_norm	4329,000000		W	internally linked	4329,000000	thermal
42	Power loss eclipse	pwr_E1_Ploss_safe	4788,000000		W	internally linked	4788,000000	thermal
43								
44	Power Equipment Details							
45	Unit 1							
46	Unit Name	pwr_E1_unit1_name	Solar Array			internally linked	Solar Array	
47	Number of Units	pwr_E1_unit1_nr	3,000000		-	internally linked	3,000000	System
48	Mass without margin	pwr_E1_unit1_mass	2,170500		kg	internally linked	2,170500	
49	Mass margin to be applied	pwr_E1_unit1_massmargin	10,000000		%	internally linked	10,000000	
50	Dimension 1 (Length)	pwr_E1_unit1_dim1	1,184000		m	internally linked	1,184000	Length

Figure 4 - 14 – Outputs worksheets of the Power workbook

These elements constitute the working principles of the Concurrent Design Tool, and ESA suggests and encourage the circulation of this common “system”, in order to create a network of institutions capable of exchanging data using this common mean: on the contrary, the logic, sizing criteria, formulas, database and so on that are not this workbook and the worksheets present in any of the design workbooks, are responsibility of the institutions, that can act independently from any suggestion or obligation

4.1.2.2.2. Concurrent Design Tool core

A detailed description of the working principles of the CDT was necessary in order to have a brighter understating of the functioning of the entire system; while the working principles are the same, as already underlined, for all the institutions that wants to implement such a design methodology for concurrent design, the core of the tool needs to be developed by the institution, and in the next pages will be shown the main features of the tool I’ve developed.

The workbooks developed are:

- AOCS (Attitude and Orbit Control System)
- Communication
- Data Handling
- Instruments (Payload)
- Mission
- Power
- Propulsion
- Structures
- System
- Thermal

In the following, some of the workbooks here listed will be described more in details, with the aim of giving a more depth look into the functioning of the *Concurrent Design Tool*. For the workbooks do not described here, is anyway available a dedicated user guide.

AOCS (Attitude and Orbit Control System)

This workbook provides a preliminary design of satellite AOCS. Approximation and conservative simplification are actuated to produce components sizing. The purpose of the AOCS workbook is that of an enabling tool to facilitate the design of an AOCS, to pre-phase A level, for any given mission in an efficient manner. The workbook is provided with all the features necessary for easy exchange of information to and from other workbooks in a controlled manner.

The whole workbook is composed by the following *worksheets:*

- **Input:** This worksheet loads data from the *data exchange* workbook and provide inputs for the AOCS workbook.
- **Input Highlights:** This worksheet shows some main inputs in a simplified mode.
- **AOCS Summary:** This worksheet presents overall results and provides the AOCS bit rate.
- **External Torques:** This worksheet calculates external perturbation torques acting on the S/C.
- **Attitude Maneuvers:** It calculates the required pulses and the required wheels momentum capability to perform slew and de-spin maneuvers.
- **Attitude Control Wheels:** This worksheet provides the sizing in case of control using wheels. It also calculates the total required pulse for slew maneuvers using thrusters.
- **Attitude Control Magnetic:** This worksheet provides the sizing in case of control using magnetic torquers.
- **Sensor Selection:** This worksheet provides an interface to select required sensors and actuators.
- **Equipment summary:** It contains a list of all selected units and provides their main features.
- **Magnetic coil design:** It provides the sizing of a magnetic torquer.
- **Data base:** Many worksheets provide lists of different components with their features and performances.
- **Output:** It loads output data from the core of the AOCS workbook and sent the results to the *data exchange* workbook.

In this section each worksheet abovementioned is analyzed, considering its structure and functionalities. Many figures are useful to understand how to use the whole tool.

1. Input

The Inputs sheet acts as the source sheet for all of the main AOCS requirements and variables for the study. The format, layout and functionality are identical to those of the standard CDF input sheet. The purpose of the Inputs sheet is to collect and update the AOCS required inputs from other subsystem workbooks via the *data exchange* workbook. Inputs are received mainly from the *structure* subsystem and from *mission* and *system* workbooks.

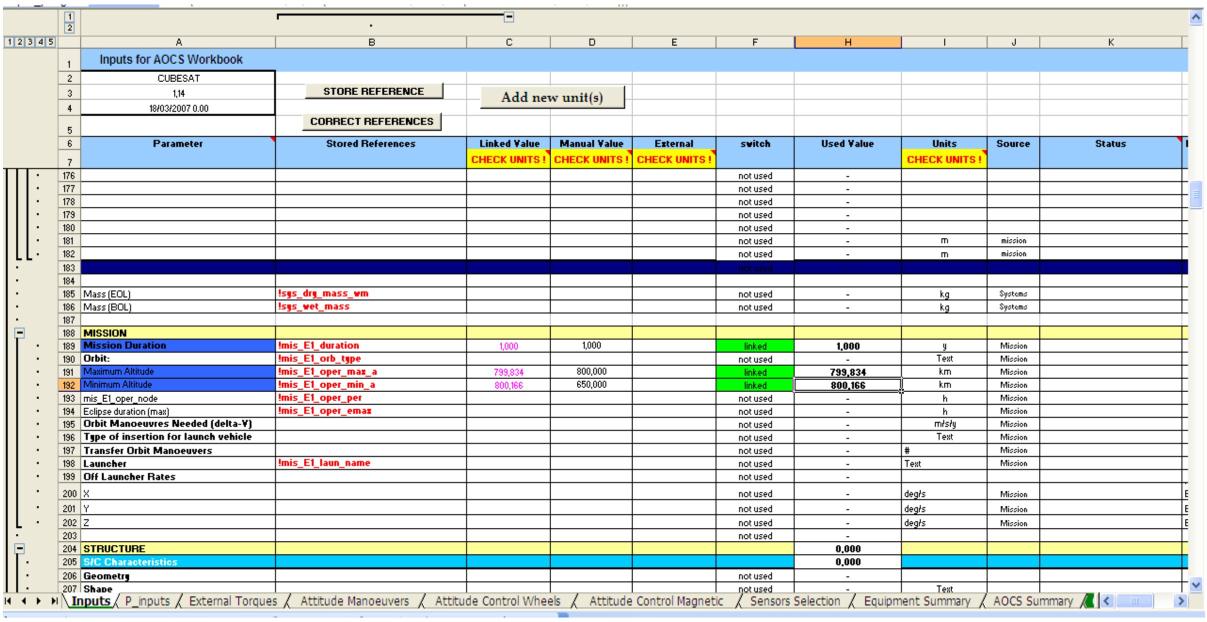


Figure 4 - 15 – Input worksheet.

2. Inputs Highlights

This worksheet shows some main inputs in a simplified mode.

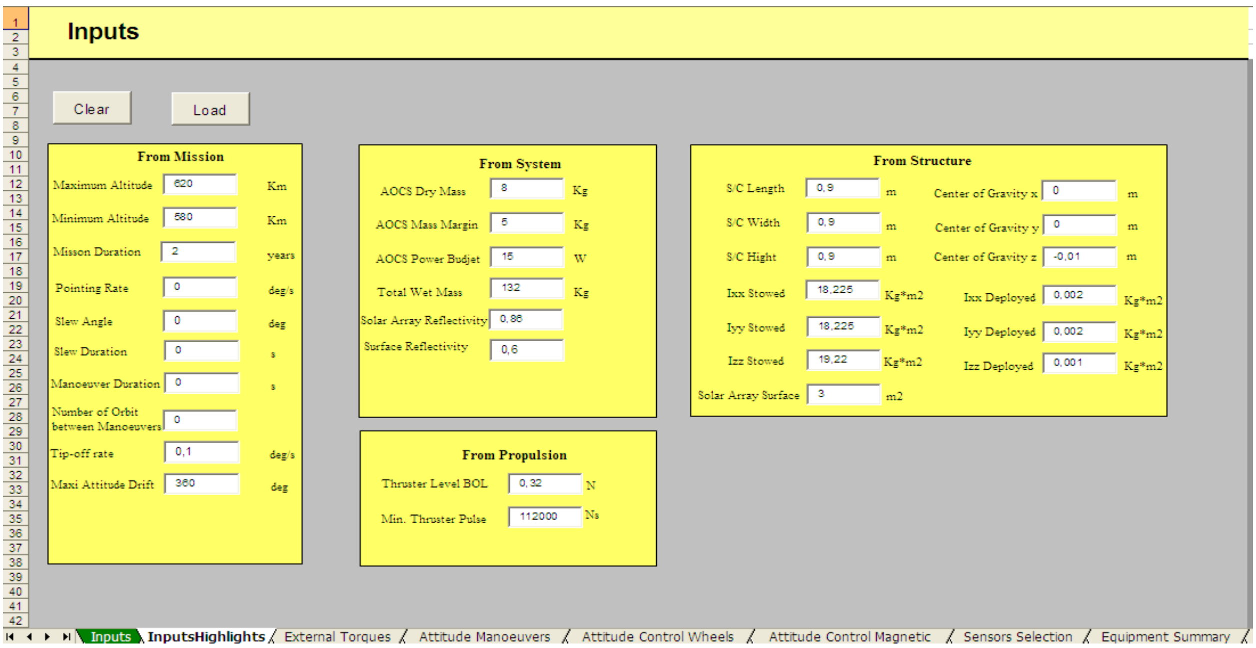


Figure 4 - 16 – Inputs Highlights worksheet

3. Output

The Outputs sheet acts as the source sheet for all of the main AOCS data passed to the Data Exchange for use by other subsystems. The purpose of the *Outputs* sheet is to collect and update the AOCS summary information that is required by other subsystem workbooks, sending this information to the data exchange; as such it also acts largely as a summary of the AOCS main details. Note that most, but not all, of the details on the output sheet are linked to values provided in one of the other core worksheets.

Data are sent to this worksheet, mainly from the *AOCS summary* and *equipment selection* worksheets.

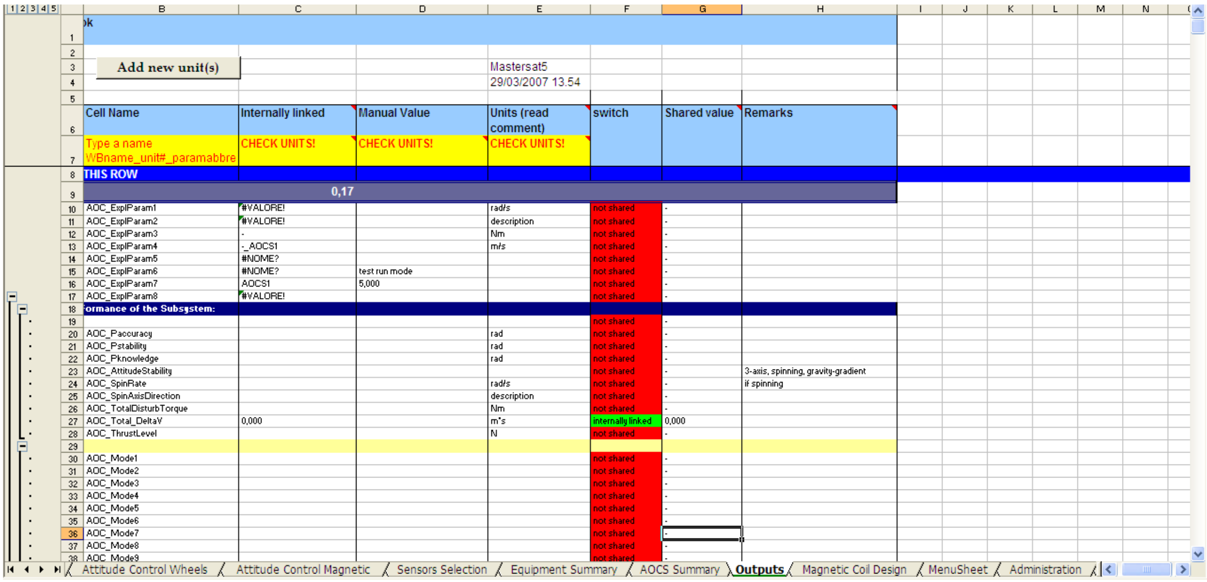


Figure 4 - 17 – Output worksheet

4. AOCS Summary

The purpose of this worksheet is to show the AOCS budgets, which are calculated in the whole workbook. This sheet summarizes all the results that are provided in the workbook.

Mass and power budget are calculated with their margin. Margin factors are calculated using the ESA procedure: a 5% margin is applied on the mass and required power for totally developed units, while a 10% and 20% margin factors are applied respectively for to be changed and to be developed units. The whole margin is then calculated as a pondered average of the units' margin factors.

Total number of elements and of supply lines is also provided. This worksheet also calculates the power consumption in safe mode.

Finally the summary worksheet estimates the bit rate and the computational through-put for sensing and control operations.

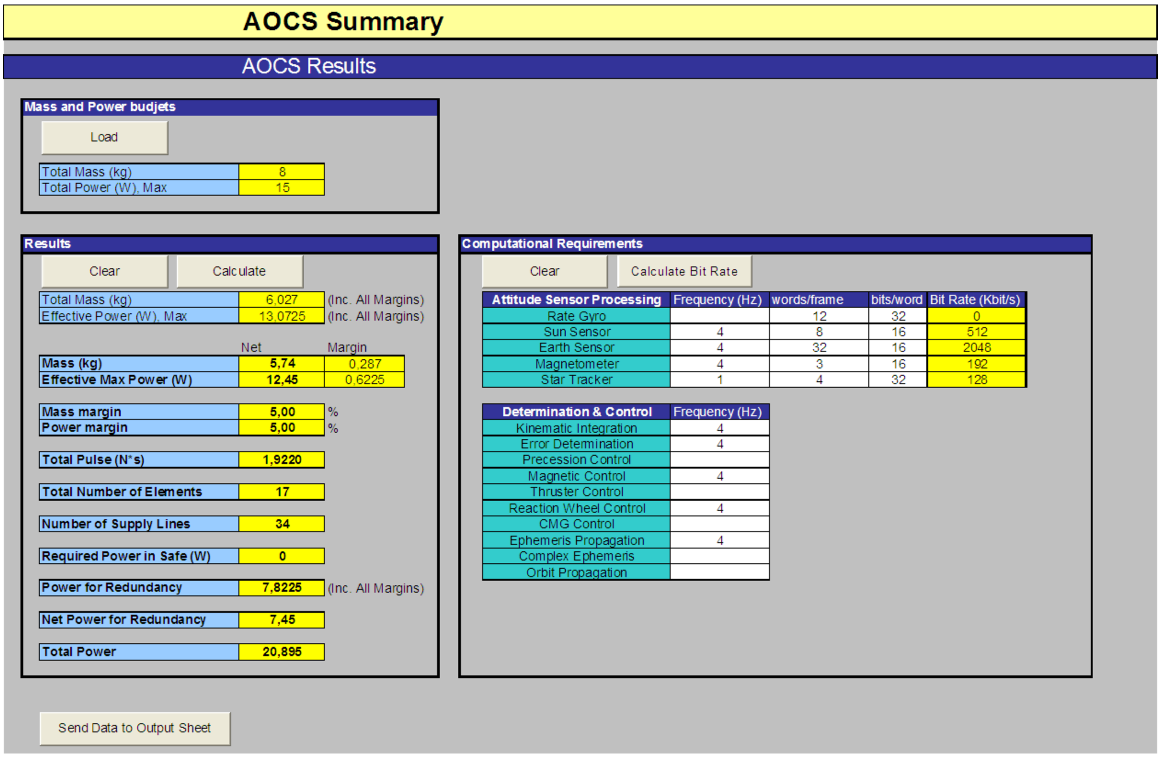
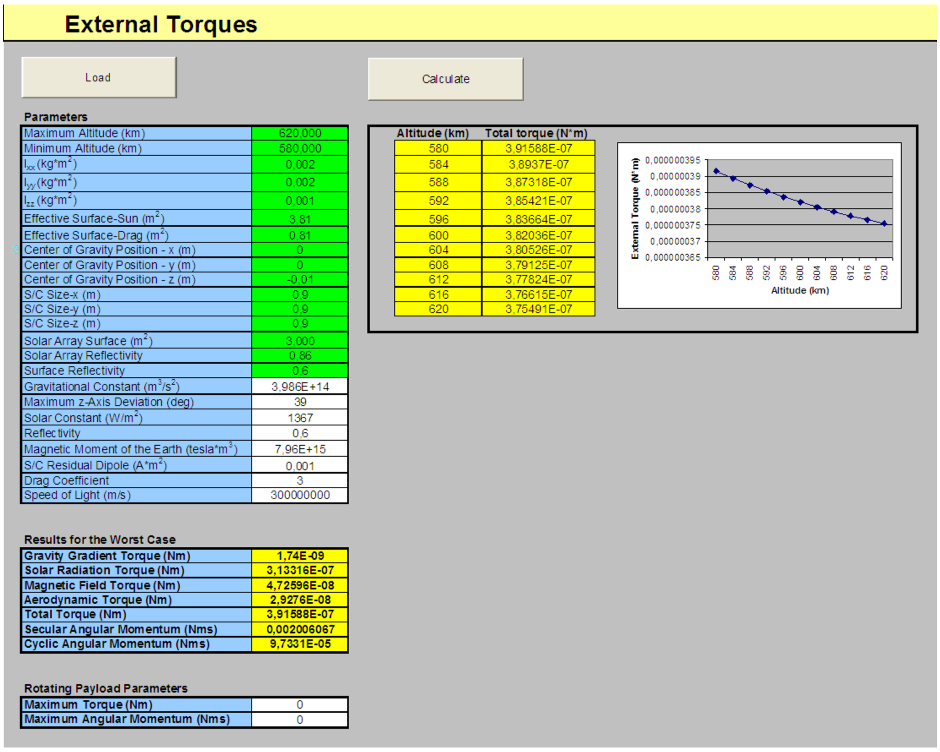


Figure 4 - 18 – AOCS Summary worksheet

A command button at the bottom of the worksheet allows sending data to the output worksheet.

5. External Torques

In this worksheet all external perturbation torques are analyzed.



External Torques

Load

Calculate

Parameters

Maximum Altitude (km)	620,000
Minimum Altitude (km)	580,000
I_{xx} (kg*m^2)	0,002
I_{yy} (kg*m^2)	0,002
I_{zz} (kg*m^2)	0,001
Effective Surface-Sun (m^2)	3,81
Effective Surface-Drag (m^2)	0,81
Center of Gravity Position - x (m)	0
Center of Gravity Position - y (m)	0
Center of Gravity Position - z (m)	-0,01
S/C Size-x (m)	0,9
S/C Size-y (m)	0,9
S/C Size-z (m)	0,9
Solar Array Surface (m^2)	3,000
Solar Array Reflectivity	0,86
Surface Reflectivity	0,6
Gravitational Constant (m^3/s^2)	3,986E+14
Maximum z-Axis Deviation (deg)	39
Solar Constant (W/m^2)	1367
Reflectivity	0,6
Magnetic Moment of the Earth (tesla*m^3)	7,96E+15
S/C Residual Dipole (A*m^2)	0,001
Drag Coefficient	3
Speed of Light (m/s)	300000000

Results for the Worst Case

Gravity Gradient Torque (Nm)	1,74E-09
Solar Radiation Torque (Nm)	3,13316E-07
Magnetic Field Torque (Nm)	4,72596E-08
Aerodynamic Torque (Nm)	2,9276E-08
Total Torque (Nm)	3,91588E-07
Secular Angular Momentum (Nms)	0,002006067
Cyclic Angular Momentum (Nms)	9,7331E-05

Rotating Payload Parameters

Maximum Torque (Nm)	0
Maximum Angular Momentum (Nms)	0

Altitude (km)	Total torque (N*m)
580	3,91588E-07
584	3,8937E-07
588	3,87318E-07
592	3,85421E-07
596	3,83664E-07
600	3,82036E-07
604	3,80526E-07
608	3,79125E-07
612	3,77824E-07
616	3,76615E-07
620	3,75491E-07

Figure 4 - 19 – External Torques worksheet

The tool takes S/C parameters from the input worksheet by clicking the control button "Calculate". It also receives the lowest and the highest altitude during the whole mission.

The altitude box is then sampled in ten points and the total perturbation torque, and each perturbation torque is calculated for in these points. The tool considers the worst case and prints the corresponding perturbation torques.

We consider the Earth magnetic field as a cyclic perturbation, whereas the other ones are considered secular actions. So cyclic and secular angular momentum, integrated over one orbit, is then calculated. Total torque corresponding to different altitudes is then plotted.

It also possible to counteract, as disturbances, the Torque and the Momentum due to a rotating or moving Payload.

6. Attitude maneuvers

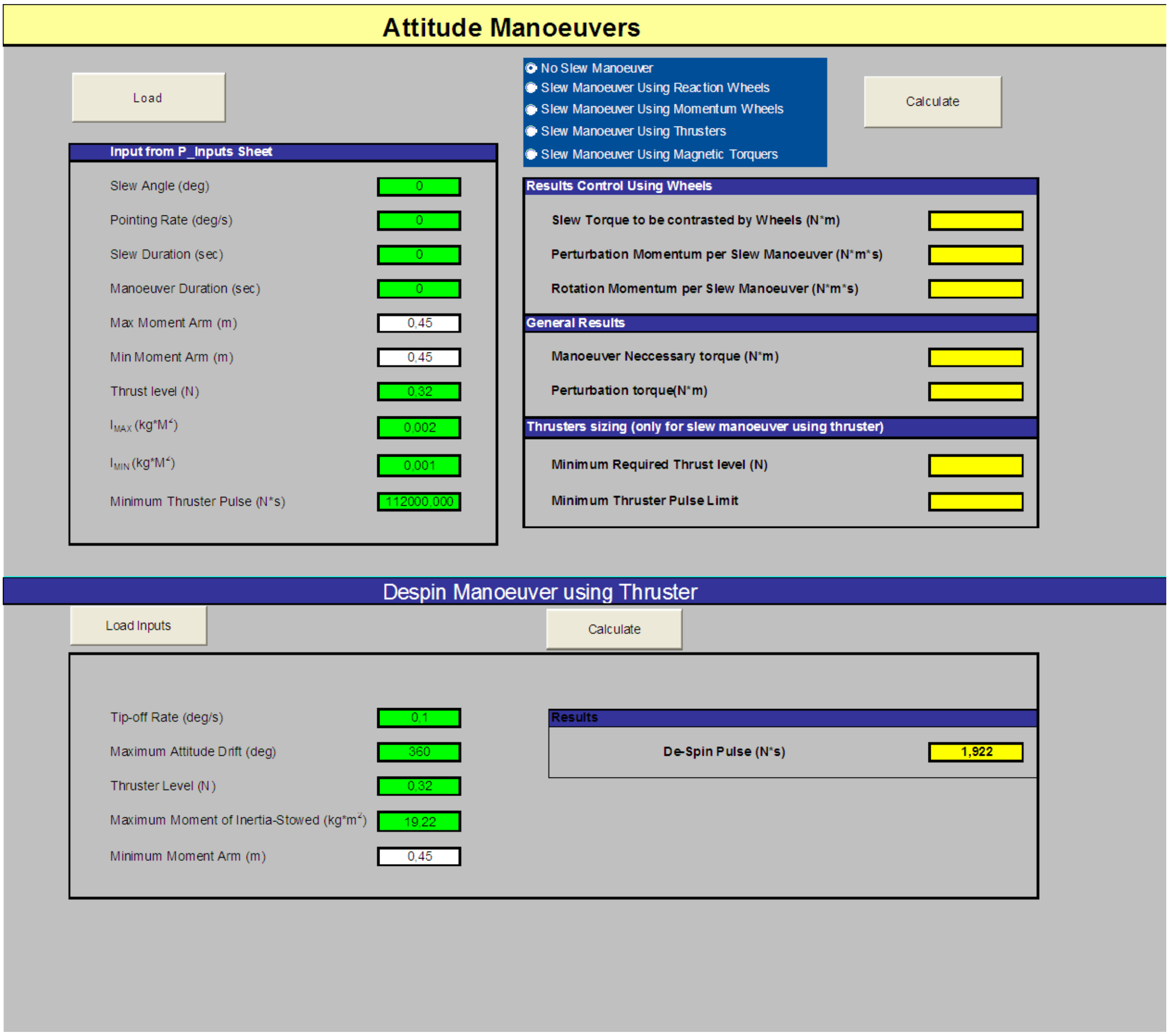


Figure 4 - 20 – Attitude Maneuvers worksheet

In this worksheet a computation for attitude manoeuvres performances and requirements are performed.

It starts from the analysis of slew manoeuvres. You can choice, by means of the *option buttons*, the actuators to be used to perform the required manoeuvre. If you select thrusters, the suitability of the used actuators will be analyzed, and a *message box* will be shown if this is not a good choice. Thrusters' minimum impulse bit and maximum thrust are considered to make this evaluation. Manoeuvre parameters are taken from the input worksheet. Additional angular momentum to be stored by reaction wheels, required torque authority, and propellant consumption are then calculated.

De-spin manoeuvres, actuated using thrusters, are then considered. The launcher *tip-off rate* and the *maximum angular drift* are the most important input parameters. If the torque produced by thrusters is not enough to perform the required manoeuvre, a *message box* will suggest to choose a larger *maximum angular drift* Propellant consumption is then estimated.

7. Attitude control using wheels

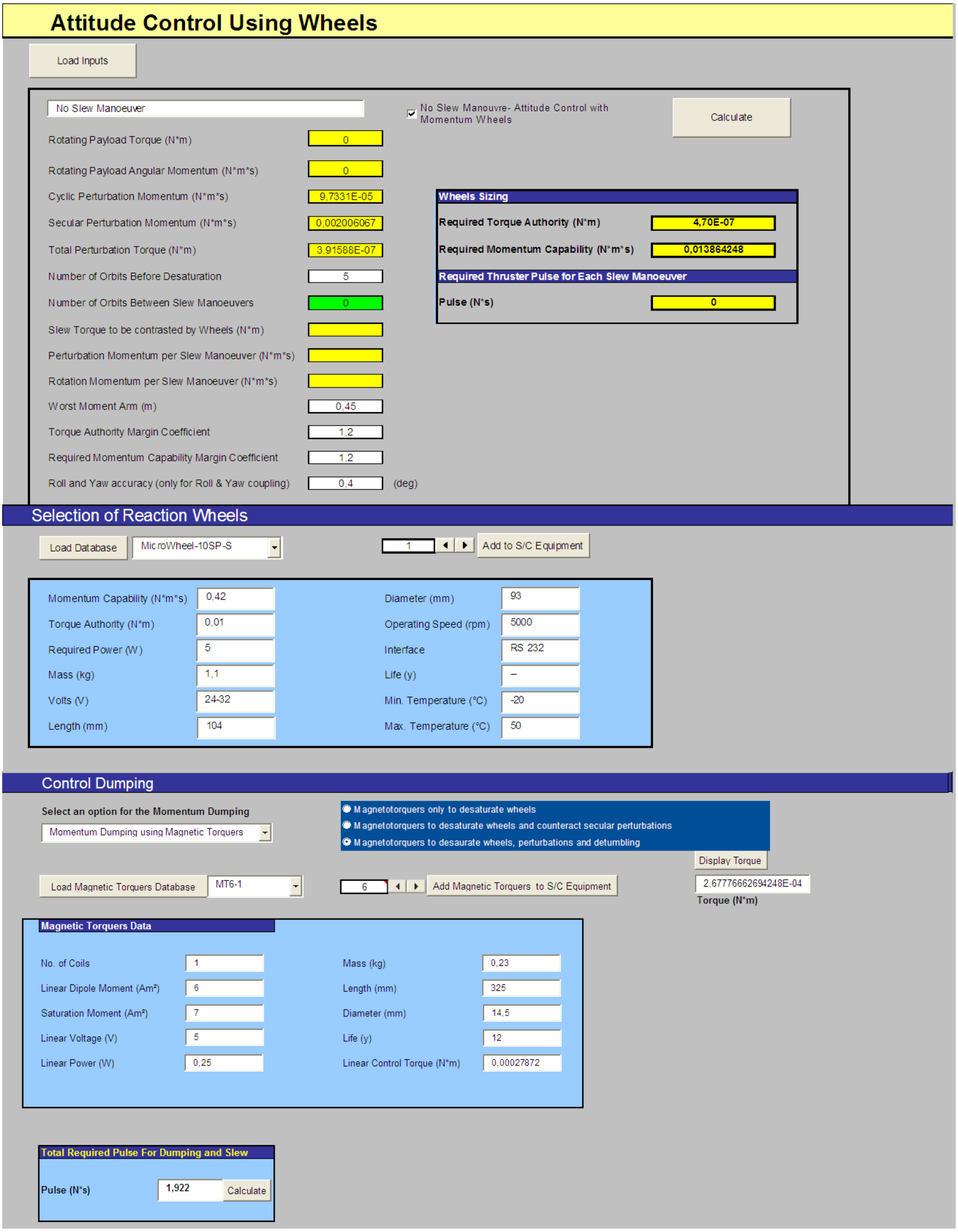


Figure 4 - 21 – Attitude Control Wheels worksheet - requirements calculation

This worksheet calculates the actuators requirements, taking information from the previous worksheets (previous figure).

Angular momentum, and torque required for both perturbation countering and manoeuvring are considered. The principal *trade-off* is the selection of the number of orbit between desaturation manoeuvres. Using *option buttons* you can select how to perform slew manoeuvres if they are required. The manoeuvres frequency is also required. After inserting margin factors for both torques and angular momentum, the wheels requirements can be calculated.

This dynamics requirements lead to the selection of the hardware, which are suitable to control the spacecraft attitude (next figure).

The first step is the selection of the *reaction wheels*. By clicking the command button Load Database", a list of the reaction wheels listed in the database worksheet is loaded in the *combo-box*. Selecting a model in the *combo- box*, all the features and performances of the selected wheel will be shown. If it fit the requirements you can select the number of wheels to be used and send data to the *Equipment Summary* worksheet.

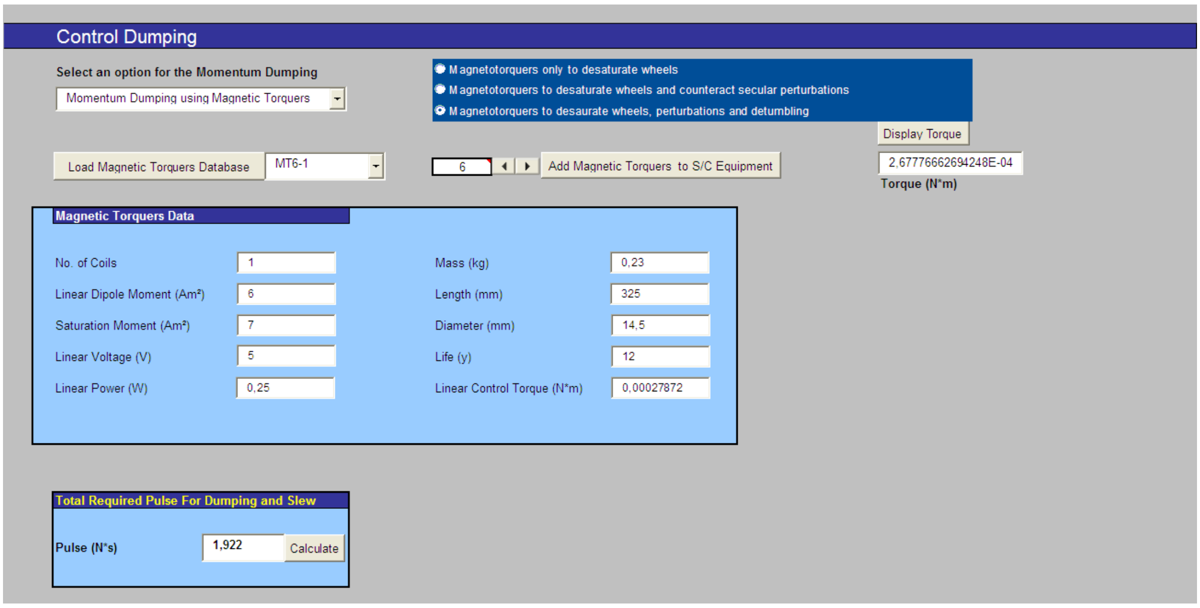


Figure 4 - 22 – Attitude Control Wheels worksheet - hardware selection

The following step is the selection of the actuators to be used to perform the desaturation manoeuvre. If you select Magnetic torquers, you must select the model to be used and then you can send data to the Equipment Summary worksheet. You can also decide if magnetic torquers are used only for desaturation (typically in a zero-momentum system of control); for desaturation and control of secular torques (typically for momentum bias system of control); also to perform the detumbling (in case of no thrusters are considered). Finally the total required pulse is calculated.

8. Attitude Control Magnetic

This worksheet provides the sizing in case of control using magnetic torquers. Inputs are loaded from the previous sheets, and then the required torques are calculated. Then it should be selected the desired magnetic torquers from the combo (it is filled reading data from the datasheets).

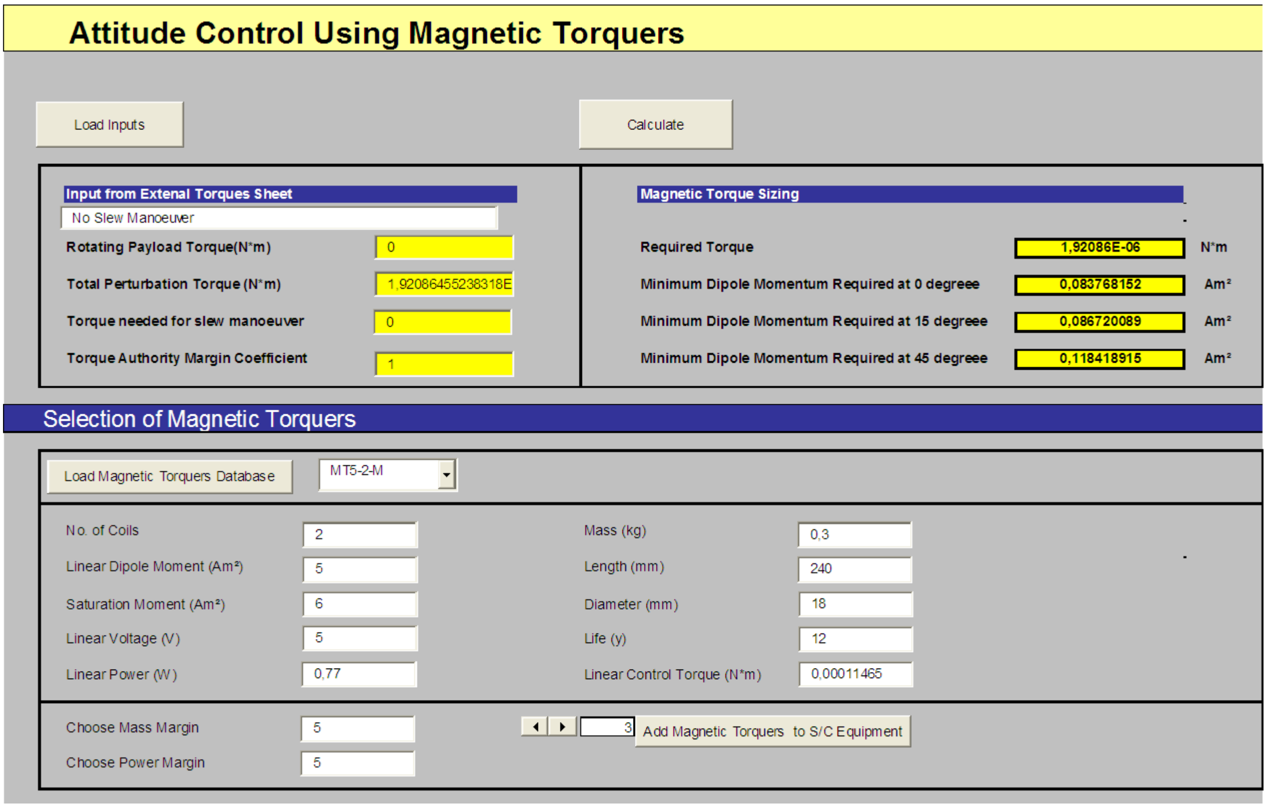


Figure 4 - 23 – Attitude Control Magnetic worksheet

9. Sensor selection

This worksheet provides an interface for the equipment selection.

Component class can be selected in the first combo–box. The data format changes by selecting the unit class. On the left side the unit features are presented, which will be sent to the equipment summary worksheet. On the right side the unit performance are provided.

After selecting the unit model, you must select the unit's number and sent data to the equipment summary.

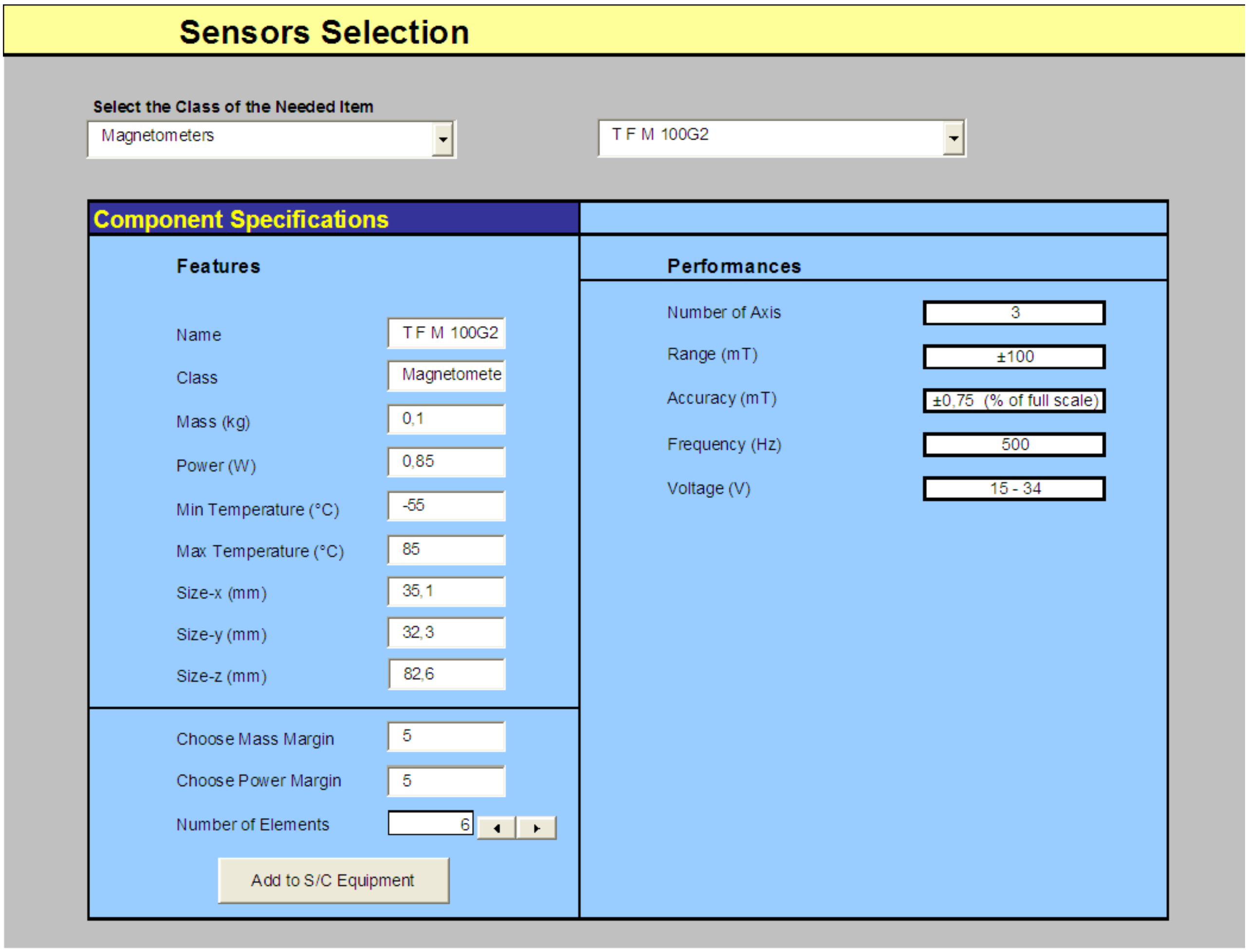


Figure 4 - 24 – Sensor Selection worksheet

10. Equipment summary

Selected units are sent to this worksheet and are listed with their features. When a row is full the next component is print in the following row, for a maximum of 10 elements kind. In the last column the required power in safe mode must be inserted manually.

If you want to remove an item from the equipment you must click the button on the corresponding row.

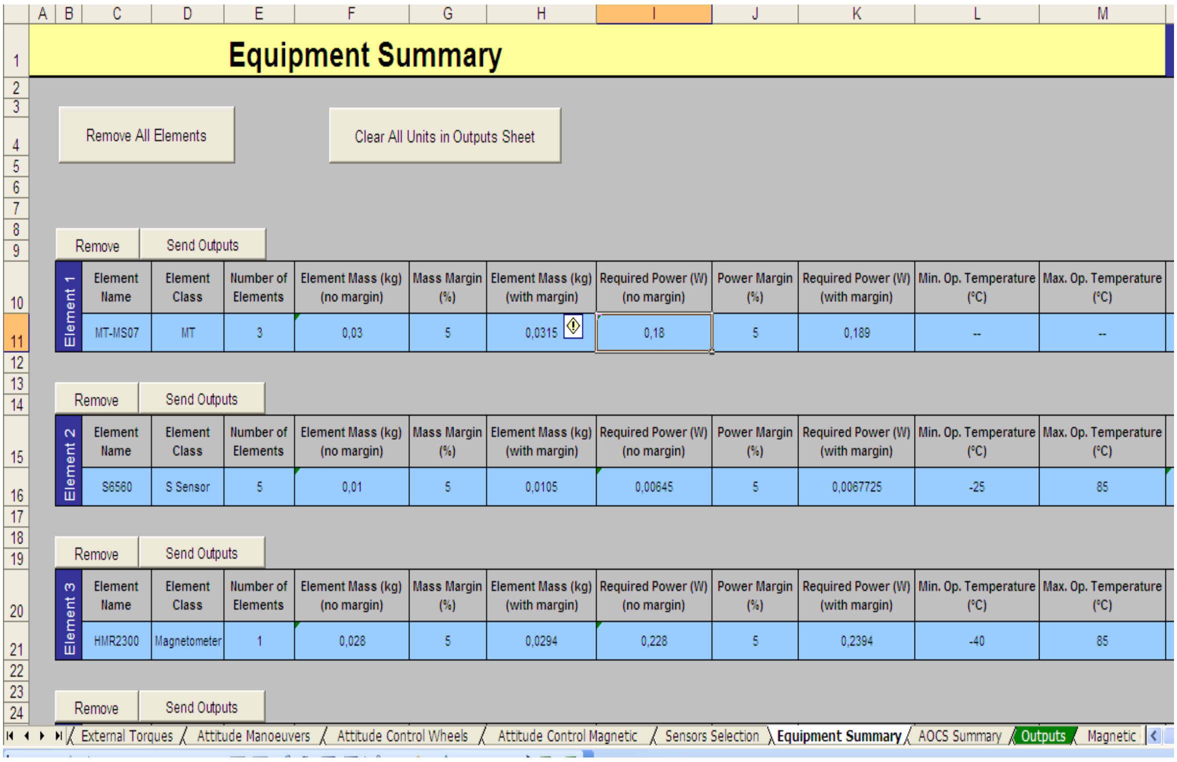


Figure 4 - 25 – Equipment Summary worksheet

By clicking the buttons “Send Outputs” all features of a unit are sent to output worksheet. By clicking the button “Clear All Units in The Outputs Sheet” all the equipment will be removed from the output worksheet.

11. Magnetic coil design

It provides the sizing of a magnetic torquer. The required data are loaded from the previous sheets, and then, depending on the mission needs, it’s possible to design appropriate magnetic coils (it should be useful in the pico satellites’ design).

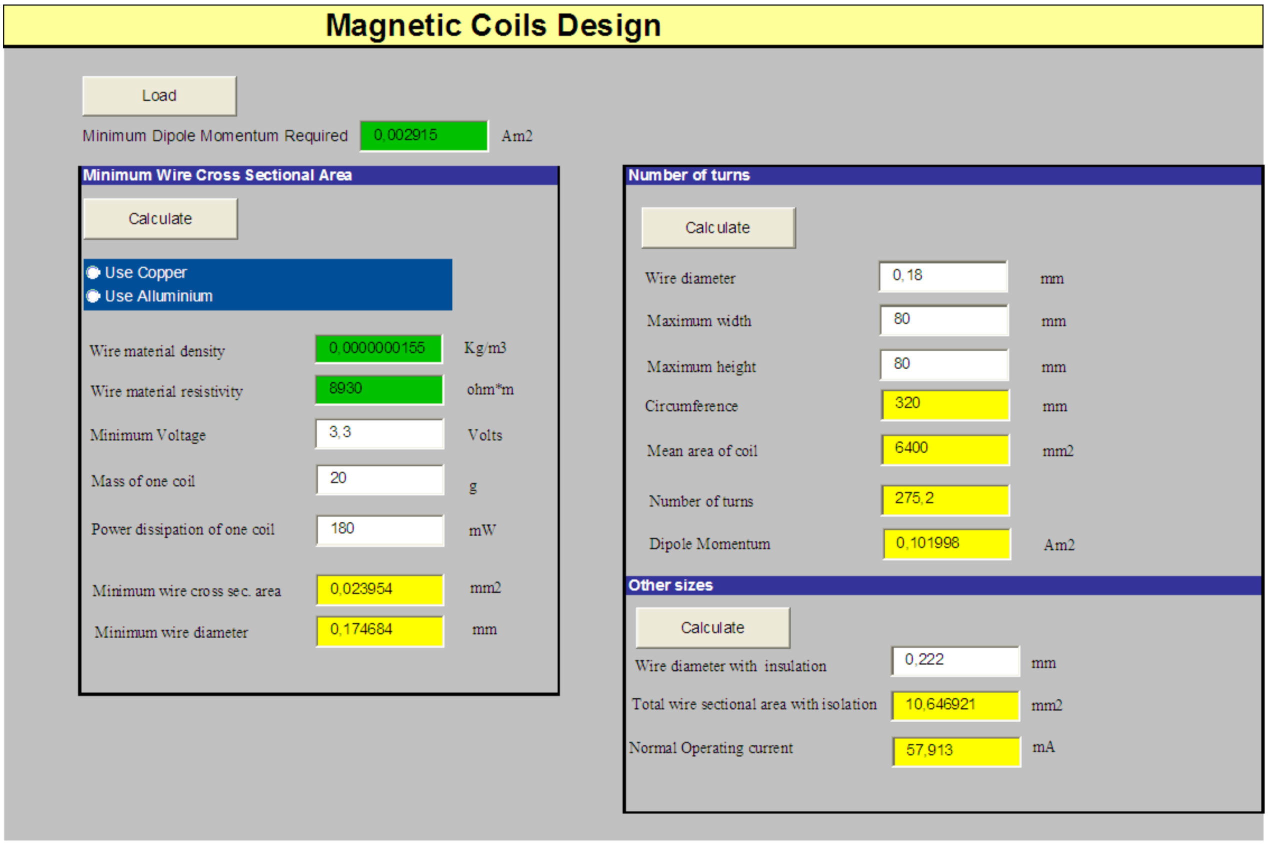


Figure 4 - 26 – Magnetic Coil Design worksheet

12. Datasheets

For each components class a database was built, containing the units' features and performances. The realized datasheets are listed below.

