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# TOWARDS THE AUTOMATION OF CONCURRENT SPACE SYSTEMS CONCEPTUAL DESIGN THROUGH MULTIDISCIPLINARY DESIGN OPTIMIZATION

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***Abstract.*** *The development of space systems is a complex multidisciplinary endeavor. It is carried out in a sequence of phases, which comprises the main activities of conception, development and operation, including closeout. In the conception phase, the construction of engineering candidate solutions for the mission architecture elements, such as the payload, spacecraft bus, orbit and ground segment, is carried out by a team of engineers with the main aim of finding at least one viable candidate, in terms of performance, cost and schedule, which meets the mission primary objectives. In recent years concurrent engineering methods have been applied very successfully for decreasing the time, and at the same time increase the technical quality, when building conceptual solutions during Phase 0 of the development of new space systems. One of the key components of such approach is the use of discipline models integrated in such a way that information generated by a discipline is delivered to others that depend on that information as soon as it is available. These models may be codified in tools such as spreadsheets, in house or commercial software packages, data bases, or a combination of them. The use of optimization methods coupled to the discipline models can further improve the efficiency of building the candidate conceptual solutions, either in time as well as in quality. In the first case by making automatic the process of generating candidate solutions, and in doing so, in the second case, significantly improving the exploration of the design space, and hence the probability of finding better solutions. In this paper the potential of such approach is investigated through a case study where design parameters of the payload, propulsion subsystem and orbit, such as spacecraft mass, orbit inclination and height, camera field of view, spatial resolution, revisiting time and propellant mass, of a hypothetical Earth observation spacecraft, are taken into account simultaneously.*

***Keywords:*** *Concurrent engineering, Multidisciplinary design optimization, Space mission design, Spacecraft design , Optimization*

## 1. INTRODUCTION

Concurrent engineering have been used for space system conceptual design since the mid-1990's, with the establishment of the Project Design Center (PDC), or TeamX (Warfield, 2012), in the Jet Propulsion Laboratory (JPL) of NASA. Design centers such as the PDC were built in other government space organizations such as the European Space Agency ESA (ESA, 2015) and German Space Agency DLR (Braukhane, 2011), and also in industry and even academia (ESA, 2015). The proliferation of these facilities is the result of the recognition of significant gains of time and system quality, by using a concurrent approach during the design process. In fact, reductions on design time from months to weeks were reported (ESA, 2015). The main reason for such achievements is that in a concurrent environment engineers from all disciplines pertinent to the system, work together in an integrated environment such that the system is designed almost in "real time". In a traditional approach, meetings are used to present and discuss results, after analysis made offline by the discipline specialists. In a concurrent approach all specialists participate in, or are aware of, all aspects of the design while it is evolving, what leads to significant reduction in information latency and improvement of systemic awareness. Much of the analysis work is performed during team meetings, so changes in the design, and their impact in the system performance, can be assessed very quickly by all team members.

The application of numerical optimization methods to the design of complex multidisciplinary problems is the realm of a field of knowledge which became known as Multidisciplinary Design Optimization (MDO). Many MDO methodologies have been developed to address more efficiently specific kind of problems (Martins and Lambe, 2013). The use of MDO in a concurrent design environment may lead to more gains in designing time, by making automatic the process of generating candidate solutions, at least for some parts of the system, and also improve the quality of the solutions created, with a better exploration of the design space.

Early examples of investigations on the use of optimization methods for spacecraft conceptual design are the works of Fukunaga *et al.* (1997), Mosher (1999), and Taylor (2000). Fukunaga *et al.* (1997) proposed a general framework for integrating different disciplines in the optimization process using heuristics. Mosher (1999) proposed an Excel® based numerical tool for spacecraft concept system design based on the selection of components of the power, propulsion and structure subsystems, and launch vehicle. This study showed the potential of design trade exploration provided by automating the process of spacecraft concept design with optimization methods, which was also confirmed by the work of Taylor (2000). Increasing computation power over recent years and the availability of optimization software packages, which can be integrated to CAE and CAD tools, has made appealing the applicability of MDO to spacecraft design, for example, for the layout of equipment (Pühlhofer *et al.*, 2004; Zhang *et al.*, 2008; Lau *et al.*, 2014; Cuco *et al.*, 2015). In fact, it has been shown that the design of many spacecraft subsystems, with several design variables, can be tackled simultaneously with MDO techniques (Wu *et al.*, 2013; Hwang *et al.*, 2014)).

