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Multidisciplinary optimization framework for satellite conceptual design

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Abstract. *In this paper a Multidisciplinary Optimization (MDO) framework is proposed to solve a satellite conceptual design problem. First the models for each discipline are explained in detail, followed by the architecture and its assembly, demonstrating the usefulness of this tool on solving space application optimization problems while being simple to implement with team interaction through the entire process.*

Keywords: Multidisciplinary Optimization; Satellite Design; Space Mission Concept.

1. Introduction

The concept design of satellites and its optimization has been real challenging to engineers, specially in early phases of development [Fliege et al. 2012, Wertz et al. 2011]. Proof of this is that the major part of the works found in the available literature that proposed tools to optimize the spacecraft design are devised for advanced stages of a space mission, in which the systems requirements and some equipment are already specified (e.g [Huang et al. 2014, Jafarsalehi et al. 2016, Wu et al. 2012]). However, it is also possible to use satellite optimization techniques in the very first phase of the project, also called pre-phase A or conceptual design phase, as done in [Chagas et al. 2014, Chagas et al. 2015]. In this case, there is little information about the mission and the design team must find feasible solutions based solely on the high-level mission specification provided by the stakeholders. Hence, MDOs[1] can be used to search the design space for optimal solutions, giving the design team a starting point for the analysis [Chagas et al. 2014].

MDOs integrate models of several disciplines and optimize, through numerical computation, the selected design variables chosen to achieve objectives, which are given as mathematical functions [Martins and Lambe 2013]. Finally, the goal of this paper is to show an implementation of an MDO applied to satellite concept design on early phases of development.

2. Models and Framework

Implementing an MDO requires to model the problem into smaller problems, non-multidisciplinary ones if possible. After creating these models for each discipline and defining

[1]For more information about MDO and its architectures read [Martins and Lambe 2013].

its variables, an analysis of how they interact together is conducted. In this stage it is defined the transition variables (variables shared between multiple disciplines), which later connect all the models together. Finally, insights about the most advantageous MDO architectures can be extracted.

Having the MDO architecture and a model to be implemented, the last step is to define an algorithm to be used as optimizer. The next subsections cover and exemplify all the stages previously described.

2.1. Problem Definition

Extending the aforementioned work of [Chagas et al. 2014, Chagas et al. 2015], a simplified case study is shown here in which a simple MDO is implemented to solve the following problem: **Given an Earth observation device, or camera, and a desired revisit, find the orbital parameters and satellite specification that minimizes the total system mass.**

2.2. Nominal Orbit Model

The Sun-synchronous orbits (SSOs) are the most used ones for Earth observation missions [Wertz et al. 2011]. A SSO is an orbit in which the gravitational perturbation precesses the right ascension of the ascending node (RAAN) by 0.9856473598948°/day [Wertz et al. 2011]. In this case, the orbit plane will ideally have the same orientation with respect to the Sun during the entire year.

The RAAN time derivative considering perturbation terms up to the Earth second gravitational zonal harmonic $J2$ is given by

$$\dot{\Omega} = -\frac{3}{2}\frac{R_0^2}{a^2(1-e^2)^2} n_0 J_2 \cos i, \tag{1}$$

where n_0 is the the non-perturbed orbit angular velocity, R_0 is the Earth equatorial radius, a is the semi-major axis, e is the eccentricity, u_0 is Earth standard gravitational parameter, and i is the orbit inclination.

Earth observation missions usually require that the satellite ground track repeats after a certain period, which is called revisit. This is accomplished by selecting the number of orbital revolutions per day as a rational number [Vallado and McClain 2007]. Hence, the camera can be designed so that an image of a target can be obtained periodically according to this revisit time.

Let the number of orbital revolutions in a day be

$$rev = I + \frac{Q}{D} \qquad I, Q, D \in \mathbb{Z}^+, \tag{2}$$

where $Q < D$. Notice that after D days, the satellite will complete $D.I + Q$ revolutions, which is an integer number, leading to the ground-track repetition.

