Multidisciplinary Design Optimization Approach to Conceptual Design of a LEO Earth Observation Microsatellite

Ali Ravanbakhsh[1] and Mahdi Mortazavi[2]
Amirkabir University of Technology, AUTSAT Research Center, Tehran, Iran

Jafar Roshanian[3]
K.N.Toosi University of Technology, Tehran, Iran

This paper presents the conceptual design of a sun-synchronous LEO Earth Observation microsatellite incorporated with Multidisciplinary Design Optimization (MDO) approach. The objective is to develop a structured system level for the design process including mission design parameters such as required Revisit Time (RT), and accessible Ground Sampling Distance (GSD). In order to apply MDO with the design process a sizing tool has been developed based on both mission and system design-estimating relationships. In addition, the conceptual design data and concepts of the microsatellites with the similar mission have been used to achieve a reliable sizing tool. The objective is to minimize the total mass of the satellite. A Genetic Algorithm (GA) is coupled with the developed sizing tool to obtain optimized design parameters for the microsatellite.

Nomenclature

MDO	=	Multidisciplinary Design Optimization
RT	=	Revisit Time
GSD	=	Ground Sampling Distance
GA	=	Genetic Algorithm
ML	=	Mission Life
SS-O	=	Sun-Synchronous Orbit
EPS	=	Electrical Power Subsystem
ADCS	=	Attitude Determination and Control Subsystem
C&DH	=	Command and Data Handling
TT&C	=	Telemetry, Tracking, and Command
TCS	=	Thermal Control Subsystem
SS	=	Structure Subsystem
t-Img	=	Imaging duration per orbit

I. Introduction

DESIGN process of an earth observation microsatellite usually starts with the definition of requirements such as RT, Mission Life (ML), and accessible GSD. RT can be defined as the repeat cycle days between two successful monitoring of the imaging spot which for a Sun-Synchronous Orbit (SS-O) it mainly depends on the orbital altitude. ML can affect satellite subsystems parameters especially Electrical Power Subsystem (EPS) requirements. In addition, other interrelated parameters like: accessible GSD, required imaging duration per orbit (t_Img), and the type of solar cells or battery cells which can be used in EPS influence the design out puts such as: satellite total mass or its dimensions. During the conceptual design phase, the variation of all these parameters creates a design space including a large number of discrete design points. In order to find an optimum design point

[1] MSC Student, Department of Aerospace Engineering, ravanbakhsh.a@gmail.com, AIAA student member.
[2] Assistant Professor, Department of Aerospace Engineering, mortazavi@aut,.ac.ir
[3] Associate Professor, Department of Aerospace Engineering, roshanian@kntu.ac.ir

regarding the lowest satellite mass in such a design space a GA which is a global search technique can be used. In this paper a MDO approach that is applied to the early conceptual deign process of a LEO earth observation microsatellite is investigated.

II. Satellite Design

Satellite design at any class is a complex, iterative process that involves multi disciplinary engineering expertise. In microsatellite projects, after determination of top-level mission requirements such as communication or earth observation, the design process will be initiated with the objective of achieving an optimal design which at early steps it can be interpreted as one that meets mission requirements as well as having lowest possible launch mass.

The design of earth observation microsatellites, which are commonly used for remote sensing applications, usually starts with mission design. The major point at this part is the decision about required RT which can affect the orbital parameters. The next step is the payload design. In this step, although the type of the payload is the most important driver of its design, the parameter like accessible GSD due to allocated aperture can affect the payload mass and power.

The design then proceeds with bus design, which for a LEO earth observation microsatellite includes six major subsystems. ADCS (Attitude Determination and Control Subsystem) which measures satellite attitude data by means of appropriate sensors and processes them using a control law then if needed sends out control commands to actuators in active mode, C&DH (Command and Data Handling) which includes an onboard computer and act as a satellite manager by controlling different relations between subsystems, EPS (Electrical Power Subsystem) is responsible for providing satellite equipments with the necessary power during sunlight as well as eclipse time using solar arrays and chemical batteries respectively, TT&C (Telemetry, Tracking, and Command) which receives uplink command signals, and transmit downlink signals such as imaging payload data or house keeping information of different parts, TCS (Thermal Control Subsystem) is responsible for maintaining different on-board equipments with in their operating temperature limit as their requirements and SS (Structure Subsystem) that carries and protects the spacecraft and payload equipment through the launch environment and operational life in orbit.

III. Multidisciplinary Design Optimization

Multidisciplinary Design Optimization (MDO) is a methodology for the design of systems in which strong interaction between disciplines motivates designers to simultaneously manipulate variables in several disciplines [1]. According to the International Council on Systems Engineering (INCOSE), 70-90% of the development cost of a large system is predetermined by the time only 5-10% of the development time has been completed. Thus, considering MDO approach in the conceptual design phase of a complex system such as satellites will give the designer greater power to control the whole design process.