A characteristic present in the actual conceptual design of spacecraft is the fact that some subsystems are design based on the selection of existing components, or sets of these (Hassan, 2004). For example, type of solar cells, batteries, structure, sensors and actuators, etc. These components are not usually designed from scratch, rather, they are selected from a set of available commercial hardware. Though built over selected commercial components, these subsystems must operate complying with a required functional performance. Parameters such as orbit design, pointing accurancy for optical payloads and propulsion sizing, usually drive the selection of subsystem components. The applicability of MDO in practical spacecraft design must take the dynamics of system designing, as performed by the engineering team, into account. It also may be more effective when applied to disciplines that are more coupled to each other, rather than trying to apply it to design the whole spacecraft.

One of the objectives of a R&D project (PJESOPrOM) being carried out at the Space Systems Division (DSE) of the Brazilian National Institute For Space Research (INPE) is to investigate the use of optimization techniques to improve the performance of system design on a concurrent environment. As a result of case studies done in the context of PJESOPrOM, a tool for rapid concept orbit optimal design has already been developed (Chagas *et al.*, 2014). In this paper other design parameters are coupled to this tool, such that it can address simultaneously design requirements applied to orbit, an optical payload and a satellite's propulsion subsystem.

## 2. PROBLEM DEFINITION

In a previous work (Chagas *et al.*, 2014), a tool called OrbGEO was developed to solve the following problem:

- Given an Earth observation device, an analysis interval, and a region of interest, search the satellite constellations that **minimizes** the number of spacecraft and the mean orbital altitude and **maximizes** the accessible area (or **minimizes** the not accessible area).

It can be seen that these optimization objectives are conflicting, since a smaller number of satellites at lower altitudes clearly decreases the accessible area (Chagas *et al.*, 2014). Thus, it is expected that the Pareto frontier will not collapse into a single point. Instead, it will be a set of points in which the designer should choose the optimal solution for each case. OrbGEO was the tool used to solve this problem using the Multi-Objective Generalized Extremal Optimization Algorithm (M-GEO) (de Sousa *et al.*, 2003) and plot the Pareto frontier.

It is known that the payload complexity is decreased for lower orbits. However, the satellite total mass, which roughly defines the total mission cost, tends to increase exponentially for very low orbits due to the amount of fuel needed to counterbalance the air drag force (Wertz, 1978). Thus, in this work, it is proposed to extend OrbGEO algorithm to minimize the satellite total mass instead of the orbit altitude. Hence, the problem to be solved is:

- Given an analysis interval, and a region of interest, search the satellite constellations that **minimizes** the number and the total mass of spacecraft and **maximizes** the accessible area (or **minimizes** the not accessible area).

In the early phases of the mission analysis process, it is not possible to obtain with high accuracy the satellite mass. However, it is, indeed, possible to obtain an estimated value based on previous missions (Wertz *et al.*, 2011). It was verified that the satellite dry mass, which is the satellite mass without the propellant, has a high dependency on the payload total mass (Wertz *et al.*, 2011). On the other hand, the payload total mass will depend highly on the orbit, given a specific mission (Wertz *et al.*, 2011).

Finally, if it is possible to find a relation between the satellite orbit, payload mass, and the satellite total mass, then OrbGEO can be extended to compute the Pareto Frontier of the aforementioned optimization problem. Thus, in the following sections, the models for the disciplines orbit analysis, payload, and propulsion are presented. It is verified that these three disciplines can provide a first estimate of the satellite total mass.

## 3. MODELS

In this section, it is presented the models for the orbit analysis, payload, and propulsion.

### 3.1 Orbit Analysis Model

The orbit analysis model is responsible to provide the coverage of the region of interest by the optical payload and total velocity variation $\Delta V_t$ budget that the satellite's propulsion subsystem shall be capable to provide. The later will be used to compute the fuel mass.

The algorithm to compute the coverage of the region of interest is the same as the one presented in Chagas *et al.* (2014).