A Sun-synchronous, ground repeating orbit (SSGRO) can be designed by finding a semi-major axis and inclination that provide the necessary RAAN time derivative, given by Equation 1, whereas the orbital angular velocity is given by

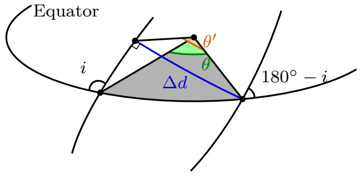
$$n_\mathrm{i} = rev\frac{2\pi}{86400}. \tag{3}$$

Finally, it can be shown that a SSGRO is obtained by selecting I, Q, D, and e, and simultaneously solving the following system of equations for the semi-major axis a and the inclination i

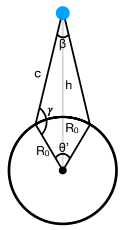
[Wertz et al. 2011]:

$$\begin{cases} 1.9910638534437197 \times 10^{-7} - \dot{\Omega} = 0 \\ n_{\rm i} - n_0 - \frac{3}{4}\frac{R_0^2 J_2}{a^2(1-e^2)^2} n_0 \{\sqrt{1-e^2}[3\cos^2(i) - 1] + 5\cos^2 i - 1\} = 0. \end{cases} \quad (4)$$

The next step is to find the minimum field of view (FOV) that the camera must have to obtain images of the entire world in the selected orbit every D days. The minimum FOV is the angular distance as seen by the satellite between two adjacent ground tracks on the Equator. Figure 1(a) shows the geometry of the problem, where θ' is the angle between two adjacent passages measured from the Earth center and considering the orbit inclination. In this case, the minimum swath width of the optical instrument, which is the length of the imaged area on the ground, is the distance Δd.



(a) Geometry of the distance between two adjacent ground tracks.



(b) Geometry for calculation of the minimum FOV.

Figure 1: Geometries of the orbit and the field of view of the satellite.

Since the orbit cycle is D days, then the satellite will cross the equator $D-1$ times between every two consecutive passages during the orbit period [Wertz et al. 2011]. Looking from the satellite point of view where β is the FOV of the instrument, shown in Figure 1(b), it follows that the minimum FOV can be computed by

$$FOV_{\rm min} = 2\sin^{-1}\left(\frac{R_0}{\sqrt{R_0^2 + (R_0 + h_p)^2 - 2R_0(R_0 + h_{\rm p})\cos\frac{\theta'}{2}}}\sin\frac{\theta'}{2}\right), \quad (5)$$

where h is the satellite altitude. If the selected orbit is eccentric, then one must use the orbit perigee altitude $h_{\rm p}$ to account for the worst-case scenario.

2.3. Orbit Acquisition, Maintenance and Disposal Model

Trajectory deviations occurs due to the injection accuracy of the launcher. This places the satellite into an erroneous orbit, called injection orbit (which will be considered circular), that needs to be corrected to the nominal orbit.

The spacecraft velocity is computed using [Wertz et al. 2011]:

$$V_{\rm a,r} = \sqrt{\mu\left(\frac{2}{r} - \frac{1}{a}\right)}, \quad (6)$$

where μ is the Earth gravitational constant, a is the semi-major axis of the orbit, and r is the norm of the satellite position vector. The maneuvers to correct the spacecraft orbit adjust the semi-major axis and the orbit inclination by changing the spacecraft velocity

[Wertz et al. 2011]. Two propellant burns are necessary to correct the semi-major axis. The first burn is executed at the angular position of the nominal orbit perigee so that the orbit apogee is placed on the desired location. This put the satellite into a transfer orbit, which is an intermediate, elliptical orbit used to transfer the satellite from the injection to the nominal orbit [Wertz et al. 2011]. The second burn shall be executed in the orbit apogee and places the perigee in the desired position. The velocity variation of the two burns is computed as follows [Wertz et al. 2011]:

$$\Delta V_1 = V_{\mathrm{at},\mathrm{r_i}} - V_{\mathrm{r_i},\mathrm{r_i}}, \tag{7}$$

$$\Delta V_2 = V_{\mathrm{a_n},\mathrm{r_{p_n}}} - V_{\mathrm{a_t},\mathrm{r_{at}}}, \tag{8}$$

where a_n is the semi-major axis of the nominal orbit, $r_{\mathrm{p_n}}$ is the perigee of the nominal orbit, r_at is the apogee of the injection orbit, r_i is the injection orbit radius and a_t is the semi-major axis of the transfer orbit.