- Magnetic Torquers
- Reaction Wheels
- Magnetometers
- Permanent Magnets
- Sun Sensors
- Earth Sensors
- Star Tracker
- Gyros
- GPS receiver and antenna

Reaction Wheels Data Sheet

Company	Model name	Momentum Capability (N*m*s)	Torque Authority (N*m)	Power Max (W)	Power Nom. (W)	Weight (kg)	Volts	Length (mm)	Diameter (mm)	Operating Speed (rpm)	Interface	Life (y)
Valley Forge Composite Technologies	VF MR 0.1	0,1	0,012	14	3	1,5	24-34	100	140	3000	Mil 1553 B	5
	VF MR 0.3	0,3	0,012	14	3	1,7	24-34	100	170	3000	Mil 1553 B	5
	VF MR 4.0	4	0,02	19	3	2,6	24-34	100	200	6000	Mil 1553 B	5
	VF MR 4.0A	4	0,025	25	5	3	24-34	110	220	4500	Mil 1553 B	5
	VF MR 5.0	5	0,025	25	5	3,5	24-34	120	230	4500	Mil 1553 B	5
	VF MR 10.0	10	0,03	30	5	5	24-34	150	250	4500	Mil 1553 B	5
	VF MR 1.0	1	0,00002	27	--	1,5	24-34	100	150	--	RS 422	5
	VF MR 2.0	2	0,00002	17,5	--	2	24-34	100	170	--	RS 422	5
	VF MR 8.0	8	0,00002	29	--	3	24-34	100	190	--	RS 422	5
	VF MR 14.0	14	0,0001	75	--	5	24-34	150	260	--	RS 422	5
	VF MR 19.6	19,6	0,00026	70	--	10,5	24-34	170	390	--	RS 422	5
	VF MR 29.4	29,4	0,00035	100	--	14	24-34	170	390	--	RS 422	5
Honeywell	HR04	0,7	0,028	6	--	1,3	12-34	54	130	9000	RS 422	5-9
	HR12 (1)	12	0,2	195	22	6	14-80	159	316	6000	RS 422	15
	HR12 (2)	25	0,2	195	22	7	14-80	159	316	6000	RS 422	15
	HR12 (3)	50	0,2	195	22	9,5	14-80	159	316	6000	RS 422	15
	HR14 (1)	25	0,2	195	22	7,5	14-80	159	366	6000	RS 422	15
	HR14 (2)	50	0,2	195	22	8,5	14-80	159	366	6000	RS 422	15
	HR14 (3)	75	0,2	195	22	10,6	14-80	159	366	6000	RS 422	15
	HR16 (1)	50	0,2	195	22	9	14-80	178	418	6000	RS 422	15
	HR16 (2)	75	0,2	195	22	10,4	14-80	178	418	6000	RS 422	15
	HR16 (3)	100	0,2	195	22	12	14-80	178	418	6000	RS 422	15

Figure 4 - 27 – Reaction Wheels datasheet

Magnetic Torquers Data Sheet

Company	Model Name	No. of Coils	Linear Dipole Moment (Am²)	Saturation Moment (Am²)	Linear Voltage (V)	Linear Power (W)	Mass (kg)	Length (mm)	Diameter (mm)	Life (y)	Minimum Operational Temperature (°C)	Maximum Operational Temperature (°C)
Zarm Technik	MT2-1	1	2	2,5	5	0,5	0,2	157,5	15	12	-30	70
	MT5-2-M	2	5	6	5	0,77	0,3	240	18	12	-30	70
	MT6-1	1	6	7	5	0,25	0,23	325	14,5	12	-30	70
	MT10-2-H	2	10	12	10	1	0,35	330	17	12	-30	70
	MT15-1M	1	15	20	14	1,11	0,43	329,5	17	12	-30	70
	MT30-2-GRC	2	30	35	12,5	1,5	1,4	404,5	29	12	-30	70
	MT70-1	1	70	75	28	3,8	2,6	400	39	12	-30	70
	MT70-2	2	70	75	24	2,6	2,2	581	30	12	-30	70
	MT80-1	1	80	100	10	3	4,12	380	50	12	-30	70
	MT110-2	2	110	120	12	2,9	3,8	600	40	12	-30	70
	MT140-2	2	145	170	10	1,9	5,3	680	43	12	-30	70
	MT250-2	2	250	300	28	4,8	5,5	883	37	12	-30	70
	MT400-2-L	2	400	500	18,5	9	7,8	952	41	12	-30	70
	MT400-2	2	400	550	21	11,4	11	750	56	12	-30	70

Figure 4 - 28 – Magnetic Torquers datasheet

Communication

In the following, a practical guide to use the Excel Workbook developed for the Concurrent Design Tool will be provided.

1. Inputs

The first worksheet is called Inputs. It directly follows the standard ESA CDF guidelines, in fact input variable values can be updated from the central *Data Exchange* Excel file by means of the native macro called *Update Input values*, in *CDF Functions* menu. Obviously corresponding switch buttons must be set to *linked* and the cells in the column *Linked Value* must contain the correct links to the variable names collected in the *Data Exchange* file. Other values for switch are: *manual*, *not used*, and *external* that must be selected when the desired input value is contained in an external file or in another database.

#RIF!							
FLORAD	STORE REFERENCE	Add new unit(s)					
1,02							
01/04/2008 0.00	CORRECT REFERENCES						
Parameter	Stored References	Linked Value CHECK UNITS !	Manual Value CHECK UNITS !	External CHECK UNITS !	switch	Cell Name	Used Value
SYSTEM							
Element Information							
Constants							
General Characteristics							
Target Launch Mass					not used	sys_target_launchmass	
Satellite Dry Mass (with margins)					not used	sys_dry_mass_wm	-
Satellite Launch (Wet) Mass					not used	sys_wet_mass	-
Launch date					not used	sys_Launch_date	-
Design Life time					not used	sys_Miss_duration	-
On Station Duration					not used	sys_nomsolorbit_dur	-
Preliminary S/S Mass		4	8,6		linked	sys_ss_mass	4
Preliminary Power		30	26		linked	sys_ss_power	30
Orbit type		Floreale	LEO		linked		Floreale
Orbit height		600	600		linked		600
Semi-major-axis		6970	6970		linked		6970
Eccentricity		0,02			linked		0,02
Inclination		63,4	63,4		linked		63,4
TT&C Frequency		2,05	2,05		linked		2,05
Telemetry BER		0,00001	0,000001		linked		0,00001
ELEMENT 1							
Mode Information							
MISSION							
STRUCTURES							
AOCS							
GROUND SYSTEMS							
DATA HANDLING							

Figure 4 - 29 – The Inputs worksheet

2. TT&C input

This worksheet reports a summary of subsystem input. TT&C receives input from Mission and System. Most important inputs are *Mass and Power Budgets*, which are maximum values designer can attains. Each input value is linked to corresponding cell of *Inputs* worksheet.

In the bottom part a table of principal Constant used in workbook is presented.

INPUTS FROM SYSTEM

MACRO INPUT	INPUT	VARIABLE NAME	VARIABLE VALUE	MEASURE UNIT
Voltage	Minimum Voltage	V_{min}	22,00	V
	Maximum Voltage	V_{max}	34,00	V
System Budgets	Mass Budget	$M_{TT\&C}$	4,00	kg
	Power Budget	$W_{TT\&C}$	30,00	W
Bit rate (Telemetry +Data)	Bit Rate	R_b	2,00E+03	Kb/sec
Telemetry BER	Bit Error Rate	BER	1,00E-05	adim
Telemetry Bit Rate	Bit Rate	R_b	2,00E+00	Kb/sec

INPUTS FROM MISSION

MACRO INPUT	INPUT	VARIABLE NAMES	VARIABLE VALUE	MEASURE UNIT
Orbit	Average Orbit Height	H	591,86	Km
Frequency	TT&C up-link Frequency	f_{lin}	2,05	GHz
Ground Station	Average Time of Visibility	Δt	638,00	sec

Figure 4 - 30– TT&C Inputs worksheet

3. Orbit and Frequency

Before starting it is necessary to reset worksheet pushing button on upper right hand corner. In this worksheet designer must enter the minimum acceptable Elevation Angle. Then, from input Orbit Height, maximum Slant Range and Orbit Period are calculated. This is the range used by the link model in determining path loss. In addition, the elevation angle is used in subsequent worksheets to estimate the atmospheric losses. In fact, any elevation angle can be entered and the corresponding slant range to the satellite is calculated. This will allow an investigation of link performance as a function of elevation angle or slant range.

Routine has been designed for a generic elliptic orbit (for a circular orbit Height of Perigee and Height of Apogee are linked to the same input cell). In the second part of worksheet, Carrier Downlink Frequency and Path Loss are computed. Designer must enter the appropriate Turn-around Ratio (transponder is supposed to work in coherent mode). In bottom part of this worksheet a figure is provided to help the operator to see the geometry associated with the link

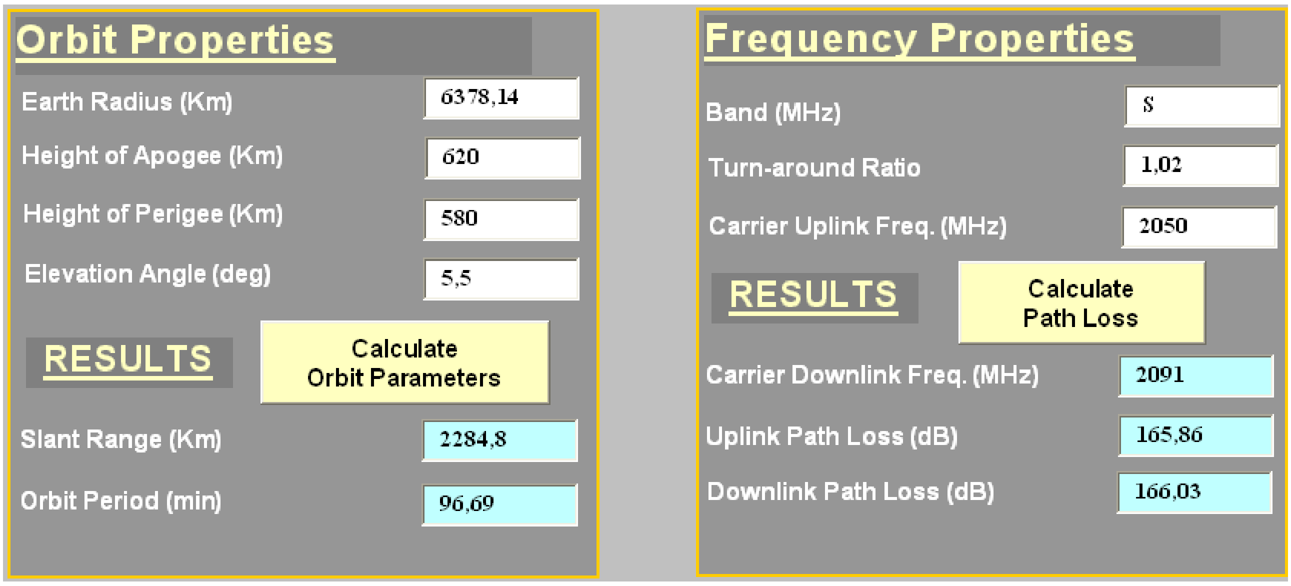


Figure 4 - 31 – ORBIT & FREQUENCY worksheet

4. Equipment Selection

In this worksheet designer selects equipments units of TT&C subsystem. Pushing the *Select Unit* button a window is opened and designer can select equipment type (transponder, RFDU or antenna), model and number of elements.

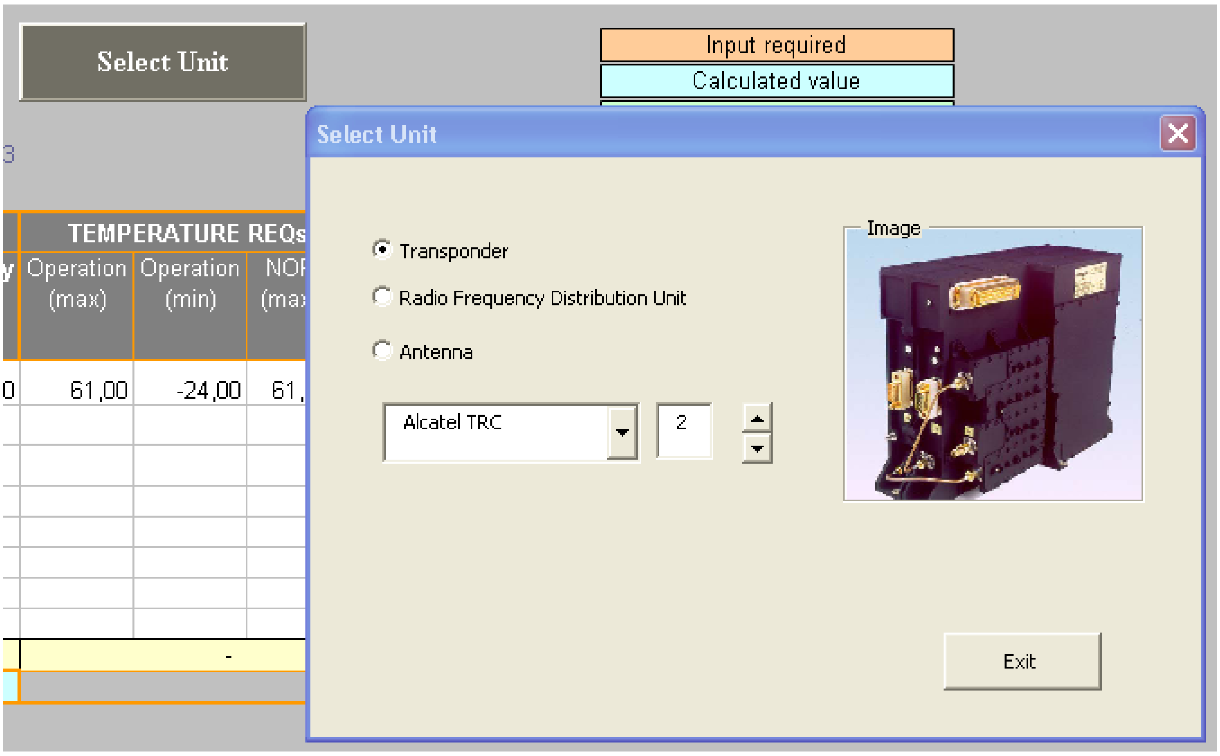


Figure 4 - 32 – EQUIPMENT SELECTION User Form

Automatically principal physic characteristics are loaded in the table. Then it's possible to see these physical parameters pushing on buttons on the left hand of worksheet.

Element 1: -			PPE			
Unit	Element 1 Unit Name Click on button below to **insert new unit**	Quantity	Ppeak	- Pon	- Pstby	- Dc
1	Alcatel TRC	2,00	26,00	26,00	6,00	10,0
2	model 2	1,00				100,0
3	SAAB Helix Quadrifilar Antenna	2,00				100,0
4						
5						
6						
7						
8						
-						
ELEMENT 1 SUBSYSTEM TOTAL		5	26,0	26,0	6,0	

Element 1: Insert New Row

Figure 4 - 33 – EQUIPMENT SELECTION worksheet

5. Ground Station Selection

This worksheet selects Ground Station Antenna and shows to designer more important parameters of transmitting and receiving chain. After antenna selection it necessary to push Load button to upload Ground Station parameters in the following sheets.

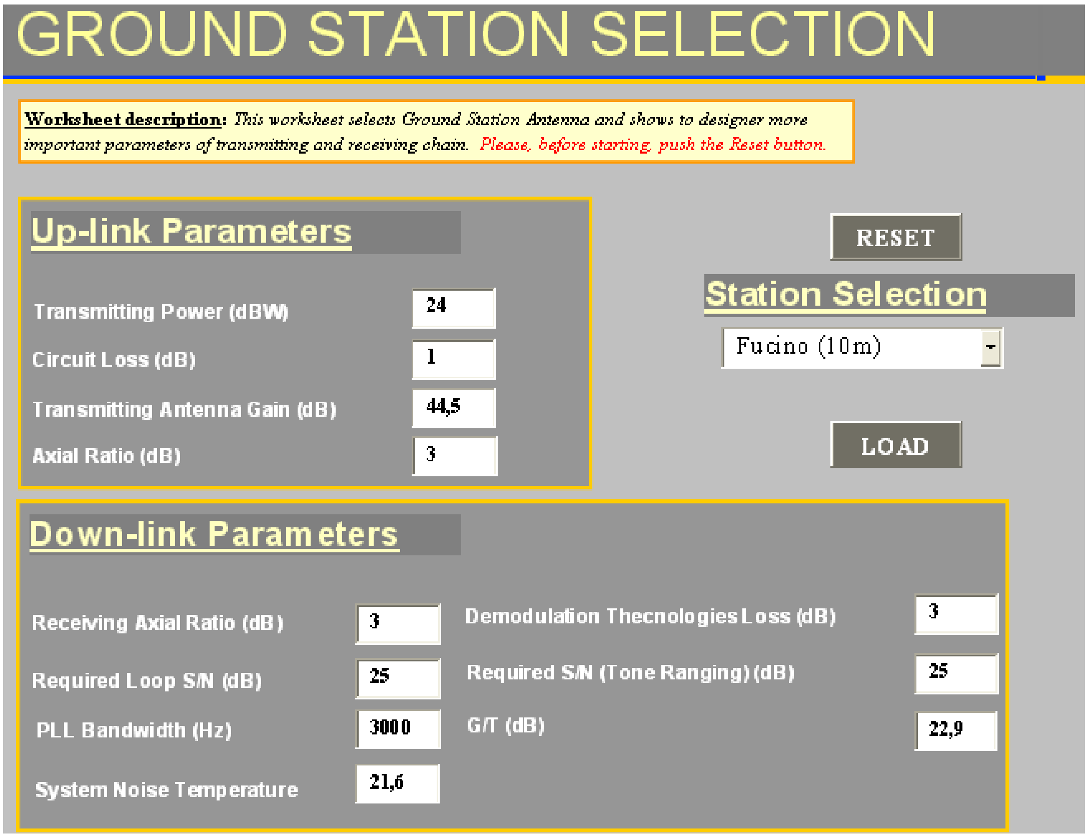
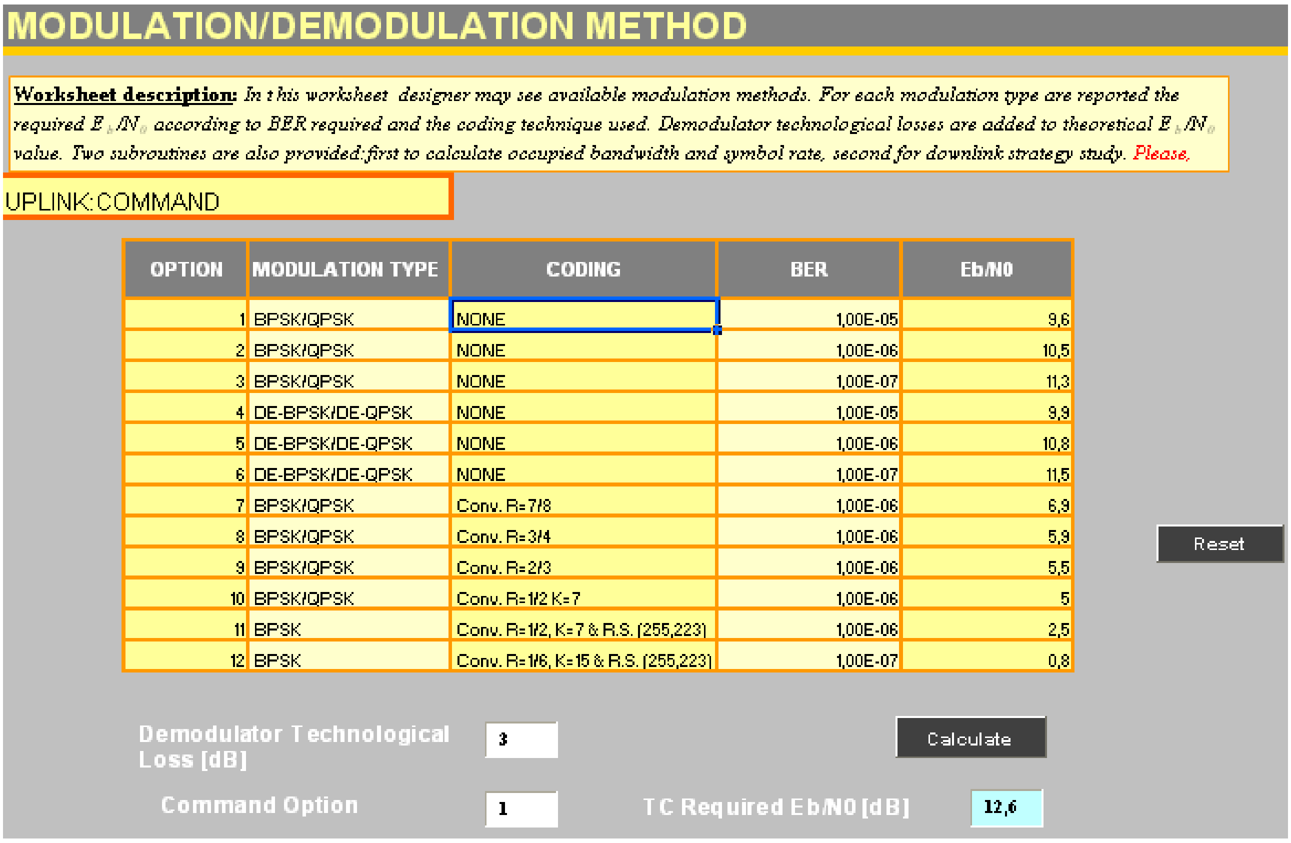


Figure 4 - 34 – GROUND STATION SELECTION worksheet.

6. Modulation Demodulation Method

In this worksheet designer may see available modulation methods. For each modulation type are reported the required E_b/N_0 according to BER required and the coding technique used. These two tables allow the link model operator to select a modulation process and bit error rate appropriately matched to the satellite being developed. Designer must enter the *Command Option* number in corresponding textbox and push *Calculate* button. *Demodulator technological losses* are added to theoretical E_b/N_0 value. In downlink section, designer must select a modulation option according with Telemetry BER value (this is one of System requirements).



MODULATION/DEMODULATION METHOD

Worksheet description: *In this worksheet designer may see available modulation methods. For each modulation type are reported the required E_b/N_0 according to BER required and the coding technique used. Demodulator technological losses are added to theoretical E_b/N_0 value. Two subroutines are also provided: first to calculate occupied bandwidth and symbol rate, second for downlink strategy study. Please,*

UPLINK:COMMAND

OPTION	MODULATION TYPE	CODING	BER	Eb/N0
1	BPSK/QPSK	NONE	1,00E-05	9,6
2	BPSK/QPSK	NONE	1,00E-06	10,5
3	BPSK/QPSK	NONE	1,00E-07	11,3
4	DE-BPSK/DE-QPSK	NONE	1,00E-05	9,9
5	DE-BPSK/DE-QPSK	NONE	1,00E-06	10,8
6	DE-BPSK/DE-QPSK	NONE	1,00E-07	11,5
7	BPSK/QPSK	Conv. R=7/8	1,00E-06	6,9
8	BPSK/QPSK	Conv. R=3/4	1,00E-06	5,9
9	BPSK/QPSK	Conv. R=2/3	1,00E-06	5,5
10	BPSK/QPSK	Conv. R=1/2 K=7	1,00E-06	5
11	BPSK	Conv. R=1/2, K=7 & R.S. (255,223)	1,00E-06	2,5
12	BPSK	Conv. R=1/6, K=15 & R.S. (255,223)	1,00E-07	0,8

Reset

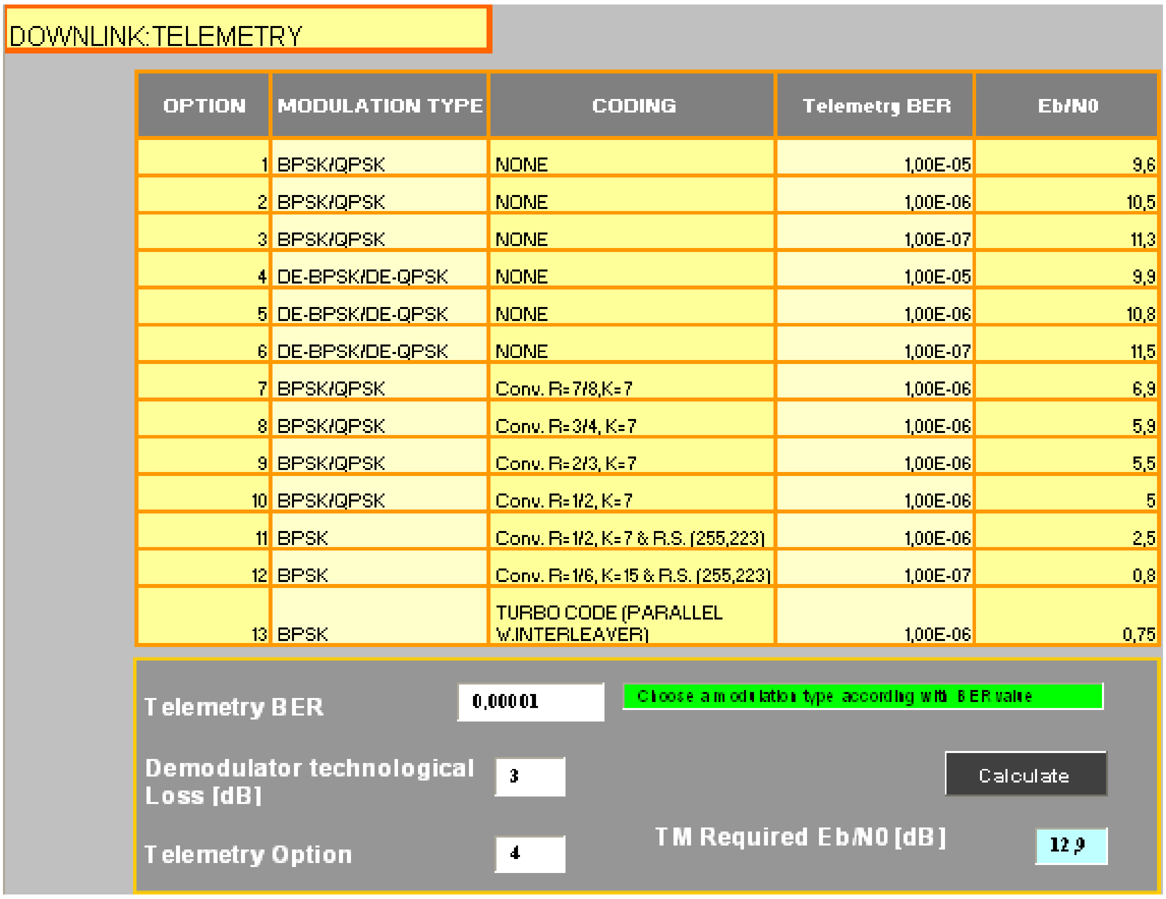
Demodulator Technological Loss [dB] 3

Calculate

Command Option 1

TC Required Eb/N0 [dB] 12,6

Figure 4 - 35 – Modulation method worksheet, Telecommand choice



DOWNLINK:TELEMETRY

OPTION	MODULATION TYPE	CODING	Telemetry BER	Eb/N0
1	BPSK/QPSK	NONE	1,00E-05	9,6
2	BPSK/QPSK	NONE	1,00E-06	10,5
3	BPSK/QPSK	NONE	1,00E-07	11,3
4	DE-BPSK/DE-QPSK	NONE	1,00E-05	9,9
5	DE-BPSK/DE-QPSK	NONE	1,00E-06	10,8
6	DE-BPSK/DE-QPSK	NONE	1,00E-07	11,5
7	BPSK/QPSK	Conv. R=7/8,K=7	1,00E-06	6,9
8	BPSK/QPSK	Conv. R=3/4, K=7	1,00E-06	5,9
9	BPSK/QPSK	Conv. R=2/3, K=7	1,00E-06	5,5
10	BPSK/QPSK	Conv. R=1/2, K=7	1,00E-06	5
11	BPSK	Conv. R=1/2, K=7 & R.S. (255,223)	1,00E-06	2,5
12	BPSK	Conv. R=1/6, K=15 & R.S. (255,223)	1,00E-07	0,8
13	BPSK	TURBO CODE (PARALLEL W.INTERLEAVER)	1,00E-06	0,75

Telemetry BER 0,00001 Choose a modulation type according with BER value

Demodulator technological Loss [dB] 3 Calculate

Telemetry Option 4 TM Required Eb/N0 [dB] 12,9

Figure 4 - 36 – Modulation method worksheet, Telemetry choice

Two subroutines are also provided: *Bandwidth* and *Downlink strategy*. The *Bandwidth* subroutine computes symbol rate and occupied bandwidth. Operator must enter the *Information Bit Rate* of data in Mbps, the *Overhead % info Rate* (the amount of "overhead" added to the information data rate to account for miscellaneous signalling requirements and initialization of modems. This is typically expressed as a percentage of the information rate. This parameter may be given as a fixed value, in which case it must be converted into a percentage before input. You can enter zero here if unknown), *FEC Code Rate* (the inner code rate used with forward error correction), *RS code* (the Reed Solomon outer code, if you are not using RS coding simply enter a value of 1) and *Roll off Factor*.

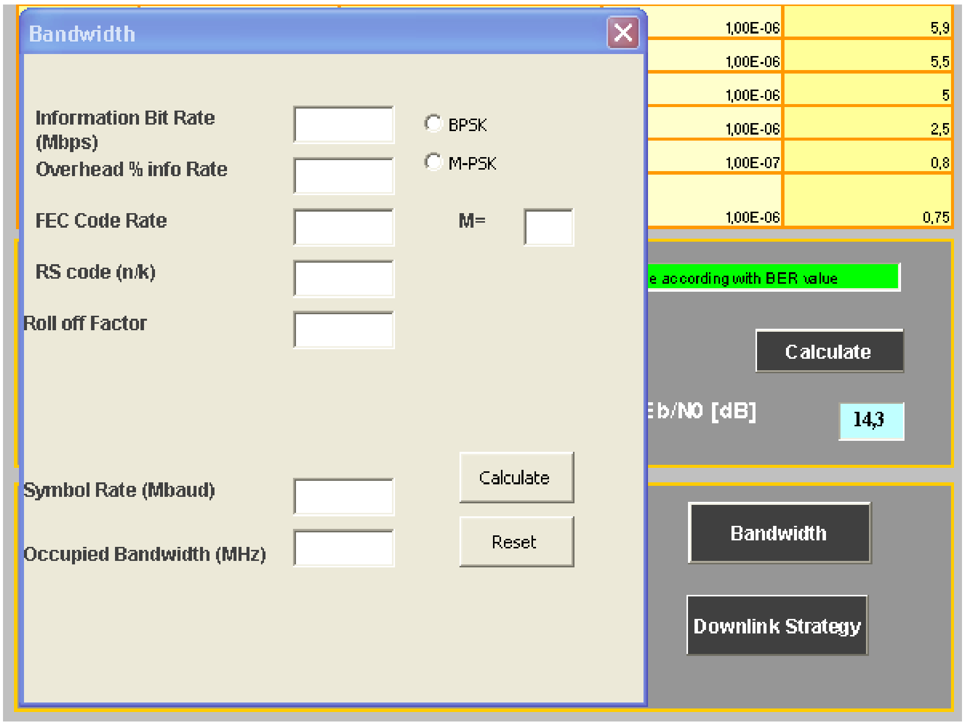


Figure 4 - 37 – Bandwidth User Form

Second subroutine has been implemented for downlink strategy study. Operator can select number of on board telemetries and corresponding sampling rates. Subroutine prevede la possibilità di considerare telemetrie a bassa e alta velocità di campionamento. We suppose telemetry data are quantized at 16 bits. Then, operator chooses number of orbit without telemetries downloading and transfer data rate. Subroutine computes the on board mass memory necessary, time of download and number of satellite passages on ground station necessary to complete downloading.

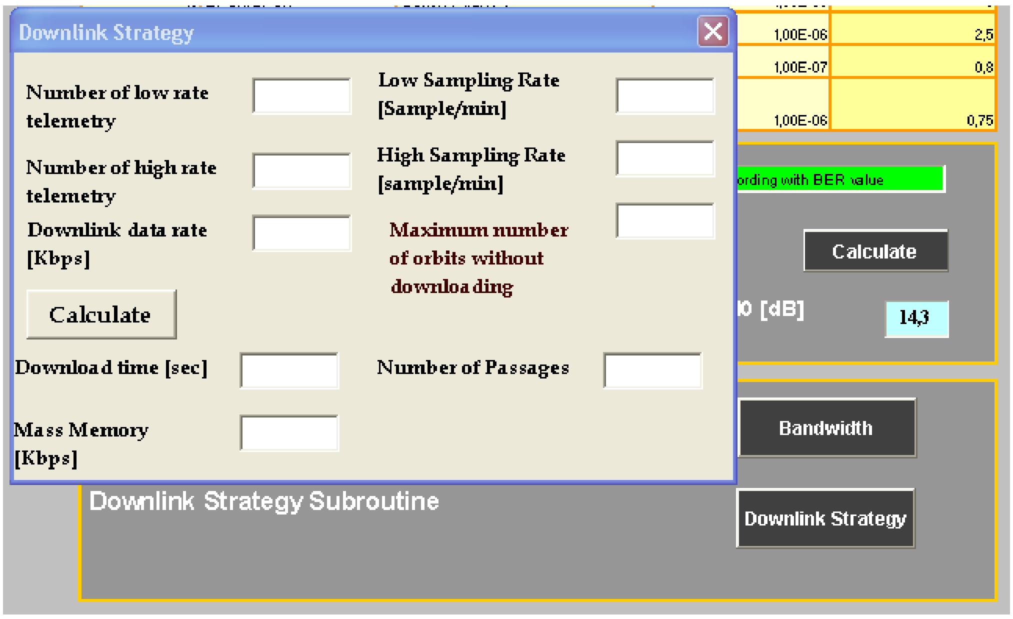


Figure 4 - 38 – Downlink strategy User Form

7. Losses

This worksheet allows determining polarization loss between satellite and ground antennas. It is also provided isolation between two cross polarized circular antennas using the same axial ratios. Designer must enter *Polarization Angle between Antennas* (worst case is obtained for 90 degree).

In the second routine *Atmospheric Loss* is calculated for the minimum acceptable elevation angle (Keep attention: it must be the same used in the Orbit and Frequency WS) and for frequency band used.

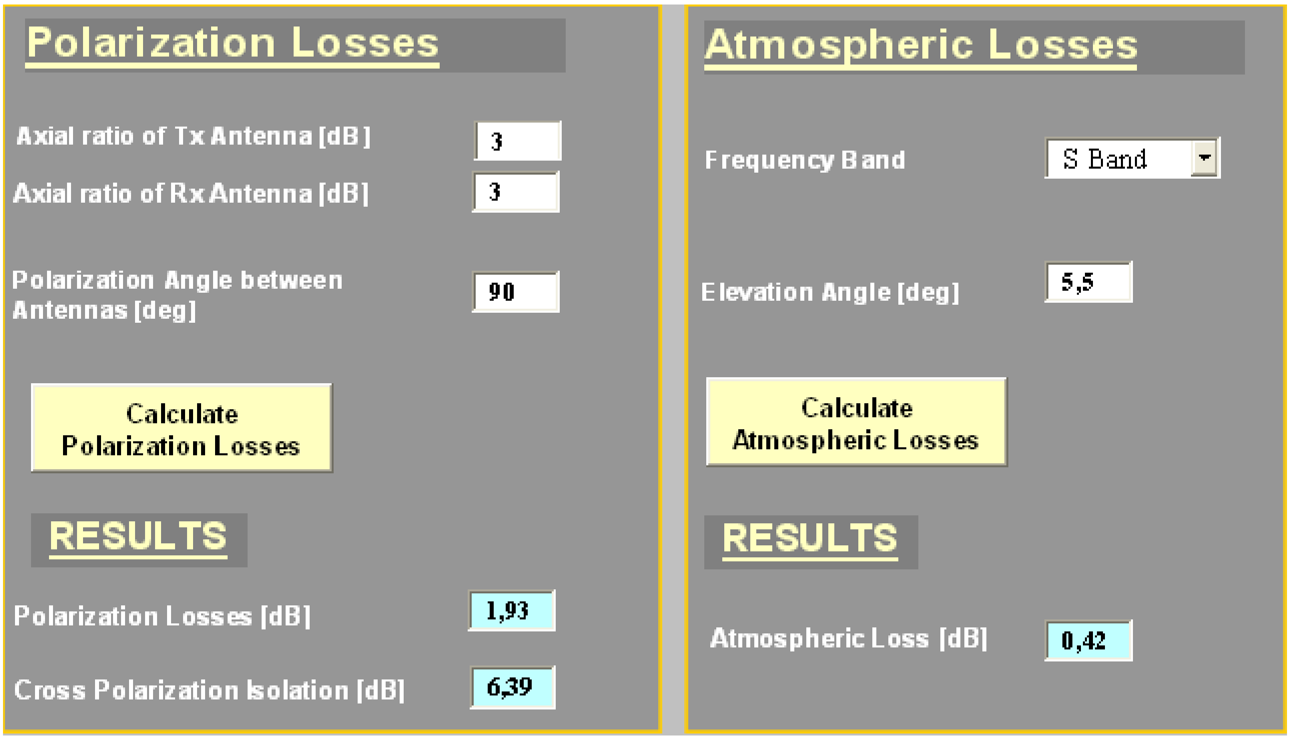


Figure 4 - 39 – Losses worksheet

8. Link Budgets

This is main link budget worksheet. Most of data needed for link has been provided in prior WS. In the left column there is a basic uplink budget estimate. *Ground Station* EIRP, *Total Path Loss*, and *G/T* of receiving system are calculated. Then we calculated *Rx Power* and S/N_0 *margin*. In the right column, we considered the modulation losses due to sharing of power between carrier and subcarrier (to view theoretical notes provided on the worksheet it's enough to move mouse on cells). It allows to calculate margin on carrier and telecommand recovery. Margin thresholds are fixed on 10 dB (Respected margin thresholds are green highlighted). On upper side of worksheet is summarized most important features of orbit, selected ground station site, telecommand and telemetry rates.

Downlink budget has the same structure. *Satellite* EIRP, *Total Path Loss*, and *G/T* of receiving system are calculated. Then we calculated S/N_0 margin. In the right column, we considered the modulation losses due to sharing of power between carrier and subcarrier. It allows to calculate margin on *carrier*, *telemetry*, and *tone recovery*.

TC BIT-RATE (Kbps) :	4,00	
TM BIT-RATE (Kbps):	4,00	
CODING :	Yes	

Link Guaranteed

Link NOT Guaranteed

Basic Uplink 1/2

Ground Station		
G/S TX POWER	21,20	dBW
CIRCUIT LOSS	1,00	dB
TX ANT GAIN	37,80	dBi
G/S ANT TX AXIAL RAT	3,00	dB
POINTING LOSS	0,00	dB
EIRP G/S	58,00	dBW
Propagation Losses		
FREQUENCY	2050,00	GHz
SLANT RANGE	1976,58	Km
PATH LOSS	164,60	dB
ATMOSPHERIC LOSS	0,23	dB
IONOSPHERIC LOSS	0,00	dB
POLARISATION MISMATCH	2,84	dB
TOTAL PROP. LOSS	167,67	dB
POW. FLUX at S/C	-48,91	dBm/m^2
Spacecraft Receiver		
RX ANT GAIN	-2,46	dBi
POINTING ANT LOSS	0,00	dB
S/C ANT RX AXIAL RAT	4,50	dB
ANTENNA NOISE TEMP	60,00	K
ANTENNA/FEED VSWR		
VSWR LOSS	1,00	dB
CABLE PHYSICAL TEMP	290,00	K
CABLE LOSS	0,00	dB
CIRCUITS TEMPERATURE	290,00	K
RFDU CIRCUIT LOSS	3,30	dB
TOTAL CIRCUIT&CABLE LOSS	4,30	dB
DIPL. CIRCUIT LOSS	0,00	dB
RECEIVER NOISE FIGURE	5,00	dB
REF SYSTEM TEMP	917,06	K
RX SYSTEM TEMP	29,62	dBK
S/C RX G/T	-36,38	dB/K
RX POWER	-86,43	dBm
CAR ACQ THRSH	-128,00	dBm
TC RX THRSH	-118,00	dBm
REQ RX POWER	-118,00	dBm
RX POWER MARGIN	31,57	dB
RX S/N$_0$	82,55	dBHz

Uplink 2/2

RX S/N$_0$	82,55	dBHz
Modulation Indices		
TELECOMMAND	0,70	rad pk
RANGING (RNG)	1,00	rad pk
Carrier Recovery		
CARRIER SUPPRESSION	3,42	dB
PLL-BDW 2Bl$_0$	2000,00	Hz
THRSH C/N in 2Bl$_0$	25,00	dB
Effect PLL-BDW 2Bl	2000,00	Hz
Effect PLL-BDW 2Bl	33,01	dBHz
IMPLEMENTATION LOSS	3,00	dB
REQ C/N in PLL BDW	25,00	dB
CARRIER MARGIN	18,12	dB
Telecommand Recovery		
MODULATION LOSS	8,97	dB
IMPLEMENTATION LOSS	3,00	dB
BIT RATE	4,00	kbps
BIT RATE	36,02	dBHz
REQ E$_b$/N$_0$	14,3	dB
TELECOMMAND MARGIN	20,26	dB

Figure 4 - 40– Uplink Budget worksheet

9. Summary Budgets

This Worksheet shows subsystem *Mass and Power budgets* and compares results of link designer with *Target Mass and Power budgets* of System. Moreover are reported geometrical dimensions of selected units, and dissipated power in different mode. In the bottom you can see a schematic diagram of subsystem architecture. On Power Budget table designer must enter a valuation of thermal losses of subsystem components. Then results are directly linked to output worksheet.

TT&C MASS BUDGET										
	Unit	No. of Units	Unit Mass (kg)	Raw Mass (kg)	Maturity Margin to be applied (%)	Predicted Mass (kg)	Remark	Dimensions (mm)		
								Length	Width	Heigh
S/S	S-Band Transponder	2	3,00	6,00	10	6,60		275	110	197
	Radio Frequency Data Unit	1	0,04	0,04	10	0,04		32	38	13
	S-Band Helix Low Gain Antenna	2	0,26	0,51	10	0,56		152	152	84
TOTAL MASS						7,20				
TARGET MASS						7,21				
DIFF.						0,00				

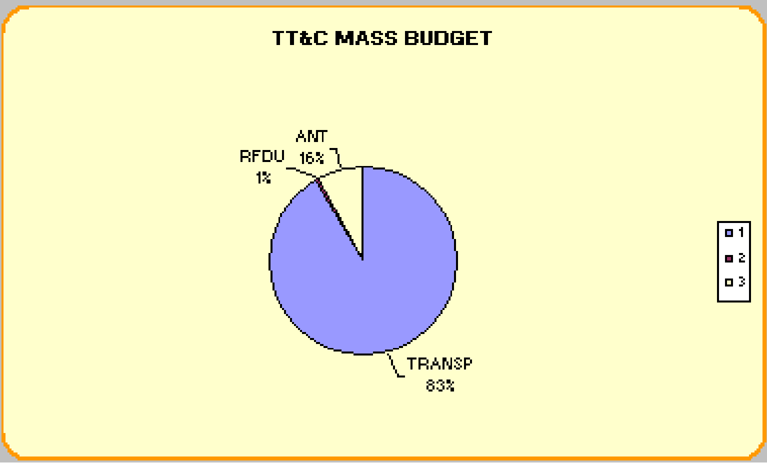


Figure 4 - 41 – Mass Budget worksheet

TT&C POWER BUDGET										
	Unit	No. of Units	Active Unit Power (W)	Standby Unit Power (W)	Maturity Margin to be applied (%)	Predicted Power (W)	Remark	Estimated Dissipated Power (W)		
								Pon	Peclipse	Psafe
S/S	S-Band Transponder	2	26,00	6,00	10	35,20		28,21	6,00	6,00
	Radio Frequency Data Unit	1	0,00	0,00	10	0,00				
	S-Band Helix Low Gain Antenna	2	0,00	0,00	10	0,00				
TOTAL POWER						35,20				
TARGET POWER						38,72				
DIFF.						3,52				

Figure 4 - 42 – power Budget worksheet

10. Data Base

The last sheets of workbook give a compendium of equipment and ground station parameter collected during this.

EQUIPMENT DATA BASE

Worksheet description: *Equipment data base are scheduled here. For each unit are reported principal RF, electric, mechanical, thermal and geometrical caractheristic.*

Model	L3_CXS-610	L3_CXS-765	L3_CXS-810	Alcatel TRC	L3_CXS-2000
Receiver Frequency (MHz)	2025 to 2120	2000 to 2110	1760 to 1840	2025 to 2120	2025 to 2120
Carrier Acquisition Threshold (dBm)	-124,00	-119,00	-120,00	-128,00	-121,00
Tc. Threshold (dBm)	-118,00	-110,00	-110,00	-118,00	-118,00
Noise Figure (dB)	5,00	6,00	5,00	5,00	5,00
Tc Modulation Index (rad)	1,35	0,70	1,00	0,70	1,00

Figure 4 - 43– data base worksheet

4.1.2.2.3. Concurrent Design Tool supporting software

The workbooks I've developed during my Ph.D. are the core of the CDT; such workbooks collect equations, formulas, sizing criteria and all the necessary information for performing pre-phase-A studies. Being these studies early investigation of space missions feasibility, most of the equation, formulas and so on are approximated and conservative as requested to such assessment studies, with margin ranging from 5% to 20% for any components; the margins that the engineering teams have to assign to the subsystem's components depends on the Technology Readiness Level (TRL) of the components/systems/hardware chosen for the design: if devices have a high TRL, the margins should be smaller, while for lower TRL the margins have to be larger, as later design phase can reveal un-feasible the solution adopted, and a certain margin is necessary in order to do not compromise the overall design.

Contemporary, together with these workbooks, the use of dedicated software for a more accurate evaluation of essential parameters could be of outmost importance; in fact, even if the design is at feasibility or pre-feasibility level, parameters like the mean contact time with the ground station, the revisit time over a certain area, the coverage the payload can provide according to its field of view, can drastically affect satellite design.

In order to evaluate with a good accuracy such parameters, the use of professional software is necessary; two software in particular were added to Concurrent Design environment, STK (Satellite Tool Kit), and Matlab.

The former is used to evaluate all the parameters related to satellite orbital configuration, while the latter is used for complex calculation, that are more effectively performed through this software, instead of using the Visual Basic for Application functionalities included in the Excel software and used for simpler calculations.

4.1.2.3. Additional Feature: specific Configuration Workbook

In the original version of the workbooks, those containing just the Inputs/Outputs worksheets and the capacity of exchanging data, the workbook devoted to design of Structures subsystem contained also the functions and the tasks related to spacecraft configuration; even if such aspect are strictly related, the fusion of the functions related to designing the primary and secondary satellite structure, and those related to configuration, components assembly, and those related to configuration they need to have, in order to be both capable of performing their necessary application while fitting at the same time into the satellite envelope, was not considered by the author of this work as profitable. The suggestion made was that the configuration aspects should be considered as a separate field of expertise, for the complexity that the geometry of arrangements and interfaces among all the parts of the spacecraft could reach. The work I've done was that of extrapolating aspects, functionalities and operations related to space system configuration from the Excel workbook devoted to structural design, and, adding a series of important capabilities, implementing an additional engineering unit with the related workbook, the *Configuration* one.

The successive UML diagram is not extensive like those presented earlier, but has only the reason of underlying the separation of the main activities that the two different subsystem has to perform: the structural engineering unit will be devoted to choose the satellite configuration (presence or not of central cylinder, presence or not of middle platform and/or shear panels, and so on), size the structural components, carry out the frequency analysis; the configuration engineering unit will collect all the information regarding mass, dimension, shape, position denied and preferred, will create the satellite 3D model, returning at any iteration the updated value of the satellite centre of gravity, the momentum of inertia, and other information related to component placements.

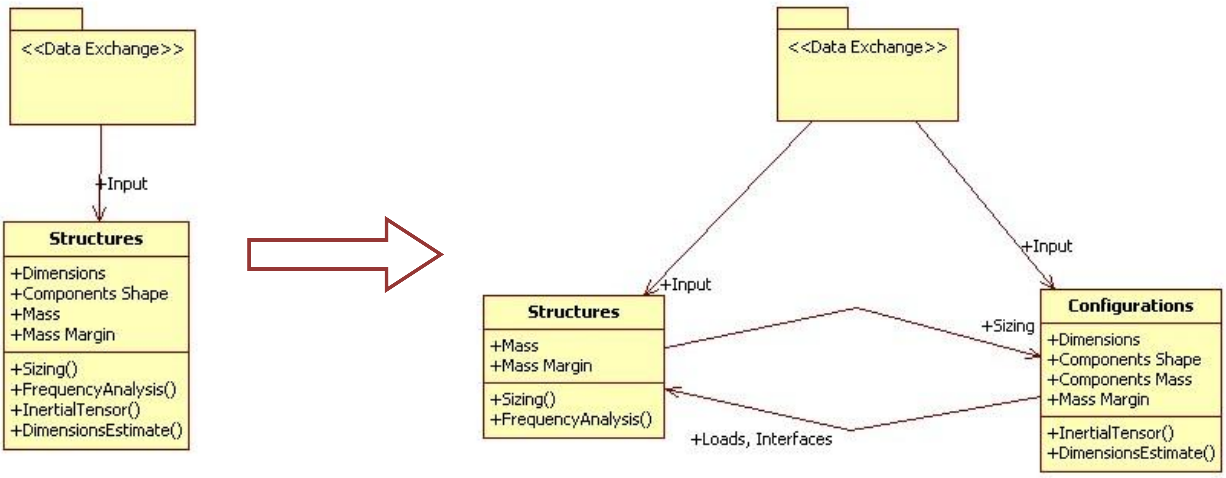


Figure 4 - 44 – Splitting of the original Structure workbook into Configuration and Structure

In such a way, the engineering unit of Configuration performs its own functionalities, and is integrated into the concurrent environment with the same logic of all the remaining ones. This process allows the configuration engineering team to enter into the loop of the concurrent iteration, not only being relegated to drawing, once the design iterations terminate, the satellite 3D model, in order to have an idea of how the satellite designed looks, but with the aim of being another piece of the jigsaw, an actor of the design iteration process: in fact, at the end of any iteration, the configuration unit will receive information, will perform its own activities, giving its outputs back to the structures subsystem.

The configuration tool comprises an Excel workbook, where the configuration specialist collect all the information coming from the other subsystems, perform preliminary operations, and a CAD 3D software, where the configuration specialist can draw the satellite model, arranging in the appropriate position the components that the different engineering team sized during the current iteration.

Once the satellite configuration is terminated, the CAD software will calculate the satellite inertial properties, and will upload such information inside the structure workbook, that, in turns, will share these data with the subsystems which needed them.

In the following figure, a screenshot of one worksheet of the Configuration workbook is provided. The list of components that the engineering unit has to draw is provided, together with components name, dimensions and density.