The $\Delta V_t$ budget is defined as the amount of change in the orbital velocity that the spacecraft must apply to 1) transfer itself from the injection orbit to the nominal orbit, 2) maintain the orbit altitude during its lifetime, and 3) transfer the satellite from the nominal orbit to the disposal orbit. Each phase is described in detail as follows.

#### 3.1.1 Orbit Acquisition

The launcher has errors and consequently will not place the satellite on the desired orbit. The orbit in which the launcher injects the spacecraft is called **injection orbit**. Since it is not desired to remove energy of the orbit, it is usually specified an nominal injection point such that it would be very unlikely that the semi-major axis of the injection orbit is larger than that of the nominal orbit (Wertz *et al.*, 2011). For the sake of simplicity, in this work both injection and nominal orbits have been considered circular.

There are two types of maneuvers that shall be executed to place the satellite on the mission or nominal orbit (Wertz, 1978). The first is the in-plane maneuver in which the semi-major axis is corrected and the second is the off-plane maneuver in which the inclination is corrected (Wertz, 1978).

To compute the velocity variation to correct the semi-major axis, notice that the velocity of the spacecraft considering a Keplerian orbit is given by (Wertz, 1978)

$$V_{a,r} = \sqrt{\mu\left(\frac{2}{r} - \frac{1}{a}\right)} \; , \tag{1}$$

where $\mu$ is the Earth gravitational constant, $a$ is the semi-major axis of the orbit, and $r$ is the norm of the satellite position vector. Hence, assuming that the launcher will place the spacecraft on a circular orbit with radius $r_i$, the transfer to the nominal orbit with radius $r_n$ is optimally accomplished by two burns (Wertz, 1978):

$$\Delta V_1 = V_{a_t,r_i} - V_{r_i,r_i} \; , \tag{2}$$

$$\Delta V_2 = V_{r_n,r_n} - V_{a_t,r_n} \; , \tag{3}$$

where $a_t$ is the semi-major axis of the Hohmann transfer orbit, which is given by (Wertz, 1978)

$$a_t = \frac{r_n + r_i}{2} \; . \tag{4}$$

The off-plane maneuver to correct the inclination is accomplished by a single burn normal to the orbital plane at the ascending or descending node (Wertz, 1978). The velocity variation can be computed by

$$\Delta V_3 = 2 \cdot V_{r_n,r_n} \cdot \sin\left(\frac{|\Delta i|}{2}\right) \tag{5}$$

in which $\Delta i$ is the inclination error.

Finally, the total $\Delta V$ budget for the orbit acquisition phase can be computed as

$$\Delta V_{oa} = \Delta V_1 + \Delta V_2 + \Delta V_3 \; . \tag{6}$$

#### 3.1.2 Orbit Maintenance

Remote sensing mission with optical payloads are placed on a Low Earth Orbit (LEO), which has altitudes from 400 km up to 1200 km (Wertz *et al.*, 2011). In these orbits, the atmospheric density applies a drag force to the satellite that removes energy from the orbit and, consequently, decreases the orbit altitude (Vallado and McClain, 2004). Thus, the satellite shall be capable to apply a force to circumvent this effect. For near-circular orbits, the velocity variation necessary to counterbalance the drag force for the entire mission lifetime can be computed using (Wertz *et al.*, 2011)

$$\Delta V_{om} = \pi \cdot \frac{C_d \cdot A}{m_t} \cdot \rho(h) \cdot r_n \cdot V_{r_n,r_n} \cdot \frac{T_m}{T_{orb}} \; , \tag{7}$$

where $C_d$ is the drag coefficient assumed to be 2.5, which is the worst case (Wertz *et al.*, 2011), $A$ is the cross-sectional area of the satellite in the direction of the velocity vector, $\rho(h)$ is the atmospheric density in the satellite altitude $h$, $T_m$ is the expected mission lifetime in seconds, and $T_{orb}$ is the orbital period in seconds, which is given by (Wertz, 1978)

$$T_{orb} = 2\pi\sqrt{\frac{r_n^3}{\mu}} \; . \tag{8}$$

#### 3.1.3 Deorbiting

There is an international concern about the number of debris in space (Wertz *et al.*, 2011), and today a kind of international rule says that every satellite in LEO must reenter the atmosphere within 25 years after the end of its expected lifetime. This can be accomplished by lowering the orbit perigee to $r_d$ in which its altitude must be less than 550 km. Hence, the velocity variation to transfer the satellite from nominal orbit to the disposal orbit is given by