The inclination correction maneuver requires a single burn normal to the orbital plane at the ascending or descending node [Wertz et al. 2011]. It can be calculated by

$$\Delta V_3 = 2V_{\mathrm{a_n},\mathrm{r_{an}}} \sin\left(\frac{\Delta i}{2}\right), \tag{9}$$

where Δi is the inclination error, and r_an is the nominal orbit apogee. Finally, the total velocity variation for the orbit correction is given by:

$$\Delta V_\mathrm{oa} = |\Delta V_1| + |\Delta V_2| + |\Delta V_3| \tag{10}$$

Considering that remote sensing missions with optical payloads are placed on a Low Earth Orbit (LEO), the atmospheric density applies a drag force on the satellite and decreases its orbit [Chagas et al. 2015]. Thus the spacecraft should be able to balance this force to sustain itself through its mission lifetime in the nominal orbit. Considering near circular orbits, the velocity variation necessary to maintain the satellite orbit, taking into account only the drag force, can be computed as in [Chagas et al. 2015]:

$$\Delta V_\mathrm{om} = \pi \frac{C_\mathrm{D} A_\mathrm{d}}{S_\mathrm{m}} \rho a_\mathrm{n} V_{\mathrm{a_n},\mathrm{r_{p_n}}} \frac{T_\mathrm{m}}{T_\mathrm{orb}}, \tag{11}$$

where C_D is the drag coefficient, A_d is the satellite cross-sectional area, S_m is the satellite total mass, ρ is the atmospheric density, T_m is the expected mission lifetime in seconds, and T_orb is the orbital period in seconds.

Finally, the satellite disposal needs to be designed. This is accomplished by decreasing the perigee to an altitude in which the spacecraft will reenter after 25 years from the mission end-of-life. The velocity variation of such maneuver is given by:

$$\Delta V_\mathrm{od} = V_{\mathrm{a_n},\mathrm{r_{an}}} - V_{\mathrm{a_d},\mathrm{r_{ad}}}, \tag{12}$$

where a_d is the disposal orbit semi-major axis found by an empirical model developed using Debris Assessment Software (DAS) v2.0.2 data, and r_ad is the disposal orbit apogee radius. At last, the total velocity variation for the entire mission lifetime is given by:

$$\Delta V_\mathrm{t} = |\Delta V_\mathrm{oa}| + |\Delta V_\mathrm{om}| + |\Delta V_\mathrm{od}| \tag{13}$$

2.4. Propulsion Model

Given the total velocity ΔV_{t} and the satellite total mass S_{m}, the propellant mass m_{p} can be obtained by the rocket equation [Wertz et al. 2011]:

$$m_{\mathrm{p}} = S_{\mathrm{m}}\left(1 - e^{-\frac{\Delta V}{g_0 I_{\mathrm{sp}}}}\right), \tag{14}$$

where g_0 is the gravity acceleration at Earth surface and I_{sp} is the propellant specific impulse.

2.5. Payload Model

The payload model estimates the instrument mass for a desired mission. The mass of an optical head can be obtained by the method described in [Wertz et al. 2011]. This method gives the mass of the new payload by scaling a similar instrument. The scaling is performed based on the payload optical aperture ratio R as in:

$$R = \frac{A}{A_0}, \tag{15}$$

where A_0 is the aperture of the reference instrument. The desired instrument optical aperture A is a function of the orbit height h, inclination i and, the focal length f [Chagas et al. 2015]:

$$f = \frac{P_{\mathrm{size}} h}{Res_{\mathrm{N}}}, \tag{16}$$

$$A = A_0 R_{\mathrm{v}} \frac{f}{f_0}, \tag{17}$$

where R_{v} is the ratio between the ground velocity at the Equator for the current mission and the payload reference mission, both measured at the descendant node, P_{size} is the pixel size of the instrument, and Res_{N} is the desired mission resolution.