	A	B	C	D	E	F	G
1-2	Unit of Measure	Dim1	Dim2	Dim3	Component Name	Inventor Name	Density* [g/cm^3]
3	mm	1850	25,3182	2700	Closure Panel (Top or Main)	Closure_panel	0,11417
4	mm	425	7,21	2700	Shear Panel (Top)	Shear_Panel	0,90337
5	mm	1850	25,3295	1850	Platform (top)	Platform_top	0,09984
6	mm	1850	25,3295	1850	Platform (bottom)	Platform_bottom	0,09984
7	mm	1000	1,62	2700	Cylinder	Cylinder	1,40186
8	mm	960	950	3,933	Cone (adapter)	Cone_adapter	5,16407
9	mm	178	418	418	HR16 (3)	Reaction_Wheel	0,12288
10	mm	160	140	195	VF 1 Earth Sensor	Earth_Sensor	0,67537
11	mm	105	149	140	BOKS-M	Sun_Sensor	0,75332
12	mm	177,8	177,8	476,2	RRGU	Star_Tracker	0,76491
13	m	17,4731	1,85	0,05	Solar Array	Solar_array	0,02618
14	m	0,68	0,795	0,2592	Battery	Battery	1,82559
15	m	0,59	0,4	0,29	PDU	PDU	0,14611
16	m	0,02	0,02	0,001	PCU	PCU	25000
17	m	-	0,159	-	Press. tankage (fuel)	Press_Tank	0,48434
18	m	-	0,89	-	Tanks (fuel)	Fuel_Tank	0,00334
19	mm	250	300	100	RAD6000	SMU	0,26667
20	mm	90	96	20,4	FM430F1612	PC	0,41984
21	cm	8	20	1,2	AT49BV040B	RAM	0,00222
22	m	2,03859	2,03859	0,05	Radiator	Radiator	0,008

Figure 4 - 45 – Equipment List worksheet

Following, the screenshot of the worksheet that is used to choice the satellite configuration: both existing configuration (listed on left) or a completely new configuration (neglecting this worksheet) can be selected, together with one-wing/two-wings deployable solar panels.

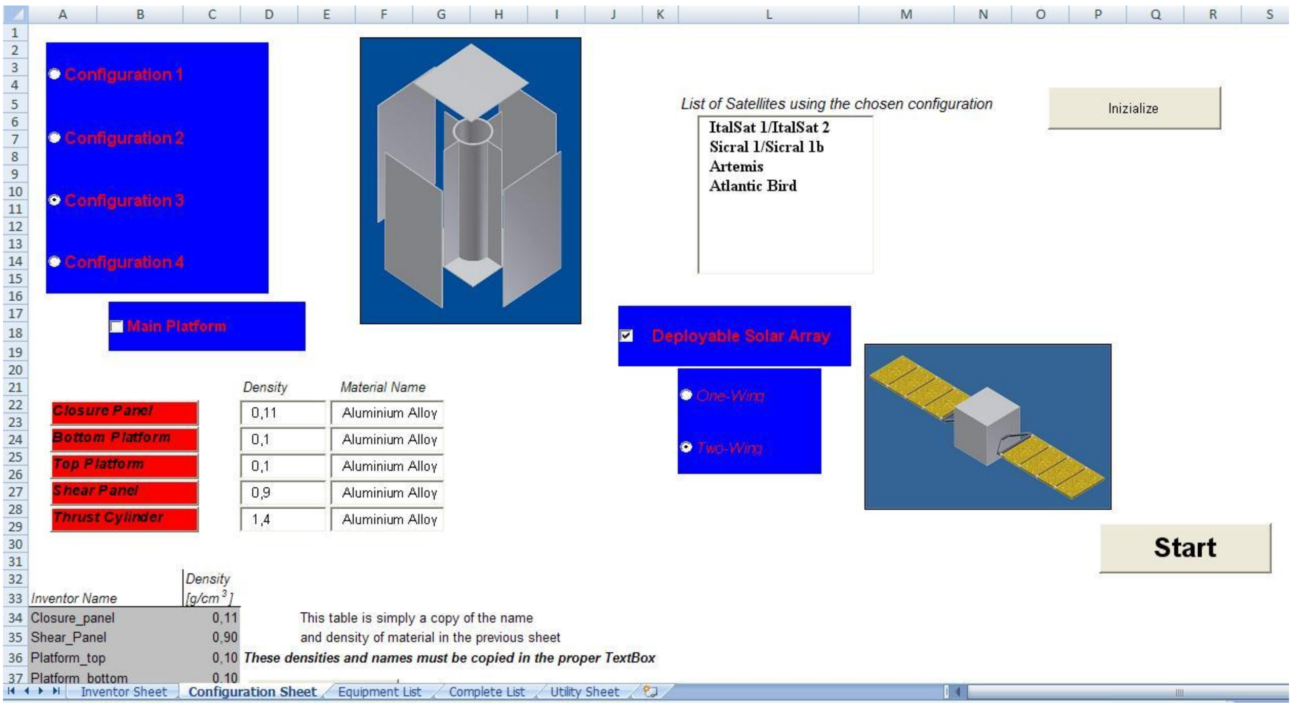


Figure 4 - 46 – Initial Satellite Configuration

Once all the preliminary operations are performed, the CAD 3D software (Autodesk Inventor Professional) is automatically initiated, loading the choices made in the workbook, and allowing the engineering team to create the satellite 3D model.

Once components have been drawn, they can also directly linked to Configuration workbook: in such a way, after any design iteration, if components are updated, for example, larger batteries are chosen, the software will perceive components updating, and the properties (mass, dimensions) will be automatically updated, without the necessity of drawing the components again.

The implementation of a separated Configuration workbook, and the use of CAD 3D software, together with all the related functionalities, will lead to a more precise and fast design of the structural and Configuration workbook.

Figure Index

REFERENCE

[1] – CDF Info Pack 2011 - http://esamultimedia.esa.int/docs/cdf/CDF-INFOPACK-2011.pdf

[2] – R. Findlay, A. Braukhane, D. Schubert, J.F. Pedersen, H. Müller, O. Essmann, DLR – Institute of Space Systems, Bremen, *"Implementation Of Concurrent Engineering To Phase B Space System Design"*, CEAS Space Journal, 2011

5. Case Studies

The methodologies, concepts and the processes introduced in the previous paragraphs, together with the constant research of the most innovative technologies, has been used to perform and complete three extensive case studies, in which has been demonstrated how such concepts can be tailored and exploited for designing real space missions, while successfully responding to real needs.

The possible mission that a microsatellite, or a constellation or formation of microsatellites, can perform are numerous and extremely diversified: there are, in fact, several possibilities for the implementation of such platform due to the fact that traditional medium and large satellite, but also "upper limit" small satellite (with a mass of approximately 400-500 kg) are limited in the implementation for space service both from a temporal point of view and from an economical point of view; in fact, the time-to-delivery, necessary to place in orbit of one of these platforms is significantly high, and the cost related to such operations tremendously high, so many applications cannot be performed since they are considered not affordable; for these reasons, there are a plethora of missions that highly technological microsatellites can perform, providing a real service. Some of these missions are:

- Constellation for dedicated (and limited in time) observation of target areas;
- Enhancing space service through microsatellite(s) performing missions in support of larger platforms;
- Missions designed to filling currently existing gaps;
- High Repetitive Ground Track missions;
- "Tailored" Data Relay microsatellites constellation (or formation);
- Creation of a network of distributed sensors.

These are just few example of how the microsatellite can be used for missions that larger satellites can hardly perform, due to, mainly, to their high cost and long development time. As an example of possibilities in which microsatellite can be used is that of Earth Observation satellite, mounting an electro-optical payload working in the visible band of the electromagnetic spectrum of radiation. Such platforms can better accomplish their mission when the illumination condition are the best possible: if the Sun is not close to the local nadir, observation in the visible band can be characterized by umbra that could be really "long" in case high buildings or mountains are in (or close to) the scene to observe. For such a reason, the tendency is that of flying platform along Sun-Synchronous Orbit with LTAN equal or close to Zenith, exploiting 10:00, 11:00, 12:00, 1:00, 2:00 am/pm Sun Synchronous Orbits. Such a choice, if on the one hand assure profitable illumination condition, severely limits the revisit time, imposing harsh constraints on the capacity of acquiring images repeated over a certain area: in addition, such capacity are strongly affected by the weather condition: in case of adverse meteorological condition, like rain or clouds, that can severely affect the image acquisition, the time necessary to have updated images of the area of interest is even further enlarged. Such situation can be improved through the implementation of microsatellite, mounting an optical payload, flying over SSO complementary to those actually flying, that can be used for having images with an higher repetitivity, even if such "additional" images cannot have the same spatial resolution of those provided by large satellite, as the platform is simpler than those actually flying (like Pleiades) and also in illumination condition that are not optimal. This possibility could be implemented for improving present service of image provision, launching satellite over orbits where they can survive for a long period, or through on demand launch of some of these microsatellites, exploiting future air-launching capabilities, with the precise aim of improving service over a dedicated area.

Several possibilities, therefore, exist for such a typology of platforms: in the following, I’m going to present the three studies that have been conceived and investigated:

The first mission is an hypothetical responsive mission, where the aspect of responsiveness is of outmost importance: the mission consist in the necessity of monitoring a site (and its proximity) where a natural (or man-made) disaster recently occurred: such event could be what recently happened in Haiti, where a catastrophic earthquake hit the Caribbean region or, even more recently what happened in Japan, where an earthquake occurred in the open seas has been the origin of a series of dramatic events that lead to the catastrophe of the Fukushima nuclear plant. The case study has been conceived focusing on two fundamental aspects: the responsiveness of the service to provide, and the demanding mission requirements that the service impose. The aspect of the innovative technologies has been considered less important in a mission like the first one presented, as it could exists situation like that where no reason for risking a dysfunction must be encountered. In both the two mentioned cases, the necessity of images for coordinating the rescuers, or for monitoring the site and its immediate proximity are crucial, helping saving many lives.

In this first case study, the spacecraft design phase will appear extremely more intensive and detailed: the formulas, equations, rules of thumb used for defining the single spacecraft components has been performed for any of the three missions presented, but presented and described really in detail uniquely in this first case study.

The reason for such a behavior is simple: even if the results and the outcome of different studies, carried out bearing in mind different mission objectives and respecting different constraints, are different, but the logic for reaching these results is the same: regardless of the fact that a satellite could face or not an eclipse period, and regardless of the fact that the payload has to work 24/7 or only for a short duty cycle, the logic for designing solar panel (body mounted or deployable, steerable or not), storage devices, and any other components of the Electrical Power Subsystem, as well as the other subsystems, are quite the same; of course, the logic will be described, the reasons for the choice of a particular device instead of another will be investigated and explained, but the formulas will not be repeated in an exhausting repetition of known (as already described) concept

The second space mission depicted has been instead identified with the aim of providing a real space service, able to guarantee a useful service that is really a need in the nowadays world, with several governmental and institutional actors interested in the development of such a capability. As such need is actual and present, the three pillars constituting this research effort has been reduced to two, providing a complete mission and system design neglecting the aspect of the air-launch; this is justified by the fact that the mission in object has a responsive aspect not so strongly marked and not of fundamental importance for the provision of the service. The mission in object is the extension of a ground-based system into space: a space-based AIS, Automatic Identification System, has been studied.

The third mission presented is a study of a microsatellite that supports a larger, existing mission, providing an added value to the original spacecraft, without any kind of “interference” to its original mission and operations. This kind of mission has been studied and presented with the precise aim of demonstrating once again the potentiality and intrinsic capabilities of such a small platform, that could be also implemented as reinforcement for enhancing the service provided by existing platform, or for the purpose of filling existing gap on service provided by large satellite, being this an added value to the existing service, that is not so important to justify investments like those afforded for the “main” satellite and service, but that could be anyway a solution for which someone could pay for it.

The mission chosen is an interferometric mission performed using a receiving Synthetic Aperture Radar working on the same bandwidth of the COSMO-SkyMed constellation of satellite. In such a way, mounting one or more receiving antennas on space platforms that flying in formation with the main satellite, without any interference on the operations performed by the main satellite, interferometric service can be provided. The fundamental aspect of the mission, which must be in any study, the goal for spacecraft design, is in this mission accompanied by the aspect of exploitation of innovative technology: in fact, the recently demonstrated capabilities of formation flying are essential to be able to implement interferometric mission capable of providing real service.

5.1. Case Study 1: Tailored Earth Observation System

5.1.1. Responsive Earth Observation System: Description

The first case study, as already explained in the introduction of the chapter, will be focused on a space system that will be designed to responsively respond to sudden arisen mission needs: these needs could be correlated, for example, to the necessity of monitoring an area where recently a disaster occurred. The mission is the primary driver of this case study, and for the typology of the mission objectives, the needs that such a mission entails cannot be carried out in other way but implementing the logic of the *Responsive Space.*

For this case study, the concept of seeking for the most advanced technology will be partially disregarded: the payload has been identified as one of the most innovative and performing devices for Earth Observation that has been recently developed for application on small space platform, but the design of the spacecraft bus has been done following the traditional concept, and making use of commercially available space components, provided with a certain space heritage. The reason for doing is that a similar mission needs to designed to be fully operative and able to perform the prescribed service without any kind of uncertainty, for the remarkable importance of the mission needs.

The study has been conducted using as guidelines the following mission statement:

> *"Due to the noteworthy impact that environmental disaster, natural or man-made, had on territory, has been decided to develop a responsive capacity to respond to such a sudden need: the space system design will start immediately after the disaster occurs, and it will designed, developed, integrated and launched in a really short timeframe, with the aim of providing the required monitoring for a period of several months (6-12)"*

5.1.1.1. Definition of Target Area

The definition of the target area is a choice that impact both the design of the satellite constellation, aimed at maximizing the revisit time over the area, and, consequently, the design of the space platform: in fact, according to the orbital parameters used there are consequences, for example, on the eclipse time, which in turns impacts the design of the Electrical Power Subsystem, and on the orbital height, which is one of the factors defining the amount of propellant necessary to keep the satellite on the right altitude. As always happens in any field of engineering, there is not a best answer suitable for any situation: on the contrary, all the condition have to be properly evaluated, identifying the most suitable orbital configuration; in fact, if the area to observe is located to low latitude, orbits with low value of inclination can help having a good revisit time (at least, one per orbit – independently from illumination condition that can affect the acquisition – if area located over the equator has to be observed and equatorial orbit are chosen), while polar orbits has to be borne in mind in case high latitude area must be examined. This means that the choice of the target area can lead to different results.

Different choice can be done, therefore, on the target area; but also the extension of such an area affect the resulting constellation performances (and, consequently, the choice of the orbital parameters). For the purpose of such study, several areas hit by real catastrophes could be used;

Haiti or Fukushima are just two example of the important possibilities that constantly updated images can provided to rescuers. But, in this case, a larger area has been considered: the constellation has been designed in order to observe the entire European territory.

5.1.2. Orbit Design Criteria

The first trade-off to face concerns the eccentricity. The advantage of having an elliptical orbit, once the semi-major axis has been identified, lies in the possibility to get closer to the target area at the perigee with respect to a circular orbit, thus increasing the spatial resolution for the same sensor. But the gravitational perturbation due to the non-uniformity, non perfect roundness of the Earth induces temporal variation of orbital parameter: the critical issue for this kind of orbit, where the closer position of the perigee is used to observe an area of interest, is to keep the argument of the perigee upon such an area. It is possible to freeze the argument of the perigee by choosing a frozen orbit.

The variation of eccentricity and argument of perigee is due to the non-sphericity of the Earth (J_2 and J_3 effect); the relationship describing the temporal variation of this two parameters are:

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

$$\dot{\omega} = \frac{3 J_2 n}{(1-e^2)^2} \left(\frac{R_E}{a} \right)^2 \left(1 - \frac{5}{4} \sin^2 i \right) \cdot \left[1 + \frac{J_3}{2 J_2 (1-e^2)} \left(\frac{R}{a} \right) \frac{\sin i \sin \omega}{e} \right]$$

where e is the orbit eccentricity, J_2 and J_3 the second and third gravitational harmonics, R_E the radius of the Earth, a the semi-major axis, i the inclination and ω the argument of perigee, and $\dot{e}$ and $\dot{\omega}$ their derivative with respect to time. A frozen orbit is defined as an orbit that keeps these two parameters (eccentricity and argument of perigee) constant in time. This can be achieved in two ways:

1) Choosing inclination equal to i = 63.4° or i = 116.6° (critical inclinations), in order to set $\left(1 - \frac{5}{4} \sin^2 i \right)$ equal to zero.
2) Choosing $\omega = 90$ in order to have $\dot{e} = 0$, and imposing equal to zero the term $\left[1 + \frac{J_3}{2 J_2 (1-e^2)} \left(\frac{R}{a} \right) \frac{\sin i \sin \omega}{e} \right]$, so that $\dot{\omega} = 0$. This condition determine a relation among eccentricity, semi-major axis and inclination of a frozen orbit, given by $e \cong -\frac{J_3}{2 J_2} \left(\frac{R}{a} \right) \sin i$.

The second possibility is not suitable for any possible observation mission, as, relationship $e \cong -\frac{J_3}{2 J_2} \left(\frac{R}{a} \right) \sin i$ imposes condition that could not be compliant with the mission requirements.

Elliptical orbits should have an apogee altitude not very high, in order to do not pay a significant loss in revisit time on target area, that is a fundamental mission requirements. But an orbit with a relatively low eccentricity cause an extremely limited gain in resolution, as the decrease in altitude at the perigee is very modest. On the other hand, choosing an orbit with a more accentuated eccentricity, capable to give a real gain in resolution, would result in the already mentioned reduction in revisit and dwell time, which are the main features of our mission. Furthermore, another problem would arise if an high eccentricity is chosen, that is to keep the argument of the perigee fixed on the target area, that means to choose a frozen orbit, which is impossible if a critical inclination is not chosen and the argument of the perigee is different to 90° because the target area is not placed at polar altitudes.

The orbit's design criterion is the following: the orbital inclination is chosen in a proper way, aiming at having the dwell time as large as possible, tailoring the value of *i* on the target area.

An airborne launcher has the necessary flexibility that allows any orbital inclination to be reached, since it is possible to put a satellite into an orbit with an inclination at least equal or larger than the latitude of the launch site.

The final decision for the orbit felt on a circular orbit; this choice has been made according to the following consideration about elliptical orbit:

- if low eccentricities are chosen, there is not any significant gain in resolution or lifetime respect to a circular orbit with the same semi-major axis;
- if high eccentricity are chosen, apart from the reduction of revisit time, that is considered as the main requirements for a such a kind of mission, there is the necessity to fix the perigee on the target area, avoiding that the argument of the perigee drifts; but this is not always possible, and in particular is not for an orbital inclination like those used for our purpose.

Chosen the circular orbit (e=0), the altitude is selected taking into account both considerations about atmospheric drag, which is the most significant cause of orbital decay for LEO orbits, and payload resolution, since it decreases with the altitude. A third element, therefore, entered in the process of evaluating the altitude, and this element is the satellite lifetime; the lower the altitude, the higher is the atmospheric drag, the faster the satellite decay, ceasing any operations. The lifetime expected for the microsatellite is 1 or 2 years, and the altitude has to be as low as possible to increase payload performances. The requirement for the panchromatic camera is to achieve a sub-metric spatial resolution.

According to the atmospheric density model MSISe-90, shown in next Figure, the orbital decay of a microsatellite having a mass of 100-150 kg and a frontal surface of 90x90 cm^2 is computed, as shown in the successive figure, in order to assess in a preliminary way, the possible altitude range allowing the microsatellite lifetime to last as long as required. It is observed that if the surface is kept constant, and the mass is increased, the orbital decay gets slower, as witnessed by the successive figure.

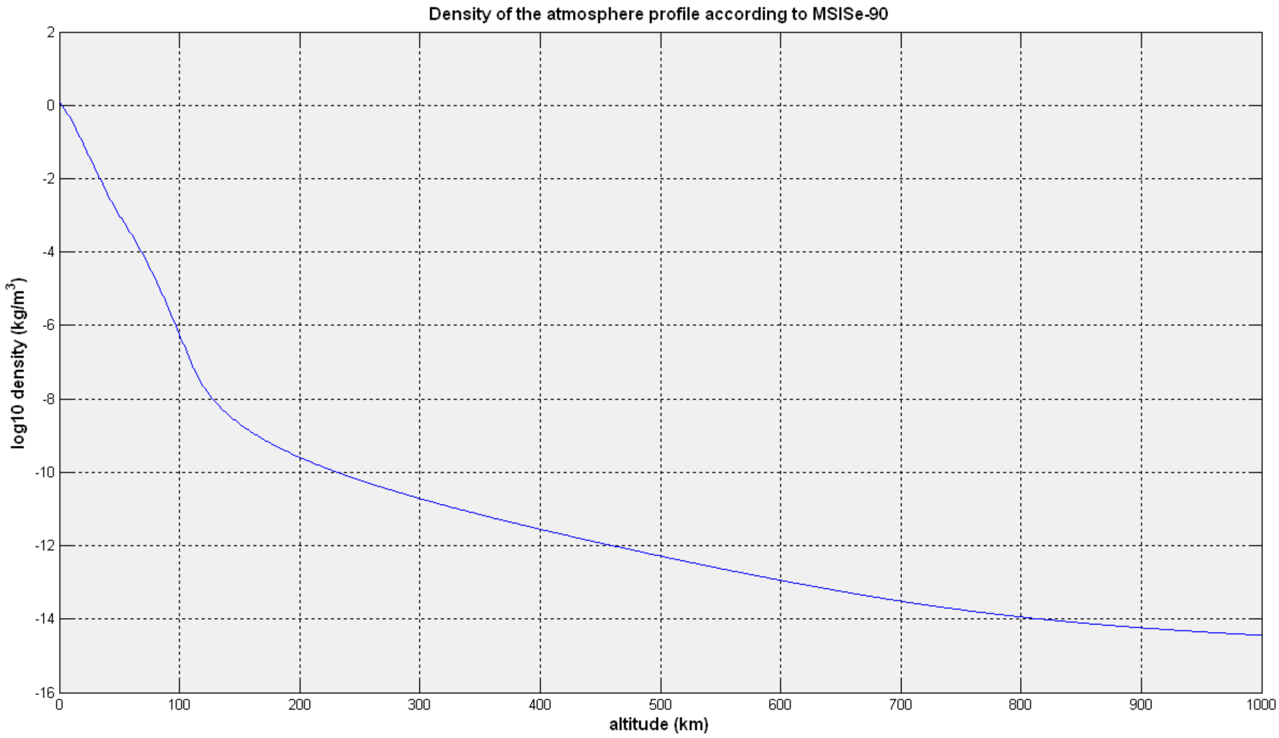


Figure 5 - 1– Density of the atmosphere as a function of the orbital height

The total energy for elliptical (and circular) orbits can be expressed as:

$$E = -\frac{1}{2}\frac{GMm}{a}$$

Where a is the semi-major axis, G is the gravitational constant, M is the mass of the Earth and m is the mass of the satellite. By deriving respect to time:

$$\dot{E} = \frac{1}{2}GMm\frac{\dot{a}}{a^2} = -E\frac{\dot{a}}{a} = P = F \cdot V$$

Where P is the energy variation per unit time, that is due to the product of a force F (mainly drag force) and the satellite velocity V. The variation of the semi-major axis is given by:

$$\dot{a} = -\frac{FV}{E}a = -\frac{a}{E}\left(\frac{1}{2}C_D\rho(a)SV^2V\right) = -\frac{a}{E}\frac{1}{2}C_D\rho(a)SV^3$$

Where C_D is the drag coefficient, $\rho(a)$ is the atmospheric density, that vary with altitude, and S is the frontal surface of the satellite. Since, for a circular orbit:

$$V = \left(\frac{GM}{a}\right)^{1/2}$$

Then:

$$\dot{a} = -\frac{a}{E}\frac{1}{2}C_D\rho(a)S\left(\frac{GM}{a}\right)^{3/2} = C_D S\rho(a)\frac{a^{1/2}(GM)^{1/2}}{m}$$

Assuming a typical C_D = 2.2, a microsatellite mass m equal to 150 kg and a frontal surface of 90x90 cm, it is possible to derive the orbital decay, that is the altitude decrease in time due to the atmospheric drag.

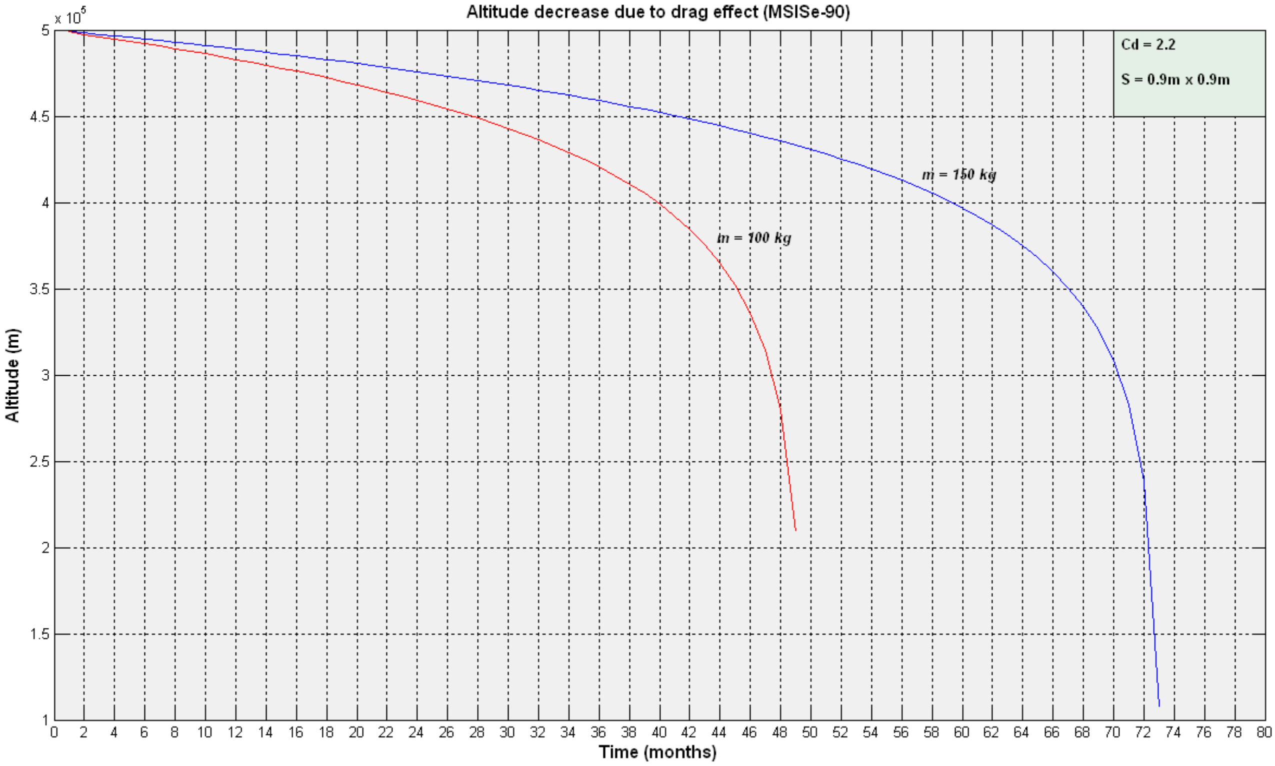


Figure 5 - 2 – Spacecraft lifetime as a function of the initial altitude

Once the influence of the atmospheric drag on the expected lifetime has been assessed, the choice over the satellite altitude has been made: the driving concept has been that of choosing an orbit as low as possible, in order to increase the payload resolution, but, at the same time, higher enough to satisfy the required satellite lifetime: the choice felt on an altitude of 400 km. A trade-off will be deepened in the next chapter about the possibility to implement a propulsion system in order to maintain a fixed orbit altitude, with additional mass, volume and power needs, or to let the microsatellite to experience a range of altitudes approximately from 450 to 350 km, without orbit control, sizing the system in order to guarantee the selected lifetime, and assessing if there are some issues, such as payload focal adjustment, or different spatial resolution images, since taken at different altitudes.

5.1.3. Scenarios

For mission of Earth Observation (EO), Sun-Synchronous Orbits has been deeply investigated, studied and, most of all, implemented. A large number of EO missions flies along these orbits (Cosmo SKYMED, ERS1 and ERS2, as well as Envisat); the SSO are characterized by a series of aspects that constitute strong advantages with respect to inclined, not Sun-Synchronous orbits: such advantages derive from the main characteristic of these orbits, which is the “quite constant” illumination condition that a satellite flying over such orbits faces during its entire lifetime. Such condition allows an easier and more efficient design of the Power and Thermal subsystems, as immediate consequence; illumination condition constant helps and improves also the acquisition of visible image; another powerful reason for being orbits suited for EO are the capability to provide global coverage, even if the revisit time cannot be particularly significant.

Repeat Coverage Orbits are another typology of orbits well suited for Earth Observation, which have their strong point in the feature to provide repeat coverage on the same over several consecutive orbits.

The following table shows SSO/RCO pros & cons.

	Advantages	**Disadvantages**
SSO	- Constant illumination conditions - Good for global coverage	- Poor revisit time - Propellant consuming (launch)
RCO	- Periods with repeat coverage over several consecutive orbits - Takes greater advantage of Earth's rotation than SSO (launch)	- Changing illumination conditions - Coverage limited to latitudes, allowed inclinations

Table 5 - 1 – Sun-Synchronous vs Repeat Coverage Orbits

5.1.4. Mission Performances

The performances resulting from the mission analysis performed will be now described and discussed in this paragraph. The target area selected for assesses the mission performances has been chosen and expressed in a previous paragraph, describing also the reason for such a choice. The target area needs to be identified in this preliminary phase as the orbits are to be tailored to maximize payload performances as well as to define the most suitable orbital parameters.

5.1.4.1. LEO Repeat Coverage Orbit

Mission performances for a LEO Repeat Coverage Orbit will be analyzed for constellation of four microsatellites, highlighting the main characteristic of such a typology of constellation. In the following figure, a screenshot of how a constellation of four satellite can look like if a LEO RCO constellation will be implemented for such a mission.

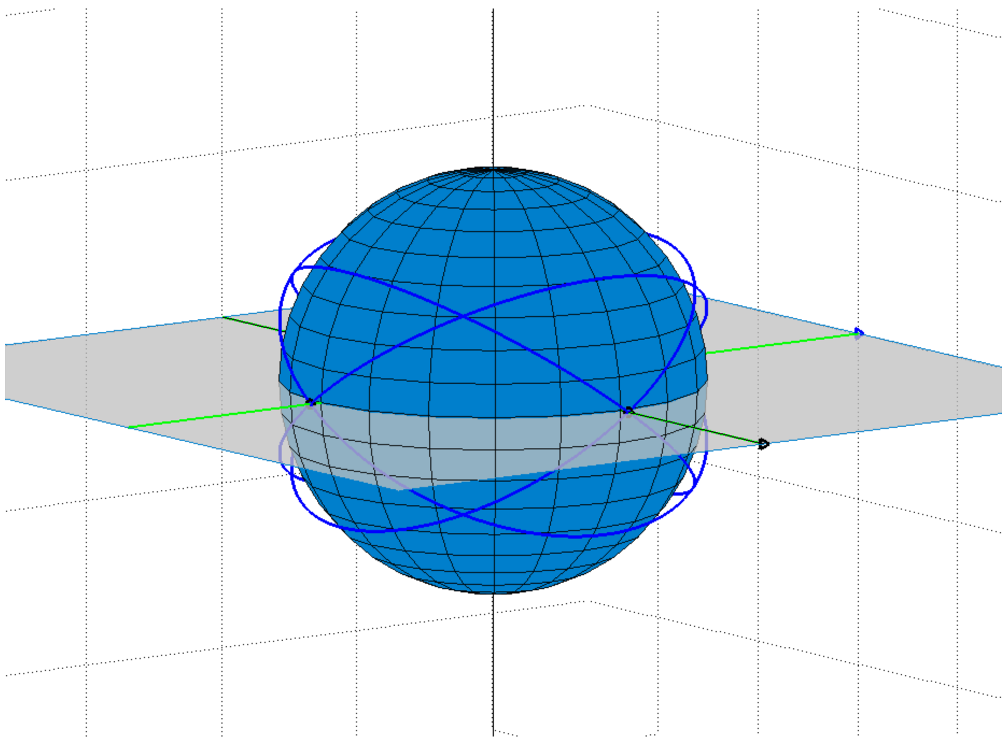


Figure 5 - 3 – LEO RCO configuration

The successive figure shows the "*Percent Coverage*" graph. With a constellation of four microsatellite, after one day the 60% of the target area is covered, and the maximum coverage is obtained after 10 days. The temporal frequency of the accesses on the target area is obviously increased, and the percent coverage does not properly reflect this increase. The reason is that the covered zones are the same, so that there is just a little improvement in time required to cover the overall target area, but the temporal resolution of the system drastically improves.

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5.1.4.2. LEO Sun-Synchronous Orbit

The results presented for LEO SSO mission refer to a constellation of four microsatellites, as numerous as the previous . the next figure shows the "*Percent Coverage*" graph. It is possible to note that after one day the 80% of the target area is covered, and the maximum coverage is obtained after only 2 days. This results is largely better then the result obtained with RCO, but the reason is that this better percent coverage is related to a better separation of the areas covered during subsequent accesses, penalizing, on the other hand, the temporal resolution because the same area is not overlooked in the same short time as for RCO option.

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In the following figure, a screenshot on how a constellation of 4 space platform flying over SSO properly spaced will look like.

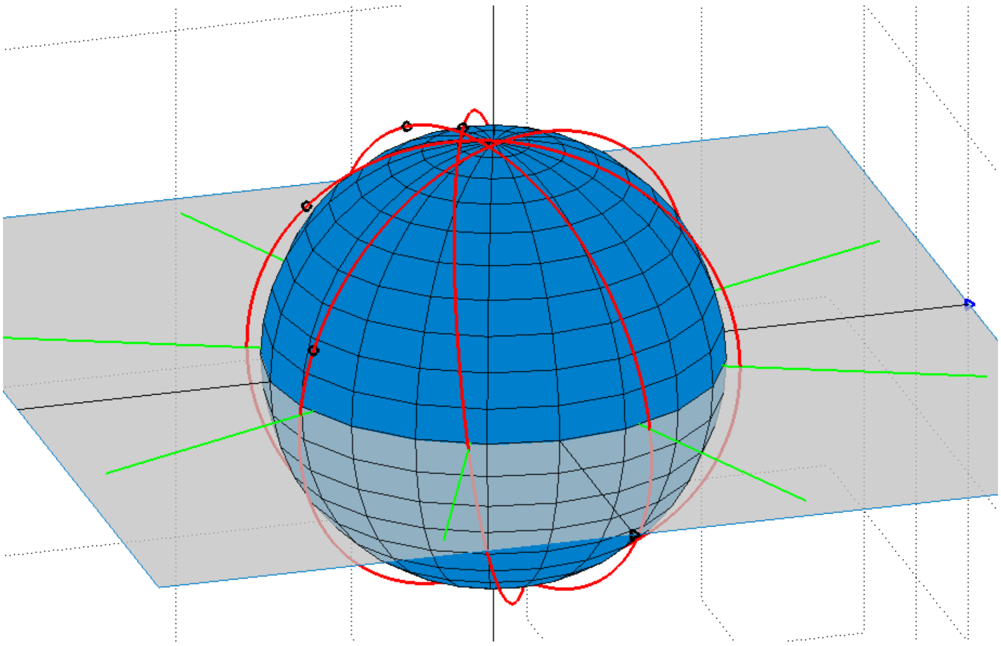


Figure 5 - 4 – LEO SSO configuration

It is important to observe that the maximum response time is lower in this case respect to the Repeat Coverage Orbit option, but also the temporal resolution is lower, because the number of accesses per day is lower (4 respect to the 7 of the RCO). Therefore, the RCO option is more suitable for monitoring dynamic situations, quickly evolving. The reason why despite the more numerous accesses, the RCO provides a higher max response time, is related to the consideration that what is being computed is the maximum response time, and not the mean response time, so it is sufficient just one larger gap to affect in a negative way the result, even if the orbit provides more accesses.

5.1.5. Requirements List

The user requirements of such a system are hereinafter specified:

- Target area: Circular area with a radius of 50 km centered in the Fukushima power plant;
- Payload: Pan cameras for Remote Sensing;
- Spatial resolution: sub-metric resolution (0.65 m at 400 km with a 12 km swath; Pleiades HR satellite has 0.7 m spatial resolution);
- Lifetime: 12 months;
- Acquisition capability: 6 images/day each microsatellite;
- Maximum response time: the microsatellite should supply the information in a very short delay. Maximum response time is given essentially by the time elapsing between two subsequent accesses of the system over the target area, must be shorter than 2 hours, at any time of the day.

According to user requirements, it has been chosen as a payload a panchromatic camera, that allows higher spatial resolutions to be achieved. In an scenario like that we need to face, multispectral images have not the same importance of panchromatic images, because of larger times necessary to complete the processing and, also the lower spatial resolution is not a feature as important as in different application. The very low acquisition capability (as a reference, Pleiades is conceived to acquire 450 images/day) comes from the necessity to concentrate our attention over a single, localized area; this helps also in minimize as much as possible the resources required for accomplishing the mission. Differently from Pleiades, which weigh about 940 kg, the microsatellite designed are a weight of approximately 120kg, providing (theoretically) the same resolution capability, with a significant saving of money, which means, apart from obvious speeches over the reduced resources necessary to implement and operate this class of platform, an easier reproducibility and a more probable diffusion for similar missions.

This requirement relaxes memory (on board storage) and transmission capability (low data rate) constraints, and allows less performing, expensive and heavy components to be chosen.

5.1.5.1. Agility and TDI Manoeuvre

Since the optical payload needs to have extremely demanding performance in terms of spatial resolution, seeking for a value lower than 1 meter at an altitude of 400 km, a suitable pointing control accuracy is required, and accurate device and their performances like micro reaction wheels has been considered, in order to provide the microsatellite with a capability in the pointing accuracy of 0.1°. In fact, an error in attitude of 0.1° produces a linear shift in the captured image of approximately 700 m respect to the wanted image, that is acceptable, being the swath width of 12 km. This pointing accuracy that ADCS has to provide is good enough to allow the success of the mission, since the TDI manoeuvre, shown in the next figure 2.7, is required to achieve an acceptable signal to noise from the camera. With TDI, successive rows of imagery are averaged as they pass over the same scene. If applied correctly, matching successive rows exactly, this process has the potential to increase the SNR by the square of the number of rows used. In order to compensate for the relatively small optics and aperture, an imaging manoeuvre is used to effectively slow down the rate at which the camera boresight scans over the ground, allowing longer integration times and so boosting signal. This is known as a *time delay integration* (TDI) manoeuvre, and the factor by which the ground speed is reduced, compared to the pushbroom case, is called the “TDI factor”. A TDI factor of up to x 4 is required for the mission to meet signal to noise requirements, with a factor of 8 being available if required. The manoeuvre takes the form of a pitch rotation in a direction which slows down the motion of the boresight over the ground. For short time periods, a constant rate rotation is sufficiently accurate to give an effectively constant ground rate.

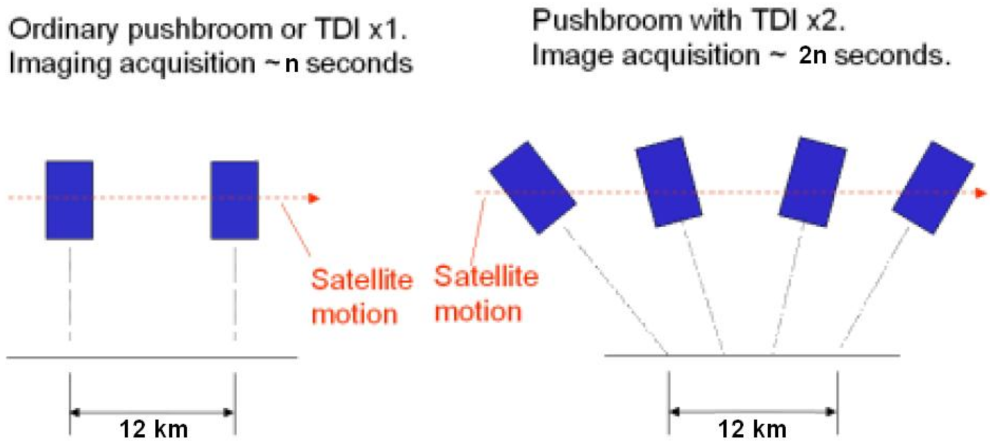


Figure 5 - 5 – TDI manoeuvre

A satellite in a 400km altitude orbit has a speed of 7.67 km/s, and its sub satellite point on the Earth’s surface has a speed of 7.22 km/s, ignoring Earth rotation. The boresight of a pushbroom imager will travel, and so image, a swath 12 km in along track direction in 1.66s. A pitch rate (microsatellite agility) of the order of 0.54 deg/s, supposing no pointing accuracy errors, is able to increase this time by a factor of 2, giving a TDI factor of 2 (next figure).

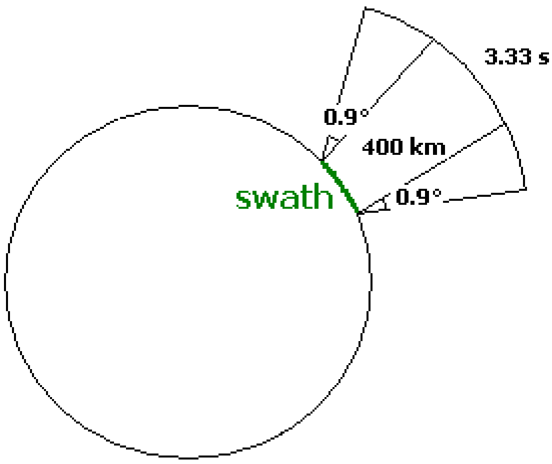


Figure 5 - 6 – Agility computation

The proposed system is a constellation of four microsatellites for electro-optical observation designed to achieve a sub-metric resolution and to offer a high revisit time on a specified target area to satisfy needs related to disaster monitoring. System responsiveness is an important design driver. Such system could provide crucial information for optimizing the execution of rescue operation in areas where natural or man-made disaster occurred. The main characteristics of the system are listed in Tab.

Mass	120 kg
Mission life	1 years
Orbital altitude	400 km
Spatial resolution	0.65 m – 12 km swath (Pan camera)
Georeferencing accuracy	20 m (without GCPs or any other device)
Acquisition capability	6 images per day per microsatellite
Maximum response time	Less than 2 hours

Table 5 - 2 – System main parameters

Each microsatellite, in order to satisfy the user requirements, mounts on board a panchromatic camera (RALCam-4) which provides a 0.65 m spatial resolution at 400 km with 12 km of swath. The communication subsystem is composed of S-band transmitter, receiver, patch antenna and high-power amplifier for data downlinking, and UHF antenna and transceiver for TT&C. This architecture supports a 4 Mbps downlink data rate. The ADCS is composed of a GPS Receiver, a Star Tracker and a 3-axis Magnetometer for attitude determination, with an accuracy of 10 arcsec in 3-axis attitude estimation; three micro-reaction wheels and 3 magnetorquers for attitude control, with an accuracy of 0.1° in 3-axis. The attitude control system provides each microsatellite with a very good agility (pitch rate 0.54 deg/s), allowing TDI manoeuvres to be performed increasing spatial resolution. On board data handling system is based on Intel 386 computer and an high speed solid state data recorder that provides a modular storage capability up to 16 GBytes. The electrical power subsystem is designed in order to fulfill the power needs in each possible condition, it is composed of a lithium-ion two-cells battery and a 0.53 m^2 body-mounted solar array with *triple junction GaAs/Ge* solar cell technology. The thermal subsystem is composed of Honeywell's HEL-700 Platinum RTD temperature sensors and Thermofoil^{TM} heaters by Minco. The heat dissipation on the microsatellite is performed by placing radiator panels, conductive strapping and shear panels in the spacecraft. Conductive strapping and shear panels will transfer heat from the subsystems to the radiator panels where it will then be released into space. The structure meets the modular concept, being composed of "building blocks" to accommodate different payloads.

The system *product tree* is shown in the next figure. The six subsystems integrated in each microsatellite are:

- Payload
- Communications
- Attitude Determination & Control
- On Board Data Handling
- Power
- Thermal

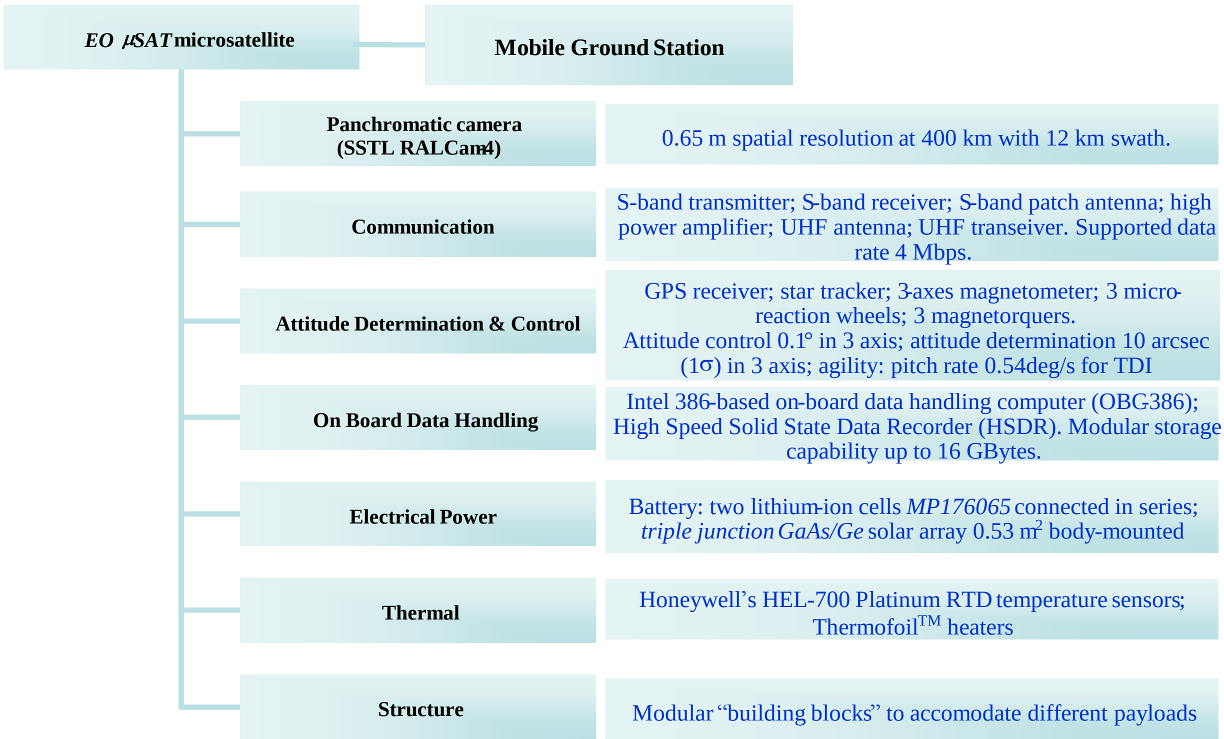


Figure 5 - 7 – System product tree

5.1.6. Electro-optical Payload Design

A key mission motivation for implementing the Operationally Responsive Space (ORS) (traditionally associated to military operation) methodology also to "welfare" scenarios is the optical observation of disaster zone. Minimizing the cost and delivery schedules of optical observation payloads having true operational capability will be a fundamental asset for the success of these missions. One of the challenging issues during sizing phase of an optical payload for Earth Observation is the calculation of the focal length. The relation that allows the focal length to be determined is written accordingly to the following figure:

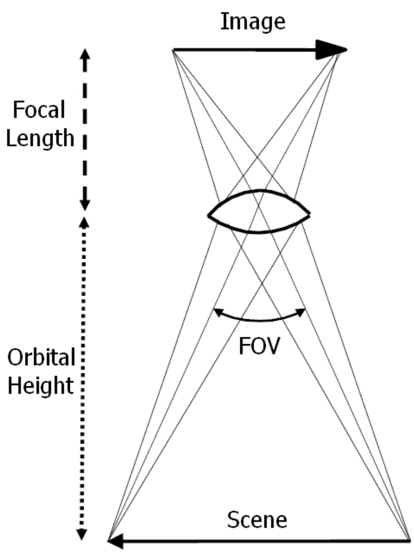


Figure 5 - 8 – Optics geometry

$$f = H \cdot d / L$$

Where f is the focal length, H is the orbital height, L is the resolution of the image and d is the camera pixel size. This relation means that in order to achieve a better spatial resolution it is necessary to increase the focal length. Considering an orbital height of 400 km, a pixel size equal to 10 μm, and an image resolution of 0.65 m, corresponding to RALCam 4 parameters, the focal length required is:

$$f = \frac{400 \cdot 10^3 \cdot 10 \cdot 10^{-6}}{0.65} = 6.15m$$

The focal length needed to achieve high spatial resolution is too large compared with the dimension of the microsatellite. This problem is solved by using a system of lenses and mirrors able to increase the focal length even in small volumes.

In order to achieve a sub-meter resolution in a small satellite with an optical payload, the main optical systems which are suitable for this purpose are listed in Table 2.3, with pros and cons of each option [1].

Optical system	**Pros**	**Cons**
Cassegrain-derived	Comparatively simple and cheap spherical mirrors. Easy to build and align.	Secondary mirror limits light gathering capacity. Large diameters needed for sub-1m resolution.
Three-Mirror Anastigmatic (TMA)	No aperture obscuration. Relatively compact & lightweight format.	Off-axis aspherical, expensive mirrors. Complex alignment.
Korsch	Smaller hole in primary mirror.	Complex to manufacture. High mounting sensitivities.
Newtonian-derived	Comparatively simple mirror shapes. Very small obscuration.	Long. Relatively complex relay lens needed.