MDO methodology can be applied to different levels of a system design project and regarding the application level it consists of different components[1]. In order to have a basic MDO approach to the problem of conceptual design of a microsatellite, a framework may be suggested as shown in Fig.1.

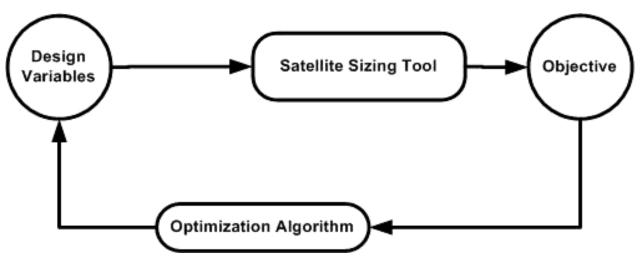

Figure 1. Suggested MDO framework

MDO framework divides the whole design problem to four parts: *Design Variables* are made up of parameters which have a major influence on the design out put, *Satellite Sizing Tool* consists of mathematical model of different deign disciplines which are correlated to each other in order to constitute a road from design variables to design out puts, *Objective* can be defined as a design out put which we desire to optimize it, *Optimization Algorithm* is a tool chained to design space created by the sizing tool for searching between design points with the aim of reaching to the optimum design.

IV. Satellite Sizing Tool

The satellite sizing tool which is introduced here as the conceptual design model consists of three principal parts: mission sizing, payload sizing and bus sizing.

Mission sizing is made of practices which lead to determination of a circular SS-O. This orbit will be the representative of a specific repeat cycle days or RT. To achieve this, some calculations have been made in order to find out the number of revolutions that satellite must have at an integer number of days. According to the fact that the satellite will be seriously affected by atmosphere drag at altitudes bellow 500 km and taking to account Van Allen radiation belts harmful effects for orbits with altitudes below 1000 km, the chosen range for altitude is between 500 and 1000 km [2]. Also, according to satellite mission, the repeat cycle rate is chosen from 3 to 26 days [3]. The out put of the mission sizing section is a number of scenarios introduce a SS-O in which after *R* revolutions in *D* days the ground tracks are repeated.

Payload sizing is mainly based on the required resolution and orbit altitude. Firstly, the payload aperture range of variation is selected. Secondly, by considering the type of the imaging payload, Multi-spectral Mid-IR, and using some design-estimating relationships from Ref. 2, the mass and power of the payload is calculated.

Bus sizing is the last part of the sizing tool. There are six subsystems which construct the satellite bus. The total mass and average power of the satellite with design estimating relationships from Ref. 4, together are used to calculate the mass and power of all subsystems except EPS (Electrical Power Subsystem). EPS sizing is done with more detail calculations including an appropriate mission scenario and different common choices for the solar cells and battery cells.

A. Mission Sizing

The SS-O is one of the most frequently used orbits for remote sensing and earth observation microsatellites. In a simple way a SS-O is defined as a near polar orbit where the nodal precession rate is matched to the earth's mean orbital rate around the sun. This orbit has the effect of keeping its geometry with respect to the sun nearly fixed such that the sun lightning along the satellite ground tracks remains approximately the same over the mission duration. Two famous examples of earth science missions which used this type of orbit are: LANDSAT and Terra. Also, microsatellites like: DLR-TUBSAT and BILSAT-1 used SS-O to do their earth observation mission.

There are some important reasons for the frequent utility of the SS-O. Since the orbital inclination is nearly polar (96.5-102.5 degrees), the SS-O provides global coverage at all latitudes with the exception of just a few degrees from the poles. Also, because the relative position of the SS-O remains almost fixed with respect to the sun's direction, lightning conditions will be the same during the mission along the foot print areas which must be observed during day light. In addition, the other property which is often important for satellite thermal design is that the SS-O could provide a continual "dark-side" for satellite in orbit which can sometimes solve the complex thermal problems of the satellite. The complementary characteristic is that discrete altitudes can be selected to provide SS-Os ground tracks which repeat after a fixed interval of days. This regular cycle of repeat is desirable because it gives the capability of conducting periodic research on the satellite remote sensing data from a specific area.

All of the above desired orbital characteristic can be the reason for why the satellite sizing tool, developed here, includes SS-Os as its primary mission sizing driver.

SS-Os with Different Repeat Cycle Days

Ronald J.Boain[5] suggests an approach to understanding how we can use RT to create a unique choice of orbital altitude for a SS-O and here this method has been used. First, considering a simple calculation made with Kepler's Equation, the range of altitudes based on the repeatability of an orbit's ground tack in just one day is determined. This range contains five practical choices of altitudes between 250 and 1700 km which are indicated in Table.1.