Afterwards, the mission payload head mass W_{opt} is estimated using [Wertz et al. 2011]:

$$W_{\mathrm{opt}} = K R^3 W_{\mathrm{opt0}}, \tag{18}$$

where W_{opt0} is the optical head mass of the reference instrument and K is a scaling factor given by:

$$K = \begin{cases} 2 & for R < 0.5, \\ 1 & otherwise. \end{cases} \tag{19}$$

Considering the payload mass W_{p} as the sum of the optical head mass W_{opt} and its electronics mass W_{e}, which, for the sake of simplicity, is considered the same for the reference and current payload, one gets:

$$W_{\mathrm{p}} = W_{\mathrm{opt}} + W_{\mathrm{e}}. \tag{20}$$

Finally, the payload field of view FOV_{payload} can be computed by

$$FOV_{\mathrm{payload}} = 2 \arctan \frac{N_{\mathrm{px}} P_{\mathrm{size}}}{2f}, \tag{21}$$

where N_{px} is the number of pixels of the camera sensor.

2.6. Satellite Mass Model

Minimize the total satellite mass depends on the orbit analysis model, which is interdependent of the propulsion module. Since it is necessary to know *a priori* the total satellite mass S_m to compute the satellite propellant mass m_p, and the contrary is also true, a recursive algorithm to estimate the total satellite mass was developed by [Chagas et al. 2015]. The algorithm first finds the satellite dry mass by [Wertz et al. 2011]:

$$m_\mathrm{d} = W_\mathrm{p}/0.31. \tag{22}$$

Afterwards, the satellite cross-sectional area A_d is estimated by a parametric model developed using data available at the UCS Satellite Database (`https://www.ucsusa.org/nuclear-weapons/space-weapons/satellite-database#.W1oDVfZFzgl`). Next, the total satellite mass first estimative is computed by the parametric equation given by [Wertz et al. 2011]:

$$S_{\mathrm{m},0} = 1.27 m_\mathrm{d}, \tag{23}$$

and recursively iterated through the computations of the propulsion module to find the propellant mass. Finally, the satellite propellant and dry masses are added together resulting in the final total satellite mass.

2.7. Multidisciplinary Optimization Framework

Considering all the models presented, it is possible to summarize the problem in mathematical terms as:

$$\begin{aligned} &\text{Minimize} && S_\mathrm{m}(I,Q,D) \\ &\text{Subject } to && FOV_\mathrm{payload} - 1.05 FOV_\mathrm{min} \geq 0 \\ & && D - Q > 0 \\ &\text{with respect to} && I^L \leq I \leq I^U \\ & && Q^L \leq Q \leq Q^U \\ & && D^L \leq D \leq D^U \end{aligned} \tag{24}$$

where m_d (Equation 22) is the satellite dry mass, m_p (Equation 14) is the satellite propellant mass, FOV_payload is the payload field of view (Equation 21), FOV_min is the minimum payload FOV in the nominal orbit (Equation 5) and I^L, I^U, Q^L, Q^U and D^L, D^U are the lower and upper boundaries of the respective parameters. The MDO described is shown in Figure 2 using an eXtended Design Structure Matrix (XDSM)[2].

First the MDO finds a nominal orbit as described in Section 2.2, then the mission payload is designed as shown in Section 2.5. Having the mission payload, the nominal orbit and launcher errors given by the user as input parameters the MDO gives the first estimative of the satellite design as shown in Sections 2.6.

After getting the satellite area, the fuel for the orbit acquisition, maintenance and disposal maneuvers is estimated as in Sections 2.3 and 2.4. Finally, the total mass is computed by adding the satellite dry and propellant masses. If the MDO reaches the stop criteria, the results are returned and the MDO is finalized. Otherwise, a new evaluation is computed from the current point.

[2] For more information about XDSM refer to [Lambe and Martins 2012].