Table 5 - 3 – Optical devices: pros and cons

The Cassegrain-derived design form is demonstrated by the NigeriaSat-2 satellites and is shown in Figure 2.13. NigeriaSat-2 satellites have 2.5m resolution in the panchromatic channel and 5m resolution in four Landsat-type bands, R, G, B and NIR in a swath width of 20km. This represents state of the art for 300kg class satellites from an orbital altitude of 700km. The diameter of the primary mirror is 385mm, and the length around 1m.

This design form reaches its limit of viability at parameters around those for NigeriaSat-2. For further reductions in ground resolution, the first problem is that adequate signal-to-noise ratios cannot be maintained, due to the smaller pixel area on the ground and the shorter dwell time on it. This problem can be solved. The use of TDI (Time Delay Integration) CCD detectors can restore signal-to-noise ratios, and once the transition to TDI CCDs is made, the optics diameter is limited by the diffraction MTF and not by light gathering power. NigeriaSat-2 with TDI detectors could be flown in orbits down to 400km to give around 1.5m resolution. However, remote sensing customers usually require resolutions down to 0.6m, coupled with swath widths in the region of 13km, even from relatively low-cost small satellites. This requires large focal plane dimensions, since the TDI CCDs cannot be made with pixel sizes below about 0.01mm. The limit to the viability of Cassegrain-type designs comes from the fact that the large focal planes require very large holes in the middle of the primary mirror, and this pushes the primary diameter up much higher than simplistic diffraction-limit calculations might suggest, to around 500 to 600mm. This makes the telescope too large and heavy for feasible small satellite applications.

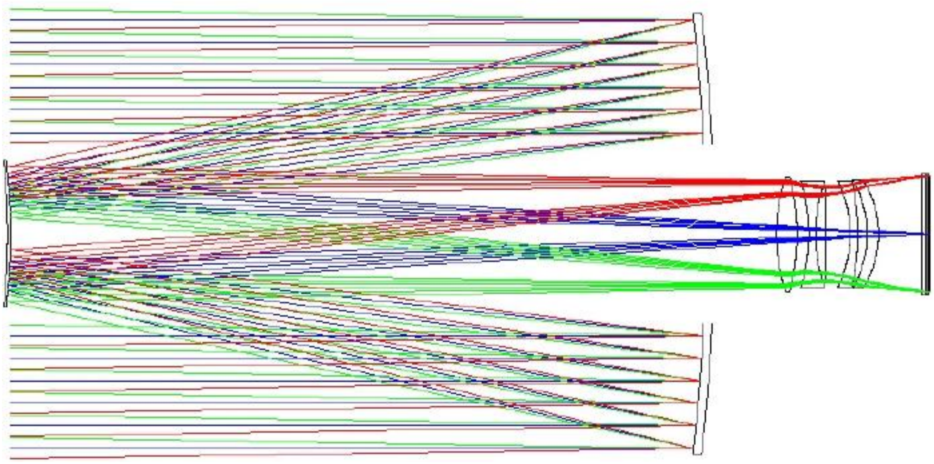


Figure 5 - 9 – Cassegrain-derived design

A TMA design which would meet the above mentioned specification (0.6m resolution in medium swath from 400km altitude) is shown in Figure 2.14. The fourth mirror shown in the figure is a flat fold mirror which can be used for focusing. The main problems with this type of design are the difficulties of manufacturing and aligning the three off-axis aspheric mirrors. Another issue is thermal control. One TMA with relatively modest 6.5m resolution supplied for an SSTL satellite demanded thermal control of around ±1 degree, which is a problem a low-cost small satellite programme could well do without, [1]. The advantages of a TMA system are the un-obscured aperture, that gives a good diffraction limit from a relatively small aperture; that it is shorter than conventional optical form; and that it is relatively lightweight.

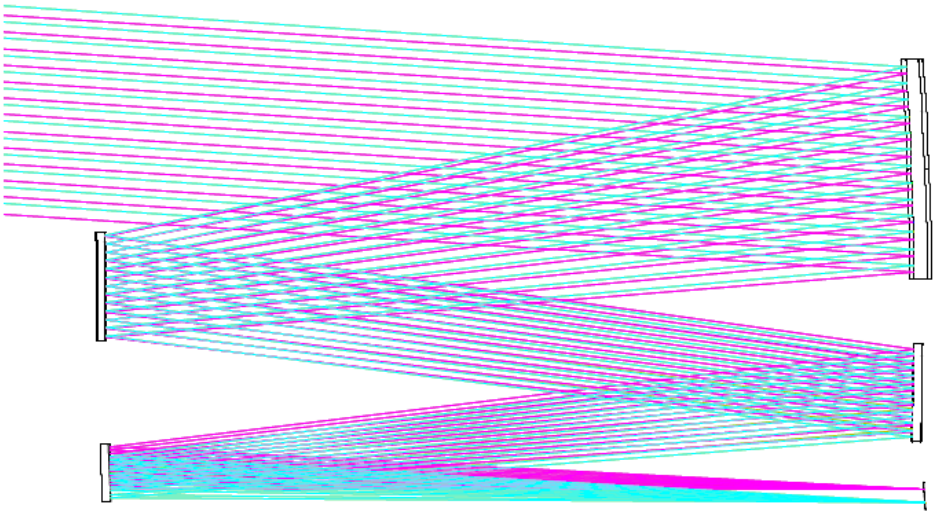


Figure 5 - 10 – TMA design

The Korsch design form (Fig. 2.15) is like a focusing TMA used close to axis so that the front-end looks like a Cassegrain system with a focused image close to the centre of the first mirror. The primary image is smaller than the final image so the hole in the first mirror is not as large as in the Cassegrain option. However, the Korsch design form still has a central obscuration, is quite complex to manufacture and some of the mounting sensitivities are high.

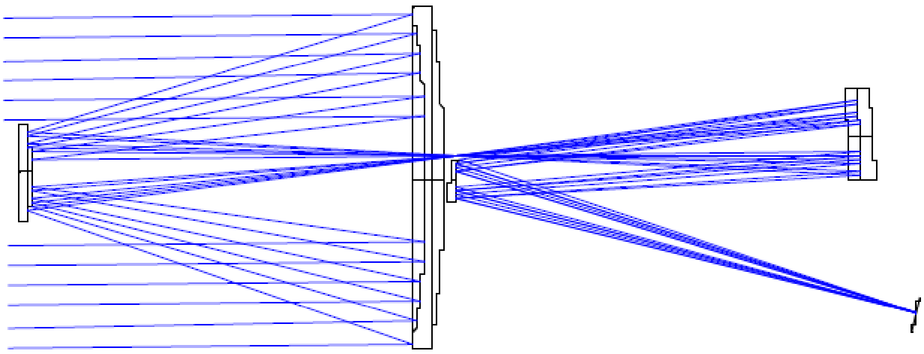


Figure 5 - 11 – Korsch design

One on-axis design which is almost un-obscured and overcomes the need for large mirrors is derived from the Newtonian form, utilising only one large curved mirror. The image is picked off by a prism or mirror at the prime focus, which can be relatively small and therefore provides only a very small obscuration of the aperture. The image is then re-imaged by a relay lens on to the focal plane. A mirror diameter of around 350mm is sufficient to give a resolution around 0.6m on the ground from an altitude of 400km. A multi-CCD focal plane using a combination of TDI CCDs for the panchromatic channel and linear CCDs for the colour channels (where the pixel size is larger) gives adequate signal-to-noise ratios. The disadvantages of this design form is that the relay lens is relatively complex, and the telescope is the longest in its class, making the satellite cross-section high in the direction of travel unless additional optical folds are applied.

Modulation Transfer Function (namely MTF) is the most widely used scientific method of describing lens performance. The MTF is a measure of the transfer of modulation (or contrast) from the subject to the image. It measures how faithfully the lens reproduces (or transfers) detail from the object to the image produced by the lens.

Diffraction is the fundamental optical limit on image quality and resolution that results from the wave nature of light and the finite diameter of lenses. A perfect lens is sometimes called a *diffraction-limited* lens, because the only thing that is limiting its performance is diffraction.
The formal definition of MTF is:

$$MTF = \frac{I_{\max} - I_{\min}}{I_{\max} + I_{\min}}$$

Where I is the intensity.

The design MTF for the optics alone is shown in Fig. 2.16, for Cassegrain-derived and unobscured (e.g. the TMA or the Newtonian, which is virtually un-obscured) design forms. The effects of optics manufacturing and assembly tolerances are not taken into account, and will worsen the MTF. Furthermore, the MTF of the detector will be typically 50% and so will cut the values by a further factor 2. The advantage of un-obscured apertures over the Cassegrain form is clear, in terms of MTF for a given aperture size.

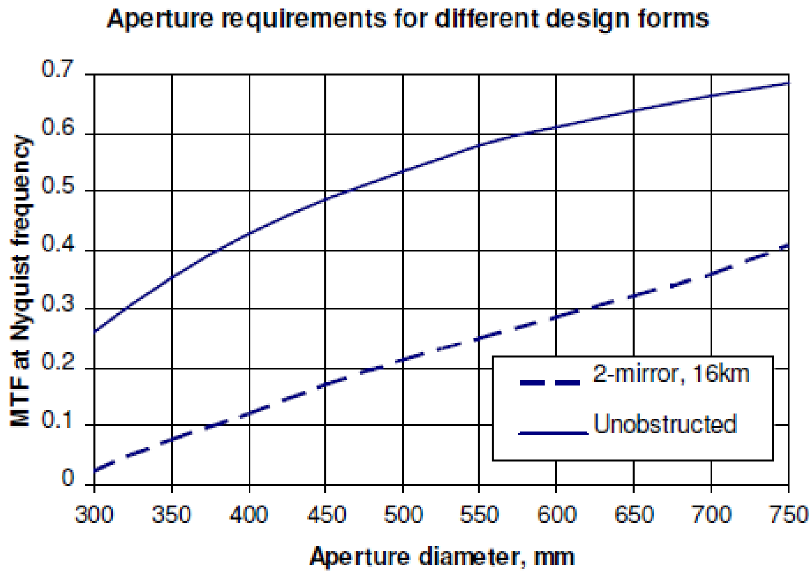


Figure 5 - 12 – Design MTF for un-obscured and Cassegrain design forms for 0.6m in 16km swath

Several trades have to be taken into account when dealing with detector design. Cameras with sub-meter resolution need signal enhancement, and one way to achieve this is to use TDI (Time Delay Integration).

The pixel size is a major factor to be considered in design trade-offs for the detector. Small pixels are desirable to keep the focal plane size as small as possible, otherwise the opto-mechanical design of the telescope becomes difficult. However, small pixels allow only a small full-well signal capacity which reduces the dynamic range of the sensor.

Another issue to be traded against signal capacity is anti-blooming. Blooming is due to the fact that lenses can never focus perfectly, and the result is that the image of the bright light appears to bleed beyond its natural borders. Anti-blooming structures on the CCD take up real estate, and whether these are used or not depend on the application.

The use of TDI CCD depends on the Earth image tracking over the CCD at the same rate as the charge steps down the CCD columns. Any mismatch in these two speeds leads to a reduction in MTF, as charge which should be in one pixel is deposited in another. This is a constraint on the quality of the attitude control system, as is the need to keep the track direction aligned with the CCD columns.

5.1.6.1. RALCam-4 Camera

Figure 5 - 13 – RALCam-4 camera

RALCam 4 combines cameras providing very high quality, high resolution imaging, with a low power Payload Controller, Processor and Memory Unit (PCPMU) that provides multi-terabit on-board data storage and high speed data output at low price.

The overall optical payload consists of two main elements:

1) the optical camera which includes the telescope and an integrated Focal Plane and Electronics Assembly (FPEA);

2) the Payload Controller, Processor and Memory Unit (PCPMU) which performs all the camera control, data storage, and data formatting.

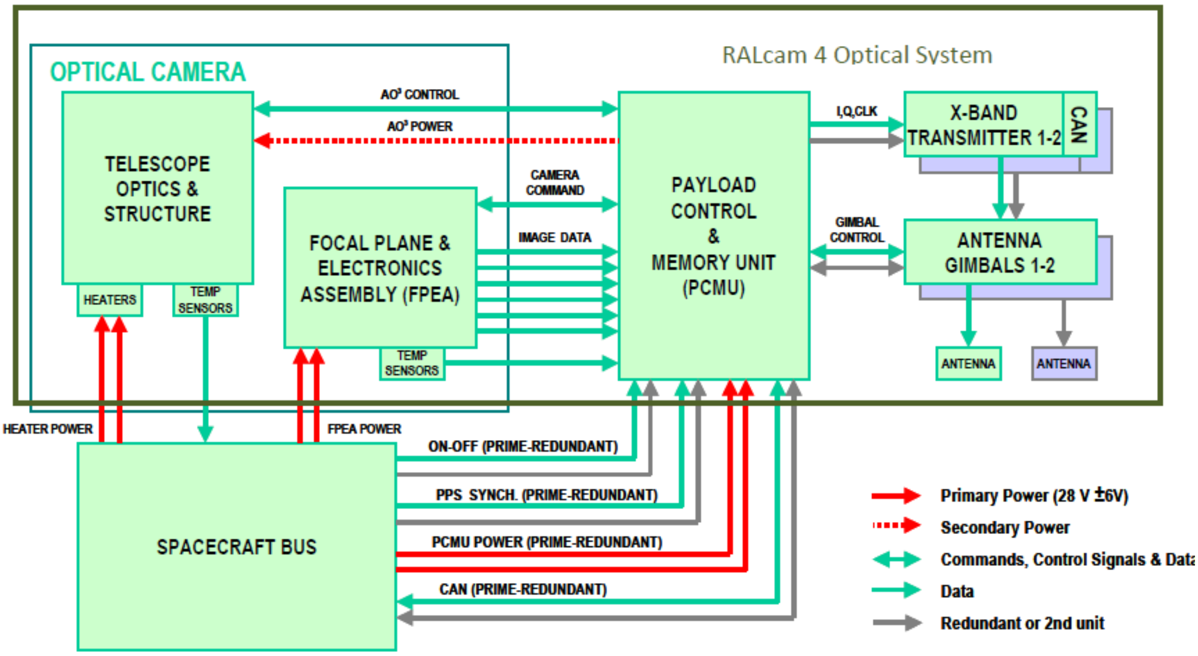


Figure 5 - 14 – RALCam-4 optical system

The optical camera design is a 'Korsch' Three Mirror Anastigmat (TMA) type with two extra fold mirrors which enable the optical path to be fitted in to a very compact envelope. The mirrors are made from Zerodur, a low expansion glass, and they have up to 70% light weighting applied to reduce their mass. The mirrors are coated with protective Aluminum to give high reflectivity in the visible wavelengths.

The telescope is used in push-broom mode with separate linear detectors and a band-pass defining filter for each channel. The SNR is controlled by TDI (Time Delay Integration) CCDs which allows the entrance pupil diameter to be only used to maintain the optical MTF. Because the telescope contains no lenses, there is no chromatic aberration and hence the focusing is the same in all bands.

The design was optimized to give a near diffraction limited performance over the whole field of view. The front section of the telescope has two on-axis conic mirrors (M1 and M2) giving an intermediate image close to M1. The final image plane is formed by M3 after folding by the two auxiliary flats. There is an accessible exit pupil between M3 and Flat #2 which, coupled with the intermediate image, gives this design excellent stray light control properties.

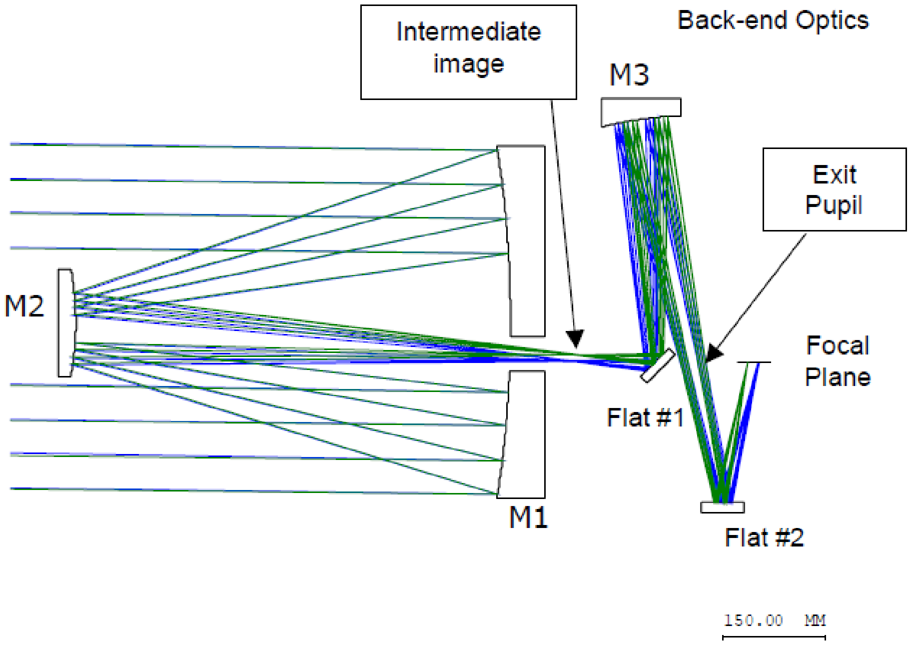


Figure 5 - 15 – RALCam-4 camera: optical system layout

A critical design aspect of this telescope (and any telescope of this performance class) is the positional instability of the optical components relative to M1 that could result from either the launch loads or material changes once in space (such as water absorption of the CFRP material on the ground which can cause small dimensional changes once in space when the water is outgassed). Maintaining high positional stability of the optical elements is a significant cost driver and is one the major impediments to attaining high image quality on orbit. To address this issue for RALCam-4 camera an approach based on the “Active Optics” techniques is used. The basic approach is that rather than relying on maintaining extremely high stability when subjected to the very large launch loads and once exposed to the vacuum of space, an MDA proprietary system (AO^3) is used to realign and focus the telescope to recover to very near pre-launch performance.

The primary structure of the camera is made from Carbon Fibre Reinforced Polymer (CFRP). This has the advantage of high stiffness combined with low mass, and through careful design of the laminate, can be tailored to give virtually zero coefficient of thermal expansion. A “camera support frame” attaches to the telescope using a single rigid titanium mount and 3 titanium flexure mounts.

This simplifies the interface of the camera to the spacecraft bus. The camera support frame mounts to the spacecraft via three vibration isolators that help reduce high frequency vibration generated by the spacecraft subsystems from reaching the mirrors and causing optical bore sight movement (jitter).

The FPEA uses two custom CCD detector designs, one for panchromatic and one for multi-spectral use. The Pan device has 10μm square pixels and is organized to operate as a TDI device with 4196 columns and 96 TDI lines. A facility is included which allows the active TDI length to be altered remotely to allow fine tuning of system sensitivity, using structures derivative from those previously implemented on other space TDI devices. Each device has 4 output amplifiers in order to output data at the required rate. For the multi-spectral application, the chip design includes 4 separate TDI devices, spaced 1.5 mm apart on the same piece of silicon. Each TDI device is covered with an optical filter. In this case, the pixel size is 40μm square, with 1049 columns and 16 lines of TDI. The data rate from each of these devices is low enough to be taken from a single output for each.

Thermal control of the camera is achieved by a combination of electrical heaters and a passive radiator that mounts directly to the camera. The radiator is located directly above the FPEA and has a high conductivity. During imaging, this FPEA subsystem requires cooling to maintain the CCD temperatures in a suitable range to achieve low noise performance. During non-imaging periods heaters on the radiator are switched on to prevent the CCD from becoming too cold. The camera is conductively and radiatively decoupled from the spacecraft by the Titanium feet & elastomeric camera isolators and multilayer insulation (MLI) wrapped around the entire body of the camera.

Camera configuration is shown in Figure 2.20:

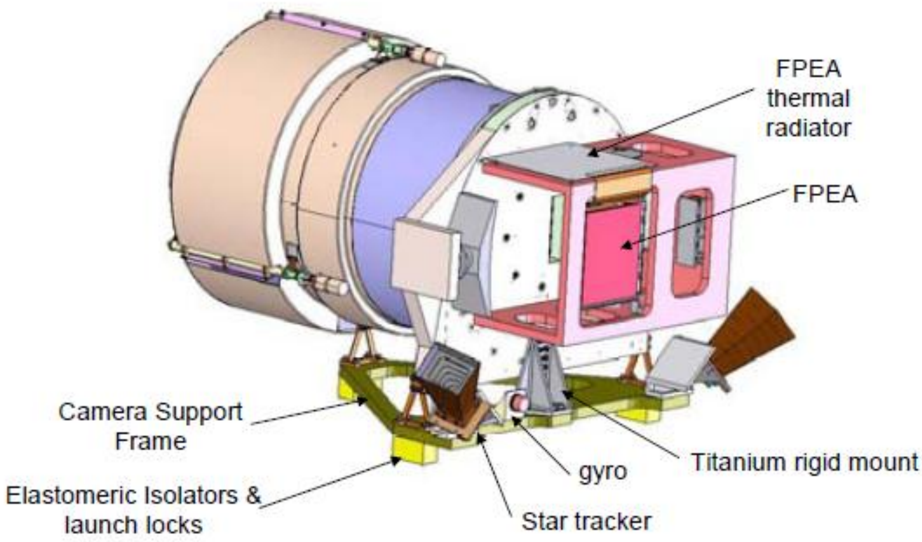


Figure 5 - 16 – RALCam-4 camera configuration

The "camera support frame" is also used to accommodate the star trackers and gyros used for the spacecraft attitude control and image processing. The camera support frame is also made of the same near-zero CTE CFRP material as used for the telescope structure. This significantly improves the systematic geometric accuracy of the system.

The PCPMU is designed as a general purpose, modular unit that can be used in a wide variety of missions. It is a payload controller, processor and memory unit that can be used for optical, radar and high bandwidth store and forward communication satellites. This modular unit supports the plug and play concept and is suitable for responsive space philosophy. The high performances consists of high speed electronics, high memory capacity (order of Tbits), and low power, but allowing for highly competitive price compared with other electronics systems of this class. For the Earth Observation tactical mission that is studied in this thesis, with a very low acquisition capability, but with very high quality products, the parameters of the RALCam 4 PCPMU are oversized. It is possible to choose a lower performance PCPMU, or to use an oversized PCPMU that is not tailored on the payload but which can be used even for different kind of missions, with the advantage that it can be kept in advance in a store and it can be available when the mission is called.

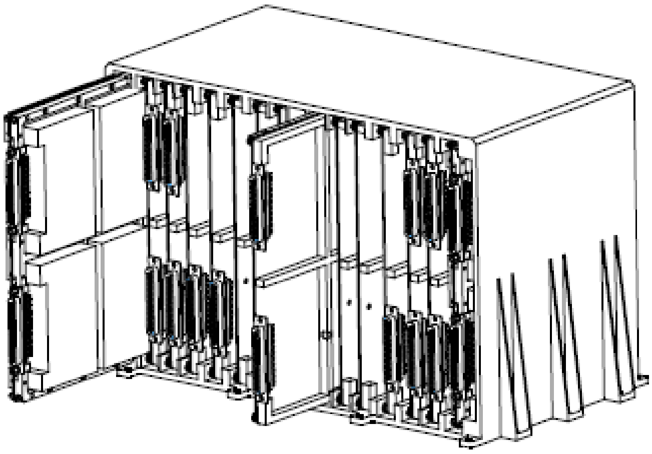


Figure 5 - 17 – PCPMU unit

For the optical payload, PCPMU provides some fundamental functions. It works as “instrument controller”, by interfacing to the optical cameras and setting operating parameters; “optical instrument data interface”, by accepting high speed image digital data from optical cameras; performing “image compression” in real time using JPEG2000, with variable compression level settings; “on-board memory”, through its data storage unit in non-volatile memory devices, that can be configured from 0.5 to 4 Terabits; “data formatting and output interface”, by formatting and encrypting image data and providing it to two downlink transmitters simultaneously at data rates of up to 400 Mbps per transmitter; “mechanism drive electronics”, by driving X-band data downlink antenna gimbals and AO^3 mechanism on the camera for optical alignment; “power controller”, by taking unregulated 28 V from the spacecraft platform, and providing various secondary voltages for the internal modules.

The PCPMU can accommodate a large number of standard 6U (eurocard) cards and is designed to be expandable to what each specific mission requires. The cards are wedge locked into an aluminum chassis for easy assembly. The configuration can be assembled quickly and avoids space consuming cables. The PCPMU is internally redundant with at least 2 of each card type to ensure single fault tolerance for all functions. For the cards that are redundant, the approach used is cold standby.

Real-time downlink capability is very useful for a range of targets close to the ground station, because the ground station is located in the target area, and the images that are acquired can be downloaded instantaneously.

Parameter	Value	Comments
Mission life	7.25 years	
Mass		
Camera	72.7 kg	Includes margin
PCMU	20.9 kg	Includes margin
Camera Envelope		
Length	1.17 m	Does not include MLI
Height	0.75 m	Does not include MLI
Width	0.83 m	Does not include MLI
Power		
Camera – FPEA	60 W	Typical duty cycle is 10%
Camera – heaters	25 W	Orbit average (estimate)
PCMU – imaging	58 W	Typical duty cycle is 10%
PCMU – imaging and downlinking	82 W	Typical duty cycle is 10%
PCMU – downlinking only	64 W	Typical duty cycle is 10%
PCMU – data retention	0 W	
Camera Parameters & Performance		
Pan CCD pixel size	10μm x 10μm	
Number of Pan pixels across track	20.580	Based on 5 CCD devices with 4196 pixels per device and 100 pixels overlap
Number of Pan TDI stages	96	Selectable
MS CCD pixel size	40μm x 40μm	
Number of MS pixels across track (per band)	5145	Based on 5 CCD devices with 1049 pixels per device and 25 pixels overlap
Number of MS band TDI stages	16	Selectable
Spectral bands	Pan: 450-700nm Blue: 450-520nm Green: 520-600nm Red: 630-690nm NIR: 760-900nm	Bandwidths used for SNR calculation. The MS spectral band widths can be tailored.

Focal length	6.000 mm	
Aperture	480 mm	
Radiometric resolution	12 bits	
Pan Ground Sample Distance (GSD)	1.0 m	At 600 km altitude and at nadir
MS Ground Sample Distance (GSD)	4.0 m	At 600 km altitude and at nadir
Swath width	20.58 km	At 600 km altitude and at nadir
Pan optical MTF	38%	Theoretical (at diffraction limit)
Pan MTF across track	> 16.1%	System level MTS – includes spacecraft velocity, bore sight jitter, thermal effects
Pan MTF along track	> 10.8 %	System level MTS – includes spacecraft velocity, bore sight jitter, thermal effects
SNR	146 (Pan) 175 – 275 (MS)	Pessimistic case. 600 km altitude based upon a Sun angle = 67.3 deg, TDI level = 24 (Pan), TDI level = 1 (MS), scene reflectance = 0.5, Solar irradiance = 1.500 $Wm^{-2}\mu m^{-1}$
PCMU Parameters & Performance		
Data storage	1.0 Tbit	At EOL (1.5 Tbit at BOL)
Real-time downlink	yes	As data is acquired, it can be downlinked in near realtime (small delay needed for buffering)
Data compression	4:1 or higher	Minimum allowable compression, uses JPEG2000
Data encryption	Triple DES or AES	Can also use other options
PCPMU interfaces to spacecraft bus - Command & Control Interface - Timing Synchronization - On-Off signal - Power Interface	MIL-1553B Pulse per Second Logic level + 28 V +/- 6 V	CAN bus or RS-422 are options (All electrical interfaces have a prime and redundant interface)
Data Downlink		
Downlink transmitter (2 units) - RF bandwidth	 8025 – 8400 MHz	

- DC power - Modulation - Mass	75 W (per unit) OQPSK 4 kg (per unit)	
X-band downlink antennas - Steerable range - Mass	Azimuth: 0-360° Elevation: 0-90° 6 kg	2 Units. Mass per antenna and gimbal assembly.
Total downlink data rate	420 Mbps	Using both transmitter and antennas simultaneously (includes formatting)

Table 5 - 4 – RALCam-4 performance specifications

Many of RALCam-4 performance specifications are oversized for user requirements. This consideration led to assess the possibility of using only few of RALCam-4 modules, and adapt them to user needs, saving in the meantime mass, power and volume. Furthermore, according to a "plug & play" concept, it is very useful to deal with modules and interfacing them in a suitable way for the specific mission.

The need of achieving high resolution images makes convenient the utilization of only Pan camera, without MS camera. Furthermore, orbital altitude of microsatellites is 400 km, and at this altitude Pan camera is able to achieve a GSD equal to 0.65 m. As it will be studied in the following paragraphs, required downlink data rate is not very high, due to the low acquisition capability necessary to support prescribed operations. This consideration led to choose a data downlink unit tailored on the particular requirements.

In the following table, it is possible to observe modified RALCam-4 performance specifications, adapted to the user requirements. Using a modular approach, the other subsystems will be tailored on the payload specifications. Since the space platform performs a real-time downloading, PCMU power required to be taken into account is about "PCMU – imaging and downlinking". In order to compute peak and average required power, it is necessary respectively to sum power absolute values and to sum power values multiplied for relative duty cycles.

Parameter	Value	Comments
Mission life	7.25 years	
Mass		
Camera	59 kg	Includes margin
PCMU	20.9 kg	Includes margin
Camera Envelope		
Length	1.17 m	Does not include MLI
Height	0.75 m	Does not include MLI
Width	0.83 m	Does not include MLI
Power		
Camera – FPEA	60 W	Typical duty cycle is 10%

Camera – heaters	25 W	Orbit average (estimate)
PCMU – imaging	58 W	Typical duty cycle is 10%
PCMU – imaging and downlinking	82 W	Typical duty cycle is 10%
PCMU – downlinking only	64 W	Typical duty cycle is 10%
PCMU – data retention	0 W	
Camera Parameters & Performance		
Pan CCD pixel size	10μm x 10μm	
Number of Pan pixels across track	20.580	Based on 5 CCD devices with 4196 pixels per device and 100 pixels overlap
Number of Pan TDI stages	96	Selectable
Focal length	6.000 mm	
Aperture	480 mm	
Radiometric resolution	12 bits	
Pan Ground Sample Distance (GSD)	0.65 m	At 400 km altitude and at nadir
Swath width	12 km	At 400 km altitude and at nadir
Pan optical MTF	38%	Theoretical (at diffraction limit)
PCMU Parameters & Performance		
Data storage	1.0 Tbit	At EOL (1.5 Tbit at BOL)
Real-time downlink	yes	As data is acquired, it can be downlinked in near realtime (small delay needed for buffering)
Data compression	4:1 or higher	Minimum allowable compression, uses JPEG2000
Data encryption	Triple DES or AES	Can also use other options
PCPMU interfaces to spacecraft bus - Command & Control Interface - Timing Synchronization - On-Off signal - Power Interface	 MIL-1553B Pulse per Second Logic level + 28 V +/- 6 V	CAN bus or RS-422 are options (All electrical interfaces have a prime and redundant interface)

Table 5 - 5– RALCam-4 adapted performance specifications

In the following paragraphs, the design of the microsatellite is shown: such design, as the request was that of being responsively carried out, has been performed using the *Concurrent Engineering* principles, and through the use of the *Concurrent Design Tool* available at “La Sapienza” University; exclusively the final results of the design iterations will be reported.

5.1.7. Microsatellite final design

In the next paragraphs, the results and outcomes of microsatellite design phase will be presented: it must be noted that the design has been performed exploiting the *Concurrent Design Tool* previously developed, and extensively described in the second chapter of this thesis. The iteration necessary for getting the final microsatellite configuration have been several, as from initial budgets of power, mass and other basic features, the design evolved towards a final, following that spiral model already introduced. Of course, also the final results of such iterations will be presented.

5.1.7.1. Communication Subsystem Sizing

The Communication subsystem transmits and receives all data and commands between the spacecraft and the receiving ground station. We will use S-Band frequency to send information, in part because X-Band, though faster, is a frequency that is over used, therefore, making it difficult to communicate to our spacecraft due to “cross talking”. The communication system needs a certain amount of power to transmit the data and the larger the transmitting antenna the smaller the required power. In this paragraph a procedure is studied to size the communication subsystem, starting from the Data Rate required to fulfil the user needs, and looking for available COTS material able to provide the wanted performances[2].

5.1.7.1.1. Required Downlink Data Rate

MDA RALCam-4 Panchromatic camera is able to produce images with a ground sample distance (GSD) of approximately 0.65 m at 400 km with a swath of 12 km (FOV equal to 2°), and a radiometric resolution of 12 bits per sample. User requirements ask for capturing a few images per day, but where and when they are necessary. Therefore, the downlink data rate is sized basing on a target of 6 images/day to download for each microsatellite. This means that a constellation of four microsatellites would produce and downlink 24 images/day. This acquisition capability is very poor, compared with that of Pleiades, which is able to acquire 450 images/day. Comparing the acquisition capability and coverage time (next Figure) of the target area, 6 images/day per microsatellite mean that a satellite captures and downloads 1 image/orbit.

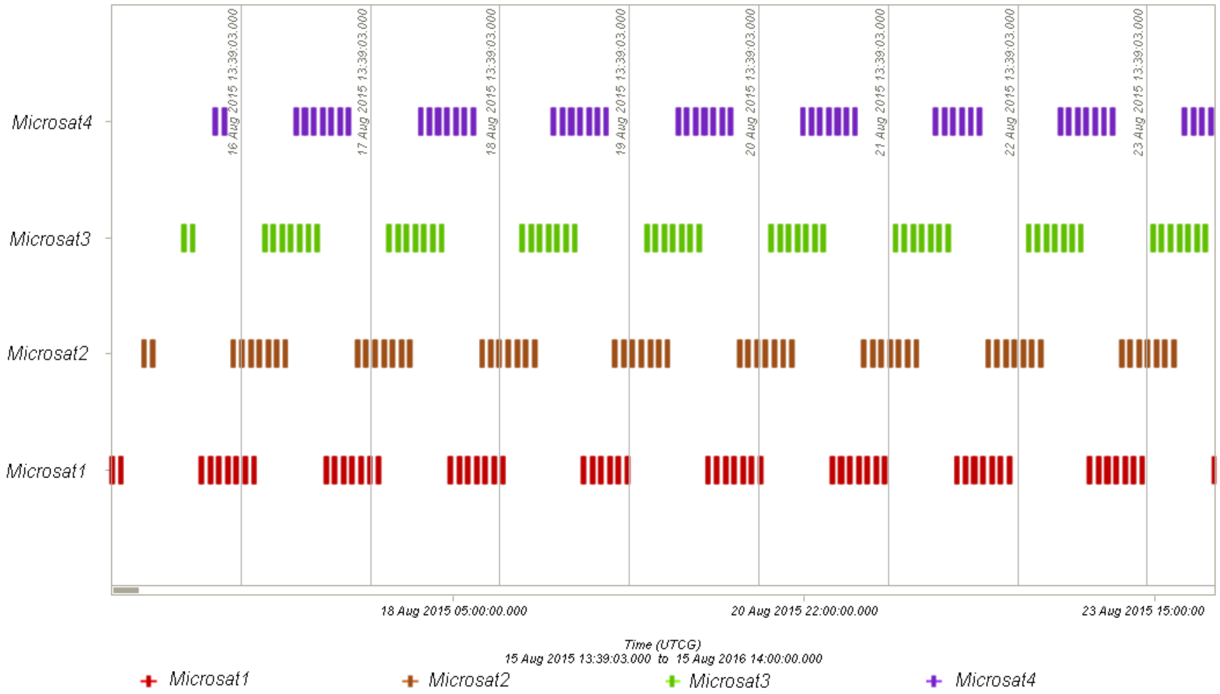


Figure 5 - 18 – Area Coverage Times

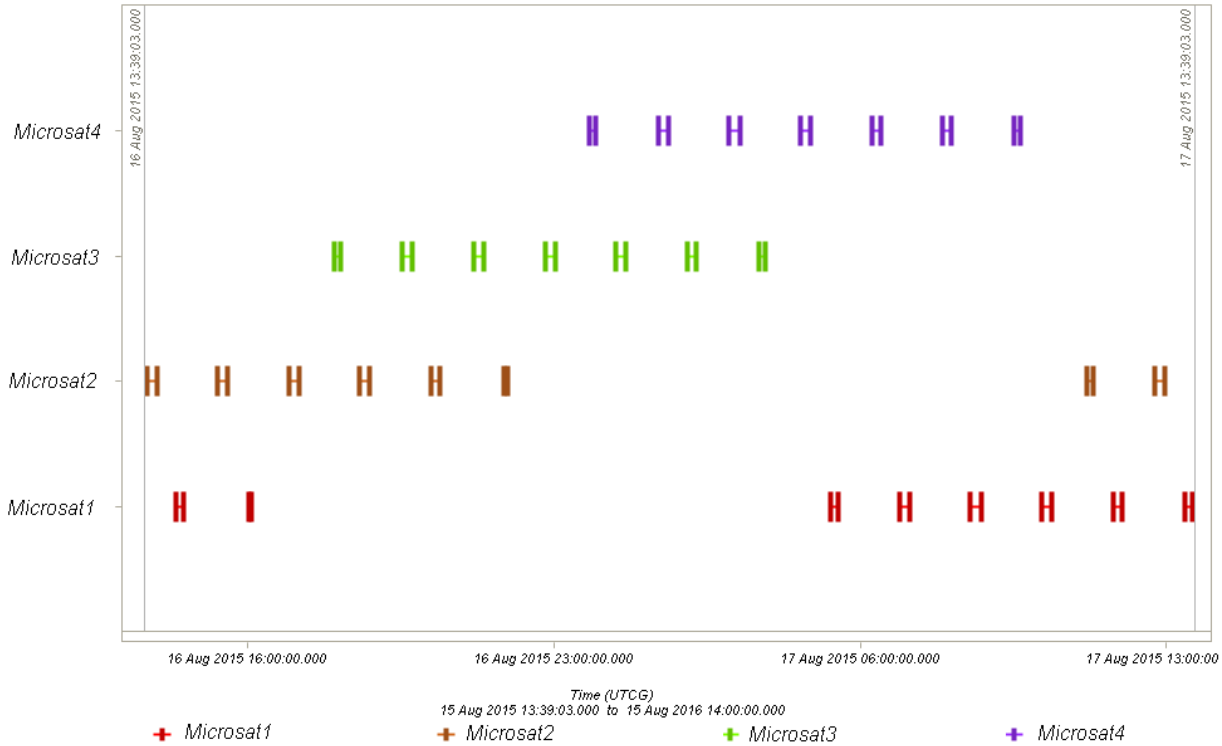


Figure 5 - 19– Area Coverage Times (smaller time scale)

Therefore, an image of 12 km x 12 km, with a GSD of 0.65 m, has 340.828.402 samples. Considering 12 bits/sample, each image has a size of 4 Gbits. This quantity of data has to be downloaded in the meantime of visibility between microsatellite's S-band patch antenna and the receiver antenna of the ground station, that is placed in the target area. This mean time has been computed through STK, and it is equal to 290 s for each orbit that allows an access to the target area. The Data Rate required to download 1 image each orbit is equal to 14 Mbps. Taking into account that RALCam-4 has a “Payload Control & Memory Unit” (i.e. PCMU) that performs an on-board data compression using JPEG2000, with a ratio 4:1, the required Data Rate, that the TLC subsystem has to support, is 3.5 Mbps. In the following paragraphs all the units of a TLC subsystem able to download a Data Rate of 4 Mbps are discussed.

5.1.7.1.2. Telecommunication subsystem architecture

The Telecommunication subsystem provides a communication link between the spacecraft and the ground station, using S-Band (commercial space band) for data downlinking and UHF (amateur space band) frequencies for TT&C. During mission, uplink is used for telecommanding, while downlink is used for telemetry, images and data downloading. The next figure illustrates a block diagram of the Telecommunication Subsystem. It includes an S-band transmitter, S-band receiver, S-band antenna, UHF antenna and UHF transceiver.

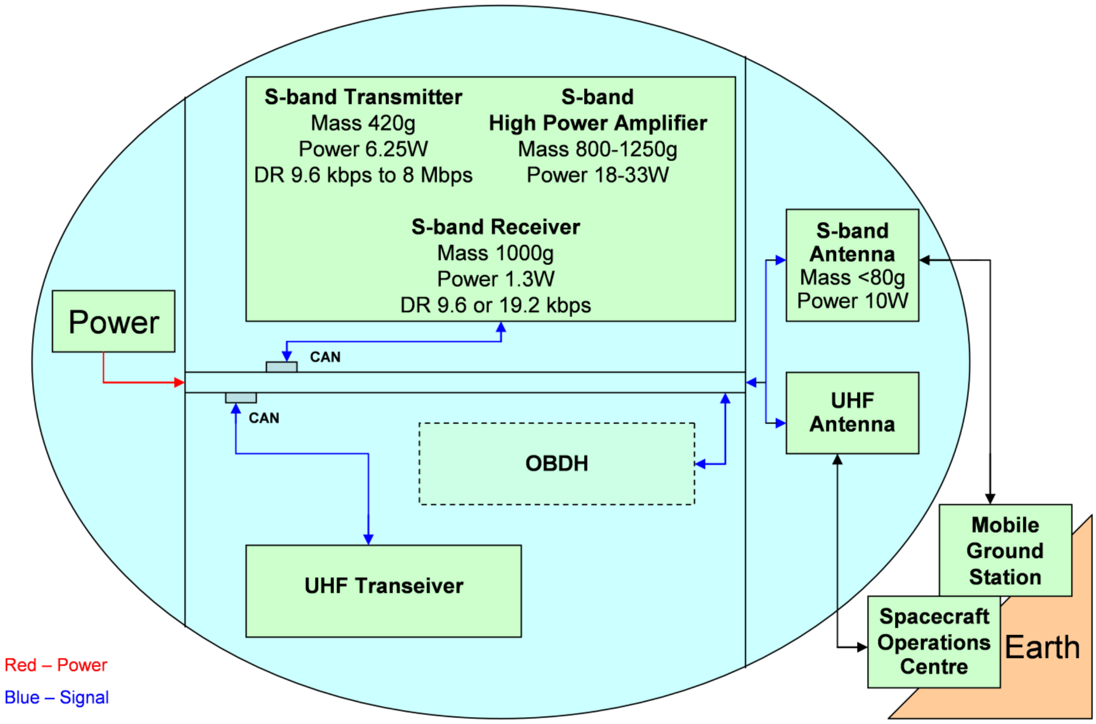


Figure 5 - 20 – Telecommunication subsystem block diagram

S-Band subsystem is composed of a transmitter, high power amplifiers, a receiver and a patch antenna. The microsatellite will be assembled with components off the shelf (COTS).

The data that is compiled from the OBDH subsystem is sent to the transmitter. This is where the data is then modulated onto a carrier. The transmitter will route the data to the antenna where it will radiate to the ground station. The ground station will be able to receive spacecraft telemetry and send command data to the spacecraft. This frequency signal will enter through the antenna and to the receiver. It will demodulate the frequency signal and route the data to the OBDH subsystem.

5.1.7.1.3. S-Band Transmitter

The transmitter relays all data received from the payloads and information concerning the status of each subsystem to the command station.

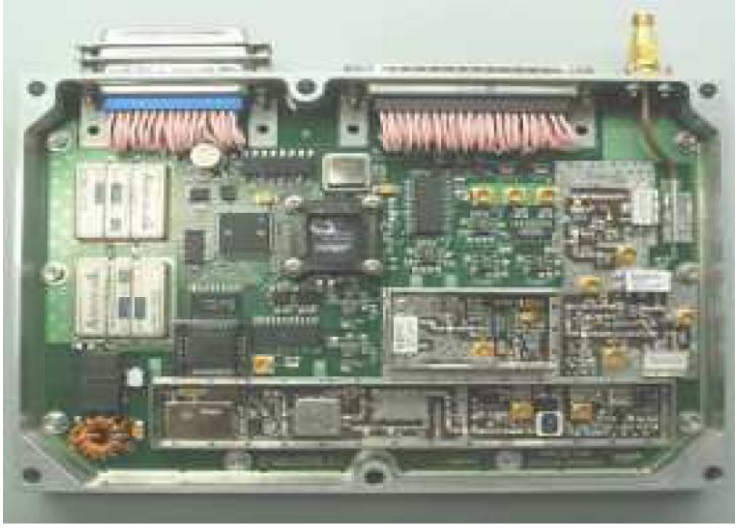


Figure 5 - 21– S-Band transmitter

The S-band transmitter shown in the previous figure, and which characteristic are depicted in the next table, is suitable for TT&C and Store-and-Forward applications. It is typically designed to support low-cost microsatellites. It can be configured for a wide range of data rates, modulation schemes and data interface options. The transmitter is based on a synthesised frequency source with a quartz crystal reference. Phase modulation is accomplished using an I/Q modulator, with bit shaping controlled via a FPGA.

S-band Transmitter Specifications	
Frequency	Pre-set in range 2.2-2.3 or 2.4-2.45 GHz
Modulation	BPSK / QPSK
Data Rate	9.6 kbps to 8 Mbps (2 rates selectable in-orbit)
Dimensions	190 x 135 x 22 mm
Mass	420 g
Connectors	15-pin D-type (Power/CAN) 44-pin D-type (Data/Clock) TT&C via CAN bus
Power	6.25W on 28V supply
RF output power	150 to 400mW with possibility to increase to 2W or 4W with additional amplifier modules
Operating temperature	-20°C to +50°C
Encoding	Differential and ½ rate k=7 convolutional encoding
Random Vibration	Qualified to 15 G_{rms} Random Vibration
Radiation Tolerance	Radiation tolerant to 5kRad

Table 5 - 6 – S-Band Transmitter Specifications

5.1.7.1.4. S-Band Receiver

The receiver's main objective is to collect commands sent from the command station on earth and relay them to the processor.

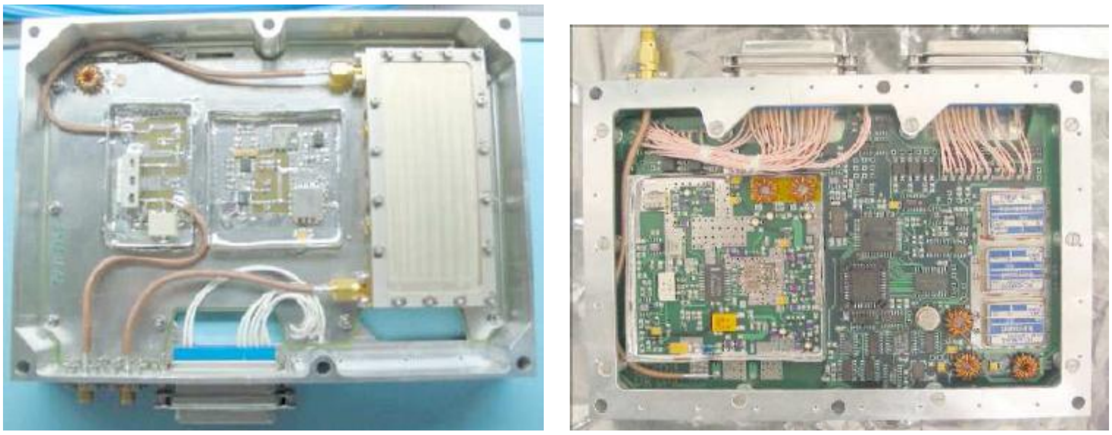


Figure 5 - 22– S-band receiver down-converter module and S-band receiver I/F module

The S-band receiver is designed for LEO TT&C applications. Uplink data is received via a 9.6 kbps or 19.2 kbps CPFSK RF modulated signal and after demodulation is forwarded to the spacecraft via TTL, LVDS or CAN data interfaces. The receiver consists of three distinguishable modules: the frontend band pass filter, the S-band down converter and the IF module. The front-end filter and S-band down converter are fitted into the same module, while the IF module is separate. The S-band down-converter PCB consists of an LNA, BPF and frequency mixer. The local oscillator port of the mixer is driven by a frequency synthesizer that allows commercial or amateur S-band frequency coverage. The IF module is divided into a RF section and a digital section with screening between the two. The RF section consists of a LPF, 20MHz band pass filter, a LNA, narrow band pass filter and FM receiver IC with associated circuitry. The IF module digital circuitry includes the CAN controller which is responsible for TT&C decoding and which interfaces with the CAN bus and down-converter synthesizer. A Miller demodulator is used to recover the 9k6 or 19k2 data. The S-band receiver has LVDS and TTL data interfaces as well as the ability to provide decoded data output via the CAN node.

S-band Receiver Specifications	
Frequency	2025 to 2110 MHz
Modulation and Data Rate	CPFSK 9.6 or 19.2 kbps
Equivalent noise figure	< 8.5 dB
Dimensions	480 x 130 x 22mm (2-module design)
Mass	1000 g
Power	1.3W on 28V supply
Connectors	44-pin DD (Power/Data/Clock) 3.5mm SMA RF TT&C via CAN bus
Operating temperature	-20°C to +50°C

Random Vibration	Qualified to 15 G_{rms} Random Vibration
Radiation Tolerance	Radiation tolerant to 5kRad

Table 5 - 7 – S-Band Receiver Specifications

5.1.7.1.5. S-Band High Power Amplifiers

Figure 5 - 23 – S-Band 2W High Power Amplifier

Figure 5 - 24 – S-Band 4W High Power Amplifier

The S-Band High Power Amplifiers provide power gain for S-Band transmitters in order to meet link budget requirements. The S-Band High Power Amplifiers are designed to boost the RF output power from the S-Band transmitter to either 2 Watts or 4 Watts as required. The amplifier is able to support commercial and amateur frequency bands of operation as well as high data rates and various modulation schemes.