$$\Delta V_{od} = V_{r_n,r_n} - V_{a_d,r_n} \tag{9}$$

in which $a_d$ is the semi-major axis of the disposal orbit that can be computed as (Wertz, 1978)

$$a_d = \frac{r_n + r_d}{2} \; . \tag{10}$$

#### 3.1.4 Total Velocity Variation

Finally, the total velocity variation for the entire mission lifetime can be computed as

$$\Delta V_t = \Delta V_{oa} + \Delta V_{om} + \Delta V_{od} \; . \tag{11}$$

### 3.2 Payload Model

The payload considered in this paper is a passive, multi-spectral, optical imaging instrument that captures images of the Earth surface in the visible and near infrared spectral range (VIS-NIR). It has been considered a push broom scanning technology to capture the images. This technology avoids moving parts that can make the instrument more complex, expensive, bigger, heavier and less reliable.

One of the most important drivers for the design of this kind of instruments is the image detector selected. For push broom technology in the VIS-NIR, the most common detectors used are linear CCDS or CMOS.

For the studies conducted herein, it has been selected a space qualified detector module that seems to be the best tradeoff in terms of number of pixels and pixel size. For a given fixed resolution and orbit height, the higher the number of pixels, the greater the strip on the Earth surface captured in a single satellite path. The smaller the pixels size, the smaller the focal length of the optical system for a given fixed resolution and orbit height, making the instrument size and weight also smaller. On the other hand, the smaller the pixel, the worse the image quality in terms of contrast and noise.

With this tradeoff in mind, it has been selected a linear CCD with 12000 pixels and $6.5 \times 6.5$microns pixel size. The butting of multiples detectors modules or multiple optical channels are used in some cases to increase the imaging strip, but this has not been considered here.

The payload model receives as input the orbit height and inclination and returns the instrument mass and field of view (FOV). For a given orbit height $h$ and desired ground resolution at Nadir $Res_N$, the instrument focal length is computed by

$$f = \frac{P_{size} \cdot h}{Res_N} \; , \tag{12}$$

where $P_{size}$ is the pixel size. In the sequence, it is possible to compute the instrument FOV as follows:

$$FOV = 2 \cdot \tan^{-1}\left(\frac{h_i}{f}\right) \; , \tag{13}$$

where $h_i$ is half of the detector sensitivity length, which is computed by multiplying the pixel size by half of the number of pixels in the detector. These equations are easily obtained by geometric relations.

The payload mass can be divided in two parts: the optical head mass ($m_{opt}$) and the electronics mass ($m_e$). Since the detector used in this instrument has been selected and it is not susceptible to change, then the electronics for the payload will not be dependent on the orbit height and inclination. Hence, it has been considered constant. In fact, the electronics clock varies with the orbit height and inclination due to the speed of the sub-satellite point on Earth's surface. However, considering the intervals of the orbit height and inclination suitable for the study proposed in this paper, these variations are small enough to assume that the electronics does not change.

The mass of optical head has been obtained by the method described in Wertz *et al.* (2011). This method consists of estimate the mass of the new instrument by scaling a "similar instrument". The scaling is performed based on the instrument optical aperture ration $R$ that is defined as

$$R = \frac{A}{A_0} \; , \tag{14}$$

where $A$ is the optical aperture of the instrument, which has been considered as function of the orbit height and inclination, and $A_0$ is the aperture of the reference instrument.

The instrument optical aperture controls the amount of energy admitted by the optical system and thus how much light reaches the image plane. For a given orbit height and inclination, the aperture $A$ is computed in order to keep the same signal to noise ratio of the reference instrument for the same target illumination and reflectance conditions.

The signal in the image plane is a function of the optical system f-number ($f/\#$) and integration time. The f-number is defined as the ratio of the optical system focal length to the optical aperture as in

$$(f/\#) = \frac{f}{A} \; . \tag{15}$$

The integration time is taken as half of the detector line time, which is the time needed for the sub-satellite point in Equator to displace the length of the resolution $Res_N$. This leads to the conclusion that the integration time is a function of both orbit height and inclination.