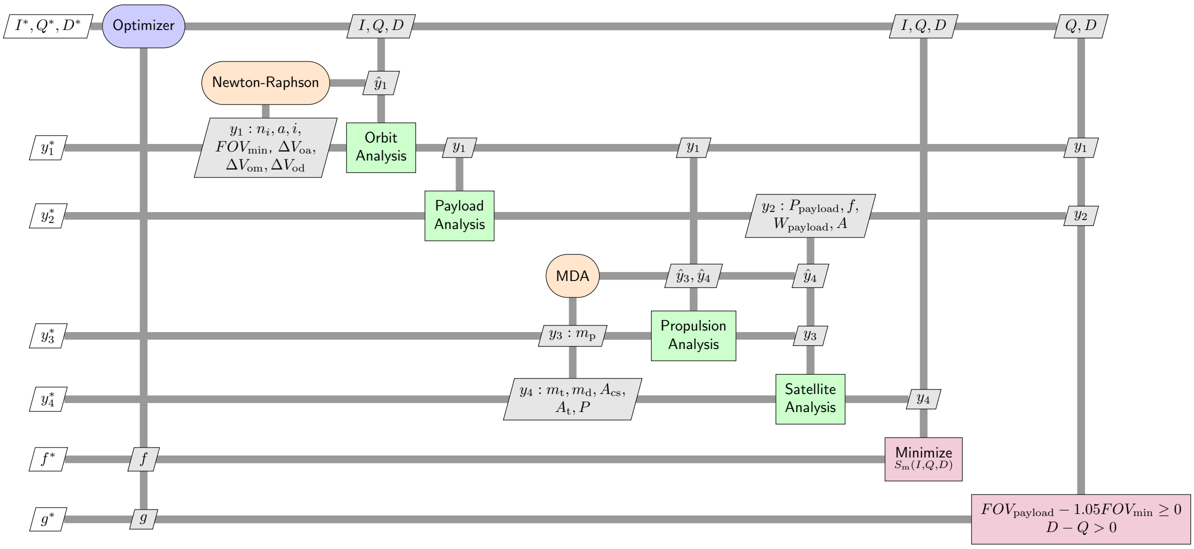


Figure 2: XDSM for spacecraft conceptual design optimization.

3. Case Study

The parameters selected for the case study can be seen in Table 1. An extensive search was performed on the problem to find its global optimum, as presented in Table 2. This was only possible because the global design variables (I, Q and D) have only 10,620 possible combinations, and it is a simplified model.

Table 1: Simulation settings derived from [Chagas et al. 2015, Wertz et al. 2011].

Parameter	Value	Parameter	Value
I	[13;15]	Q	[1;59]
D	[1;60]	Drag coefficient (C_d)	2.2
Mean cross-sectional area (A_m)[1]	180%	Atmospheric density model[2]	1 year with $F10.7 = 225$ and 3 years with $F10.7 = 175$
Mission lifetime	4 years	Inclination launch error	0.015°
I_{sp}	225 s	Mission resolution	20 m
Semi-major axis launch error	20 Km	Eccentricity of nominal orbit (e)	0.0

[1] Mean cross-sectional of the satellite with respect to the satellite cross-sectional area, in this case the mean cross-sectional area is 1.8 times the cross-sectional area.
[2] The mean atmospheric density was obtained from NRLMSISE-00 model.

The result presented in Table 2 provides an optimum solution for the preliminary design of a satellite, which can be used as the initial input for the study conducted by the design team. This can reduce the time required to analyze the mission at hand as well improve the results quality.

4. Conclusion

This work explained how to implement an MDO to solve a space application, showing the efficiency of this tool for satellite design optimization in early phases of development. Further work should be focused in implementing optimizers for the MDO, improving its performance.

Table 2: Optimal solution.

Variables	Values	Variables	Values
I	14	Inclination of nominal orbit (i)	97.656°
D	60	Total change in Velocity (ΔV_{t})	248.751 m/s
Q	59	Major semi-axis of nominal orbit (a)	6,944.284 Km
Orbit Acquisition (ΔV_{oa})	30.768 m/s	Dry mass (m_{d})	175.952 Kg
De-orbiting (ΔV_{od})	0 m/s	$\mathrm{FOV}_{\mathrm{min}}$	4.469°
Orbit Maintenance (ΔV_{om})	217.983 m/s	Mean cross-sectional area (A_{m})	6.841 m^2
Cross-sectional area (A_{d})	3.801 m^2	Propellant mass (m_{p})	20.997 Kg
Total mass (S_{m})	196.949 Kg		

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