Hardware

- Single-stage (2 watts) or two-stage (4 watts) amplifier design utilising discrete RF GaAs FET transistors providing linear RF output power.
- 2 watts amplifier is a single module configuration housing the DC power supply and RF amplifier.
- Provides telemetry of DC current consumption, temperature, forward & reverse RF power.

- Bias board ensures that the FET gate and drain voltages are sequenced correctly.
- Thick metal backed microwave substrate provides low loss and good thermal conductivity.
- RF input and output ports protected by isolators.

Environmental

- Passive thermal management via conductive thermal design features.
- Designed for missions in the LEO environment.

S-band High Power Amplifier Specifications	
Frequency	2200 - 2300 MHz
RF Output Power	2 or 4 Watts
Gain	10.75dB / 26dB
Dimensions	2 Watts: 35 x 135 x 190mm (1 module) 4 Watts: 57 x 135 x 190mm (2 modules)
Mass	800g / 1250g
Power	18W / 33W on 28V supply
Connectors	D-Type (Power/Telemetry) Input RF: 3.5mm SMA F Output RF: 3.5mm SMA F
Operating temperature	-20°C to +50°C
Random Vibration	Qualified to 15 G_{rms} Random Vibration
Radiation Tolerance	Radiation tolerant to 5kRad

Table 5 - 8 – S-Band High Power Amplifier Specifications

5.1.7.1.6. S-Band Patch Antenna

The S-Band antenna acts as a beacon for the transmitter and receiver. Data is received and released through the antenna.

Figure 5 - 25 – S-Band Patch Antenna

When searching for an S-Band antenna our main concern was to find the smallest antenna possible in order attain the most surface area available for the solar arrays.

The S-band Patch Antenna supports Telemetry and Telecommand data for Earth Observation and Space Science missions. The patch antenna produces a hemispherical pattern, with good gain along boresight. Gain drops to about 0 dBiC at approximately ±60º off boresight. The antenna is matched to a 50 Ohm system interfacing via a single coaxial SMA female connector. Data rates arrive up to 4 Mbps.

S-band Patch Antenna Specifications	
Frequency	Frequency tunable in the range 2.0 to 2.5 GHz
RF Power Handling	Up to 10 W
Polarisation	Right of Left Hand Circular Polarisation
Dimensions	82 x 82 x 20mm (included connector)
Mass	< 80g
Half-Power Beamwidth	~ ±35°
Connectors	Coaxial SMA Female
Operating temperature	-20°C to +50°C operating
Random Vibration	Qualified to 15 G_{rms} in all axis
Radiation Tolerance	Radiation tolerant to 10kRad

Table 5 - 9– S-Band Patch Antenna Specifications

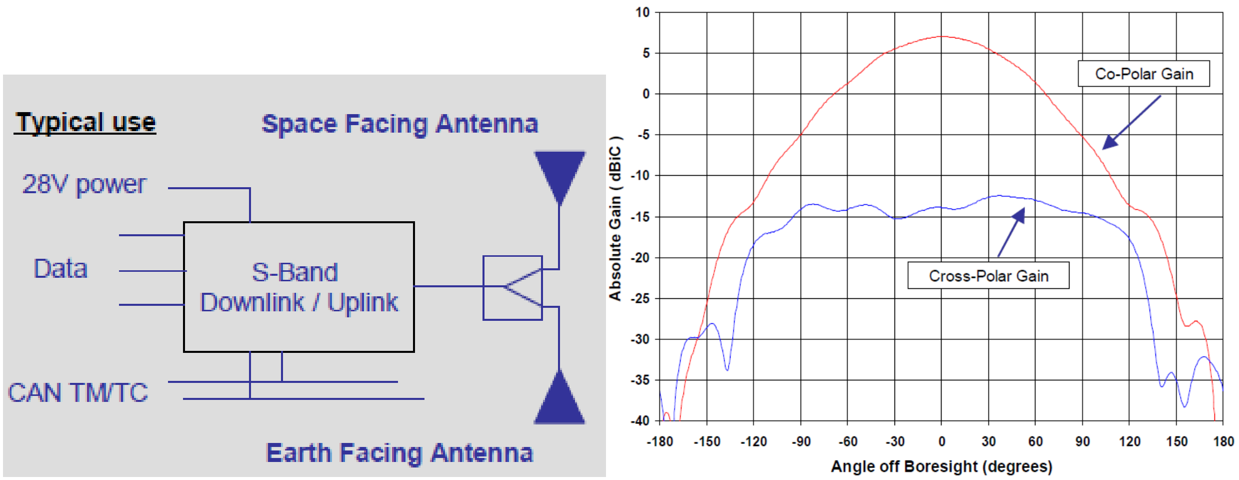


Figure 5 - 26 – S-Band Patch Antenna: typical use and performances

5.1.7.1.7. UHF Transceiver

The UHF Transceiver will collect and release TT&C data. It will also act as a backup for the S-Band transmitter and receiver if one of them fails.

Figure 5 - 27 – UHF Transceiver

UHF Transceiver Specifications	
Mass	85 g
Interfaces	RS-232/422/485
Dimensions	94 x 56 x 13.6 mm
Operating Temperature	-10°C to +60°C
Power Consumption	0.15W (receiving); 0.88W (transmitting)

Table 5 - 10 – UHF Transceiver Specifications

5.1.7.2. On Board Data Handling (OBDH) Subsystem Sizing

Even if RALCam-4 has an integrated OBDH, different solutions are analyzed in this paragraph, tailored on the particular needs of this study. The OBDH subsystem collects data and telemetry of the satellite and manipulates data sets from payload. It runs and stores all necessary software for the spacecraft and controls all functions carried out by each subsystems. The OBDH implements all commands and directions received by the ground station and relays the data to the correct subsystem. It directs the modes of the spacecraft while it orbits the Earth and ensures each one is operating at the proper time.

5.1.7.2.1. On Board Data Handling Computer OBC 386

Intel 386-based on-board data handling computer (OBC 386) is a general purpose computer for LEO applications, with triple memory redundancy (TMR) protected program memory and Reed-Solomon (software) protected ramdisk. It is the brain of the spacecraft. It includes four high speed serial communication channels and Controller Area Network (CAN) connectivity. The high speed serial links provide connections for RF modules and other systems (i.e. cameras). Other connections to the module are made through the 32 kbps CAN. A floating point processor is provided for high performance processing (i.e. for ADC). Device’s figure and scheme are shown in the followings.

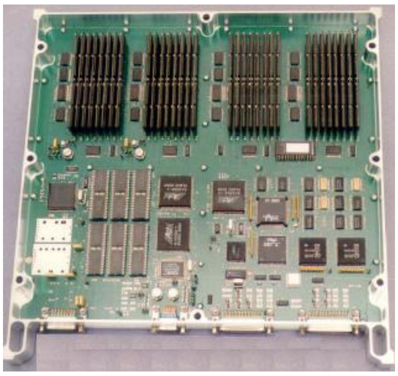


Figure 5 - 28 – OBC 386

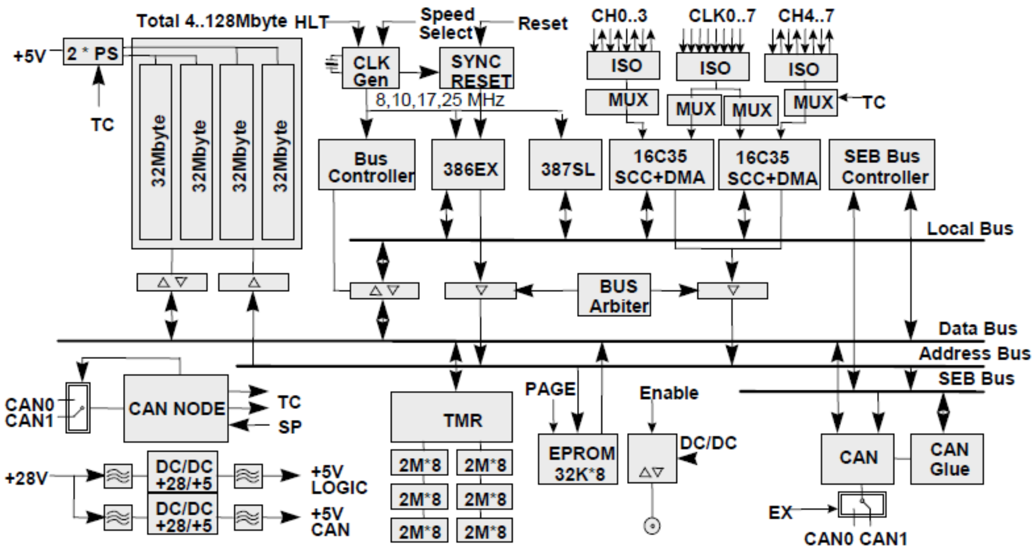


Figure 5 - 29 – OBC 386 Combines Floating Point Unit, Serial Communications and Data Storage in a single Integrated Onboard Computer.

OBC 386 specifications are listed in the next Table:

OBC 386 Specifications	
Dimensions	330 x 330 x 32 mm
Mass	1700 g
Power supply	2.5W on 5V (non isolated); 5W on 28V (isolated)
Processor	Intel 386EX 387SL co-processor 8/16/20/25 MHz clock SCOS operating system 4 Mbytes EDAC memory (TMR) Up to 128 Mbytes ramdisk (software Reed-Solomon EDAC) 4 SSC channels CAN connection with 2 buses

Table 5 - 11 – OBC 386 specifications

5.1.7.2.2. High Speed Solid State Data Recorder

A High Speed Solid State Data Recorder (HSDR) with 16 GBytes storage capability, shown in the next figure, is designed in order to support high speed data return for Earth Observation missions.

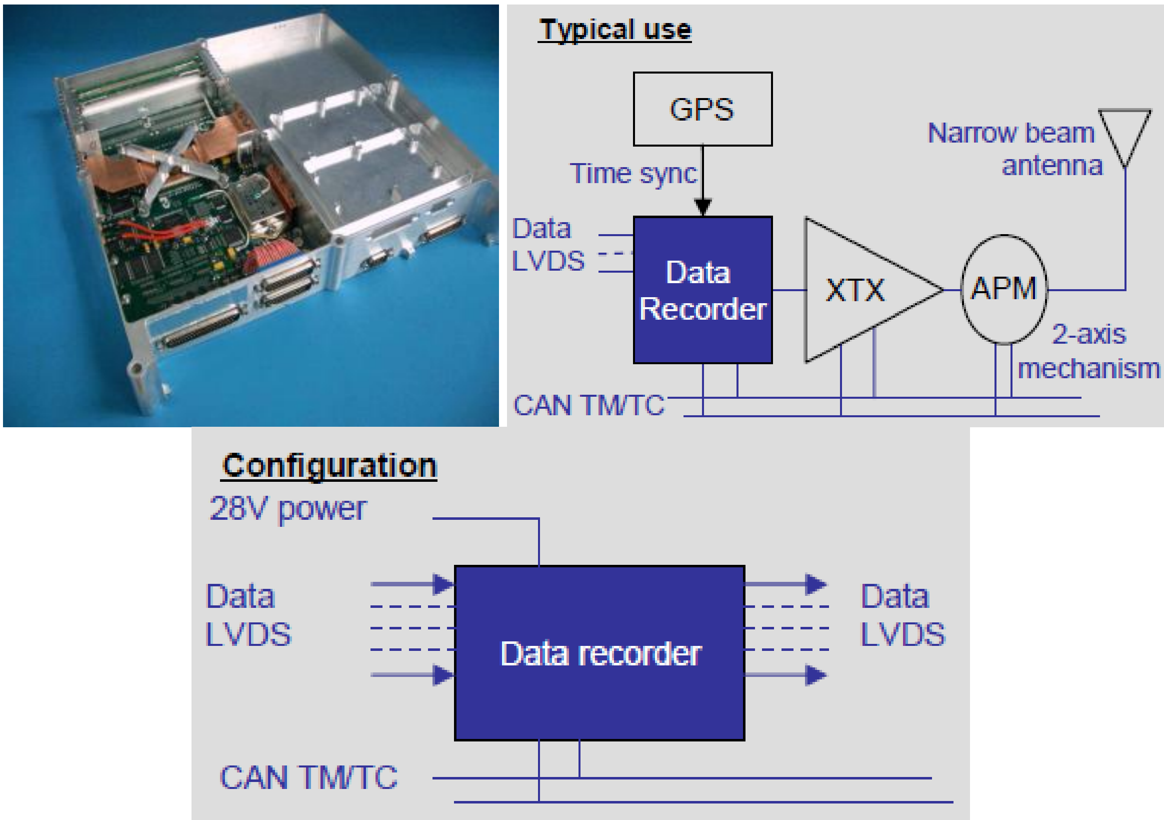


Figure 5 - 30 – High Speed Data Recorder

HSDR Specifications	
Dimensions	320 x 170 x 55 mm (half micro-tray)
Mass	1 kg
Power	15 to 50V, 5W standby, 15W peak
Storage Capacity	Modular 16 GBytes
Data I/O processing capability	> 5 Gbps
Data Input / Outputs	- 20 Low-Voltage Differential Signalling (LVDS) 150 Mbps inputs - 5 Serializers/Deserializers (SerDes) 1 Gbps inputs - 10 LVDS 150 Mbps outputs
Temperature	-20 to +50°C operating -30 to +60°C non-operating
Random Vibration	Qualified to 15 G_{rms} in all axis
Radiation Tolerance	Radiation tolerant to 10kRad

Table 5 - 12 – HSDR specifications

If RALCam4 payload is chosen, the integrated PCMU provides data storage and handling capabilities, but if another optical payload was chosen, a HSDR could be selected.

5.1.7.3. Attitude Determination and Control (ADCS) Subsystem Sizing

The ADCS determines and controls the spacecraft's location in space and its orientation relative to the Earth. The ADCS is vital for placing the spacecraft in the proper position to point for taking pictures and downlinking and uplinking data.

5.1.7.3.1. Implementing Plug & Play Concept for ADCS

The "plug and play" ADCS is designed in a modular fashion to allow it to be easily modified for an even wider variety of spacecraft, [3].

The design requirements of the "plug and play" ADCS are as follows:

- Provide attitude control to better than 0.1° in 3 axes;
- Provide attitude determination to within 10 arcsec in 3 axes;
- Operate in LEO;
- Minimum operational lifetime of 5 years.

To meet the above system requirements and to standardize components, the ADCS will use star trackers for attitude determination and reaction wheels for attitude control. Star trackers are chosen instead of an earth/sun sensor suite because they are more accurate and can be used on a wider variety of spacecraft.

A Space GPS Receiver-10 (SGR-10) is used to provide real-time PVT (Position, Velocity, Time) data. A Star Tracker is used to assess its attitude. In order to control the attitude of the microsatellite, with a pointing control accuracy equal to 0.1° as user requirement and a pitch rate (microsatellite agility) that supports the wanted TDI factor, taking into account the moments of inertia of the microsatellite, three reaction wheels with their axes in three orthogonal directions have been chosen (the fourth wheel for redundancy purposes is considered not necessary, due to the short length of the responsive mission), and three magnetorquers that are needed in order to desaturate the wheels. Finally, a three-axes magnetometer is necessary in order to measure the components of the Earth magnetic field, to allow the magnetorquers to control the attitude.

Therefore, the ADCS is composed by the following attitude sensors:

- A GPS Receiver;
- A Star Tracker;
- A 3-Axes Magnetometer.

And the following attitude actuators:

- Three Reaction Wheels;
- Three Magnetorquers.

The following COTS components are chosen, produced by SSTL.

- Space GPS Receiver-10 (SGR-10)

Surrey GPS receivers are known for providing GPS standard time, position and velocity in a compact unit.

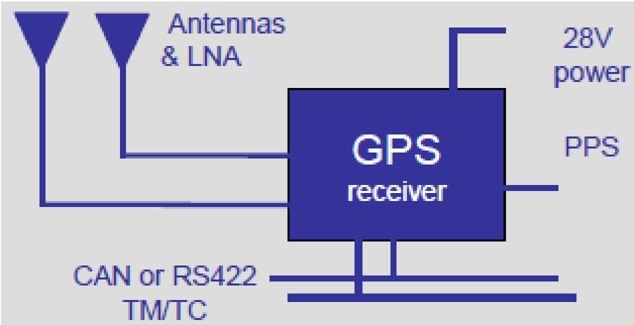


Figure 5 - 31 – Space GPS Receiver-10

SGR-10 Specifications	
Dimensions	160 x 160 x 50mm or 295 x 160 x 35mm
Mass	950g
Power	5.5W at 28V (supply 18-38V)
Orbital Position (3-D)	Typical (95%):10m; Max (95%):20m
Orbital Velocity (3-D)	Typical (95%):0.15m/s; Max (95%):0.25m/s
Time	Typical (95%):0.5μs; Max (95%):1μs
Time to First Fix	Typical (95%): 50s (warm) or 200s (cold) Max (95%): 90s (warm) or 350s (cold)
Interfaces	50 Ohm antenna interfaces (SMA) CAN and RS-422 TM/TC Interface
Operating Temperature	-20 to +50°C
Random Vibration	Qualified to 15 G_{rms} in all axis
Radiation Tolerance	Radiation tolerant to 10kRad
Typical Measurement Precision	Pseudorange: 0.9m Carrier-Smoothed Range: 0.15m Carrier Phase Noise: 1.5mm

	Doppler Velocity: 0.5m/s Carrier Range Rate Velocity: 0.03m/s Filtered Velocity: 0.01m/s

Table 5 - 13 – SGR-10 specifications

5.1.7.3.2. Star Tracker ALTAIR-HB+

A Star Tracker is able to determine the attitude of a spacecraft by comparing the stars coordinates measured trough sky observation with those of a catalogue that the system has in its memory.

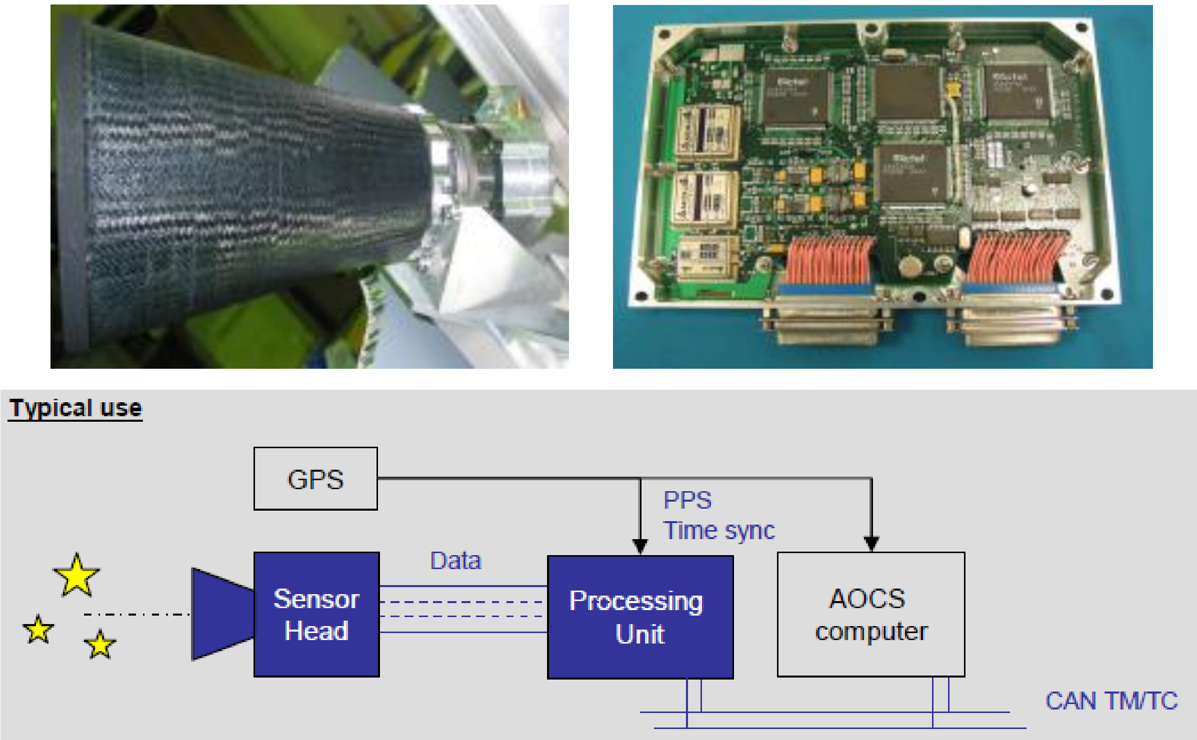


Figure 5 - 32 – Star Tracker ALTAIR-HB+

The ALTAIR-HB+ is a third generation star tracker designed specifically in support of cost-effective missions. The unit provides reliable 3-axis attitude estimation, and includes lost-in-space initialisation capability. The unit has a 99.8% matching success rate, and up to 15 stars can be tracked. The unit outputs boresight vectors in the J2000 frame as quaternions. Dynamic filtering schemes can be used to improve the attitude estimates by almost an order of magnitude. The star tracker needs a Pulse-Per-Second input from an on-board time reference such as the SGR-10 GPS receiver previously described.

ALTAIR-HB+ Star Tracker Specifications	
Dimensions/Mass	Processing module: 190 x 135 x 44 mm, 830 g Camera head: 74 x 95 x 105 mm, 530 g Baffle: 150 x 150 x 185 mm, 300 g

Power	8.5 W at 28 V unregulated Supply 16-50 V
Pointing Accuracy (1σ)	10 arcsec X-Y 55 arcsec around boresight
Operating bandwidth	0.5 Hz
Maximum tracking	0.3 deg/s
Exclusion angles (standard baffle)	Sun: 45° Earth: 35°
Interface	CAN / LVDS
Operating Temperature	-20 to +50°C
Random Vibration	Qualified to 15 G_{rms} in all axis
Radiation Tolerance	Radiation tolerant to 10kRad

Table 5 - 14 – Star Tracker specifications

5.1.7.3.3. 3-Axes Fluxgate Magnetometer

The 3-axes fluxgate magnetometer measures the magnetic field and provides readings from three sensors arranged in orthogonal axes. In addition an analogue temperature sensor provides case temperature telemetry.

Figure 5 – 33 – 3-Axes Fluxgate Magnetometer

3-Axis Fluxgate Magnetometer Specifications	
Dimensions	85 x 35 x 32 mm
Mass	140 g
Power Supply	Supply ± 10 V; Consumption < 300 mW
Sensitivity	± 10 nT
Range	± 60 uT
Bandwidth	10 Hz

Interfaces	Analogue Inputs / Outputs D-Type DC Connector
Temperature	-20 to +50°C (operating) -40 to +80°C (non-operating)
Random Vibration	Qualified to 15 G_{rms} in all axis
Radiation Tolerance	Radiation tolerant to 10kRad

Table 5 - 15 - Magnetometer specifications

5.1.7.3.4. Microwheels 10SP-M Reaction Wheels

Reaction wheels allow us to adjust the spacecraft's orientation and inertial rates. The spacecraft will have three reaction wheels one for each axis of motion.

The Microwheels 10SP-M are designed for a 3-axes control system, in momentum bias or zero bias mode. They have already been used for Deimos-1, UK-DMC2, NigeriaSat-2 and NX microsatellites.

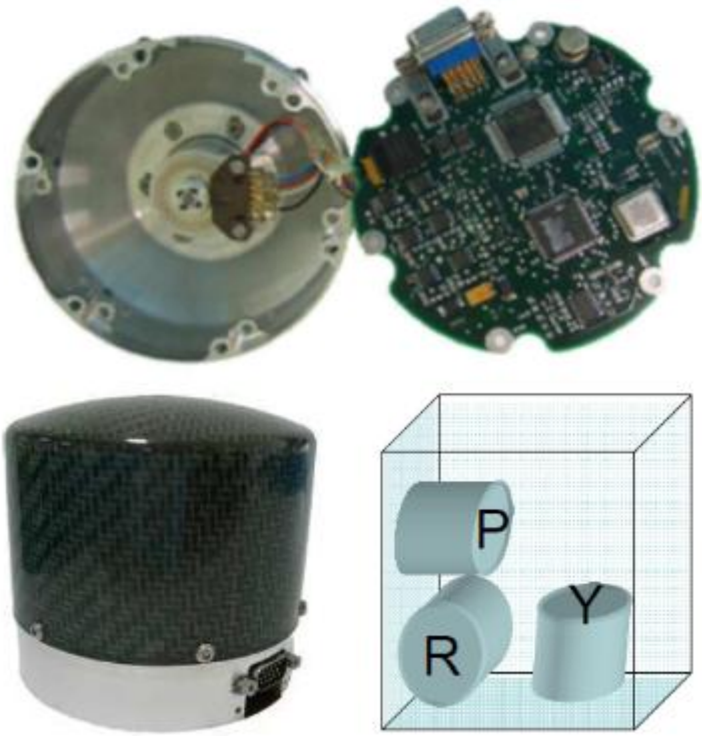


Figure 5 – 34 – 10SP-M Micro Reaction Wheel

Microwheel 10SP-M Specifications	
Dimensions	100 x 100 x 90 mm
Mass	0.96 kg
Power	5 V, 24-32 V supplies 0.65 W (0 rpm)

	0.70 W (300 rpm) 3.50 W (max acceleration)
Wheel Momentum	0.42 Nms (max)
Wheel Torque	10 mNm (max)
Wheel MOI	0.0008 kgm^2
Wheel Speed	± 5000 rpm
Speed resolution	± 0.0088 rpm
Speed control	< 0.1 rpm (rms)
Update rate	5 Hz (typ)
Interfaces	CAN / RS422
Operating Temperature	-20 to +50°C
Random Vibration	Qualified to 18 G_{rms} in all axis

Table 5 - 16 – Micro Reaction Wheel specifications

5.1.7.3.5. MTR-5 Magnetorquers

Magnetorquers are chosen in order to provide the desaturation torque for the reaction wheels.

Figure 5 - 35– MTR-5 Magnetorquers

The MTR-5 is a low cost magnetorquer capable of providing reliable attitude control to demanding missions. The MTR-5 is a Magnetic Torque Rod designed to operate from +5 V per coil and producing a minimum of 5Am^2 per coil over its operating temperature range of -30°C to +50°C. Dual Coil operation at 100mA produces 12.4Am^2.

MTR-5 Magnetorquers Specifications	
Dimensions	251 x 30 x 66 mm
Mass	500 g

Power	Peak Power Consumption: 1W Power Supply: 5V Analogue Temperature
Magnetic Moment	± 6.2 Am^2 per coil
Scale Factor	0.062 Am^2/mA per coil
Linearity	± 5% over the range operating from +100mA and -100mA
Operating Temperature	-30 to +50°C
Storage Temperature	-40 to +80°C
Random Vibration	Qualified to 15 G_{rms} in all axis

Table 5 - 17 – Magnetorquers specifications

5.1.7.4. Power Subsystem Sizing

The electrical power subsystem (EPS) provides, stores, distributes and controls power to the spacecraft during all mission phases. The Power subsystem consists of solar panels and battery that provide all the necessary power for each subsystem and payload to run properly. The EPS sizing procedure takes into account the demands for average and peak electrical power (at the End Of Life) and the orbital parameters (to determine maximum eclipse duration and sun incidence angle on array surface). The main elements of EPS are shown in Fig.:

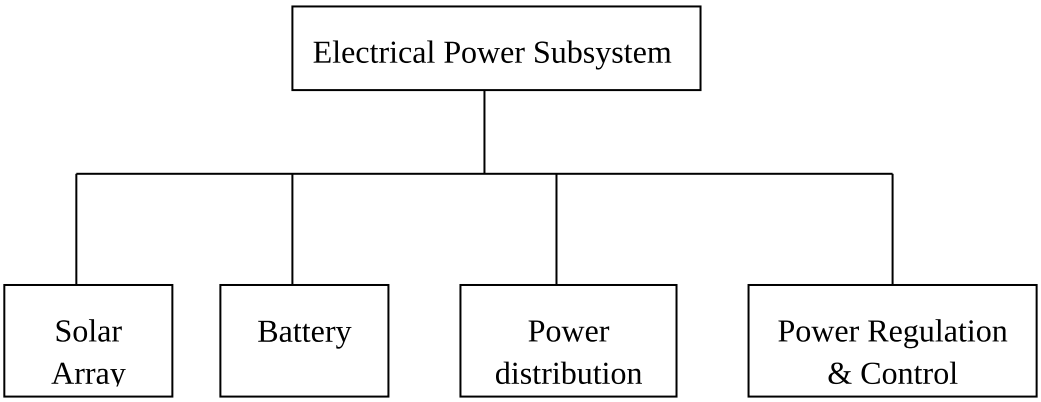


Figure 5 - 36 – EPS Functional Diagram

The EPS design process needs to identify power requirements for each subsystem (average and peak power) and to define mission life and orbital parameters. In the next table power requirements for the main subsystems are listed:

Subsystem	Average Power Required	Peak Power Required
Optical Payload	39.2 W	167 W
IR Payload	5 W	5 W
Communications	3.6 W	36 W
OBDH	10 W	20 W
ADCS	23.3 W	27.8 W

Table 5 - 18 – Power requirements

ADCS average power required is computed considering a mean rpm value for the micro-wheels. For the payload, the duty cycle by which it is necessary to multiply the peak power required, is equal to the mean coverage time, because the payload is used only upon the target area. By the knowledge of the W·h required, it is possible to choose the battery, and then to design the solar array.

The **orbital period** is **92’33’’**, while the mean **eclipse period is 34’**. The **mean coverage time upon the target area is 4’20’’**. These times have been computed through STK.

5.1.7.4.1. Battery

Battery Packs provide a way to supply energy to the spacecraft while it is eclipsed by the Earth and store power while directly in the sun.

In order to correctly size the Power subsystem it is necessary to identify the worst case and to define mission profile.

Each orbit could be split in two phases: eclipse phase and sun phase. Furthermore, during one orbit, the microsatellite could fly over the target area, where the payload acquires images and requires power, that could be illuminated or not, and could independently arrive in the eclipse or sun phases. Each of these conditions requires different power absorptions (payload and subsystems, and battery recharge) and different power providers (battery and/or solar array). Finally, an important condition is represented by the possibility that the target area is adjacent to the eclipse phase, in fact both of them require to use also the battery, which has to be designed in order to be able to provide power for the eclipse time plus the target area fly-over. All these cases are shown in the following Figure and in the successive table

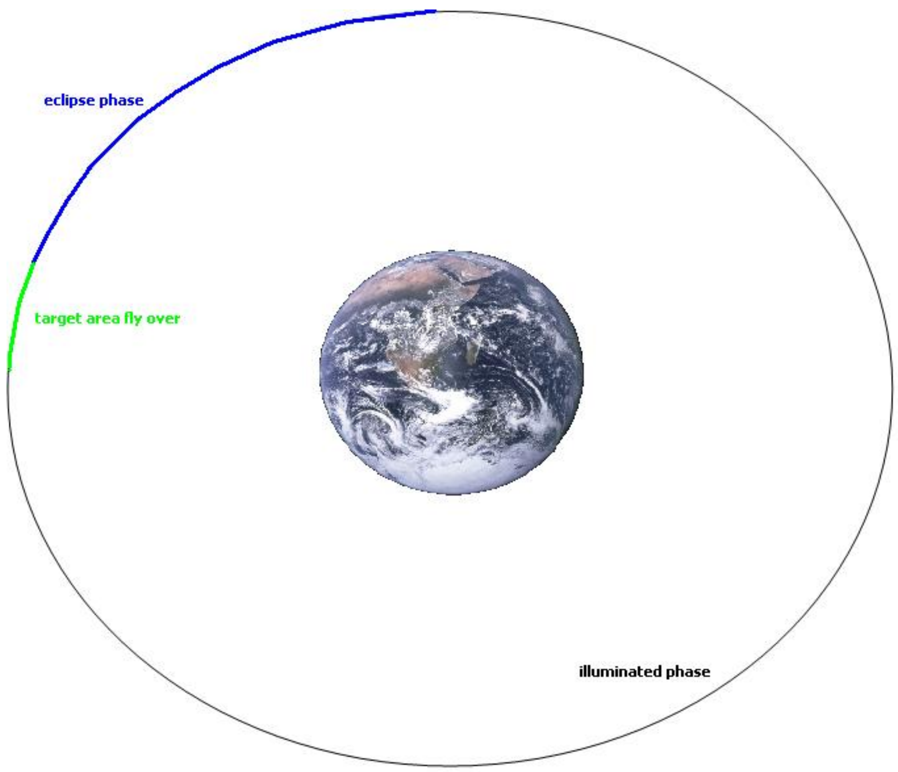


Figure 5 - 37 – Orbit phases for power subsystem sizing (figure not to scale)

In particular, all the following cases could happen:

Case	Conditions	Power providers	Power request
1	- Satellite in orbit illuminated phase; - Satellite flies-over the target area; - Target area in daylight.	- Solar array - Battery	- ADCS, Comms, OBDH - Optical payload
2	- Satellite in orbit illuminated phase; - Satellite flies-over the target area; - Target area in shadow.	- Solar array	- ADCS, Comms, OBDH - IR payload - Battery recharge
3	- Satellite in orbit eclipse phase; - Satellite flies-over the target area; - Target area in daylight.	- Battery	- ADCS, Comms, OBDH - Optical payload
4	- Satellite in orbit eclipse phase; - Satellite flies-over the target area; - Target area in shadow.	- Battery	- ADCS, Comms, OBDH - IR payload
5	- Satellite in orbit illuminated phase; - Satellite does not fly over the target area.	- Solar array	- ADCS, OBDH - Battery recharge
6	- Satellite in orbit eclipse phase; - Satellite does not fly over the target area;	- Battery	- ADCS, OBDH
7	- Target area flown-over during a sunlit orbit segment adjacent to the eclipse phase; - Target area in daylight. *In this case it is not possible to recharge the battery for a long period (eclipse plus adjacent target area fly-over).*	- Solar array (just for the sunlit orbit segment) - Battery	- ADCS, OBDH - Comms, Optical payload (just for the sunlit orbit segment)

Table 5 - 19 – Possible occurrences for battery design.

In this case study, ADCS and OBDH subsystems are supposed to work during the overall orbit; Communications subsystem is supposed to work only when the target area is flown-over, because the mobile ground station is located in the target area; the Payload works only during the time when the microsatellite flies upon the target area (the optical camera when the target area is in daylight, the IR camera when the target area is in shadow); the battery is recharged when the microsatellite is not in eclipse and the power demand does not require the battery to be used. Referring to the table in previously displayed, the worst case to consider in order to size the battery is “Case 3”: the battery has to provide the necessary power for optical camera and Comms during target area fly-over time and for OBDH and ADCS during eclipse time, without any help from solar array. In

“Case 7”, even if the duty time for the battery is larger, the power is provided not only by the battery, but also by solar array. The power request in “Case 3” is equal to $47.8 \cdot 0.56 + 203 \cdot 0.07 = 41.75 \quad W \cdot h$ and it is not so demanding.

In order to support this power demand, that is the design point for the battery, lithium-ion battery cells *MP176065* have been chosen, shown in the following figure.

Figure 5 - 38 – Lithium-ion battery cell MP176065

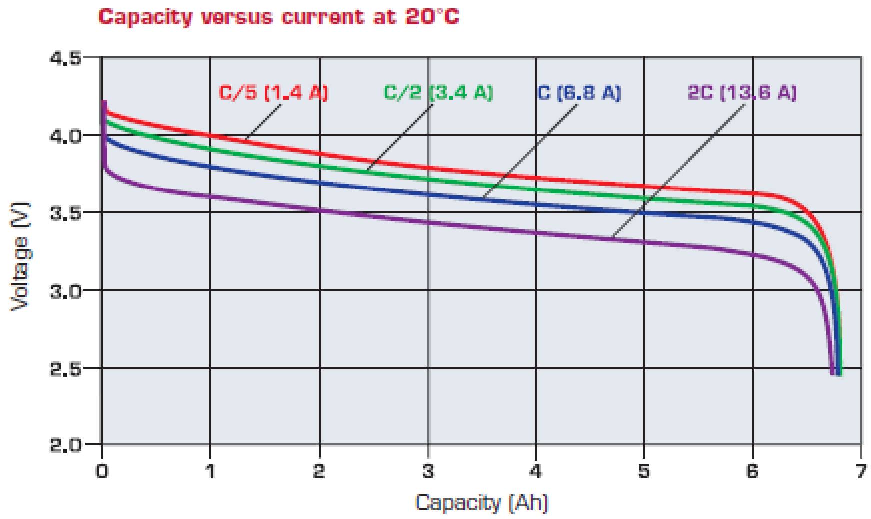
They have been chosen because of the very high energy density (165 Wh/kg) and the high operative temperature range. Typical performances are listed in the following table.

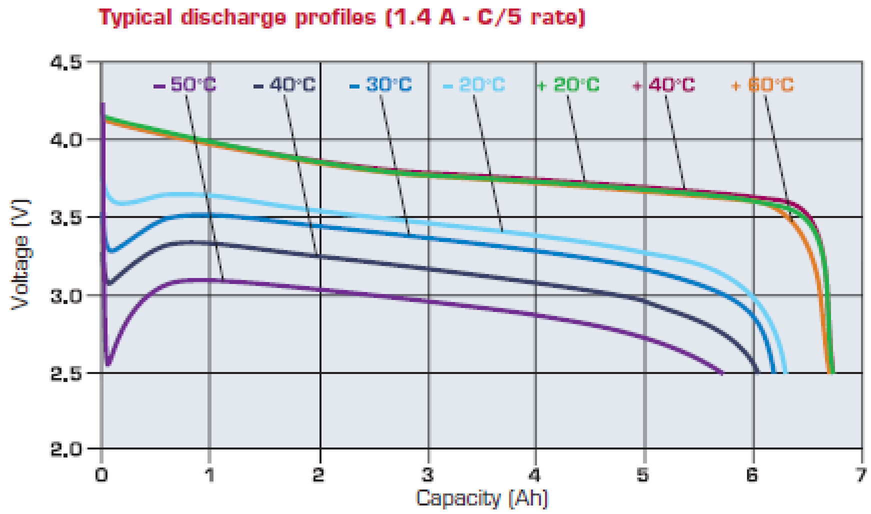
Electrical Characteristics	
Nominal Voltage (1.4A rate at 20°C)	3.75 V
Typical Capacity (at 1.4A 20°C 2.5V cut off)	6.8 Ah (when charged at 4.2 V) 6.1 Ah (when charged at 4.1 V)
Mechanical Characteristics (100% charged cells)	
Thickness max	19.6 mm
Width max	60 mm
Height max	65 mm
Typical weight	153 g
Lithium equivalent content	2.0 g
Volume	68 cm^3
Nominal energy	26 Wh
Operating Conditions	
Charge method	Constant Current / Constant Voltage
Charge voltage	4.20 ± 0.05 V
Maximum recommended charge current	6.8 A (C rate)
Charge temperature range	- 20°C to + 60°C

Time at 20°C	To be set as a function of the charge current: C rate - 2 to 3 h C/2 rate - 3 to 4 h C/5 rate - 6 to 7 h
Maximum continuous discharge current	14 A (~2C rate)
Pulse discharge current	Up to 27 A (~4C rate)
Discharge cut off voltage	2.5 V
Discharge temperature range	- 50°C to + 60°C

Table 5 - 20 – Battery cell typical performances

The battery needs to be composed of two cells connected in series in order to provide 41.75 Wh, since a cell typical capacity is 6 Ah. Two cells connected in series, with a nominal voltage of 3.75 V and a capacity of 6 Ah produce 45 Wh.





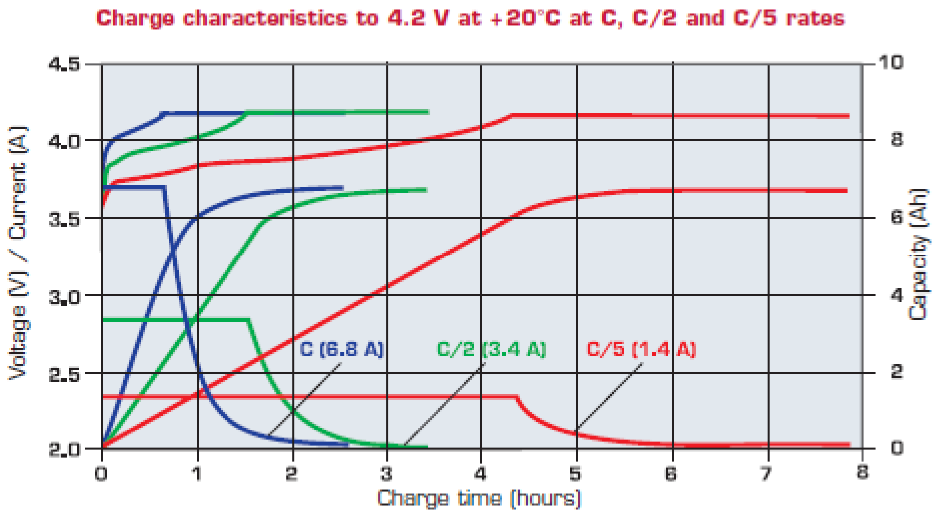


Figure 5 - 39 – Battery characteristics

Since lithium-ion cells have been chosen, it is very important to properly design a suitable Battery Charge Regulator (i.e. BCR). It is an important design phase, because a lithium-ion cell requires a charge profile CCCV (Constant Current Constant Voltage), that is a charge profile at constant current, followed by an other one at constant voltage, equal to the maximum voltage tolerated by the battery, to be performed with very strict margins in order to do not damage the cell. The end of the charge phase is detected by a current value smaller then a known threshold (i.e. 100 mA). Furthermore, the charging system has to allow the charge phase to be interrupted if the solar array does not produce the power required to feed the users, even if the battery is not charged yet.

5.1.7.4.2. Solar Array

The second step of the EPS design is to size the solar array to supply power to all subsystems and to recharge battery. The main objective when choosing the solar array is to attain the highest efficiency and the lowest density. The solar array is composed of photovoltaic solar cells, which convert solar radiation in electrical energy. In order to correctly face the problem, it is necessary that the solar array must be designed to provide the power required at the end of life (EOL). In fact the spatial environment is very stressful for the solar cells, which are exposed to radiations and to continuous changes in temperature. The result is a degradation of solar cells and a reduction of efficiency. In order to meet power requirements at EOL, solar array are oversized at beginning of life (BOL). The power that the solar array has to supply is shown in the following relation.

$$P_{SA} = \left(\frac{P_i \cdot t_i}{X_i} + \frac{P_e \cdot t_e}{X_e} \right) \cdot \frac{1}{t_i}$$

where P_i is the power required during illumination, P_e is power required during eclipse, t_i and t_e are respectively the length of illumination and eclipse periods per orbit. X_i and X_e represent respectively the efficiency of the path from the arrays directly to the loads and the path from the arrays through the battery to the loads. These values depend on the type of power regulation: Direct Energy Transfer (DET) or Peak Power Tracking (PPT). Usually, their values are respectively:

- DET: X_i=0.85 and X_e=0.65
- PPT: *Xi*=0.8 and *Xe*=0.6

The worst case to consider in order to size the solar array is "Case 7" of the previous table (that summarizing the operative condition. Referring to orbital illumination condition, the power required during illumination is split into "target area fly-over", where the solar array has to provide power for OBDH, ADCS, Optical Payload (peak power) and Comms; and illuminated phase, where the solar array has to provide power for OBDH, ADCS (during this period the solar array recharges also the battery). During the eclipse phase, the power demand is provided by OBDH and ADCS subsystems.

Considering that:

- $P_i \cdot t_i$ = (power required during target area fly-over)·(target area fly-over time) + (power required during simple illuminated phase)·(simple illuminated phase time) = 248W · 0.072h + 35W · 0.9h = 50Wh
- $P_e \cdot t_e$ = 35W · 0.57h = 19.8Wh

The power that the solar array must provide in the period t_i is given by:

$$P_{SA} = \left(\frac{50\text{Wh}}{0.8} + \frac{19.8\text{Wh}}{0.85} \right) \cdot \frac{1}{0.976\text{h}} = 88\text{W}$$

It means that during each orbit, the solar array must be able to produce a power equal to 88W. In order to determine the area of solar array, it is necessary to choose the solar cell technology. The power per unit area generated at Begin Of Life is:

$$P_{BOL} = c_0 \cdot \cos\beta \cdot \eta$$

where C_0 is solar intensity at 1 AU (1358W/m^2), η is solar cell efficiency and β is the angle between sun direction and the normal to surface of array. This angle varies continuously during the orbit, and it depends on the configuration of the solar array. In a preliminary way, a mean β value of 45° has been chosen. The solar cell efficiency η depends on the technology chosen for the cell. If *triple junction gallium arsenide on germanium* (*GaAs/Ge*) solar cells are chosen (next figure), it is possible to reach an efficiency equal to η = 28% at 25°C. The cell size is 40mm x 80mm and each cell has a voltage equal to 2.8 V.

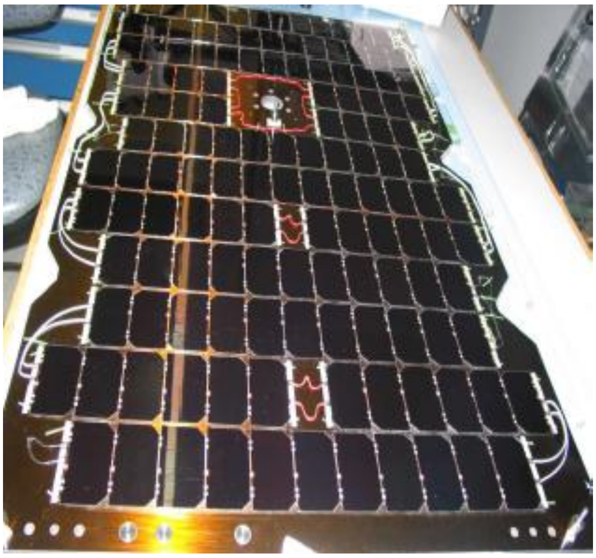


Figure 5 - 40 – Triple junction cells on carbon fibre (CFRP) panel. Courtesy of Surrey SSTL.

Therefore, P_{BOL} = 267 W/m^2. The power per unit area at EOL is given by:

$$P_{EOL} = P_{BOL} \cdot L_d \cdot F \cdot T_d$$

where L_d is the life degradation coefficient (typical value: L_d = 0.7), which takes into account the degradation due to spatial environment (radiation, plasma, solar wind); F is the assembly factor (typical value: F = 0.9), that takes into account the inter-cell spacing; T_d is the temperature degradation coefficient (typical value: T_d = 0.98), which takes into account the temperature cycles. Therefore, P_{EOL} = 165 W/m^2.

The area of the solar array necessary to support the microsatellite's power requirements P_{SA}, is:

$$A_{SA} = \frac{P_{SA}}{P_{EOL}} = \frac{88\,\mathrm{W}}{165\,\mathrm{W/m^2}} = 0.53\ \mathrm{m}^2$$

Since the density of a *triple junction GaAs/Ge* solar cell is equal to 0.85kg/m^2 [4], the weight of the solar array is 450g.

Now, the number of solar cells necessary to generate the power P_{SA} is calculated:

$$N_{cell} = \frac{A_{SA} \cdot F}{A_{cell}} = \frac{0.53 \cdot 0.9}{0.04 \cdot 0.08} = 150\ cells$$

In order to obtain the required voltage and current, the cells must be connected in series and in parallel. The number of series cells is obtained dividing the maximum bus voltage by voltage of a single cell, while the number of string connected in parallel is obtained dividing the total number of cell by the number of series. Considering a bus voltage equal to 28V (because most of the components work at this voltage, so it is necessary to put a transformer before each component that does not work at this voltage), and a cell voltage equal to 2.8V, the number of cells connected in series and in parallel is:

$$N_{series} = \frac{V_{bus\max}}{V_{cell}} = \frac{28}{2.8} = 10$$

$$N_{par} = \frac{N_{cell}}{N_{series}} = \frac{150}{10} = 15$$

The next step is to determine the dimension of solar array. Considering 10 cm for margin and 1 mm for inter-cell spacing, the width of the solar array is:

$$Array_{length} = N_{series} \cdot (cell_{length} + 0.1) + 10 = 91\, cm$$

The length is calculated at same way, considering 6 cm for margin and 1 mm for inter-cell spacing:

$$Array_{width} = N_{par} \cdot (cell_{width} + 0.1) + 6 = 67.5\, cm$$

Considering these solar array dimensions, it is possible to think a body-mounted solar panel, without the necessity to design deployable panels, which would increase design and manufacturing complexity and costs, and would reduce reliability and introduce mechanical errors.

5.1.7.5. Thermal Subsystem Sizing

The Thermal subsystem will monitor and regulate the temperature on each payload and subsystem. Temperature regulation is vital for the success of the mission to ensure that all subsystems and payloads stay operational. It is composed by temperature sensors and heaters.