Thus, for a given orbit, the optical head mass $m_{opt}$ is then computed by

$$m_{opt} = K \cdot R^3 \cdot m_{opt,0} \; , \tag{16}$$

where $m_{opt,0}$ is the optical head weight of the reference instrument and $K$ is a scaling factor that should be 2 if $R < 0.5$ or 1 otherwise (Wertz *et al.*, 2011). Finally, the payload total mass can be computed using

$$m_p = m_{opt} + m_e \; . \tag{17}$$

The reference instrument used has been selected as a conceptual design of a camera payload that uses the same detector, has a ground spacial resolution of 20 m, and was designed for a specific orbit. The other parameters for this reference instrument are: $A_0 = 0.048576886$ m, $m_{opt,0} = 26.5$ kg, $m_e = 47.71$ kg. The conceptual design of this instrument was done based on the heritage INPE has in the development of this same kind of payloads.

### 3.3 Propulsion Model

Given the total velocity variation $\Delta V_t$ provided by the orbit analysis model and the satellite total mass $m_t$, the propellant mass $m_f$ can be obtained by the rocket equation as in (Wertz *et al.*, 2011)

$$m_f = m_t \cdot \left(1 - \exp\left\{-\frac{\Delta V_t}{g_0 \cdot Isp}\right\}\right) \; , \tag{18}$$

where $g_0 = 9.80665$ m/s is the gravity acceleration at Earth's surface and $Isp$ is the propellant specific impulse, which is 225 s for hydrazine.

### 3.4 Satellite cross-sectional area

The orbit analysis model needs the satellite cross-sectional area in the direction of the velocity vector to compute the air drag. However, in the early phases of the vehicle sizing, a cross-sectional area estimate may not be available. In the absence of any information related to this parameter, it is a common practice to use an area-to-mass ratio in the range 0.005 to 0.02 $m^2$/kg, which covers around 90% of relevant satellites in LEO (IPS, 1999).

The average cross-sectional area from the LEO satellites listed in the UCS Satellite Database is 10 $m^2$ (McManus *et al.*, 2007). A subset from this database relating the area to the satellites' dry masses is shown in fig. 1. Thus, the following relation is found

$$A(m_d) = 0.019646 \cdot m_d \; , \tag{19}$$

where $A$ is the cross-sectional area in $m^2$ and $m_d$ is the satellite dry mass in kg.

### 3.5 Satellite total mass

Notice that the satellite total mass is needed by the orbit analysis model. Furthermore, the output of the mission analysis model is the input of the propulsion model. On the other hand, the propellant mass, which is the output of the propulsion model, is one key parameter to obtain the satellite total mass. Thus, an recursive algorithm has been designed to provide the estimate of the satellite total mass given the orbit. The algorithm is described as follows:

- Given the orbit, obtain the payload total mass $m_p$;
- Obtain the satellite dry mass estimate using the parametric relation in Wertz *et al.* (2011): $m_d = m_p/0.71$;
- Obtain a satellite total mass estimate using the parametric relation in Wertz *et al.* (2011): $m_{t,0} = 1.27 \cdot m_d$;
- Compute the satellite cross-sectional area using the model in section 3.4 with the dry mass $m_d$;
- FOR $k = 1 : max$ DO
  - Compute the total velocity variation $\Delta V_t$ using the orbit analysis model using the total mass $m_{t,k-1}$;
  - Compute the fuel mass of the $k$-th iteration $m_{f,k}$ using the propulsion model;

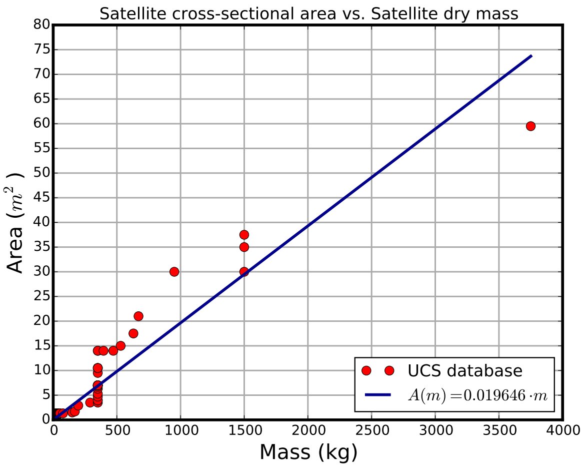


Figure 1. Satellite cross-sectional area vs. satellite dry mass

- Compute the new estimate for the satellite total mass: $m_{t,k} = m_d + m_{f,k}$;
- If $|m_{t,k} - m_{t,k-1}| < \epsilon$, stop.