5.1.7.5.1. Temperature sensors

The temperature of each payload elements and subsystem device will be monitored through temperature sensors placed on every board and payload. Operating temperature requirements are listed for each subsystem. The most stringent temperature requirement is -10°C to +50°C. The required accuracy in temperature estimation is fixed in 1°C.

We will use Honeywell's HEL-700 Platinum RTD to provide linearity accuracy, stability and interchangeability. Resistance on HEL-700 changes linearly with temperature. The HEL-700 has a large temperature sensing range of -200 to 540 ºC.

Figure 5 - 41 – HEL-700 Platinum RTD temperature sensor

Temperature Sensors Specifications	
Sensor Type	100 Ohm Platinum RTD
Temperature Range	-200°C to 540°C
Temperature Coefficient	0.00375 Ohm/Ohm/°C
Packaging Type	Radial chip, SMT axial flip chip
Self Heating	> 0.3 mW/°C

Table 5 - 21 – Temperature sensor specifications

5.1.7.5.2. Heaters

Thermofoil™ heaters by Minco will be controlled by the OBDH systems to ensure that none of the subsystems or payloads freezes while the Earth eclipses the spacecraft during its orbit. Such heaters can safely run at wattages twice those of their wire-wound equivalents. Thermofoil™ heaters transfer heat more efficiently, over a larger surface area, than round wires.

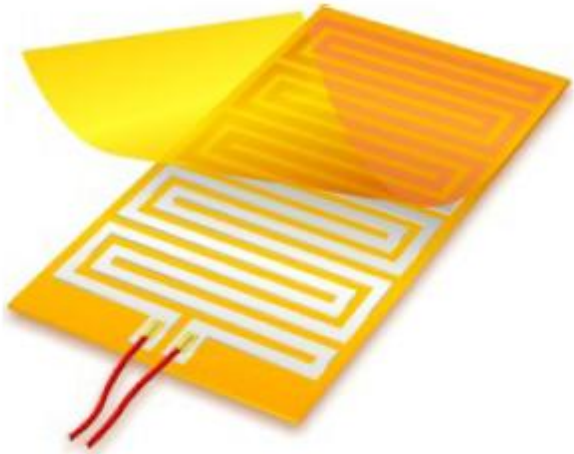


Figure 5 - 42 – Thermofoil™ heaters by Minco

Thermofoil™ Specifications	
Material	Kaplan/FEP
Temperature Range	-200 to 200°C
Max Resistance Density	8-70 Ohms/cm^2
Thickness	0.25 mm
Mass	0.04 g/cm^2

Table 5 - 22 – Thermofoil™ heater specifications

The heat dissipation on the microsatellite is performed by placing radiator panels, conductive strapping and shear panels in the spacecraft; the two latter devices will transfer heat from the subsystems to the radiator panels where it will then be released into space.

5.1.7.6. Mass & Power Budget

A mass and power budget is provided in this paragraph, basing on the results illustrated in previous paragraphs.

Mass budget	
Optical Payload	80 kg
Communications	2.5 kg
OBDH	2.7 kg
ADCS	7.2 kg
Power	300 g (battery) + 450 g (solar array)

Thermal	2.8 kg (2% of total weight)
Structure & Mechanisms	21 kg (15% of total weight)
Margin	21 kg (15% of total weight)
Total	**138 kg**

Table 5 - 23 – Mass budget

Power budget		
Power absorbed		
Subsystem	**Average Power Required**	**Peak Power Required**
Optical Payload	39.2 W	167 W (max duty cycle 5%)
IR Payload	5 W	5 W (max duty cycle 5%)
Communications	3.6 W	36 W (max duty cycle 5%)
OBDH	10 W	20 W
ADCS	23.3 W	27.8 W
Power sources		
The power subsystem is designed in order to fulfil the power demands. The power is usually provided only by battery during eclipse phase; by both battery and solar array during target area fly-over; and only by solar array during the illuminated phase if the target area is not flown-over. The detailed utilization of power sources is described in a previously depicted table		

Table 5 - 24 – Power budget

5.1.7.7. Configuration Analysis

The following two figures show the configuration of microsatellite. The colours correspond to the subsystems as indicated in the legend below

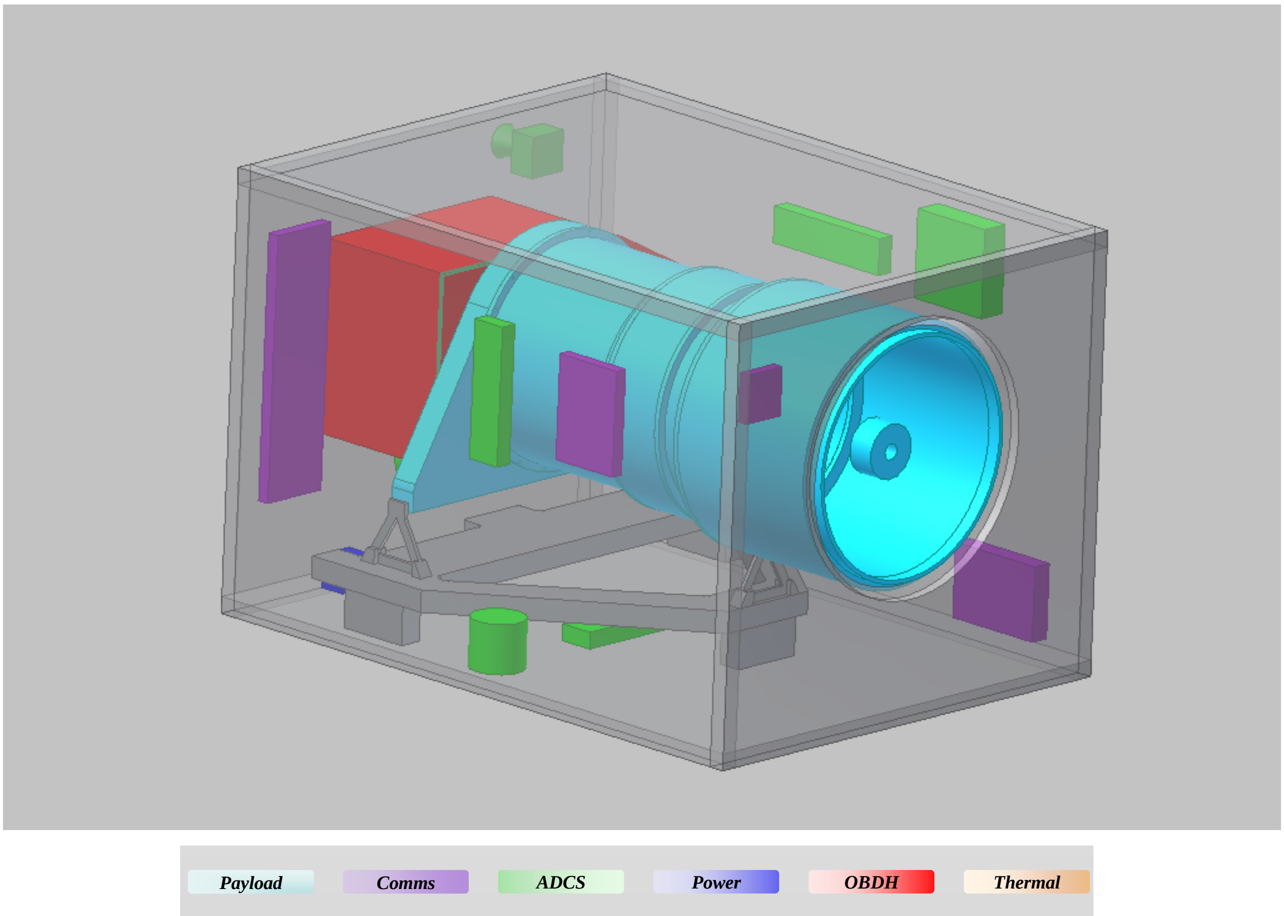


Figure 5 - 43 – Microsatellite configuration

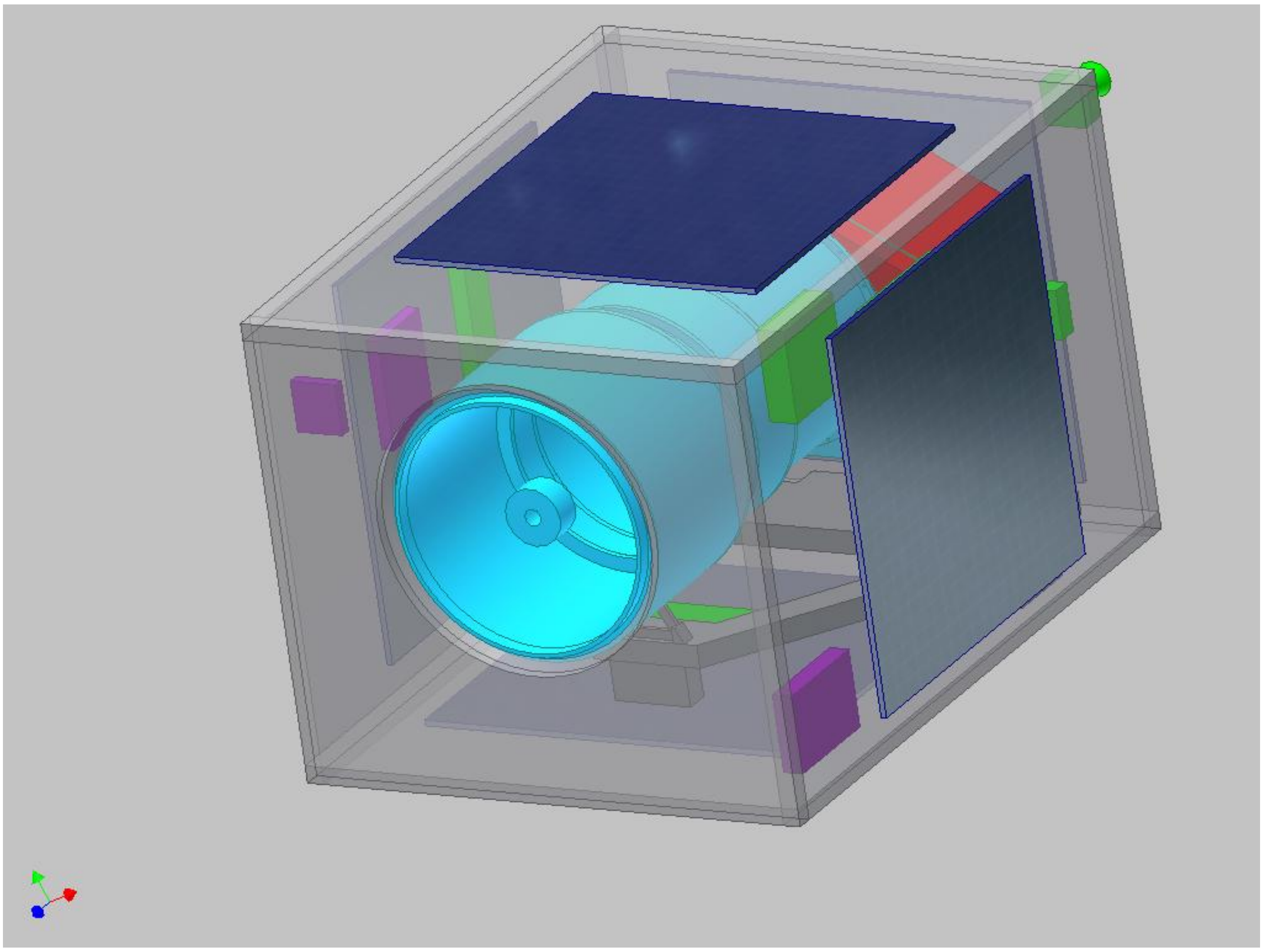


Figure 5 - 44 – Microsatellite configuration

The free spaces are necessary to house the cables. All the dimensions of each subsystem come from the detailed results of previous paragraphs. The sizes of the microsatellite are: **120 x 90 x 80 cm**, corresponding to a volume equal to **864 dm^3**. These volume and sizes of the microsatellite are small enough to be compatible with the volume capability of common launchers.

5.1.8. Ground Architecture

The constellation of microsatellite is designed to capture high resolution images of the target area, and to direct deliver data to local and authority users via the use of a fully mobile ground station, located in the target area. QinetiQ, one of the world's leading defence technology and security companies, has already built and provided a mobile ground station for TopSat mission[5], as shown in two following figure. QinetiQ developed a fully mobile ground station based on the real time acquisition and processing-integrated data system (RAPIDS). The system consists of fully transportable 2.7m trailer mounted antenna (hydraulically pointed) together with PC based tracking, control and image processing equipment mounted in a customised land rover vehicle. Data can be processed in near real time to provide rapid image products to local users.

The advantages of such a kind of architecture consist of:
- possibility of in-situ tasking and downlinking;
- only 2 hours set up time;

– data available on ground within 2 minutes of imaging, directly to users and, eventually, to "*in-situ*" operator;
– local image processing.

Figure 5 - 45– Mobile Ground Station (courtesy of QinetiQ)

The microsatellite can acquire an image and downloads it in near real-time (after few seconds) both to a mobile ground station located in-situ (if needed), and to traditional Ground Station, according to the necessity dictated by the needs. The telemetry and telecommand of the microsatellite is managed through a different TT&C ground station. Both "User Ground Segment" (UGS) with mobile ground station, and TT&C ground station exchange telecommand (TC), telemetry (TM) and Image data with the "Satellite Ground Control Segment" (SGCS), which has a "Mission Control Center" and a "Satellite Control Center", and could be co-located with the TT&C ground station.

5.2. Case Study 2: Space-Based AIS

5.2.1. The Automatic Identification System: Description

The Automatic Identification System (AIS) is a system developed by the International Maritime Organization (IMO) and the International Telecommunications Union (ITU): AIS is a signaling system among ships, and ships and coastal stations for traffic control that aims at the safety of maritime traffic and the control of vessels routes. Such a system is used to automatically identify and locate vessels: the information exchanged among two or more AIS devices can help in several critical conditions, like providing crucial information in situational awareness, or assisting during the maneuvers of collision avoidance; such system is also used for normal navigation operation, making available, for example, location of buoys and lights. Initially designed as anti-collision system, land-based AIS became largely widespread also for maritime traffic monitoring and guidance, which requires ships to be equipped with proper AIS transceiver, that transmit relevant data, while receiving information from surrounding devices.

The International Maritime Organization's International Convention for the Safety of Life at Sea requires AIS to be fitted aboard international voyaging ships with gross tonnage (GT) of 300 or more tons, and all passenger ships regardless of size. In 2010, the most commercial vessels operating on the EU inland waterways were mandated to fit an approved AIS device. The entire EU fishing fleet over 15 meters was given until 2014 to do the same.

Automatic Identification System signals are centered over two VHF channels (161.975 and 162.025 MHz), using two 25 kHz band; the signals transmitted are modulated according to the Gaussian Minimum Shift Keying (GMSK) and make use of the Self-Organized Time Division Multiple Access (SOTDMA) protocol. This protocol partitions every “frame”, equal to one minute, into multiple time slots, in which information are sent. Every one-minute frame are divided into 2250 time slots for each AIS channel, which means that up to 4500 different time slots can be used to send information. The baud rate is 9600 bps, which means that 256 bits are transmitted in the 26.7 seconds length of any time slot. The next figure shows a schematic of the how the frame is organized.

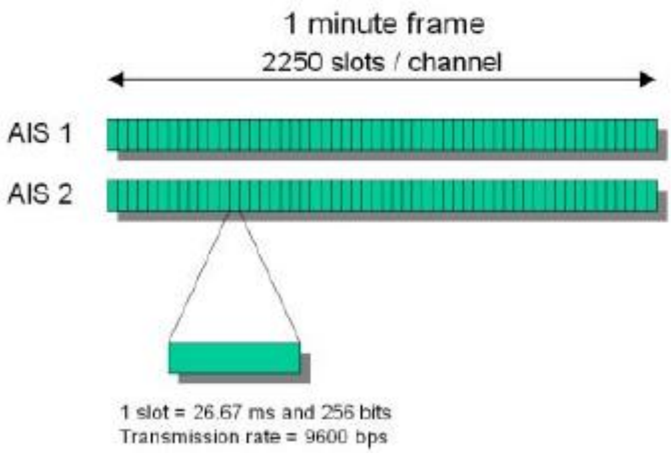


Figure 5 - 46 – AIS communication technique

In the next figure, instead, is depicted a representation of the how an AIS signal is composed.

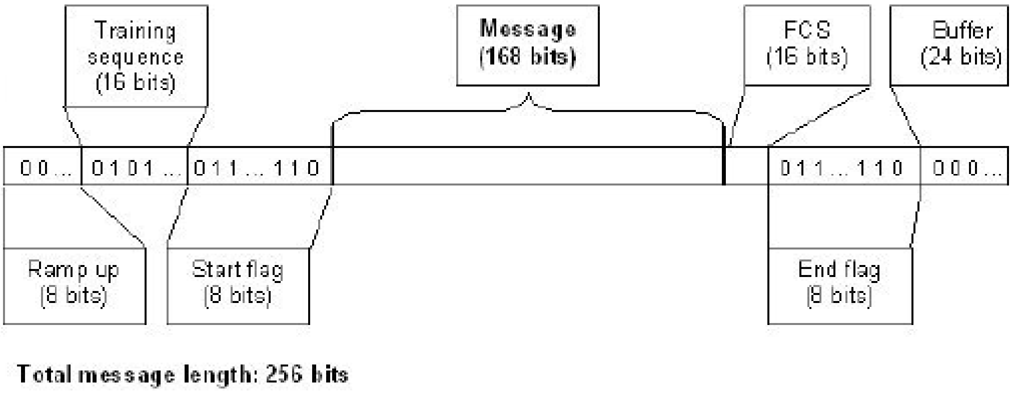


Figure 5 - 47 – AIS Signal

An AIS transceiver integrates a standardized VHF transceiver with a positioning system, with other electronic navigation sensors, such as a gyrocompass or rate of turn indicator. The information that typically are transmitted on AIS channels includes ship's information, like its identification, latitude, longitude, speed, heading, rate of turn etc, but also other kind of data, such as destination, or information related to crew or cargo; AIS standard calls up to 26 different AIS message types [6] that contain varied information.

Ships (and shore-stations) alternate the transmissions on the two channels, and the SOTDMA protocol guarantees that no superimposition between two or more signals occurs, automatically allocating the time slots when transmitting. A position report from one AIS station fits into one of 2250 time slots established every 60 seconds. AIS stations continuously synchronize themselves to each other, to avoid overlap of slot transmissions. When a station changes its slot assignment, it pre-announces the new location. The next figure briefly shows how this automatic system works

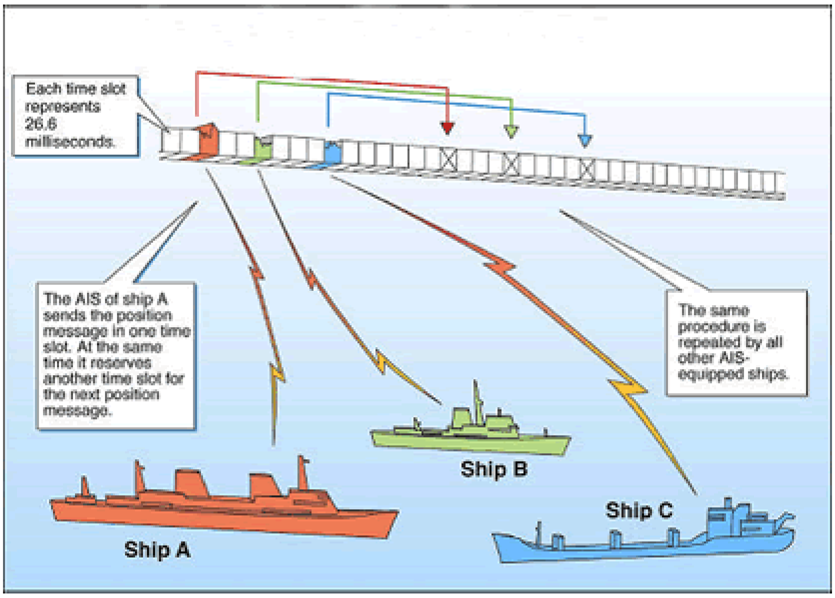


Figure 5 - 48 – Illustration of how SOTDMA protocol works

In this way, new stations, including those stations which suddenly come within radio range close to other vessels, will always be received by those vessels. In theory, a large number of ships can be accommodated using this scheme without congestion. The system is also scalable: in the event of system overload, targets further away will be subject to drop-out, in order to give priority to closer targets, that are a primary concern to ship operators. In practice, the capacity of the system is nearly unlimited, allowing for a great number of ships to be accommodated at the same time.

The following parameters summarize/characterize AIS signals as transmitted and received on the surface of the Earth:

- Centre Frequency: 161.975 MHz (AIS channel 1), 162.025 MHz (AIS channel 2)
- Bandwidth: 25 kHz per channel
- Power: 12.5 Watts (Class A), 2.5 Watts (Class B)
- Transmission Range: 50 nautical miles radius (i.e. to the horizon)
- Modulation: Frequency modulation, GMSK at 9.6 kbps
- TDMA Frame and Slot: 1 minute frame timing extracted from GPS signals, 26.7 ms slot (256 bits)
- Typical Antenna Mounting: Vertically polarized monopole or dipole antenna mounted at some height above the ship's deck, typically to a mast.
- Propagating Energy of Transmission: Primarily line of sight and along the surface of the Earth (i.e. perpendicular to the antenna).
- Carrier to Co-channel Interference Ratio Threshold C/I : 10 dB
- Typical Maritime AIS Receiver Sensitivity: -108 dBm for 20% packet error rate

As being a system designed for terrestrial use, the actual AIS is limited to identify vessels and organize the transmission into a range of a value that is highly variable, but that typically do not exceed the 50 nautical miles: this value represent the maximum separation between two ships in the open sea for which there's still visibility between the AIS transceiver, that are typically mounted on top of a mast. Such distance characterizes the size of a "cell" of ships that are self- organized, i.e. the area in which transmissions among ships can occur and are organized according to the SOTDMA scheme. In the next figure the global ship density map of all class A vessels is displayed. The map clearly depict the high vessel density areas around the world.

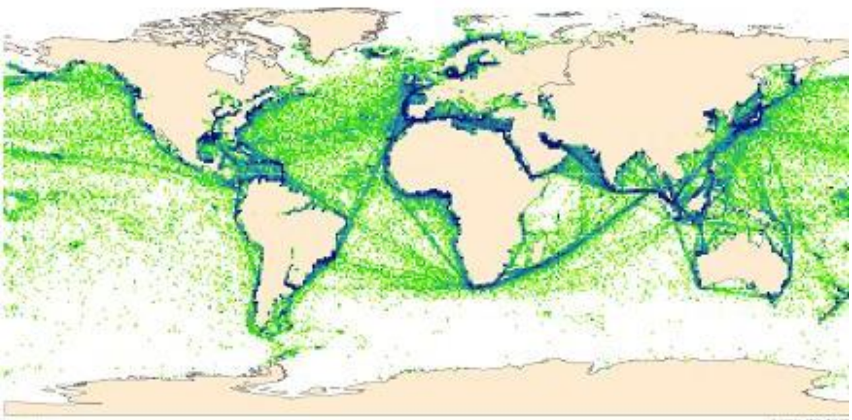


Figure 5 - 49 – Illustration of the class A vessels

5.2.2. Improving the Automatic Identification System service: Space Based AIS

As many entities rely on data coming from AIS, and being more and more urgent the need of managing a secure maritime traffic, exploiting AIS information not only in the coastal and inland waters, but also in the open seas, the possibility of implementing a worldwide, space-based AIS system has been investigated.

The personal contribution is represented, firstly, by a deep investigation of the problematic related to the transposition of such a terrestrial application into an "Earth-Space dimension"; then the study proceeded with the design of a microsatellite capable of performing the operations required to a space-based AIS-station, exploiting components compatible with the characteristic of the existing Automatic Identification System; ultimately, the study of a constellation of space platform capable of providing a real service, accomplishing the mission needs, and complying with requirements set by ITU and maritime institutions.

5.2.2.1. Space-Based AIS: transition from Earth to Space

Such mission is a worthwhile example of how can be profitable to exploit the new concept for micro-satellite: the mission, in fact, reveals itself to be challenging from a technological point of view: many past studies [7,8] demonstrated how many technical difficulties must be faced to set up a space-based AIS system: such difficulties mainly come from the fact that the system has been designed for terrestrial use, and a series of technical decision taken during the system development, made for augmenting the terrestrial-based AIS performances, reveals to be extremely self-defeating for a space-based system.

A bright example consists in the principal direction in which AIS mast-mounted antennas propagate energy: being designed for terrestrial use, the antenna is mounted with the dipole axis pointing the local zenith, so that the most of the available energy will be radiated along the surface of the Earth, as shown in the next figure.

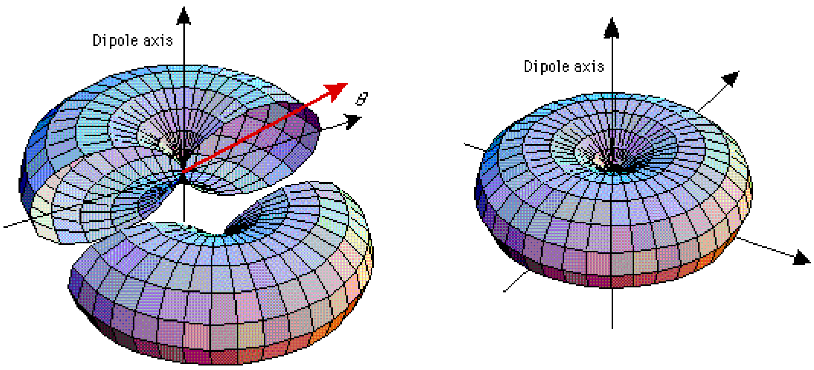


Figure 5 - 50 – Radiation pattern of a dipole antenna

Such decision, the most logic to undertake when a terrestrial-based system is being studied, implies that little energy are radiated (and thus wasted) towards the sky. But this decision also implies that a space-based receiver needs to be extremely sensitive and performing to be able to collect such signals.

But, apart from technical and technological issues, the necessity of a global coverage recommends that a valuable service could be only provided if a dedicated constellation of satellites is deployed. These two crucial aspects suggest that a high performing, innovative microsatellite can be the best solution for carrying out this mission, providing the necessary level of service through implementation of current technologies.

For making the space-based AIS service available to the authorities, and to make it compatible with the traditional, terrestrial-based service, has been decided the use of equipment compatible with those on board the ships (at least, in some cases, could be used exactly the same); such decision entails, hence, that no impact occurs on the vessels side, making this system completely "passive" (in the sense that no actions need to be undertaken on vessels and ships to have this service active) and immediately available.

5.2.3. Space Based AIS: Mission Requirements

Mission and users requirements are derived from needs of Governmental maritime organizations and commercial owners, which specify the desired service that should be provided by an advanced Space-AIS system. The service to provide should be worldwide, as institution are interested in monitoring entirely the maritime traffic, and, even if continuous coverage is not necessary, the time update interval should be in order of magnitude of hours; this two requirements impose that the service must be provided by a constellation of satellites in order to assure the global coverage required, while satisfying the temporal constraint imposed. As we said, continuous coverage is not requested: the ships' velocity is relatively low, and the wide distances between the ships in open seas routes do not require rapid traffic data refreshments.

Traffic information could be updated at fixed intervals, and not constantly: one of the relevant parameter for the constellation design is the time update interval, i.e. the time elapsing between two successive update of the ship information available to the ground users. The time update interval is calculated considering:

- the maximum time necessary for detecting a ship in a certain area;
- the time necessary to download such data to the first available ground station;
- the time necessary to (eventually) process such data in order to make data available to users.

As the ships have a low velocity, a time update interval of 3 hours has been identified as profitable for the application in exam, while a performance of 1 hour should be desirable, if economically affordable in terms of satellite to launch and operate. The second parameters that must be accounted for is the data latency, i.e. the time passing from the ship detection by the AIS receiver and the availability on ground of this data. Since the users would like to have information as updated as possible, the maximum period between detection and data availability has been set in 1 hour, with a desirable performance of 30 minutes. Such constraints directly impact the constellation design, as well as the constellation geometry and ground stations placements are impacted by the restrictions set by data latency. In order to have a data latency of less than 1 hour, any satellite has to pass in proximity of ground station any orbit.

The contacts with the Ground Stations are calculated under the assumption of a minimum elevation angle of 5°, a typical value for such situation, while the minimum angle for ship detection has been set to only 2°, due to the fact that in the open seas there are no obstacles that obstruct the line of sight, and due to the characteristic of the AIS transmitting antennas.

System Parameters	**Required Performance**	**Desirable**
Time update interval	3 hours	1 hour
Timeliness	1 hour	30 minutes

Table 5 - 25 – System Performances

5.2.3.1. Space Based AIS: Payload Design

The payload is the device that usually drive the design of a space platform: in fact, being the spacecraft only the module that provide the payload adequate service and survival in space, the payload is the reason for implementing a space mission, driving the design, setting the requirements for power demand, pointing accuracy and stability (if any), volume needed and so on. So, the payload will be the central device of the satellite. Being the system conceived for terrestrial application, all the aspect related to such a service (as already underlined also in the previous pages) will be terrestrial-based.

The choice of the payload will be done considering those commercially available, looking for the most suitable one, and evaluating its performance from a space-based system. From the commercially available device, the most sensitive has been chosen: in fact, the main difference between the land- and the space- based systems, is the incredibly higher distance that the AIS signals sent from a transceiver mounted on a ship have to cover before getting captured from a receiver orbiting in space. The selection of the payload has been done looking for the most sensible device: a sensitivity of -119 dBm has been reached, with a PER (Packet Error Rate) of 20% [9], which is the upper value of “tolerable” error rate.

Figure 5 - 51 – Terrestrial AIS component

	Radar Plus SM1610-2
Width:	8 cm
Length:	17.5 cm
Height:	5.7 cm
Weight:	900 g
AIS Data Rate:	9,600 bits/s
Sensitivity:	≤20% PER @ –119 dBm

Table 5 - 26 – Terrestrial AIS component performances

Figure 5 - 52 – AIS Antenna

Such device necessitates an antenna, and a proper antenna has been identified, checking the compliance with requirements and constraints of the AIS receiver.

Once identified the payload and the requirements settled over the mission, the constellation design started: keeping the spirit of the microsatellite revolution, in particular the rule of “single purpose” platform, no additional payload has been added to the platform. No additional system that could improve the satellite mass has been taken in consideration. The platform design has been performed trying to keep the mass of the microsatellite as small as possible, in order to save money at launch, to enable multiple launches (if orbit constraints would have allowed such possibilities). Only the basic function has been considered, to avoid increasing in mass not functioning to the primary objective.

Once the payload has been identified, the consequence of implementing such a system in a space-based platform has been investigated. Two major issues have been identified:

- Receiver Saturation
- Message collision

5.2.3.1.1. Receiver Saturation

The first concern for a space based AIS system so conceived is mainly due to the large viewing angle of the antenna from space, which, in turns, leads to an enormous "ground track": it could happen, in fact, that too many vessels are in visibility of the satellite, any of them sending a message in the same frequency, at the same time, leading to a satellite receiver saturation, and a consequent complete blackout, and inability to collect any other message. The next figure compares the dimensions of a single SOTDMA cell, with is a circular area with a radius of about 25 nautical miles, with the footprint of the AIS antenna mounted on board satellite. Today, institutes like US Coast Guard and ESA estimate a probability of 50% of detect additional signal when roughly 1500 - 2000 vessels (and the relative AIS signal) are detected[8].

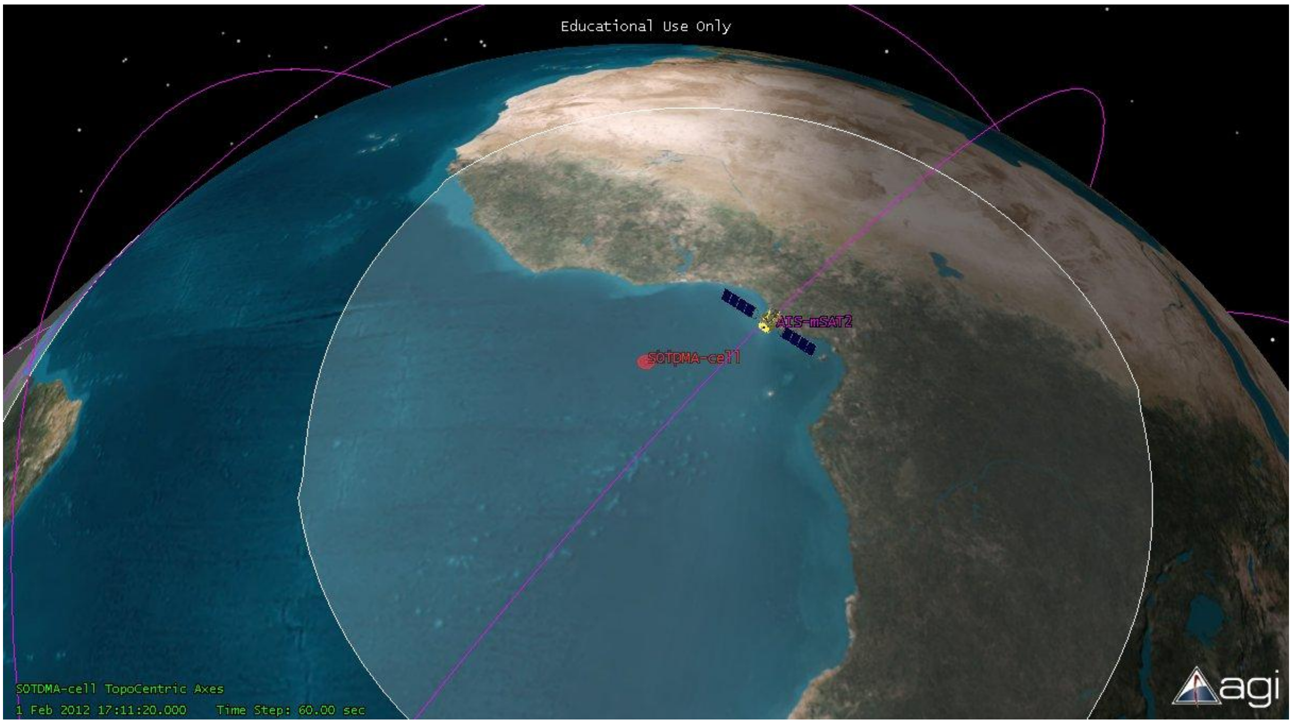


Figure 5 - 53 – Comparison between antenna FOV and dimension of a single SOTDMA cell

5.2.3.1.2. Message collision

Another problem affecting such a system is that related to message collision: as the AIS system has been designed for terrestrial use, also the use of SOTDMA protocol has been designed for such a purpose. This protocol ensures that AIS transmitters are organized into cells, in which is dynamically assigned, to every transmitter, a time slot for message transmission: the message collision occurs when operating a space-based AIS platform for message reception, which can collect data from many different cells, how illustrated in the following figure.

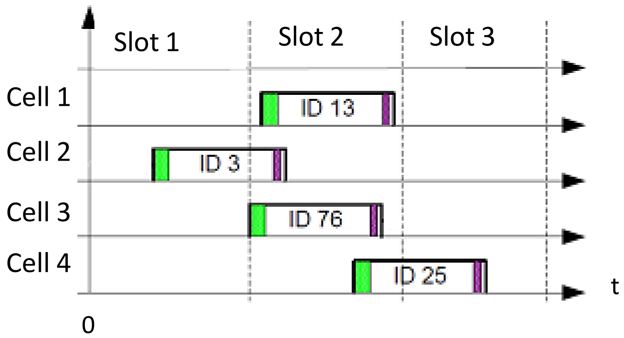


Figure 5 - 54 – Messages collision

This phenomenon is caused by two types of collisions:

- Type I: AIS transmitters in different SOTDMA cells transmitting in the same time slot
- Type II: AIS transmitters emitting in different time slots, which overlap due to distance caused time delays

Such a problem affects the probability that a satellite can detect ships, and such probability decrease as the number of ships in the field of view of AIS receiver increases. This is obvious, as increasing the number of ships increase the number of signals that the satellite receive at the same time, increasing the difficulty in detecting each of them.

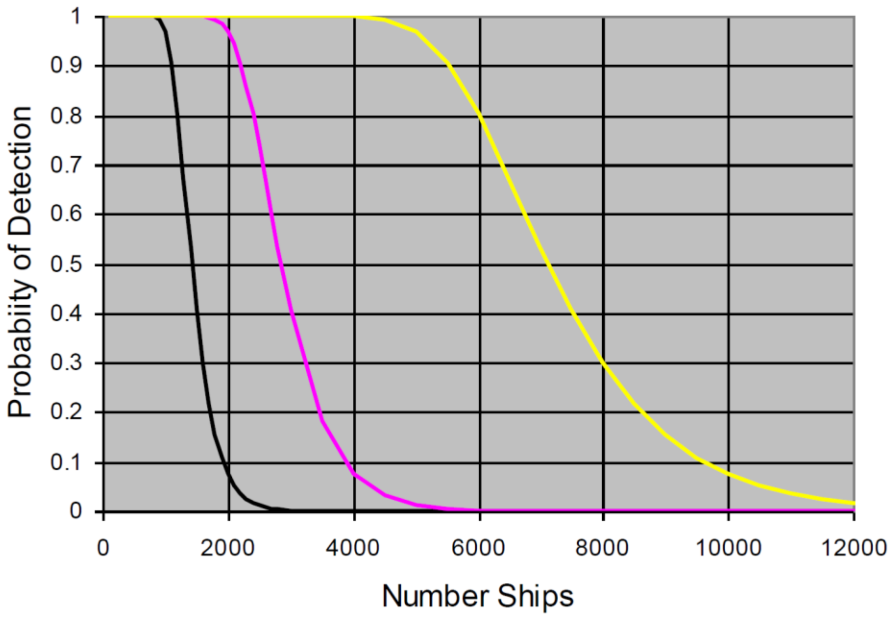


Figure 5 - 55 – Probability of message detection as a function of the number of ship sending AIS signals

In the so-called high traffic zones (HTZ), the zones where the maritime traffic is more pronounced, these problems can have severe consequences, because the antenna can be illuminated by up to 20,000 different signals coming from as much vessels. However, the problem of message collision can be mitigated by applying innovative technologies[11].

Anyway, it is possible to exploit several factors that can lead to an improved rate of detection:

1. Entering a High Traffic Zones the receiver will not be immediately saturated, so it is able to detect a large number of signal before being saturated;
2. All vessels send messages in vertical polarization. Due to the so-called Faraday rotation, polarization received at satellite level, after the signals flied several hundred kilometers, may be vertical, horizontal or even transversal. Such diversity can be used, with proper receiving architecture, to improve ability of decode messages.
3. The vessels transmit message every 5 seconds; the satellite is in visibility of any vessel for about 10 minutes; this means that the receiver has 120 "attempts" to collect the signal of any ship;
4. The use of more directive antenna can help reducing the antenna viewing angle, reducing the number of vessels which signals illuminate the antenna.

The listed factors can be implemented by use of the following technological solutions.

Antenna

The AIS antenna is necessary in every spacecraft in AIS constellations. The choice of antenna technology can lead to positive effect on the probability of detecting an AIS message in a signal band with interference. This is accomplished by exploiting the following phenomena, which are related to the choice of antenna technology:

- Polarization diversity
- Frequency diversity (i.e. Doppler shift)
- Power diversity
- Geometric diversity (radiation pattern)

For the choice of antenna, various technology options are available. These include:

- Monopole /dipole
- Helix
- Patch

The monopole /dipole antenna and the patch antenna offer linear polarization, and are therefore able to introduce polarization diversity. The helix antenna suffers from a 3 dB gain loss due to polarization mismatch in addition to losing the benefit of polarization discrimination. Nevertheless, a helix offers a steep gain slope in its antenna pattern, which improves the power diversity in the received signals.

Receiver Architecture

The AIS receiver is responsible for receiving and demodulating the AIS messages. Therefore, the use of a Software Defined Radio (as defined in the third paragraph) can allow the implementation of powerful signal detection algorithms. Two kind of receiver architecture can be used for the overall demodulation of AIS messages:

- On-board processing
- Digital bent-pipe / digital sampling

In the first architecture, the algorithms for decoding the AIS signal are performed on board the satellite, and the decoded data downloaded, while the second envisage the download of the original data (using the satellite as a store-and-forward platform) and executing on ground the algorithms. The former solution requires a large on board processing capacity, while the latter needs large download capacity to transfer on ground the large amount of non-decoded data.

Digital Signal Processing

Digital signal processing algorithms have the task to separate the multiple AIS messages in a particular time slot, so that they can be successfully decoded. These include:

- Core GMSK demodulation algorithm
- Multi-user detection algorithms
- Multi-receiver combination algorithms

A core GMSK demodulation algorithm is used for demodulation of the AIS GMSK modulated message. Coherent demodulation provides the best performance, but is most complex. Slightly less performance is gained with a non-coherent demodulation algorithm, which is simpler though. A multi-user detection algorithm is used to separate multiple AIS messages occupying a single slot. This will be done in an iterative way by subtracting previously demodulated messages from the original noisy signal containing multiple AIS messages. Although these algorithms are capable of decoding a low amount of messages (between 2 and 5), their performance quickly diminishes when large amount of messages are present in a single slot. Performance will significantly increase by introducing a large amount of discrimination between the signals.

A multi-receiver algorithm can be used when multiple receivers are flown on a single satellite. This algorithm combines the received AIS signals from all receivers in an attempt to increase the probability to demodulate individual AIS messages.

5.2.3.2. Space Based AIS: Constellation Design

The constellation design started from the user requirements, bearing in mind payload performances. The constellation characteristics, in terms of orbits and number of satellites, will be presented: the requirements drive the definition both of orbital altitude, defined in close correlation with payload performances, and of a range of orbital inclinations. Issues related to cost suggest that unique spacecraft design and multiple launches could be the best solution: the number of launches will drive the design of the constellation more than the number of satellite will do. Hence less orbits with more satellites are preferable than more orbits with just one (or few) satellite(s) in.

The orbits have been chosen circular, in order to provide homogeneous, quite global coverage and service. The satellites are considered operative 24/7, with an Earth pointing attitude.

The Payload performance drive the orbit altitude selection, identified at 600 km taking into account several issues:

- the strength of the signal upcoming from the ships;
- the FOV of the payload from orbital altitude: the wider the FOV, the smaller the number of satellites for covering the Earth in the same period, which lead to a significant benefits in terms of system cost;
- the possible interferences in the signal, and the number of ships in the Field Of View (FOV) to be detected;

- the orbit maintenance, which requires a lower ΔV for higher altitude, while ΔV required for EOL de-orbiting increase with altitude.

When evaluating the payload performances and the access duration, it is worth noticing that the satellite visibility from ships is considered almost equal to the geometrical visibility (the elevation angle used was 2 degrees), due to the complete absence of obstacles when in open seas, and also due to the ships antennas characteristics and beams.

The table summarizes the constraints on the orbit characteristics as derived from the performance preliminary analysis:

Mission Requirements	Global Service
	Continuous service
	Max reporting time < 3hr
System Requirements	> 4 satellites
	Circular orbits
	Nadir Pointing

Table 5 - 27 – Mission and System Requirements Table

Bearing in mind that a larger number of satellites and orbital planes increase the cost of the mission, the constellation has been designed seeking at the smaller possible numbers of satellites: in any approached case, the constellations the upper limit is 16. Several constellation patterns are analyzed, aiming at global, non-continuous coverage, according to mission requirements settled.

Configurations traditionally used in case global coverage must be achieved, like Walker Delta Pattern and Streets Of Coverage (SOC), were investigated. As global coverage is required, orbital inclination *i* that assure such conditions are considered: according to the geometry shown in the next figure, values comprised between 90°-θ and 90°+θ allow covering also the Polar Regions, where θ is the geocentric angle of the satellite sensor visibility.

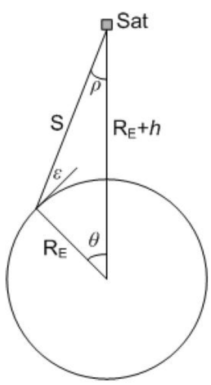


Figure 5 - 56 – Geometrical configuration

where:

- Re, Earth equatorial radius;
- h, mean satellite altitude;
- S, maximum slant range;

- θ, geocentric angle of the antenna maximum visibility semi-arc on the Earth surface;
- ε, minimum visibility angle, related to the antenna characteristics and mission objectives; it represents the satellite minimum elevation angle to the local horizon;
- ρ, Field Of View, or scan angle.

Considering the orbital height of 600 km, the minimum value of inclination that allowed for global coverage is slightly less than 67.5°. Orbital plane inclined of 67.5°, 90° and 97° (Sun synchronous orbits) are considered, coupled with the abovementioned 600 km of orbital height. The choice of involving Sun Synchronous Orbit (SSO) in the study is due to the interesting advantage that such option would noticeably simplify the satellite architecture: in fact, SSO guarantees constant illumination condition during the satellite lifetime, which would lead to simpler Power and Thermal subsystems. Another important advantage of such orbit (but also of other highly inclined orbits) is the constant (one per orbit) passing over the Poles: establishing polar ground stations, twice per orbit (at the North and South Poles), will assure the shorter data latency time along the entire mission. Differently, for constellation characterized by lower inclination, a different solution must be considered: such option should be combined with a more widespread network of G/S.

The different constellations have been investigated and analyzed, together with the related number of satellites that any constellation requires: the cost for any of this solution, coupled with the performances they can guarantee, will be the driver for the choice of the solution to implement.

5.2.3.3. Proposed Configuration: Walker

Walker constellation has been recognized since a long time to be extremely promising for application that requires global and continuous coverage: constellations providing services like telecommunication (Iridium, with a Star Walker constellation) and monitoring (GPS and the incoming Galileo, with a Delta Walker constellation) rest on such typology of constellation To guarantee global continuous coverage, particular orbital condition must be achieved[12]: in these patterns, the satellites are close enough in order to generate on ground a kind of circles, where the sub-satellite points are on the circumference; global coverage is attained when the centre of circles are the point of contact of the antennas footprints. The geometry of the proposed architecture sets the circumcentre as the point with the minimum elevation of the satellite from the local horizon. To guarantee the required coverage, every point of the Earth surface shall see a specific amount of satellites at the same time and at least at the minimum angle of elevation ε. Walker Star Pattern and Walker Delta Patterns are two different kind of Walker constellation, which differs from orbital parameters slightly diverse. This study is focused on the Walker Delta Pattern only[13], and, since the continuous coverage is not required, the references about the circumference and circumcentre decay.

The Walker Delta Pattern constellations are characterized by p orbital planes showing the same value δ of inclination with respect to the reference plane (when the reference plane coincides with the equator, then δ is equal to the inclination i); the ascending nodes of the p orbital planes are equally spaced on the equator of an angle equal to $2\pi/p$, while the n satellites flying along each orbit are uniformly distributed, spaced each other by an angle of $2\pi/n$.

A Walker constellation can be univocally identified by T/P/F code: T is the total number of satellites; P is the number of orbital planes; F is a non-dimensional value, identifying the relative phase angle among satellites on different planes. Such value indicates the angle swept out by a satellite over a certain orbital plane when the satellite in the adjacent plane is at the ascending node: the angle is calculated with the formula *F * PU*, where *PU* (Pattern Unit) is *360°/T* and *F* is included between *0* and *P-1*.

5.2.3.3.1. Walker 6/6/0

The first proposed 6/6/0 patterns have inclination of *67.5°*, and the satellites are distributed along a RAAN spread of 360°. This first constellation is focused on the optimization of the High traffic zones of the oceans, concentrated in middle-low latitude areas (Atlantic Ocean, Mediterranean Sea, and equatorial regions like Africa-Indian seas and Gulf of Mexico). The use of medium orbital inclination is due to the increase in the repetition frequency of the passes over those areas, to be preferred with respect to passes over the Polar Regions, where the high frequency is not justified, due to the low number of vessels (as for near polar orbits).

The 6 orbital planed are spaced of 60° in RAAN from each other, as imposed by Walker constellation requirements. The revisit time is always less than 90 minutes, except for polar latitudes, where the revisit time exceeds the one and half hour by few minutes. The configuration provides a very good coverage with respect to the number of satellites implemented. A screenshot of the constellation is depicted: it is evident how large is the antenna viewing angle.

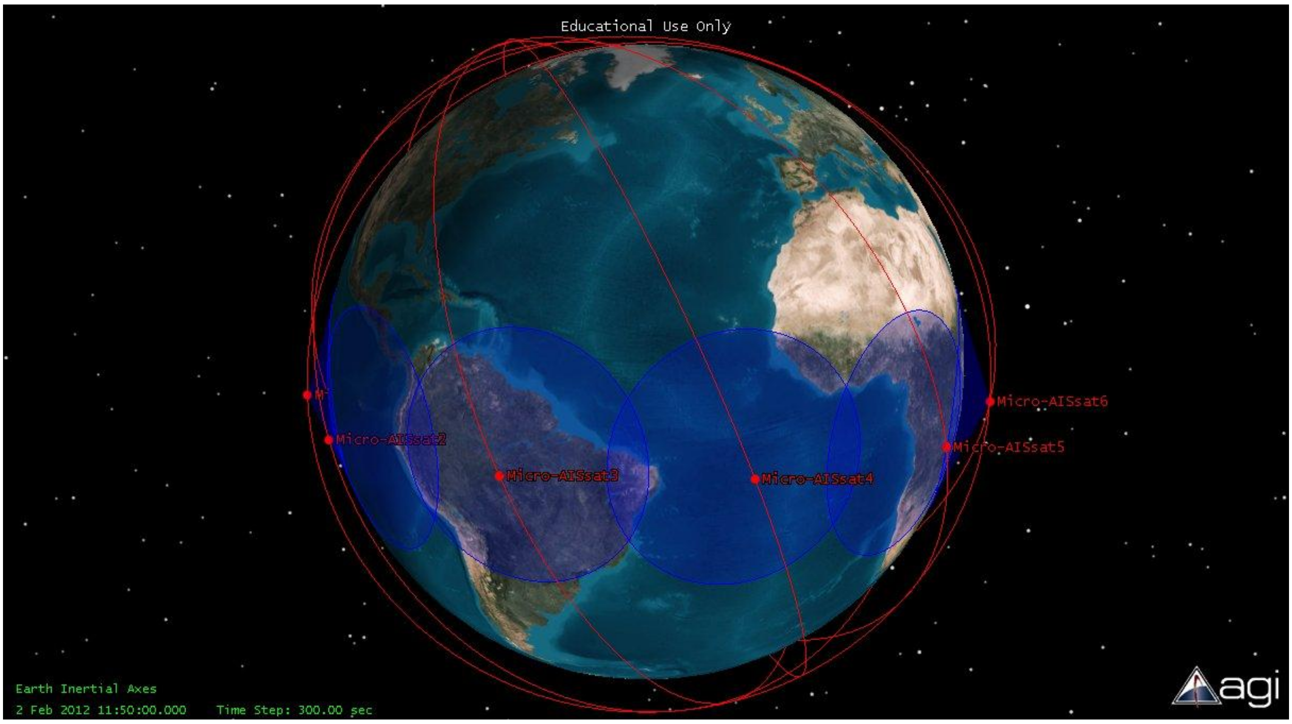


Figure 5 - 57 – Walker 6/6/0 configuration

The revisit time achieved is shown in the next figure. This configuration leads to have a maximum value of revisit time really close to one and half hour, exceeding this value only for polar latitude, where it is not anyway so dramatic, being the number of vessels really reduced; the minimum is instead obtained for latitude in correspondence to HTZ (North Atlantic and Mediterranean Sea).

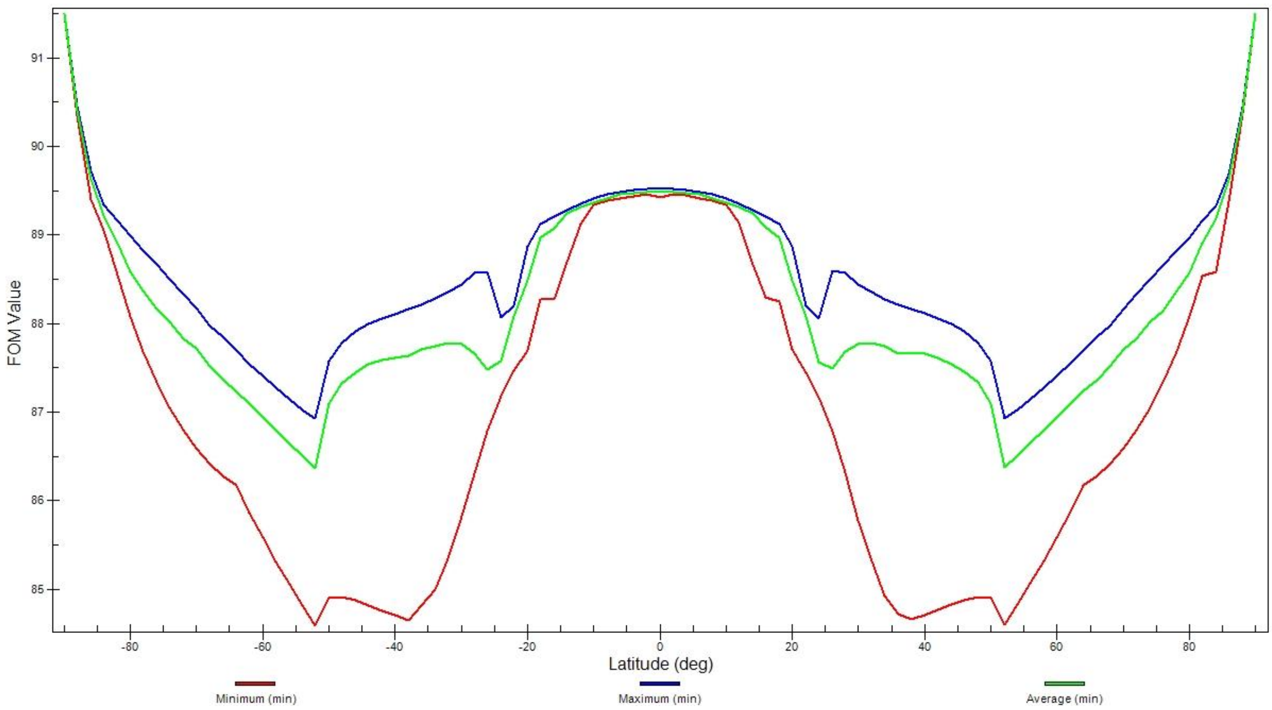


Figure 5 - 58 – Revisit Time vs. Latitude

The requirements of timeliness shorter than 1 hour can be achieved in case 4 ground stations, located two close to the South and two close to the North Pole, are taken in consideration: these ground station could be Svalbard and TrollSat from ESA network and PokerFlat and McMurdo from ESATRACK network. The maximum gap between consecutive accesses to the Ground Station Network is always less than 51 minutes.

The main negative aspect of this configuration is related to the constellation deploy: in fact, placement of 6 satellites over 6 different orbital planes so different one another requires a big effort in terms of number of launches.

5.2.3.3.2. Walker 9/3/2

The second pattern is 9/3/2, a Walker constellation consists of 9 satellites distributed on 3 near polar orbits, to cover all the Earth - from the Equator up to the Poles - with few planes; the inclination has been set to $i = 97.8°$ in order to satisfy the requirements for SSO. The satellites are spaced in phase among each other because of turns in the Ground Stations accesses.

Once selected the Sun synchronous orbits, the Local Time of the Ascending/Descending Node (LTAN) shall be set: in fact, as prescribed by the condition for setting a Walker constellation, the orbital plane must be spaced of 120°, but a constraint over the value to set doesn't not exist; so the initial value can be varied with the aim of improving specific performance or exploiting particular advantages. As the choice of the LTAN influence the platform's EPS (Electrical Power Subsystem) sizing, the most convenient choice from this point of view has been set. Several options for the initial value of the LTAN have been made, and the final choice fell on the following option:

	1° plane	*2° plane*	*3° plane*	
Nr of satellite	3	3	3	
Altitude	600	600	600	
Eccentricity	0	0	0	
Inclination	97.79	97.79	97.79	
Argument of perigee [deg]	0	0	0	
Local Time	8:00 am/pm	12:00 am/pm	4:00 am/pm	
Mean Anomaly [deg]	0	40	80	Value of the first satellite on any orbital plane
satellite spacing [deg]	120	120	120	Angle between successive satellite on the same orbit

Table 5 - 28 – – Walker 9/3/5 constellation parameters

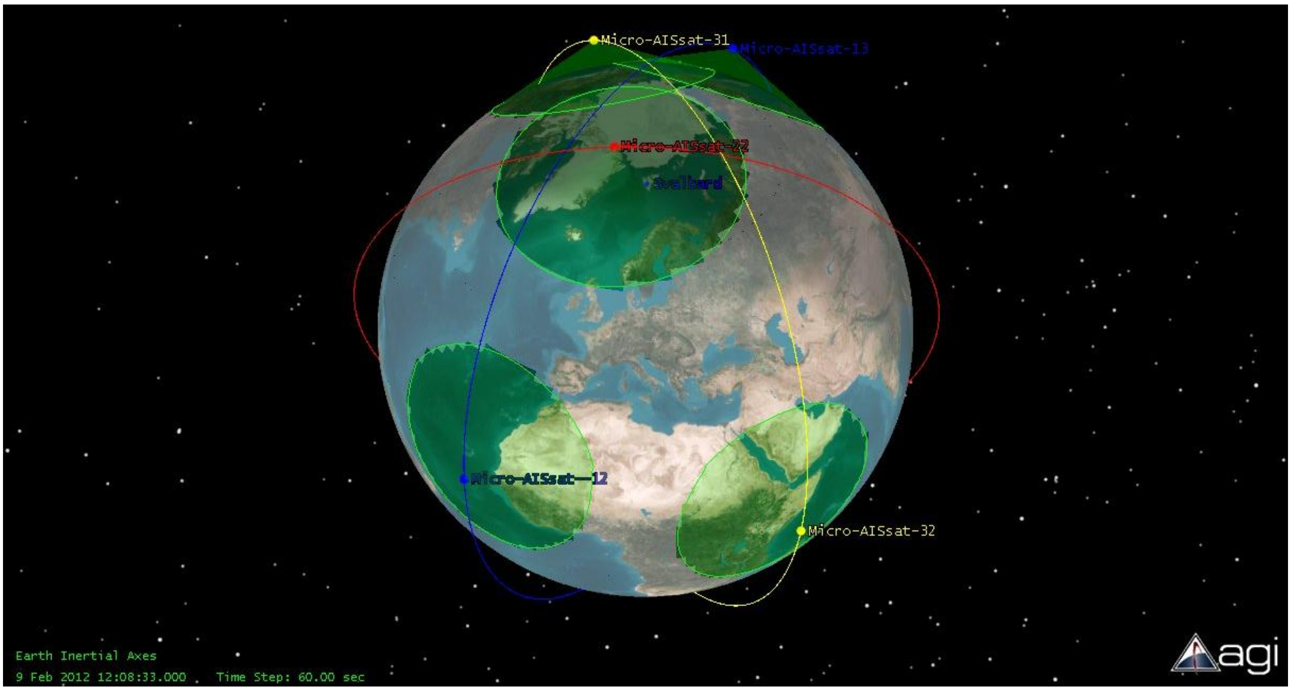


Figure 5 - 59 – Walker 9/3/2 configuration

The value of mean anomaly is calculated through the formula previously introduced. The satellite configuration has been settled is this way not only for distributing and alternating satellite passages over the same area, for coverage purposes, but also for allocating the needed number of Ground accesses to each satellite. As the orbits used are near polar, only two or three Ground Stations close to the Poles are necessary. Such Ground Stations could be, for example:

- Svalbard (78ºN 15ºE);
- Troll Sat (72ºS 2.5ºE);
- Mc Murdo (78°S 166°E).

It could sometimes happen that concurrent contacts (in the order of a couple of minutes) with two satellites occur; according to deeper successive analyses, can be chosen to disregard concurrent contacts, losing a couple of minutes of contacts, if the performance provided are anyway acceptable, or to use two antennas for each G/S.

The difference in using just one G/S per Pole instead of two, as in the previous constellation, is related to a trade-off between costs of G/S management and the performance the system is able to guarantee in terms of data latency: if only contacts longer than 4 minutes are taken into account, a single ground station per Pole assures a 75% of the orbits characterized by a data latency lower than 60 minutes, while the remaining 25% are characterized by a maximum value of 90 minutes. The value of 4 minutes has been chosen as a reference value for data download, disregarding the shorter contacts, as we cannot be sure that shorter contacts allow for the entire data to be downloaded. If a third G/S (the second in the South Pole) is added to the network, the percentage of orbits characterized by data latency larger than 60 minutes would be reduced to 8%.

Further analyses could led to the conclusion that also contacts shorter than 4 minutes could be sufficient for downloading the complete amount of data; if contact longer than 2 minutes (instead of 4) are considered as valuable, the data latency for any orbits will be lower than 60 minutes. If such assumption cannot be done, in order to be compliant with the 60 minutes constraint, a fourth ground station on the North Pole can be added to the network (e.g. Poker Flat G/S).

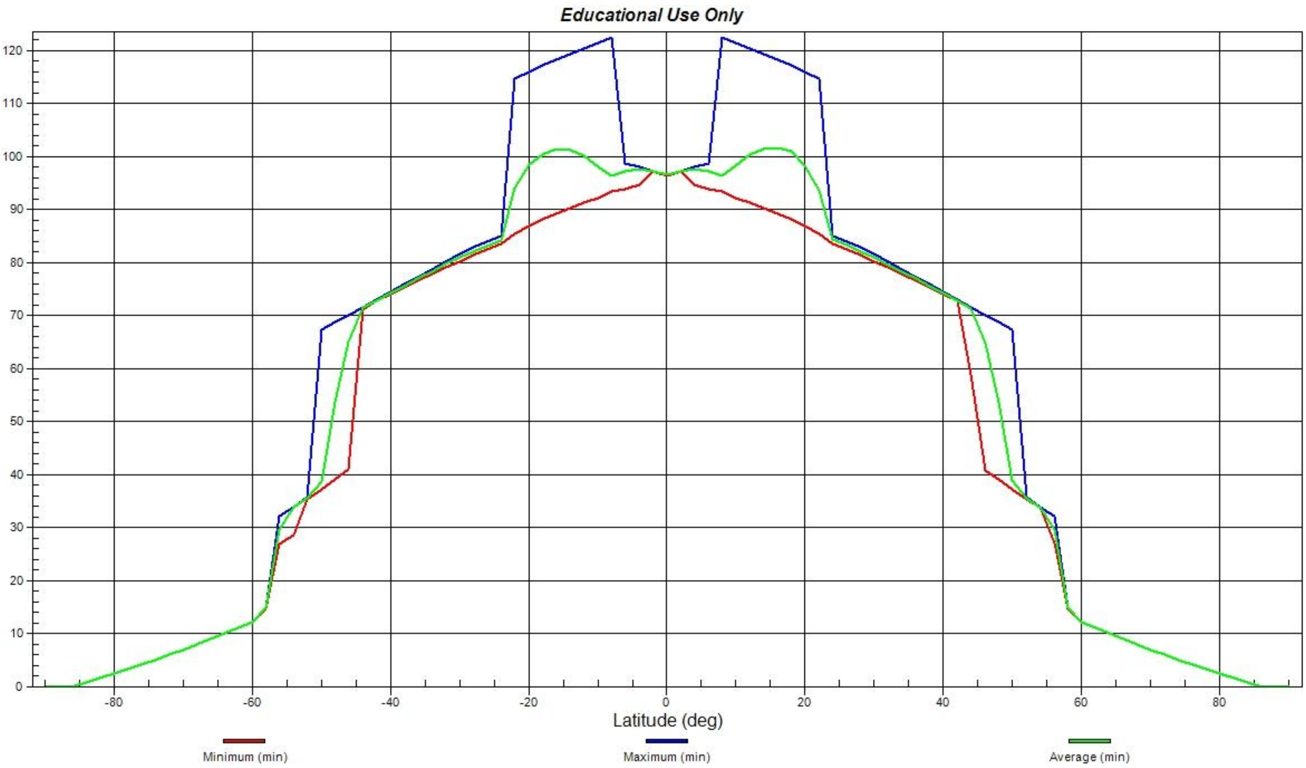


Figure 5 - 60 – Revisit Time vs. Latitude

In the previous figure the revisit time, as a function of the latitude of the scene to observe, is depicted. In this second case, the increment in satellite number lead to really positive effect for latitude close to the Poles: this is due to the fact the 9 satellites are spaced along the anomaly, and the highest and lowest value for the latitude are often overpassed by one or more satellite. The more the latitude is close to the Equator, the high the revisit time; the value of latitude which gives revisit time close to those obtained in the previous constellation is around ± 20°. From these values to the equator, the revisit time grow, reaching also value of 120 minutes of revisit time in the worst (otherwise rare) cases.

One of the advantages of such configuration is an optimization (w.r.t the Walker 6/6/0) of the constellation deployment, keeping the required performance of worldwide coverage; this is due to the smaller number of orbital planes such solution envisage. Satellites flying over Sun-Synchronous orbits encounter more constant illumination condition during their lifetime, being SSO designed for this exact reason: this implies that the spacecraft's power system is smaller and simpler, (smaller solar panels and battery pack, easier battery life cycle and power management). In the same way, also the thermal subsystem shows almost the same characteristics among platforms on different SSO planes.

In this case, however, the satellites flying on the 8:00 am/pm and 4:00 am/pm orbits face similar illumination condition, while the satellites flying over the 12:00 am/pm orbit face illumination condition significantly different from other satellites. In such a way, at least two different spacecraft architectures need to be designed. Differently, the previous constellation can be characterized by very similar satellite, being the illumination condition that the satellites face during the operational lifetime similar

Furthermore additional pro of Sun-synchronous orbits is a really good performance in terms of coverage.

5.2.3.4. Proposed Configuration: Streets Of Coverage

The Streets of Coverage (SOC) is a methodology for designing constellation aiming at global or regional coverage which uses the minimum number of satellite[14]. The terms "Streets of Coverage" derive from the bounded surface around the satellite ground track, drawn from the satellites intersection of the FOV as depicted in next figure.

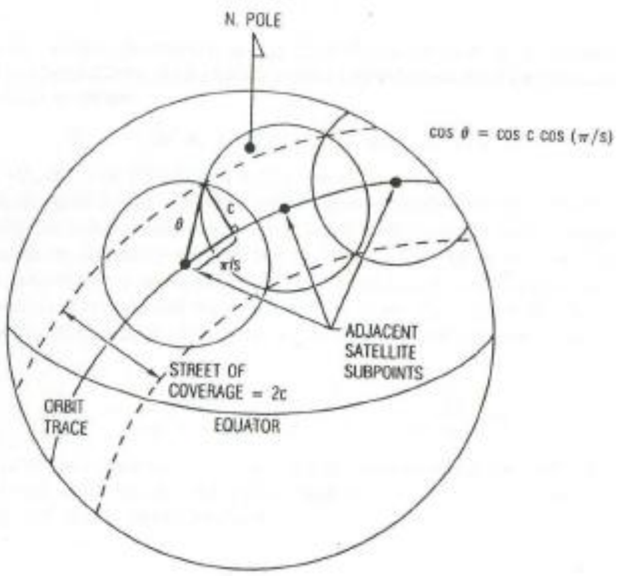


Figure 5 - 61 – Geometrical Configuration of a Street of Coverage constellation

The orbits of a SoC constellation have all the same inclination, and an eccentricity equal to zero; along the same orbital plane satellites are uniformly distributed, and the different orbital plane (in case more than one is necessary) are equally spaced along the equator. No constraints are set on the inclination, so different value can be set; polar orbits can guarantee the global coverage (also for very narrow FOV), are characterized by higher efficiency and major symmetry with respect to inclined ones, and are less subjected to perturbation: this lead to traditionally identify such orbits as the better solution[15]. Such typology of constellation can guarantees also continuous coverage, but the number of satellites to launch in order to provide continuous service should be enormously high, involving very high cost, exceeding, in addition, the constraint of solution made up with less than 16 satellites; furthermore, the AIS service doesn't require continuous coverage.

This technique assures the single coverage of any latitude bound up to global coverage that is between 0° and 90° of latitude. The relative phasing between orbital planes is obtained in accordance with logic of optimization of ground accesses and coverage.

Several attempts have been done, varying the number of satellite and the number of orbital planes, and in table presented below summarize the result obtained calculating the number of satellite necessary for deployment of a SoC constellation able to fulfill the revisit time proposed. Revisit time higher of 60 minutes are not taken into account because better performance (revisit time < 60 minutes) has been previously obtained through Walker constellation with lesser satellites and easier obtainable.

Revisit Time	Mimimum number of satellites on polar orbits (SOC configuration) @600 km
< 60 minutes	10 satellites
< 30 minutes	16 satellites
< 20 minutes	24 satellites
< 15 minutes	32 satellites
null	54 satellites

Table 5 - 29 – Revisit Time vs. Number of Satellites

The global continuous coverage can be achieved if 54 satellite are launched on 6 orbital plane (the usual value of orbital height, 600 km, are used). Since the service, as already underlined, does not require continuous coverage, this option will be immediately disregarded; the option of 16 satellites, that can provide the desirable requirements of revisit time, and that of 10 satellites, that guarantees reasonable performance, will be presented.

The first option envisages 16 satellites over 4 different orbit plane; the plot of revisit time demonstrates the good performances of such a configuration. The four orbital planes are shifted of 45° in RAAN, while the angle between two successive satellites along the same orbit is 90°. These conditions assure constant performances.

As happens for different constellations, a phasing between satellite of adjacent orbital plane are set: in fact, being the orbit polar, if the satellites at a time show the same value of mean anomaly, there will be concurrent passes (of four satellite each time) in correspondence to the Pole-based G/S. So, a difference in mean anomaly has been introduced. For sake of symmetry, an angle of 22,5° has been chosen.

The second option envisage 10 satellites flying along 5 different orbit planes: such configuration guarantee a revisit time lower than 60 minutes, in accordance with the mission requirements. The satellites along the same orbit are shifted of 180°, while the difference in RAAN is 35°. With the aim of facilitating the accesses to the ground stations, any couple of satellites of a particular orbit is spaced of 36° in anomaly with respect to the couple of the adjacent orbits, as shown in the following table. This optimization modifies the SOC pattern, but improves the design applicability and the cost effectiveness of the constellation.

	1° plane	***2° plane***	***3° plane***	***4° plane***	***5° plane***
Nr of satellite	2	2	2	2	2
Altitude [km]	600	600	600	600	600
Eccentricity	0	0	0	0	0
Inclination [deg]	97.79	97.79	97.79	97.79	97.79
Argument of perigee [deg]	0	0	0	0	0
RAAN	35	70	105	140	175
Mean Anomaly [deg]	0 – 180	36 – 216	72 – 252	108 – 288	144 – 324

Table 5 - 30 – SoC Constellation with 10 satellites over 5 different orbital plane

The Polar orbits are characterized by changeable illumination condition during the year, due to the Sun and satellite relative motion. This variability in the illumination condition mainly impacts on the platform design of several subsystems, but mainly on:

1. the Electrical Power Subsystem, in terms of Solar panels design and Battery charge/discharge cycles;
2. the Thermal Control Subsystem.

To evaluate the impact of this variable condition, the trend of the angle between the Sun vector and the Orbital plane during a year is analyzed for each plane. The interval of degrees is included between -79° and +79°.

Hence, during a year, the Sun position varies very much, passing from an almost perpendicular condition with respect to the orbital plane to a position near to the Zenith of the satellite and again to the other side of the orbital plane up to a near perpendicular direction. Keeping a face of the satellite pointing the Earth, the thermal conditions on the satellite faces vary: the eclipse length and frequency change during the year; the succession of the lightened faces of the satellite means the succession of hot and cold faces; variation in illumination condition means variation in power produced by solar array, which has impact on the battery charge and discharge depth; variation in illumination has also impact on design of radiator, which has to face harder life cycle. All these varying conditions impact on the thermal design, making more difficult to face up the worst case. The variation in the position of the Sun from one side to the other of the satellite could be partially compensated with a yaw manoeuvre; but the introduction of this yaw manoeuvres require more AOCS equipment and some adaptation to the Payload management. Moreover, the manoeuvre is not sufficient to point the solar panels towards the Sun in every condition. While the satellite should not rotate along other axis to maintain the Nadir pointing, at least another solar panel on the satellite face opposite to Nadir should be foreseen. An alternative solution could be the use of deployable and adjustable panels. However, even if it simplifies the design avoiding attitude manoeuvres or wide solar panels, it introduces very high technological risks due to the mechanisms, and it may involve heavier structure to support the deployed solar panels. Other orbits provide similar performance with lower risks, so those mechanisms are avoided.

5.2.4. Opportunities for improving services

The solution to this problem may be to use a more directional AIS antenna, limiting the field of view and thereby decreasing the number of ships simultaneously visible to the AIS sensor. The directional AIS antenna may be combined with an omni-directional AIS antenna for increased coverage over areas where the ship density is low.

5.3. Case Study 3: Microsatellite for SAR

5.3.1. The Interferometric SAR System: Description

5.3.1.1. Synthetic Aperture Radar

Radar is an active sensor that emits and measures returned pulsed signals in order to locate and track objects. The spatial resolution of a traditional radar system can be expressed as $R{\cdot}\lambda/L_a$, where R is the range from the radar antenna to the target, λ is the wavelength of the signal, and L_a is the length of the antenna. While this relationship between antenna size and resolution may be acceptable for a ground-based radar system, it quickly becomes obvious that the same relationship is not practical for space-based radar. A desired 1 km image resolution from an orbiting radar sensor in an 800 km orbit using L-band (20 cm wavelength), for example, would require an antenna length of 160 meters. SAR uses post-processing analysis of the Doppler frequencies of returned pulses to increase resolution and allow smaller antenna sizes. The range (cross-track direction) and the theoretical azimuth (along-track direction) resolutions of a SAR sensor are shown in the next two equations, respectively, where c is the speed of light, τ is the time width of the radar pulse, θ is the incidence angle, R is the range, λ is the wavelength, V_p is the platform velocity, and L_a is the antenna length.

$$\Delta_r = {}^{c\tau}\!/_{2\sin\vartheta}$$

$$\Delta_a = \left({}^{R\lambda}\!/_{2V_p}\right)\cdot\left({}^{L_aV_p}\!/_{R\lambda}\right) = {}^{L_a}\!/_{2}$$

The theoretical azimuth resolution of $L_a/2$ would seem to imply that an arbitrarily high azimuth resolution is achievable by reducing antenna length[19]. This is true, but not the actual case, however; a lower bound on the pulse repetition frequency of the signal prevents pulse ambiguity in the identity of returned pulses, and places a corresponding lower bound on antenna area. This lower bound is expressed in the next equation, where A_a is the antenna area, M is an additional design margin that is often added, and R_m is the slant range to the mid-swath, as depicted in the figure.

$$A_a > \frac{4MV_p\lambda R_m \tan\vartheta}{c}$$

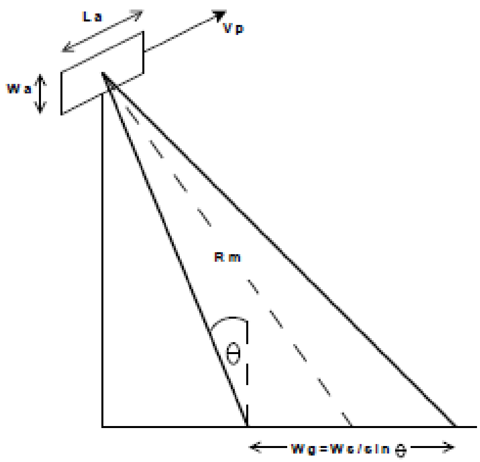


Figure 5 - 62 – System geometrical configuration

In the next figure, the SAR geometry is illustrated, being L_a and W_a the antenna length and width, V_p the platform velocity, θ the incidence angle, R_m the slant range to mid-swath, and W_g the illuminated swath on the ground, as related to θ and the measured swath width W_s.

However, the limitation in the formula of A_a is only applicable if the maximum swath width is desired. Reducing swath width allows a faster pulse repetition frequency, which in turn allows for smaller antenna sizes. The relationship between antenna size, swath width, and azimuth resolution is governed by the next formula, where Δ_a is the azimuth resolution and W_s is the swath width[20].

$$A_a = \frac{2\lambda R_m \tan\vartheta\, \Delta_a}{c}$$

The relationship between antenna size, swath width, and azimuth resolution is illustrated in the next figure for the X band, most commonly used in high-resolution mapping.

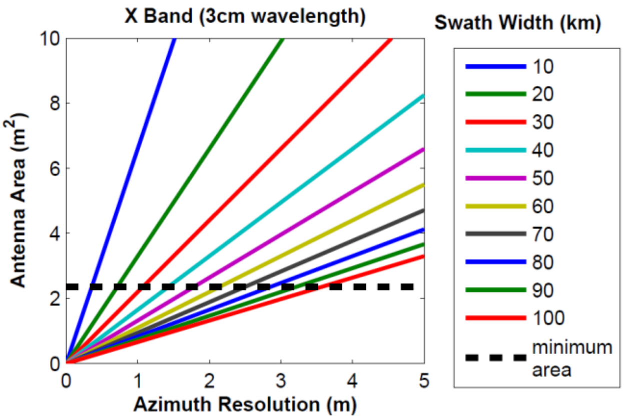


Figure 5 - 63 – Antenna Area vs. Azimuth resolution, at different value of Swath width

The restrictions on microsatellite mass and size place a stringent upper bound on antenna size. The largest possible non-deployable antenna area that could fit diagonally in the Ariane Structure for Auxiliary Payload Micro envelope is 0.558 square meters[20]. Smaller antennas generally require greater transmit power, and also result in a reduced swath width. The smallest flown SAR antenna in a literature survey was the Cassini radar, and had an area of 3.3 square meters[21]. It is therefore likely that a deployable antenna design would be selected for a microsatellite SAR mission, a conclusion which has been supported in the literature thus far[22].

5.3.1.2. Formation Flying

Formation flying is a concept involving the capability, for a multitude of satellite, to work together, monitoring and controlling the respective position, in order to provide a precise and distributed network of sensors capable of accomplishing mission that cannot be provided (or provided only partially) by a single (even large) satellite. Such capability can lead to space capacity significant benefit, from an improved, unprecedented high resolution images, to the ability to observe (not necessarily with an optical payload) the same scene from multiple angles or at multiple times. These characteristic can enable new and powerful application, helping also improving the performances of some existing ones.

Formation flying (FF) includes the capacity of the space platforms composing the formation (at least one of them) to autonomously evaluate the relative position, and maneuvering in order to keep the attitude within well-defined boundaries, that depend on the application. Presently, no mission capable of exploiting formation flying has been realized, if we exclude just a couple of missions for testing FF capability[22,23,24], that do not have any other objectives apart testing and proving FF.

According to the application for which the formation is intended, there are two formations possible:

- **Trailing formations** are formed by multiple satellites orbiting on the same path. They are displaced from each other at a specific distance to produce either varied viewing angles of one target or to view a target at different times. Trailing satellites are especially suited for meteorological and environmental applications such as viewing the progress of a fire, cloud formations, and making 3D views of hurricanes. Even without the capability to modify their relative position, such configuration are used by Landsat 7 with EO-1, CALIPSO with CloudSat, and Terra with Aqua.

- **Cluster formations** are formed by satellites in a dense (relatively tightly spaced) arrangement. These arrangements are best for high resolution interferometry and making maps of Earth. Example: TechSat-21[22].

In this overview, constellations of satellite like that of Global Positioning System, or Iridium, Globalstar or others are not encountered as the satellites do not have the capability to interact each other, evaluating the relative geometry and modifying their orbital configuration.

Formation flight requires precision attitude control and position determination methods, as well as new orbital maintenance algorithms; several missions attempted are currently attempting to reach such performances, looking for centimeter-level position determination and sub-meter control[22]. Such performances has been demonstrated to be achievable, and future mission capable of providing additional services with respect to those provided nowadays perfectly fits the spirit of such research effort.

5.3.1.3. Interferometric Synthetic Aperture Radar

The term interferometry refers to a class of technique consisting in overlapping electromagnetic waves in order to extract information concerning the waves; such a technique can be used in a variety of fields.

Interferometric synthetic aperture radar, also indicated with the acronym InSAR, is a radar technique used, among the other things, for purpose of remote sensing. This technique consist in using two or more antenna for acquiring a particular scene from multiple observation angle; SAR interferometry uses the phase differences in the returned signal in two SAR images to compute the elevation of points in the imaged terrain. To form digital elevation maps of a region, two SAR images taken from different angles of the target area are needed. The images are co-registered (aligned using orbit data and ground control points), and an interferogram is generated by differencing the phases of the returned signal forming the image. The interferometric phase (phase difference) ranges from 0 to 2π, and a phase unwrapping process must be used to solve the inherent integer ambiguity problem and obtain the absolute phase difference. Orbital data, specifically the attitude of the satellites, the baseline, and the orbital altitude, are used along with the interferometric phase data to triangulate the elevation of surface points in the image[23].

InSAR technique is able to measure changes in the order of centimeter in deformation. Such a technique can be used for monitoring of natural hazards, for example earthquakes, volcanoes and landslides, and also in structural engineering, in particular monitoring of subsidence and structural stability. As interferometry is a process that requires precise knowledge of spacecraft attitude and relative position, it is potentially a particularly useful application for formation-flying technology.

Once formation flight has been completely demonstrated, and capabilities and performances set up, future missions can be conceived to carry onboard payloads designed to exploit these capabilities. Earth observation, the study of the Earth’s surface using imagery from spaceborne platforms, is an area with numerous applications that can benefit greatly from the availability of multiple platforms with precise position determination and attitude control; and among the numerous applications that can benefit from such a capacity, interferometric synthetic aperture radar (InSAR) missions using microsatellite can really reveals themselves to be profitable. In fact, the use a “large” microsatellite platform for the purpose of providing the necessary power and mass support for a passive SAR payload, to use in conjunction with a large, already existing platform as main object, represents a significant cost savings, envisaging an economical effort on the order of a tens of millions of Euro per microsatellite, compared with traditional large satellites, which is instead an expense of several hundreds of millions of Euro, as the program COSMO SkyMed, or other satellite mounting a SAR (SAR-Lupe, for example), witnesses. In this paragraph, a microsatellite has been evaluated as a potential platform for mounting payloads for multistatic InSAR Earth Observation, analyzing its feasibility and potential applications exploiting already demonstrated formation-flying capability.

5.3.2. Mission Concept

SAR Interferometry is an obvious mission concept choice for a formation-flying mission, due to the availability of multiple radar receiver platforms that can be controlled with precise attitude and position determination. Traditional interferometry is performed in a repeat-pass mode, in which successive images are taken during repeat passes over an area, temporally separated by hours or days. Accurate digital elevation maps can be generated, provided that the terrain on the ground has not changed since the previous pass. Differential interferometry specifically measures terrain displacement between successive passes, as a means of mapping the effects of earthquakes or tracking glacier flow, for example. The availability of formation-flying microsatellites raises the possibility of taking near-simultaneous images for the construction of digital elevation maps, as well as images with an higher resolution, and performing differential interferometry on a much shorter time scale, which could enable mapping of phenomena such as ocean currents. One interferometry configuration that has been studied, particularly with regard to microsatellites, is the interferometric cartwheel. The interferometric cartwheel is named for the orbital configuration of one leading transmitter satellite, and two or more following receiver satellites that move in an ellipse relative to the center of the receiver formation, as shown in the next figure.

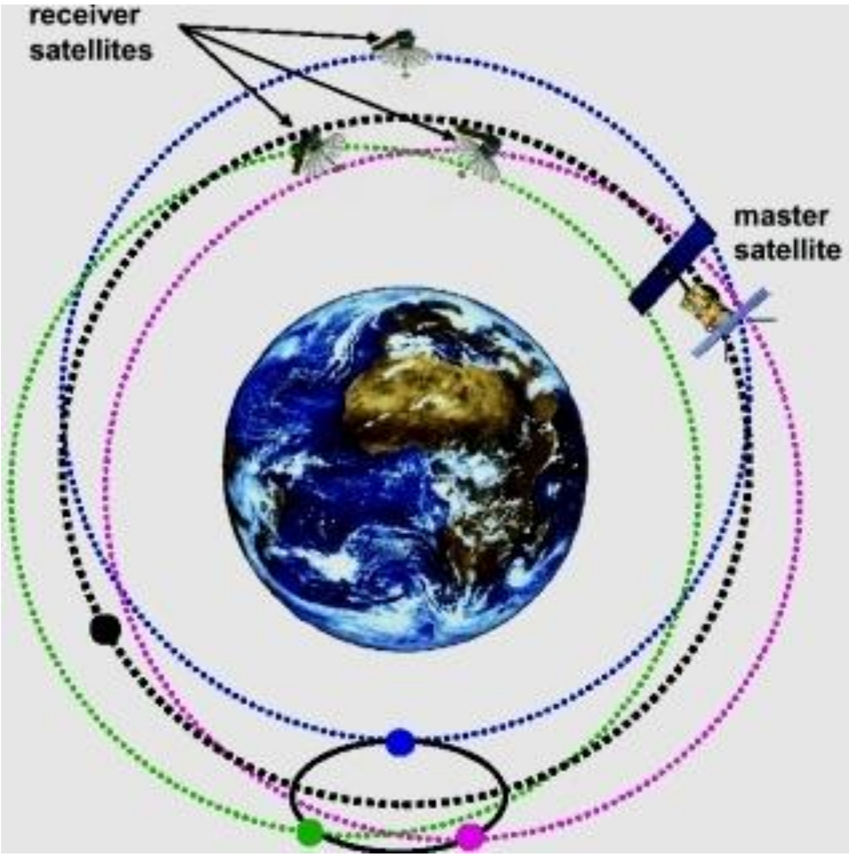


Figure 5 - 64 – Orbital configuration of receiver satellites following a master SAR spacecraft

This type of configuration is most often investigated in the context of supplementing an existing space-based radar mission with a constellation of microsatellites collecting interferometric data. Such solution offers the advantage of cost-savings over launching an additional transmitter satellite, but limits the orbital configuration and signal band of the interferometric mission to that of the existing radar mission. Other receiver configurations, apart from the Cartwheel, will be investigated to determine the feasibility, and the most appropriate solution, for a microsatellite InSAR mission using demonstrated formation flying capabilities.

5.3.2.1. Objectives

As already done for previous mission studies, the overall objective of this study is to investigate a possible concept for a future mission using innovative capabilities, presently not yet implemented for real space service, but already demonstrated in past space mission.

The mission chosen for this last study is an interfererometric mission, performed through the use of multiple receiver SAR antenna, and the capabilities related to formation flying and relative positioning are necessary to correctly set up the orbital configuration.

The work performed has been:

- To identify promising applications for multistatic InSAR data
- To choose among the identified application the one to deeply investigate
- To generate mission scenarios and orbital configuration for the application and investigate the most profitable

5.3.2.2. InSAR Application

The application investigated in this paragraph has been only some of the possible application that the interferometry of SAR data can be used for: other example of InSAR applications are traffic monitoring[24], differential terrain elevation modeling[25], and improving spatial resolution by combining multi-pass imagery[26].

Sensors used for Earth observation measure can be categorized in two different classes: those passive, that acquire the electromagnetic radiation reflected by the object of the analysis but emitted by other sources (the Sun, for example) and those active, that acquire the electromagnetic radiation reflected by the object of the analysis and emitted by the sensor itself.

Such electromagnetic radiation are acquired for the purpose of gathering information about the surface and atmosphere, yielding data used for topography, environmental and geological studies, surveillance, and many additional applications.

Remote imaging sensors such as SAR, optical, and hyperspectral imagers can be classified in terms of four types of resolution – spatial, temporal, radiometric, and spectral. Spatial resolution, sometimes also referred to as the ground sample distance, is a measure of the sensor’s ability to distinguish between point targets in the image, expressed in the distance covered by a single pixel. Temporal resolution is a measure of the frequency at which the sensor can collect images, expressed in frames per second. Radiometric resolution indicates how finely differences in intensity are measured. Each pixel contains a level of gray, determining the intensity of the image. 256 levels are possible, with 0 representing black and 255 representing white if the maximum resolution is used. The radiometric resolution reflects the number of levels used to represent intensity, expressed in the number of bits per pixel necessary to store the maximum number of levels of gray in the image. Spectral resolution is a measure of how finely differences in reflected wavelength are measured. A sensor’s spectral resolution determines whether color is reproduced in the images, and also which materials in the image can be identified by their spectral reflectance properties.

- Superresolution Imagery

Superresolution imagery is a technique that combines data from two or more images of the same location in order to produce a single image with a spatial resolution that is greater than that of the original imagery, and is useful for any application that requires a higher spatial resolution than is available from a given sensor, or if mission constraints such as cost, size, or power place a limit on the capabilities of an imaging sensor. Superresolution imagery is currently formed using images from repeat satellite passes, and has been demonstrated with ENVISAT data to produce a range resolution improvement of about 50%[26]. Superresolution imagery in the range and azimuth dimensions requires receiver separation in the vertical and along-track directions, respectively.

- Digital Elevation Modeling

Digital Elevation Modeling (DEM) generation requires multiple receivers to image the same target and measure the phase difference in the returned signal at different angles, and therefore determine the height of the target. Existing DEMs have been generated using image pairs from either repeat satellite passes, as in the case of Radarsat-1, or separate receivers co-located on the same spacecraft, as in the case of the Shuttle Radar Topography Mission (SRTM), and the upcoming TanDEM-X mission will generate single-pass image pairs along with TerraSAR-X for the purpose of DEM generation. DEM usefulness has been demonstrated in monitoring geological phenomena and land movement[25], hydrologic modeling[27], and fusion with superresolution imagery for the purpose of classifying map features and monitoring changes[28].

- Moving Object Detection

The signal phase in an interferometric image is a function of the Doppler shift of the reflected signal, and is therefore proportional to the line-of-sight velocity of the objects in the image. The time delay between two images received by spacecraft separated in the along-track direction can be used to compute the velocity of targets on the ground. Along-track interferometric measurements are currently acquired using receivers co-located on the same spacecraft; the receivers in the Shuttle Radar Topography Mission were separated by 7 meters in the along-track direction and were used to successfully track a car traveling at 48 kilometers per hour. Additionally, a single receiver antenna can potentially be configured for along-track interferometry; the “split-antenna mode” available on TerraSAR-X has been demonstrated to be capable of measuring ocean currents with speeds of 1 to 2 meters per second[26].

5.3.2.3. Mission Design

As any space program that starts only once the user requirements has been set, and one or more mission objectives are clearly defined, the starting point for the study presented needs to have a well-defined application to carry out: among the possible application that can be chosen as case study, the most important of which (in the opinion of the author) has been briefly described, that selected to be investigated and used as mission objective is the superresolution imagery.

5.3.2.3.1. Superresolution Imagery

The application chosen for this mission design is the super resolution imagery, which is a class of techniques aimed at improving the resolution of an imaging system. The following figure clarify the geometry necessary for the correct implementation of superresolution imagery technique.

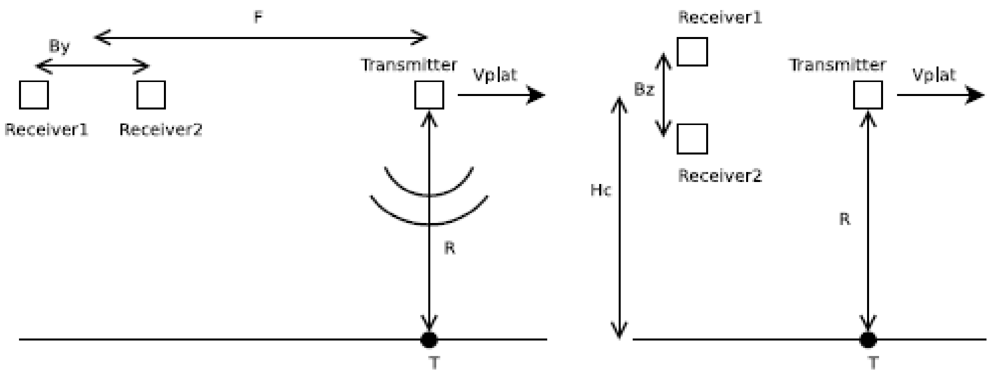


Figure 5 - 65 – Orbital configuration for superresolution imagery

In the previous figure, the relevant parameters and distances are illustrated: the range R from the transmitter to the ground target, the direction of the platform velocity V_{plat}, the following distance F between the transmitter and the center of the receiver constellation, the along-track and the vertical baseline between receivers B_y and B_z respectively, the height of the center of the receiver constellation H_c, and the positions of two receivers.

To be capable to apply superresolution imagery technique it is required that the length of the baseline between receivers is high enough such that the information in the received signals is independent, and cannot be combined to create interferograms. The value of the distance at which this occurs defines the critical baseline both for vertical and along-track directions. This critical baseline occurs when the distance between receivers is high enough such that the path to each receiver from the same ground target has changed by more than a wavelength between each received pulse. Since the received signals are independent, they can be combined to form images with an improved resolution in azimuth or in range up to a factor of two[29]. Additional receivers properly spaced can be added to the constellation, if desired, each improving the image resolution in the relevant direction by an additional factor of two. The critical baselines in the along-track and vertical directions are defined by the following formulas, respectively:

$$B_{crit-a} = \frac{\lambda}{D_p}\sqrt{R^2 + F^2}$$

$$B_{crit-v} = \frac{2 f_s \lambda R^2}{H_c c}$$

where f_s is the sampling frequency, D_p is the distance traveled by the receivers between signal pulses, and H_c is the height of the midpoint of the vertical baseline[29]. The baselines between receivers must be longer than these critical baselines in order to construct superresolution images. The following table lists the critical along-track and vertical baselines for four angle of incidence.

	Along-track baseline (km)				Vertical baseline (km)			
Incident angle (deg)	15	30	45	60	15	30	45	60
X-Band	9.7	10.8	13.2	18.7	14.6	18.2	27.3	54.6

Table 5 - 31 - Critical along-track and vertical baselines for four angle of incidence.

5.3.2.3.2. Orbital Scenarios

There are three main possibilities for the implementation of formation-flying interferometric mission:

1. the cross-track pendulum;
2. the interferometric cartwheel;
3. the Car-Pe configurations.

The selected orbital configurations represent a range of options for baseline variability in both the cross-track and along-track directions. The orbital configuration to choose depend on the service that the interferometric mission has to carry out; digital elevation modeling, super-resolution imagery, and moving object detection are the most important application for which putting into service an InSAR mission.

Each of these applications utilizes different configuration of receivers, imposing different requirements on the orbital configuration. So, in order to choose the most valuable orbital configuration, it is necessary identifying the requirements of the application identified.

The promising orbital configuration that has been proposed for multistatic InSAR formations (i.e. the cross-track pendulum, the interferometric cartwheel, and the Car-Pe – Cartwheel-Pendulum – formation) has been investigated and will be here described; evaluations about the characteristics of each one will be done, and the one most compliant with the requirements of the application chosen will be selected.

- The cross-track pendulum configuration consists of two more receiver satellites in circular orbits. The right ascension of the ascending node and (optionally) the inclination are varied to produce a stable cross-track “swinging” motion between the satellites, resulting in a cross-track baseline that varies between zero or near-zero, and the maximum desired baseline. In this case, the right ascension of the ascending node of the first receiver was held at zero while that of the second receiver was varied in order to produce the desired swinging motion. The along-track position of each satellite relative to the others can be adjusted independently of the cross-track motion. One potential disadvantage of the cross-track pendulum arises if differential inclinations are used to contribute to the cross-track motion, causing secular drifts of the ascending nodes of the orbits. As per [30], a theoretical 100kg microsatellite would require 1 kg of liquid propellant per year for each kilometer of effective baseline to correct this drift. To avoid drift, the cross-track motion in the pendulum configurations in this investigation is produced using only variations in the right ascension of the ascending node, keeping the inclinations of the receivers the same.

- The interferometric cartwheel configuration consists of two or more receiver satellites with a slight change in eccentricity from the transmitter satellite, and the same inclination. The arguments of perigee and the true anomalies are evenly spaced throughout 360 degrees; for a two-satellite cartwheel the arguments of perigee are 0 and 180 degrees. The baselines in the cartwheel configuration are coupled, and the small difference in eccentricity is varied to produce the desired baseline in the chosen direction. While the cross-track baselines in the pendulum configurations vary between zero and the maximum baseline distance, both the cross-track and along-track baselines in the cartwheel configuration remain within an envelope, as shown within the next section. However, due to the coupled nature of the baselines, the along-track baseline cannot be independently adjusted, which could limit potential along-track interferometry applications due to temporal decorrelation, and impair the quality of interferograms due to differing Doppler centroids between receivers[30].

- The Car-Pe configuration consists of two or more receiver satellites in a pendulum configuration, combined with one or more satellites in a cartwheel formation. Such an orbital configuration combines the advantages of both the pendulum and cartwheel configurations, including the decoupled baselines of the pendulum configuration and the baseline envelopes of the cartwheel. The along-track baseline of the pendulum satellites can be set up in order to have a minimum along-track baseline, and the larger along-track baseline of the cartwheel satellite(s) can be used if a longer along-track baseline is desired[28]. This configuration reveals to be particularly helpful if different interferometric application wants to be performed, and several requirements, probably contrasting each other, needs to be satisfied.

For the purpose of this investigation, each configuration involves two receiver satellites orbiting in an along-track formation, either leading or following a transmitter satellite, which could be any of the satellite mounting a Synthetic Aperture Radar, operating in either an L-band, C-band, or X-band. For the sake of completion, also a mission has been chosen, and such a choice felt on Cosmo SkyMed satellite; then, the scenario has been simulated using STK

In all three scenarios, the satellites were orbiting at the altitude of 619km, at 97.8 degree of inclination. The Cross-Track Pendulum configuration was generated by varying the right ascension of the ascending node of the receiver satellites to generate a cross-track motion with the desired baseline as the maximum cross-track distance. The Interferometric Cartwheel configuration was generated by spacing the arguments of perigee and true anomalies of the receiving satellites evenly through a 360 degree range, and adjusting the eccentricity to produce the desired cross-track baseline. The Car-Pe configuration combined the cross-track pendulum setup with an additional receiver satellite in a cartwheel orbit. In each scenario, the receiver orbits were adjusted to produce a maximum cross-track baseline equal to the critical cross-track baseline for digital elevation modeling using a 60 degree incidence angle, as shown in Table 2.2. Orbital elements of the receiver satellites for each configuration are shown in Table 2.5. Finally, for each orbital configuration scenario and each transmitter option, the available baselines (cross-track and along-track), operating areas, and bistatic resolutions were compared, and the feasibility of each selected application was evaluated.