• END FOR

The value of $\epsilon$ has been selected as $0.1$ kg and the value of $max$ has been selected as $100$. It has been verified that if the orbit altitude is higher than 360 km, then this algorithm converges. Otherwise, the stop condition is never reached and after each iteration the satellite total mass estimate is increased until $k = max$.

## 4. INTEGRATED MODEL

Theses models have been integrated in the OrbGEO algorithm. The evolutionary algorithm selects an orbit, then the satellite total mass and payload FOV is obtained using the algorithm described in this paper. Afterwards, the accessible area by the satellite is computed. Finally, the Pareto Frontier is plotted and the specialist can select optimal configurations for the mission.

The design variables available in this new OrbGEO version are the number of satellites and the 6 orbit parameters for each one of them. The objective functions are the satellite mass and the percentage of the area of interest that is not accessible during the analysis interval. For more information about OrbGEO and how the accessible area by an optical payload is computed, see Chagas *et al.* (2014).

## 5. RESULTS AND ANALYSIS

A hypothetical study case is presented here to shown the kind of results that is possible to obtain with the new tool. The scenario is similar to that studied in Chagas *et al.* (2014) but, instead to minimize the orbital altitude, it is desired to minimize the satellite total mass whereas the covered area of the region of interest is maximized. The region of interest for this study is the Amazonia rain forest and the mission is composed of a single spacecraft. For the sake of simplicity, only circular orbits were considered. The parameters of the simulation can be found in Table 1. Notice that the orbits were not constrained to be sun-synchronous in this study. Thus, the design variables are the semi-major axis, orbit inclination, right ascention of the ascending node (RAAN), and the true anomaly at the beginning of the simulation. The parameters that are not shown in Table 1 were chosen exactly the same as the ones in Chagas *et al.* (2014).

The results obtained from OrbGEO using a payload with ground spatial resolution of 20m and 30m can be seen in fig. 2. The MGEO algorithm, which was coded using C++ language and OpenMP for parallel computing, evaluated the objective functions **13,800,000 times** in less than **25 hours** in a computer with 2× Intel® Xeon® E5-2687W processors and 64 GiB of RAM. It is shown in Table 2 some selected points of the Pareto Frontiers that have a good trade-off between satellite mass and area not accessible. Hence, it is possible to conclude from the results:

- All the frontiers have a corner, thus it is easy to obtain a point with a good trade-off between area not covered and satellite total mass for each case.
- If the ground spatial resolution of 30m is acceptable, then only one satellite is capable to obtain images of almost all Amazonia rain forest within three days. Furthermore, the satellite total mass is decreased by 15.5% for 2 years of lifetime, 20.2% for 4 years of lifetime, and 20.8% for 6 years of lifetime comparing to the mass of the solutions obtained using a payload with a ground spatial resolution of 20m.
- All optimal solutions chosen in Table 2 were obtained when the inclination of the orbit is $14.2857°$, which is expected because the region of interest is the Amazonia rain forest.
- It can be seen that, in this case, the mass of a satellite designed to last 2 years is not much lower than that of a

Table 1. Input parameters for each simulated scenario.

| Parameter | Value |
|---|---|
| *Maximum number of generations in MGEO* | 100,000 |
| *Number of independent runs in MGEO* | 50 |
| *Interval of analysis* | 3 days |
| *Latitude interval of the region of interest* | [-34°, 6°] |
| *Longitude interval of the region of interest* | [-74°, -33°] |
| *Launcher error in the semi-major axis* | 20 km |
| *Launcher error in the inclination* | 0.01° |
| *Perigee of the disposal orbit* | 500 km* |
| *Camera resolution* | [20 m, 30 m] |
| *Expected mission liftime* | [2, 4, 6] years |
| *Atmospheric density* | Solar maximum** |

*: If the orbit radius is lower than 500 km, then a disposal maneuver is not needed. Thus $\Delta V_{od} = 0m/s$.
**: The atmospheric density in the solar maximum was obtained by the NRLMSISE-00 model using $F_{10.7} = 225$ and $AP = 20$ (Wertz *et al.*, 2011).

satellite designed to last 6 years. Thus, it is preferable to choose a solution that has an estimated lifetime of 6 years.
Finally, this hypothetical study case showed how the results obtained from the new version of OrbGEO are filtered to obtain candidate solutions for a specific mission. Theses solutions can then be used into the concurrent design facility to drive the design of the others subsystems.