Cross-track Pendulum

One of the characteristic of the cross-track pendulum formation was that the cross-track baseline between the two receivers varies along the orbit between two values that depend on the choice of the right ascension of the ascending node and the inclination of the receiver satellite. Since the application of superresolution imagery does not impose particular limitation to such a value, this orbital parameters can be set according to different requirements. Another characteristic is that the pendulum formation does not inherently cause an along-track separation, so it could be imposed respecting the constraints about the critical along-track baseline. As we said that the inclination will kept the same for both the receivers, the only parameter to vary in this configuration will be the right ascension.

Variation in the right ascension of the ascending node of the receivers causes the cross-track baseline to vary between near-zero and the critical baseline for the selected transmit band, while the along-track baseline remains nearly constant at a predetermined distance, and the vertical baseline is near-zero.

Interferometric Cartwheel

The interferometric cartwheel formation was designed such that the maximum along-track baseline between receivers is equal to the critical along-track baseline for a value of inclination angle that depends on the angle of observation that mission/user requirements imposes. The geometry of the cartwheel formation results in coupled baselines; the maximum cross-track and along-track baselines are identical, and the maximum vertical baseline is one-half that of the cross-track and along-track baselines

Cartwheel-Pendulum

The Car-Pe formation combines two receivers in a cross-track pendulum formation with a third receiver in a cartwheel formation with the first pendulum satellite. As such, there are three sets of baselines to consider. The baselines between the pendulum receiver pair and the baselines between the first pendulum receiver and the cartwheel receiver are identical to the baselines that has been already evaluated and exposed in the previous lines; the third set of baselines is that between the second pendulum satellite and the cartwheel satellite. The behavior of these baselines is more irregular than that of the pendulum and cartwheel configurations, as one receiver is “swinging” with respect to the first pendulum satellite and the other is “circling” in formation with the first pendulum satellite. The baseline variations between these two receivers are periodic, however, and the time period shown in the figure represents one cycle of the observed variations.

Car-Pe available baselines between pendulum satellite #2, and the cartwheel satellite: the cross-track baseline displays an irregular oscillatory behavior that is loosely centered about the critical baseline, and the along-track and vertical baselines behave roughly similarly to their counterparts between cartwheel satellites.

Among the three options here expressed, the one chosen is the cross-track pendulum configuration: it guarantees an easy accomplishment of the constraints dictated for a correct implementation of a superresolution imagery technique.

Orbit parameters: two satellite flying over an orbit similar to those flown by COSMO-SkyMed. The orbital characteristics of COSMO-SkyMed-1 are:

	COSMO-SkyMed-1
inclination:	97.8699°
perigee height:	621 km
apogee height:	623 km
right ascension of ascending node:	17.6121°
argument of perigee:	84.3348°

Table 5 - 32 - COSMO-SkyMed orbital parameters

In order to set the correct parameters for the application in investigation, a vertical baseline has to be created: the first parameter is the orbital height: the application requires two orbits with the same semimajor axis of the main satellite, since we the microsatellites must not have a shorter or longer period, otherwise they fly constantly faster or slower, causing a constant drift of the along-track baseline: the semimajor axis needs to be the same (7000 Km), but, in order to allow the existence of a vertical baseline, apogee and perigee of two satellite will be set at, respectively 628 km and 616 km. The microsatellite inclination will be kept the same, for having a simpler orbital configuration, while right ascension of ascending will set up properly for imposing a correct along-track separation. The differences between the RAAN of the main satellite and the RAAN of the two microsatellite will be, respectively plus and minus 1°. The argument of perigee of the two satellites will be separated of 180 degrees.

	COSMO-SkyMed-1	**Microsatellite InSAR-1**	**Microsatellite InSAR-2**
Eccentricity:	0.0001492	0.000857126	0.000857126
inclination:	97.8699°	97.8699°	97.8699°
perigee height:	621 km	616 km	616 km
apogee height:	623 km	628 km	628 km
right ascension of ascending node:	17.6121°	18.6121°	16.6121°
argument of perigee:	84.3348°	264.3348°	84.3348°

Table 5 - 33 – Orbital parameters of formation of 2 receiving satellites and COSMO-SkyMed

Such an orbital condition entails a cross-track separation that does not affect the superresolution imagery, a vertical separation that can be imposed

5.3.2.4. Microsatellite Design

The design of the satellite platform starts from the definition of the payload, using the information, constraints and requirements stated by the orbit designed by mission specialists. The orbit must be of course chosen according to the service that has to be provided: such a mission envisage the use of a couple of microsatellites each of which mounting a receiver operating in a certain range of frequency, and flying in formation with another satellite, that needs to mount a Synthetic Aperture Radar, of course working in the same bandwidth.

The satellite chosen to implement the application of superresolution imagery has been COSMO-SkyMed, with a SAR antenna operating in the X-band: as a consequence, the two satellite will mount an antenna receiver operating in the same bandwidth.

5.3.2.4.1. *Payload*

The Payload will be an antenna receiver; the design of this kind of antenna is much simpler than that of a conventional SAR; in fact, in such a situation, the antenna is passive, which implies a simplification also of the instrumentation necessary. Anyway, for such a space mission, the critical subsystem is the antenna.

The first point to complete is the choice of the receiving antenna; according to studies carried out by CNES[31], a suitable antenna for such application is that depicted in the next figure, a wrap-rib antenna developed by Lockheed Martin.

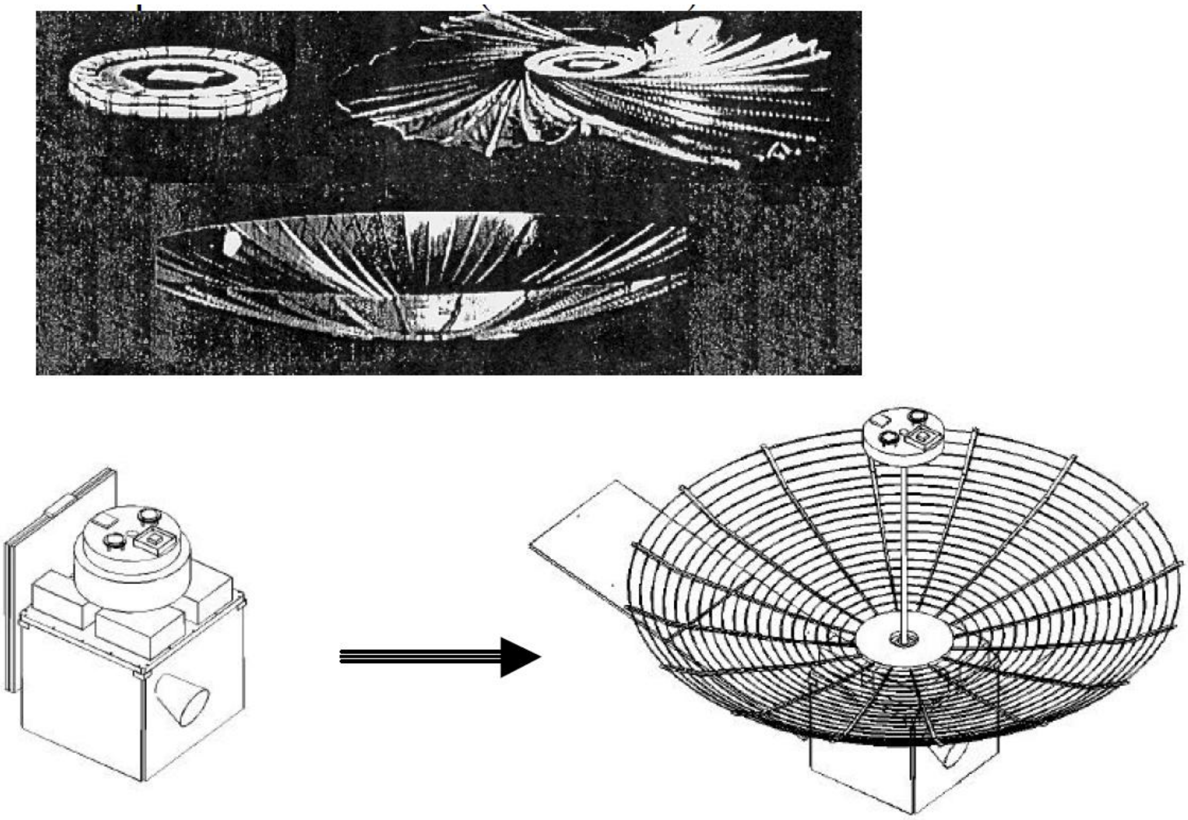


Figure 5 - 66 – Wrap Rib antenna for SAR superresolution imagery

Such configuration has been chosen as the main solution, also bearing in mind back-up solution. The electronics required by the antenna are showed in the next figure, while the following table contains the technical characteristic of the payload, together with mass, dimension, and power consumption in the operative mode.

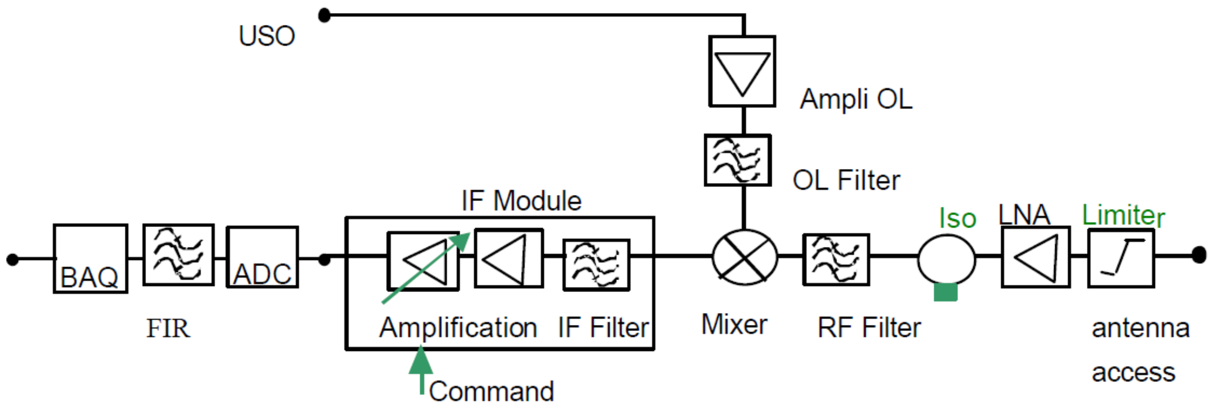


Figure 5 - 67 – Wrap Rib antenna electronics scheme

Preliminary Payload Parameters		
Antenna size		2.4 m diameter
Mast lenght		1.2 m
Antenna gain (RF losses included)		~ 26 dBi
Instrument data rate		~ 160 Mbit/s
Memory size		20 Gbit
Telemetry data rate		Up to 50 Mbit/s
Mass	Antenna + Mast	16 kg
	Receiver	12 kg
	Telemetry unit	2 kg
Power consumption	Imaging mode	50 W
	Telemetry mode	50 W
	Stand-by mode	20 W
Mission lifetime		2 years

Table 5 - 34 – Wrap Rib antenna main parameters

Starting from the characteristic of the payload, the design of the space system should start.

5.3.2.4.2. System for Formation Flying

The first (and probably more important) subsystem to define is that devoted to the control of the formation flying configuration: such a system can be the one developed for the canX-4 and CanX-5 nanosatellite, that already demonstrated the capability and the necessary performances for implementing such an application [16]. Both the microsatellite will be equipped with same devices devoted to such a capability: a GPS receiver, an intersatellite link, and a proper delta-V will be evaluated to perform manoeuvres to keep the correct relative geometry.

The GPS receiver will have the task to provide absolute position and velocity information to each satellite; the information coming from the GPS constellation will be elaborated, and information about absolute position and velocity evaluated: an accuracy between 2 and 5 meters (RMS) and between 5 and 10 cm/s (RMS) respectively will be achieved; Such information are not compliant with the requirements of superresolution imagery.

The solution has been provided by an algorithm developed at the University of Calgary[32]: this algorithm employs carrier phase and Doppler data with differential technique to determine the relative position and velocity of the satellites to within 2-5 cm (RMS) and 1-3 cm/s (RMS). Such algorithm requires that both the satellites keep contact with the same GPS satellite; one of the two satellite will mirror the attitude of the deputy satellite during formation flying manoeuvres.

The intersatellite link supposed for this project is composed by an S-Band radio transceiver mounted on board each satellite, allowing the satellite to share information regarding their absolute position, velocity but also attitude. One omni-directional antenna on each satellite will provide the necessary data rate.

Bearing in mind the mission specifications (two receiving satellite, following in a cross-track pendulum configuration the main satellite) it is necessary that both the satellite have the capability to manoeuvre, and, for such a reason, both the satellite will have a delta-V budget reserved for "configuration-keeping" manoeuvres.

The propellant necessary for the manoeuvres for keeping the attitude has been evaluated to 15 m/s per year of operation per satellite.

5.3.2.4.3. Propulsion Subsystem Sizing

As the satellites have to perform the manoeuvres necessary for keeping the orbital cross-track pendulum configuration, a propulsion system is necessary: additional purpose of the propulsion subsystem is to provide adequate forces during the mission lifetime (1 year) and to complete the following manoeuvres:

- To correct launcher dispersions;
- Orbit maintenance manoeuvres.

In terms of performance, Ion thrusters have far better specific impulse if compared to chemical propulsion systems; but the complexity of the system suggests the selection of a monopropellant/bipropellant system. Among these solutions, the monopropellant solution seems to be the more appropriate, as the complexity of a bipropellant system is not justified for the mission is exam.

The elements composing the propulsion subsystem are:

- Propellant Tanks
- Thruster
- Latching Valve (LV)
- Liquid Filter (LF)
- Fill and Drain Valve (FDV)
- Pressure Transducer (PT)

- Feeding lines (Pipework)

The Propellant Tank proposed for the study is the PEPT-420 model, developed by Rafael Advanced System. Propellant Management Device tanks are used in monopropellant systems for the control of fluid and separation of the pressurant gas from the fuel to provide gas-free propellant to the thrusters through the spacecraft life. The tank is a spherical pressure vessel; The shell material is Ti-6AI-4V.

The thrusters used for the spacecraft are CHT 0.5 thruster units manufactured by EADS Space; they were designed and tested for blow-down applications for a max inlet pressure of 22 bar. For any of the components here identified, a mass margin of 5% has been taken in consideration, as all the components are space qualified, so a small margin is sufficient. The other elements:

Unit	Unit Name	Quantity	Mass per quantity excl. margin	Margin	Total Mass incl. margin
1	Pressurant	1	0,0976	5	0,126
2	Propellant tank (PEPT-420)	1	3,5000	5	3,675
3	Fill Drain Valves (Vacco MRS)	5	0,0180	5	0,095
4	Latching Valves (LEE VHS-M/P)	16	0,0040	5	0,068
5	Filters (Vacco LP filter)	4	0,0800	5	0,336
6	Lines and fittings	1	0,0200	5	0,021
7	Temp. Transducers (National LM335Z)	10	0,010	5	0,010
8	Pres. Transducers	5	0,0750	5	0,394
9	Thrusters (EADS CHT-0.5)	16	0,1950	5	3,276
SUBSYSTEM TOTAL			**7,620**	**5,0**	**8,001**

Table 5 - 35 – Propulsion sytem

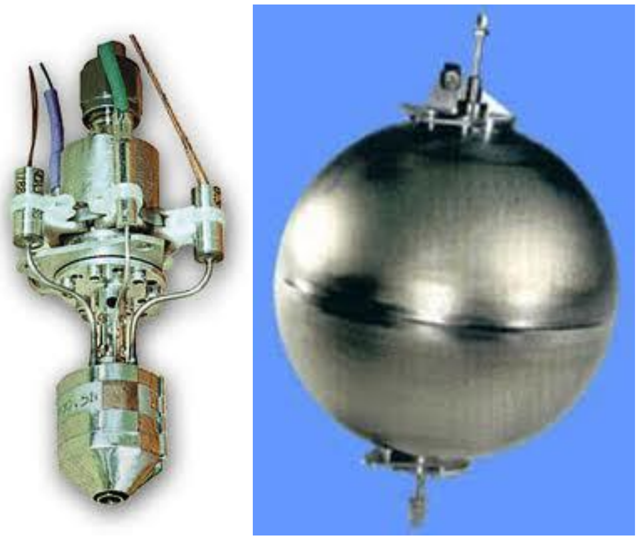


Figure 5 - 68 – Thruster and propellant Tank

5.3.2.4.4. Attitude Determination and Control Subsystem Sizing

The ADCS determines and controls the spacecraft's position in space and its attitude relative to the Earth. Such a subsystem is extremely important, as any other satellite subsystems, but gains particular importance when the payload requires a precise and stable pointing for accomplishing its mission.

The Attitude Determination and Control Subsystem needs to be provided with precise attitude sensors, to evaluate with a strong accuracy its orientation relative to the Earth (and to the acquisition area) and powerful attitude actuators for properly pointing the payload.

The fine pointing will be provided by star tracker, capable of guaranteeing the required level of accuracy; the device chosen is represented in the following figure, and the performances that it can guarantee provided in the table.

Figure 5 - 69 - RIGEL-L Star Tracker

RIGEL-L Star Tracker Specifications	
Dimensions/Mass	DPU: 155 x 210 x 45 mm, 1.2 kg CHU: 90 x 111 x 139 mm, 1.0 kg Baffle: 147 x 147 x 144 mm, 0.4 kg
Power	10 W at 28 V unregulated Supply 16-50 V
Pointing Accuracy (1σ)	<3.5 arcsec X-Y 25 arcsec around boresight
Operating bandwidth	1-16 Hz
Maximum tracking	6 deg/s
Exclusion angles (standard baffle)	Sun: 30° Earth: 24°
Interface	CAN / LVDS

Operating Temperature	-20 to +50°C
Random Vibration	DPU Qualified to 20 G_{rms} in all axis CHU Qualified to 20 G_{rms} in all axis
Radiation Tolerance	Radiation tolerant to 10kRad

Table 5 - 36 - RIGEL-L Star Tracker main specification

Table

A coarse value of the accuracy, necessary immediately after the satellite release from the launch vehicle, or when the satellite lost the attitude, will be assured by Earth sensors.

Figure 5 - 70 - Earth sensors

The satellite will be also equipped with a GPS receiver, capable to provide real-time data about position, velocity and time.

Figure 5 - 71 – GPS Receiver

SGR-10 Specifications	
Dimensions	160 x 160 x 50mm
Mass	950g
Power	5.5W at 28V (supply 18-38V)
Orbital Position (3-D)	Typical (95%):10m; Max (95%):20m
Orbital Velocity (3-D)	Typical (95%):0.15m/s; Max (95%):0.25m/s
Time	Typical (95%):0.5μs; Max (95%):1μs

Time to First Fix	Typical (95%): 50s (warm) or 200s (cold) Max (95%): 90s (warm) or 350s (cold)

Table 5 - 37 - GPS Receiver specification

The control of the satellite is guaranteed by four reaction wheels, that have to assure to the satellite the necessary agility and flexibility for pointing correctly the receiving antenna. In addition, the system needs to be provided with further actuators, that are necessary for giving the reaction wheels the possibility to be desaturated: three magneto-torquers has been chosen for such a purpose. Finally, a three-axes magnetometer is necessary in order to measure the components of the Earth magnetic field, to allow the magnetorquers to control the attitude.

5.3.2.4.5. Telemetry, Tracking & Control Subsystem

Basic requirement for TT&C subsystem is to provide a telecommunication link in S-band between space and ground segments; the information send in S-band are those related to spacecraft status, while operation and telecommand are uploaded from the Ground Station to the satellite through the same link.

As regards design constraints it is required to be able to guarantee link availability with ground station even in emergency situations. Downlink design has to guarantee a BER for telemetry rate up to 10^{-6}. Main driver in component choice is not to overcome system mass and power budgets.

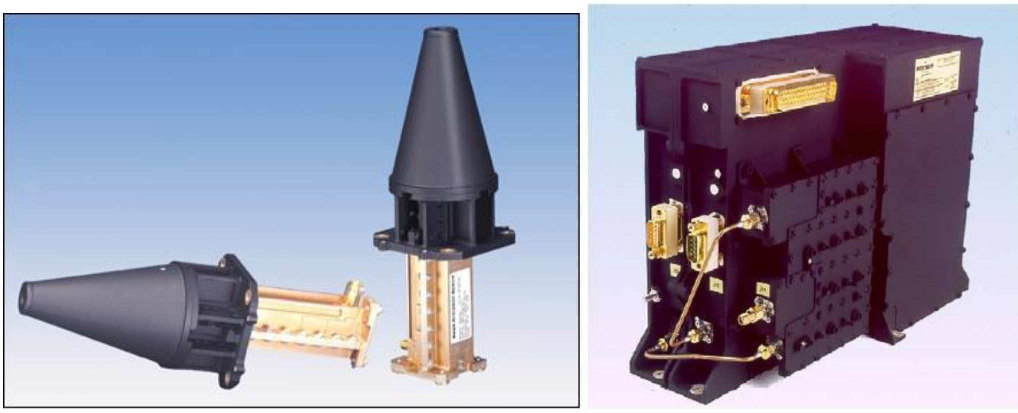


Figure 5 - 72 - S-band communication system

The communication subsystem consists of the following elements:

- Two Low Gain Antennas (LGA);
- RF Distribution Unit equipped with four microwaves switches and associated cabling;
- Two S-Band Standard transponders with coherency and ranging capabilities, including:
    - Diplexer
    - Transmitter
    - Receiver

The transmitters shall operate in cold stand-by and the receivers in hot stand-by.

Unit	Unit Name	Quantity	Mass per quantity excl. Margin [kg]	Margin [%]	Total Mass incl. margin [kg]	Power [W]
1	S-Band Transponder	2	3,00	5	6,300	6 – 26
2	Radio Frequency Distribution Unit	1	2,00	5	2,100	2
3	S-Band Helix Low Gain Antenna	2	0,24	5	0,504	0
SUBSYSTEM TOTAL			**8,48**	**5,0**	**8,904**	**8 - 28**

Table 5 - 38 – S-Band Communication system characteristic

5.3.2.4.6. Data Handling Subsystem

Data Handling subsystem has the main task of managing AOCS information, Payload scientific data, and transmit them. The sizing of spacecraft computer analysing primarily AOCS functions gave the following results:

<table>
<tr><th></th><th>Requirements</th><th>Component Chosen</th></tr>
<tr><td>Code Memory – ROM (Kbit)</td><td>1542,4</td><td>2048</td></tr>
<tr><td>Data Memory – RAM (Kbit)</td><td>1059</td><td rowspan="2">16000000</td></tr>
<tr><td>Telemetry Memory – RAM</td><td>1600</td></tr>
<tr><td>Throughput (MIPS)</td><td>2,04</td><td>4</td></tr>
</table>

Table 5 - 39 – Data Handling performances required

Moreover data handling equipment will manage other units such as thermal and power components, however these are not considered in this work because the requirements for AOCS are preponderant. Telemetry and telecommand data handling requirements are taken into account. The device I decide to use for accomplishing these tasks was the BAE RAD6000, shown in the following figure.

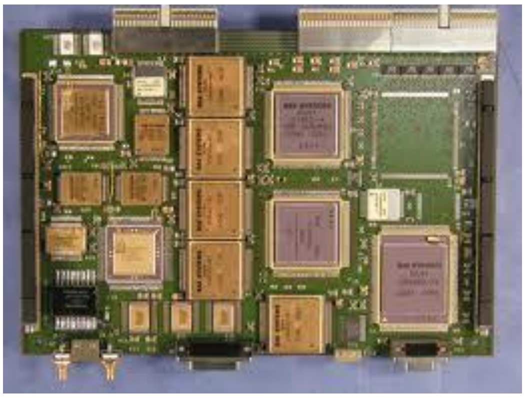


Figure 5 - 73 - BAE RAD6000

The most demanding performance that such a subsystem has to guarantees, is the capacity of storing the high amount of payload data that the Synthetic Aperture Radar acquire; as a benchmark, any of the four COSMO-SkyMed satellite has a mass memory devoted to payload data storage of about 320 Gbit, capable of storing only a limited amount of images (according to the kind of acquisition technique (ScanSAR, Stripmap, or Spotlight). Each satellite can acquire, daily:

- Up to 75 SPOTLIGHT images
- Up to 375 STRIPMAP or 150 SCANSAR images

for a total of 140GB per day that a single satellite can acquire each day.

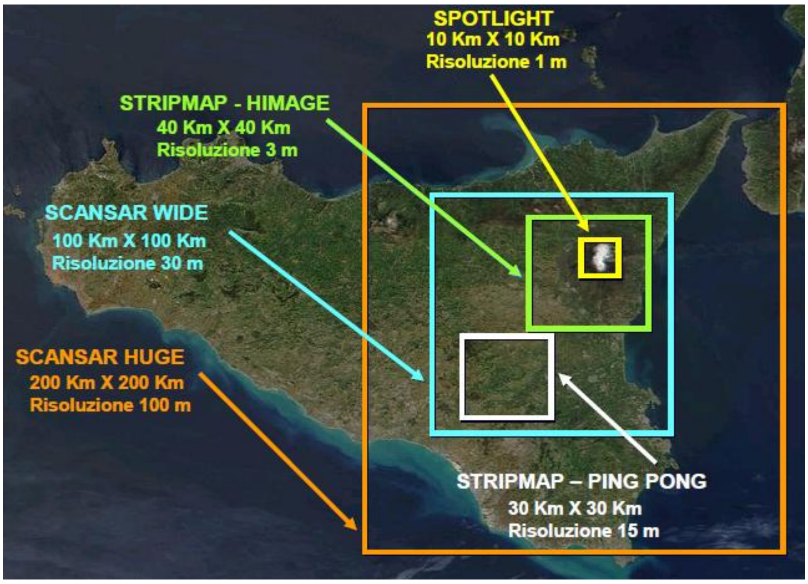


Figure 5 - 74 – Comparison between COSMO-SkyMed acquisition modes

The dimension of the mass memory that we decide to install on board the microsatellite will affect the performances of the interferometric application of superresolution imagery: the larger the memory, the larger the area where superresolution technique can be acquired. However, the mass and the volume of the components to mount on board of microsatellite are drastically limited, so the dimension of the mass memory has been evaluated making a trade-off between the will of performing a service as good as possible, and the constraints of severe, stringent, limited mass budget. The decision felt of using a couple of modular High Speed Solid State Data Recorder (HSDR), each with 16 GBytes of storage capability, providing for each of the satellite an approximate 256 Gbit storage capacity.

Figure 5 - 75 - High Speed Solid State Data Recorder

The previous picture depict one of the HSDR module, while the table that follows contains the main characteristics of the whole storage equipment.

HSDR Specifications	
Dimensions	320 x 170 x 55 mm (half micro-tray)
Mass	1 kg
Power	15 to 50V, 5W standby, 15W peak
Storage Capacity	Modular 16 GBytes
Data I/O processing capability	> 5 Gbps
Data Input / Outputs	- 20 Low-Voltage Differential Signalling (LVDS) 150 Mbps inputs - 5 Serializers/Deserializers (SerDes) 1 Gbps inputs - 10 LVDS 150 Mbps outputs
Temperature	-20 to +50°C operating -30 to +60°C non-operating
Random Vibration	Qualified to 15 G_{rms} in all axis
Radiation Tolerance	Radiation tolerant to 10kRad

Table 5 - 40 – High Speed Solid State Data Recorder specifications

5.3.2.4.7. Power Subsystem

The electrical Power Subsystem (EPS) is the satellite subsystem devoted to the generation of electrical energy, the storage of such energy into devices able to release such energy when needed, and comprises all the equipment related to the management of such energy, like power distribution unit, and power regulation unit. In the following scheme, are illustrated the main elements composing the EPS, and their functional relationship:

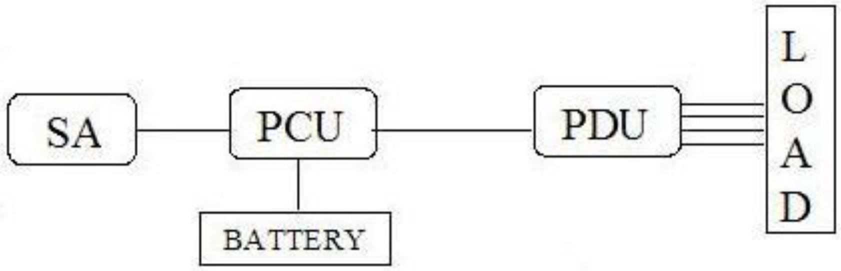


Figure 5 - 76 – EPS Scheme

Subsystem	Average Power Required	Peak Power Required
Payload	20 W (stand-by)	50 W (Imaging Mode) 50 W (Telemetry Mode)
Communications	8 W (stand-by)	28 W (Telemetry Mode)
OBDH	7 W	18 W (Imaging Mode) 18 W (Telemetry Mode)
ADCS	20.5 W	26 W

Table 5 - 41 – EPS requirements

As a storage device, the most widespread solution used for space application is the rechargeable battery, but also supercapacitor system can be used for the same purpose. Batteries (or Supercapacitors) are necessary for providing the additional electrical power required by the payload when it is working, both during the imaging mode, and the telemetry mode. The choice of having storage mechanism is necessary, as the microsatellites face eclipse periods; in addition, batteries are used during the illumination period in order to provide the additional power necessary to perform certain operation. Such peak power cannot be provided entirely by solar panels, that would have been larger if the target power to produce would have been set to be the peak power. Even if such value would have not been prohibitive, the reduced duty cycle of the Synthetic Aperture Radar would have set demanding constraints to Thermal Protection Subsystem that would have be required to dissipate a large amount of electrical power not necessary during the period the payload does not work; in addition, rechargeable batteries (or storage mechanisms in general) are required to provide the satellite with the necessary electrical power in case the it loses its attitude, entering in safe mode, and, consequently to the loss of attitude, directing the solar panel towards the deep space instead of the Sun. The battery has been chosen as storage mechanism, and in the following figure the model of battery is presented, together with its technical parameters.

Figure 5 - 77 – Lithium-ion battery cell MP176065

Electrical Characteristics	
Nominal Voltage (1.4A rate at 20°C)	3.75 V
Typical Capacity (at 1.4A 20°C 2.5V cut off)	6.8 Ah (when charged at 4.2 V) 6.1 Ah (when charged at 4.1 V)
Mechanical Characteristics (100% charged cells)	
Thickness max	19.6 mm
Width max	60 mm
Height max	65 mm
Typical weight	153 g
Lithium equivalent content	2.0 g
Volume	68 cm^3
Nominal energy	26 Wh
Operating Conditions	
Charge method	Constant Current / Constant Voltage
Charge voltage	4.20 ± 0.05 V
Maximum recommended charge current	6.8 A (C rate)
Charge temperature range	- 20°C to + 60°C
Time at 20°C	To be set as a function of the charge current: C rate - 2 to 3 h C/2 rate - 3 to 4 h C/5 rate - 6 to 7 h

Table 5 - 42 – Battery specifications

Advanced and performing solar cell has been chosen as energy generator for the microsatellite: in fact, the demanding payload (not only in terms of electrical power needed to be operated, but also for the stringent pointing accuracy that requires a sophisticated system for attitude determination and control) requires a significant capability for the EPS to generate large amount of energy.

If triple junction gallium arsenide on germanium (GaAs/Ge) solar cells are used – shown in the next figure – it is possible to reach an efficiency of energy conversion equal to $\eta = 28\%$.

As already done in the previous mission design, once the area to produce the necessary electrical power has been evaluated, such cells are to be properly connected in series with the aim to generate the required voltage.

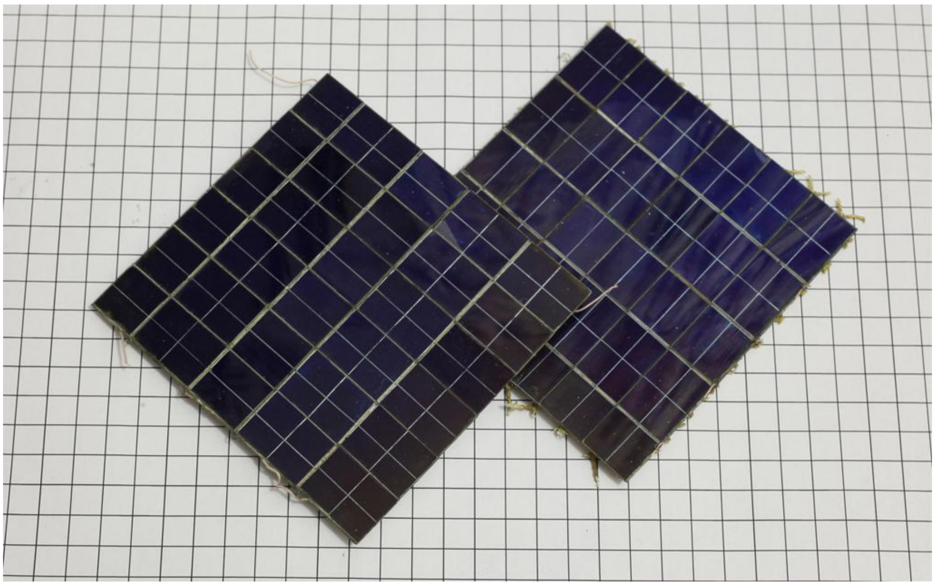


Figure 5 - 78 - GaAs/Ge solar cells

5.3.2.4.8. Thermal

Spacecraft thermal control is a process of energy management in which heating coming from the environment and on-board produced play a major role. The principal forms of environmental heating on Earth orbit are direct sunlight, sunlight reflected by Earth (albedo), and infrared (IR) energy emitted by Earth. The electronic components on board dissipate heating for several reasons.

The overall thermal control of an orbiting satellite is usually achieved by balancing the energy emitted by the spacecraft as IR radiation against the energy dissipated by its internal electrical components plus the energy absorbed from the environment.

The task of the thermal control subsystem is to maintain the spacecraft with its subsystems and payload components within their required temperature ranges during the mission. Two limits are frequently defined: operational limits, that the component must remain within while operating, and survival limits, that the component must remain within at all times, even when not powered. The following table gives typical component temperature ranges for spacecraft components.

Subsystem	**Operating Temperature**		**Non Operating Temperature**	
	Tmin (°C)	Tmax (°C)	Tmin(°C)	Tmax(°C)
Electronics	-10	50	-20	60

Batteries	15	25	10	30
Hydrazine Tanks and Lines	15	40	5	50
Antennas	-90	90	-110	110
Earth Sensors	-40	40	-40	40
Star Trackers	-30	50	-40	60
Momentum wheels	-25	60	-35	70

Table 5 - 43 – Components thermal requirements

The following assumptions have been used in the design process:

- The platform's components are supposed to be at the same temperature between 0 °C and 50 °C, in the nominal phase, and between –10 °C and 60 °C, in the safe phase;
- Only the external view factors between the platform and the solar array are taken into account: possible interaction between the internal units are neglected.
- The worst hot and cold case are considered in the nominal and safe phase for sizing the satellite thermal control;

The monopropellant propulsion system is prevented from freezing (2 °C) by a dedicated Multilayer insulation blankets (MLI), whereas the batteries are maintained within their operative range of temperatures (usually between 0 °C and 20 °C) using dedicated MLI, radiators and doublers.

Component Class	Used Component	ε_{EOL}	α_{EOL}	ε_{BOL}	α_{BOL}	Mass (kg)	Mass Margin (%)	Area [m^2]	Power (W)
MLI (21 layers)	Aluminium foil tape with 2 mil adhesive	0.04	0.17	0.04	0.15	0.505	20	-	0
	Y9360-3M aluminized Mylar	0.03	0.21	0.03	0.21				
Radiator	Silverized fused silica optical	0.0007	0.003	0.0007	0.0027	0.088	20	0.22	0

Table 5 - 44 – TCS specifications

The thermal design is mainly based on passive thermal control techniques, designing for keeping the worst case hot below the maximum operating temperature, and using heaters when the temperature drop underneath the minimum operative: the appropriate radiating area is designed for the maximum dissipation of the spacecraft; MLI blankets (21 layers) cover all other external surfaces of the spacecraft wall panels; the external side consists of Aluminized foil tape with 2 mil adhesive (2 mil), whereas the internal side consists of Y9630-3M aluminized Mylar (1 mil); The usage of radiators with a little area (0.22 m^2) allow to maintain the temperature within the limit; temperature sensors are used to monitor the temperature of each payload elements and subsystem device.

The required accuracy in temperature estimation is fixed in 1°C. Honeywell's HEL-700 Platinum RTD sensor are mounted. Thermofoil^{TM} heaters by Minco will be controlled by the OBDH systems to ensure that none of the subsystems or payloads cool excessively off,. Such heaters can safely run at wattages twice those of their wire-wound equivalents. Thermofoil^{TM} heaters transfer heat more efficiently, over a larger surface area, than round wires.

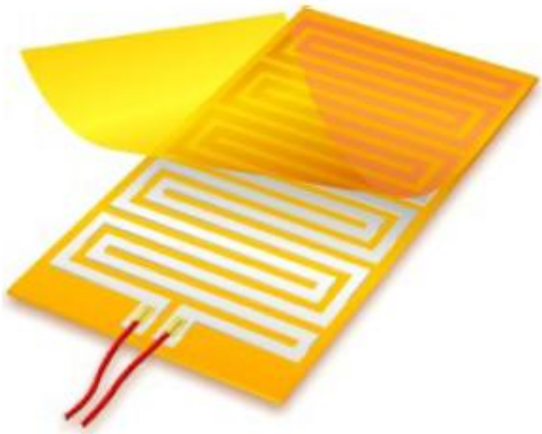


Figure 5 - 79 – ThermofoilTM heaters by Minco

5.3.2.4.9. Structures

Structures and Configuration are two closely connected subsystems. The choices adopted for one subsystem influence the analysis of the other one.

The main configuration and structure requirement is the accommodation of the spacecraft in the chosen launcher. Design drivers for the configuration are:

- Available volume in the chosen fairing:
- Baseline: Vega long fairing (cylinder part d= 2.2 m, h=5.5 m)
- Optional launchers : Rockot, Long March 2C, DNEPR, Soyuz-S. All these launchers have available dimensions larger than Vega. So the latter represents the stronger constraint for the stowed configuration.
- Structural - mechanical requirements of the spacecraft during mission lifetime
- Thermal requirement of the spacecraft elements
- Pointing direction and field of view of SAR Antenna.

The spacecraft shape is a cube with a square side of 0.7 m of side: 4 lateral closure panels withstand to the axial and lateral loads. These items are made with sandwich panels manufactured with honeycomb core in Aluminum. So the skins withstand all the axial loads while the core supply enough resistance to the lateral loads. Top and bottom panels, providing a mounting surface for the SAR payload, complete the structure. These items are made with sandwich panels manufactured with aluminum Alloy. They have a core thickness larger than the other elements since their main load is longitudinal one. So not only the skins withstand the load but especially the core. Moreover, the bottom platform has a bigger thickness, because of the attachment to the adapter. The latter is mounted to the spacecraft by means bolted attachments which form a “load paths” to the top of the spacecraft for the required stiffness during launch.

Payload module and service module are divided by a platform which supply a thermal insulation between the two environments and provide lateral stiffness to the structure. It is manufactured with sandwich panel of Aluminum alloy.

In addition to these elements it’s important to underline the requirements to gain good joints between the items in order to assure stability and prevent dangerous gaps. So bolted and welded joints add a strong contribute to the baseline design.

Preliminary design has been performed considering a Safety Factor equal 1.4; Margin of Safety is of 10%, in order to supply an efficient preliminary design, has been considered.

MASS BUDGET AND BASELINE SIZING									
Item	**# items**	**Material**	**Mass per each Item (kg)**	**Mass with margin (kg)**	**Mass Margin (%)**	**Height (mm)**	**Length (mm)**	**Width (mm)**	**Thickness Skin or Bulk (mm)**
Closure Panel	4	Al	3,58	3,7585	5	700	700	0	0,5
Platform (top)	1	Al	2,754	2,8917	5	0	700	700	0,250
Platform (bottom)	1	Al	2,754	2,8917	5	0	700	700	0,250
Total Mass of Structure (kg)			**22,57**	**23,7**	**5**				

Table 5 - 45 – Structures Subsystem

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REFERENCE

[1] A. Cawthorne, D. Purll, S. Eves, “Very High Resolution Imaging Using Small Satellites”, 6th Responsive Space Conference, AIAA, RS6-2008-4007

[2] http://www.sstl.co.uk/Products/Subsystems/Available_Subsystems

[3] D.M. Surka, M.A. Paluszek, S.J. Thomas, “The Development of a Low Cost, Modular Attitude Determination and Control System”, 11th AIAA/USU Conference on Small Satellites, SSC97-VIII-5

[4] N.S. Fatemi, H.E. Pollard, H.Q. Hou, P.R. Sharps, “Solar Array Trades between Very High-Efficiency Multi-Junction and Si Space Solar Cells”, 28th IEEE PVSC, 2000 September 17-22, Anchorage, Alaskaù

[5] - S.Cawley, “TopSat: lowcost high-resolution imagery from space”, *Acta Astronautica 56 (2005), pp. 147 – 152*

[6] – The International Telecommunications Union's, “Technical Characteristics for a Universal Shipborne Automatic Identification System Using Time Division Multiple Access in the Maritime Mobile Band”, ITU-R Recommendation M.1371-3.

[7] – Hoye et al., "Space-based AIS for global maritime traffic monitoring", pre-print from 5th IAA Symposium on Small Satellites for Earth Observation, Apr. 4-8, 2005, Berlin.

[8] – M.A. Cervera and A. Ginesi, *Satellite Based AIS Feasibility Analysis*. ESA-ESTEC, Noordwijk, the Netherlands, May 2008

[9] - RADARPLUS® SM1610-2 data sheet. http://www.shinemicro.com/docs/SM1610-2_050410.pdf

[10] – James K.E. Tunaley, “*Space-Based AIS Performance*”

[11] - Hennepe(1), R. Rinaldo(2), A. Ginesi(2), C. Tobehn(1), M. Wieser(1), Ø. Helleren(3), O. Olsen(3), F. Storesund, R. Challamel(5), L. De Vos, “*Deployment Of A Small Satellite Constellation For Space-Based Detection Of Ais Signals”* Pestana Conference Centre – Funchal, Madeira – Portugal 31 May – 4 June 2010 F.

[12] – J.G. Walker, “Satellite Constellation”, Journal of the British Interplanetary Society, Vol.37, No.12, Dec. 1984,pp.559-572.

[13] – J.G. Walker, “Continuous Whole-Earth Coverage By Circular-Orbit Satellite Patterns”, Royal Aircraft Establishment, Technical Report 77044, September 1977

[14] R.D. Lüders, Satellite Networks for Continuous Zonal coverage, American Rocket Society Journal, Vol. 31, February 1961, pp. 179-184;

[15] A. Lang, A Comparison of Satellite Constellation for Continuous Global coverage, International Workshop on Mission Design and Implementation of Satellite Constellation, Toulouse, France, Nov. 1997, pp.51-62

[16] – N.G. Orr, J.K. Eyer, B.P. Larouche, R.E. Zee, *"Precision Formation Flight: The CanX-4 and CanX-5 Dual Nanosatellite Mission"*, 21st AIAA USU Conference on Small Satellites

[17] – Curlander, J.C. Synthetic Aperture Radar: Systems and Signal Processing. Wiley, 1991.

[18] – Ariane Structures for Auxiliary Payload User's Manual.

[19] – Freeman, A., Johnson, W.T.K., Huneycutt, B., Jordan, R., Hensley, S., Siqueira, P., Curlander, J. The "Myth" of the Minimum SAR Antenna Area Constraint. IEEE Transactions on Geoscience and Remote Sensing, January 2000.

[20] – Aguttes, J.P. High Resolution (metric) SAR Microsatellite, Based on the CNES MYRIADE Bus. International Geoscience and Remote Sensing Symposium, 2001.

[21] – http://science.nasa.gov/science-news/science-at-nasa/2007/06jul_astroandnextsat/

[22] – M. Martin et all, "*TechSat 21 and Revolutionizing Space Missions Using Microsatellites*", 15th USU/AIAA, Logan, Utah, US, Aug 2001.

[23] – Hellwich, O. SAR Interferometry: Principles, Processing, and Perspectives. In: C. Heipke and H. Mayer (eds.), Festschrift fur Prof. Dr.-Ing. Heinrich Ebner zum 60 Geburtstag. Technische Universitat, Munich, Germany, pp. 109-120, 1999.

[24] – Breit, H., Eineder, M., Holzner, J., Runge, H., Bamler, R. *Traffic Monitoring using SRTM Along-Track Interferometry*. International Geoscience and Remote Sensing Symposium, July 2003.

[25] – Catani, F., Farina, P., Moretti, S., Nico, G., Strozzi, T. *On the application of SAR interferometry to geomorphological studies: Estimation of landform attributes and mass movements*. Geomorphology, vol 66, 2005.

[26] – Fornaro, G., Pascazio, V., Schirinzi, G., Serafino, F. *Improved resolution image focusing by multi-pass ENVISAT ASAR data*. In Proc. Fringe 2005 Workshop, Frascati, Italy, November 2005.

[27] – Ludwig, R., Schneider, P. *Validation of digital elevation models from SRTM X-SAR for applications in hydrologic modeling*. ISPRS Journal of Photogrammetry and Remote Sensing, vol 60, 2006.

[28] – Romero, R., Sanz-Marcos, J., Carrasco, D., Moreno, V., Valero, J.L., Lafitte, M. *SAR superresolution change detection for security applications*. In Proc. Envisat Symposium 2007, Montreux, Switzerland, 2007.

[29] – Massonnet, D. Capabilities and Limitations of the Interferometric Cartwheel. IEEE Transactions on Geoscience and Remote Sensing, vol 39, no 3, March 2001.

[30] – Krieger, G., Fiedler, J., Mittermayer, J., Papathanassiou, K., Moreira, A. Analysis of Multistatic Configurations for Spaceborne SAR Interferometry. IEE Proceedings Radar, Sonar and Navigation, vol 150, no 3, June 2003.

[31] – http://smsc.cnes.fr/PLEIADES/Fr/PDF/Interferometrie.pdf

[32] – M.E. Cannon, Q.Marji, "Relative navigation for satellite formation flying" University of Calgary, Department of Geomatics Engineering, October 2006

6. Conclusions

In this Ph.D. thesis, a deep and extensive investigation over the performances that highly technological microsatellite can exhibit has been carried out; in fact, based on a conception completely different from the traditional concept, microsatellite has to be seen as platform able to carry out space missions for which the use of traditional large, expensive satellite are not profitable: some of these missions have been investigated, together with the reason for using microsatellites.

The new conception of microsatellite is based on three fundamental pillars:

1. Technology innovation;
2. Responsive Space capability;
3. Tailored missions.

The presence of at least two of these pillars is an enabling factor for the implementation of the novel concept of microsatellite.

Microsatellites are space system that has been extensively used, with a number of these platform launched over the years that raised constantly from the beginning of the 1980s. Such platform reveals to be really profitable for technological demonstration, for testing the behavior in space of newer devices, and was often manufactured making an extensive use of commercial-off-the-shelves components, especially devoted to terrestrial application. As no particular mission were designed for such platform, there were no need for developing tailored payload, nor to adapt existing ones for this class of satellite. The absence of a demanding payload installed on board, and the consequent absence of a real space service to provide lead to a situation where such platforms do not impose particular requirements or constraints over the orbits; for this reason, the launch occurred always as piggy-back of other platforms, with the final orbits imposed by the primary launcher payload. In addition, microsatellites cannot be launched in a particular orbits, as a way for a dedicated launch does not presently exists.

In this framework, the new concept of microsatellite proposed in this thesis has the objective of drastically changing the vision for this platform: through the three pillars illustrated, a new dimension can be provided to microsatellite.

The use of technology innovation, not only intended as novel and more powerful devices, but also as methodologies for rapidly design space platform, can be of outmost importance for the diffusion of microsatellite; innovative devices, that will hardly be mounted on board of larger satellite for many reasons, the most important of which known as legacy, the lack of heritage in space mission, can help to drastically raise the performances that even really small satellite can be equipped with.

The implementation of responsive capability, ranging from the capability to develop in a timeframe in the order of months, or even weeks, a space system, until the development of a system for launching these platforms with a dedicated and responsive method, can provide time-to-delivery performances that a satellite weighing more than 500 kg will hardly be capable of reaching; such performances can enable envisioning and performing typology of mission that are currently out of space users' mind.

Finally, the identification of a set of mission that microsatellite (or a constellation or formation of microsatellites) can perform at reasonable cost, and with a significant level of reliability and confidence, comparable to those typical of larger space systems. These mission can be either responsive mission, where the constraints over the time-to-delivery performances are insurmountable limits for any other class of spacecraft apart the microsatellite, or mission where a distributed network of sensors are necessary for providing the service, being these sensors, in any case, small devices with requirements compatible with the characteristic of a microsatellite.

In fact, the mission of these microsatellites needs to be properly implemented, evaluating all the possibilities that such platforms can provide, but also bearing in mind that, as widely underlined in many instances, they cannot absolutely substitute larger platform, and the performances that these satellite can provide: value of several kW of electrical power produced, lifetime of 10 or more years, ability to assure an operative capability close to 100%, and so on are performances that microsatellite are not able to reach in the short, but probably also in the medium, period.