Table 2. Selected solutions for the hypothetical study case.

| **Altitude** (km) | **Inclination** (°) | **RAAN** (°) | **True anomaly** (°) | **Lifetime** | **Area Not Accessible** | **Satellite Mass** (kg) |
|---|---|---|---|---|---|---|
| **Ground Spatial Resolution = 20 m** | | | | | | |
| 591.942 | 14.2857 | 11.4286 | 216.0 | 2 years | 11.5932 % | 194.388 |
| 686.430 | 14.2857 | 11.4286 | 264.0 | 4 years | 9.8955 % | 211.783 |
| 686.430 | 14.2857 | 11.4286 | 264.0 | 6 years | 9.8955 % | 217.269 |
| **Ground Spatial Resolution = 30 m** | | | | | | |
| 680.131 | 14.2857 | 308.571 | 96.0 | 2 years | 2.5447 % | 164.243 |
| 717.926 | 14.2857 | 320.000 | 72.0 | 4 years | 2.3550 % | 169.063 |
| 717.926 | 14.2857 | 320.000 | 72.0 | 6 years | 2.3550 % | 172.186 |

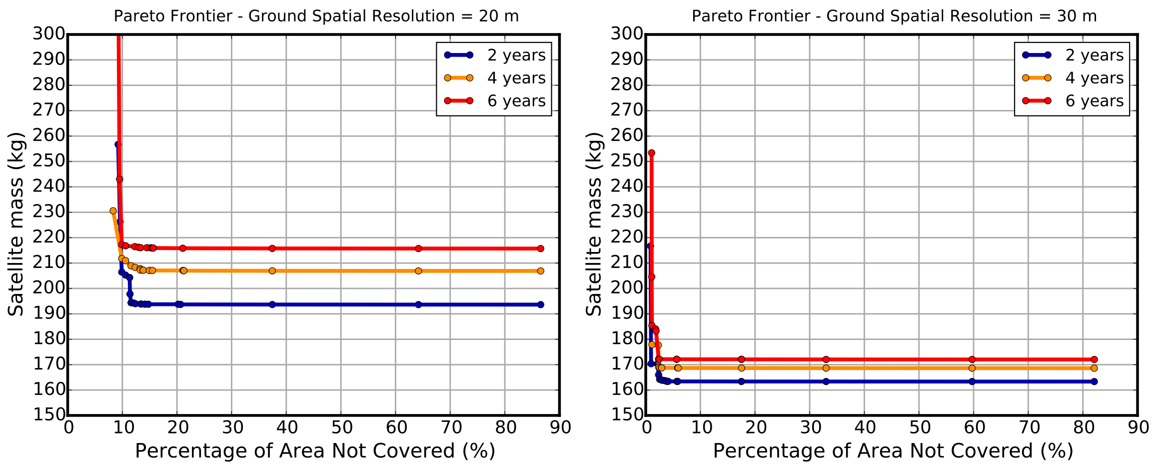


Figure 2. Pareto frontier for a payload with ground spatial resolution of 20m (left) and 30m (right).

## 6. CONCLUSIONS

In this work, the OrbGEO algorithm was extended to find optimal orbits for a given Earth observation mission, considering also the design of the propulsion subsystem and the satellite's payload. These two disciplines together with the orbit analysis model can be integrated to provide an estimate of the satellite total mass and the payload field of view.

A hypothetical study case was presented in this paper and it was verified that this algorithm provided very quickly candidate solutions for a test case space mission, since more than 13,800,000 evaluations of the objective functions were completed in less than 25 hours. Further improvements to OrbGEO include: 1) improve the algorithm to estimate the satellite cross-sectional area by adding the power as a input variable; 2) add other models, like power subsystem model, to this MDO algorithm; and 3) improve the payload model to provide a better estimate of the FOV and payload mass.

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