Table 1. Orbit parameters for SS-O with an integer number of revolutions in one-day

Revs per Day, #	Orbital Period, (Sec)	Equatorial Altitude, (km)	Distance between Adjacent GTs, (km)
12	7200.00	1680.86	3339.59
13	6646.15	1262.09	3082.69
14	6171.43	893.79	2862.50
15	5760.00	566.89	2671.67
16	5400.00	274.42	2504.69

Revs: Revolutions, GTs: Ground Tracks

This means that each orbit of Table 1, respectively lays down a ground track grid on the surface of the earth with 12, 13, 14, 15 and 16 ascending nodal positions equally spaced around the equator. Also, it can be seen that the range include and is in consistent with the range previously discussed which was between 500 and 1000 km.

In many remote sensing applications a shorter distance is needed between two adjacent ground tracks. To achieve such orbits we extend the orbit choices with RT in the range between 3 to 26 days using Eq. (1) from Ref. 5.

$$P = 86400\,\frac{D}{R} \qquad (1)$$

In Eq. (1), P is orbital period in seconds, D is repeat cycle days representative or RT in days, and R stands for number of integer revolutions during D days.

Using this method and eliminating the orbits with the same periods to another orbit, a set of SS-Os will be determined with specific RTs. Some of these orbits in RT range between 3 to 10 days are shown in Fig.2.
In Fig.2 each number besides the points indicates the number of revolutions in the repeat cycle days.

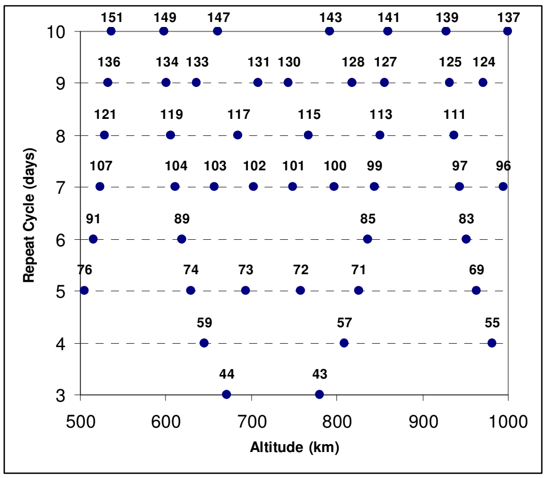

Figure 2. Sun-synchronous repeat ground track orbits

B. Payload Sizing

One of the most important parameters in payload design is the requirement about accessible GSD for remote sensing application. Accessible GSD is based both on optical and electrical parts of the payload. In the first part, aperture size of the payload, only a single parameter of optical part beside many others has a major influence on accessible GSD. This influence can be determined by some simple equations from Ref. 2. Based on Raleigh diffraction criteria[2] the angular resolution is obtained from Eq. (2).

$$\theta_r = 1.22\,\frac{\lambda}{D} \qquad (2)$$

In Eq. (2), λ is the wavelength of the electromagnetic spectrum selected for the payload, D is the aperture diameter of payload, and θ_r presents angular resolution in radiant. For a satellite at altitude, h, the linear ground resolution or GSD at nadir is a function of h, λ and D which is obtained from Eq. (3).

$$GSD = 2.44\,\frac{h\lambda}{D} \qquad (3)$$

It can be seen that the accessible GSD as one of the mission requirements have a correlation with D which is a payload parameter.

Using suggestions and relations for payload size, weight and power estimation from Ref. 2, the Multi-Spectral Mid-IR, an existing system for resources remote sensing, is selected as the reference system for this part of the sizing tool.

C. Bus Sizing

At this section, sizing proceeds with platform components mass and power estimation. The total mass and average power of the satellite and design estimating relationships together are used to calculate the mass and power of all subsystems except EPS (Electrical Power Subsystem). Table 2, contains the design estimating relationships from Ref. 4.

Table 2. Subsystems mass and power estimation

Subsystem	Mass (kg)	Power (w)
ADCS (active) ADCS (passive)	-0.0142 Mt+13.748	0.0036 Pav+18.304 -0.0152 Pav+8.858
C&DH	-0.0079 Mt+5.5627	-0.03 Pav+15.39
TT&C	-0.0103 Mt+6.5935	0.0456 Pav+25.583
TCS	0.0498 Mt+0.4785	0.0067 Pav+0.7862
SS	-0.01 Mt+31.079	-

Mt: Satellite total mass, Pav: Satellite average power

EPS Sizing

EPS sizing is done with more detail calculations including an appropriate mission scenario and different common choices for the solar cells and battery cells. The EPS sizing has two main parts: solar array sizing and battery sizing.

Solar Array Sizing

Solar array sizing is mainly affected by the power profile of different subsystems including imaging payload during per orbit. The simple Eq. (4) from Ref. 2 is used to estimate the satellite required power.

$$P_{sa} = \frac{P_e T_e / X_e + P_d T_d / X_d}{T_d} \qquad (4)$$

In Eq. (4), P_{sa} is minimum power that solar array must provide during daylight, P_e is the power requirements during daylight, P_d is the power requirements during eclipse, T_e is eclipse duration, T_d is daylight duration, X_e is the efficiency of the path from solar array to battery and then to individual loads (~0.60), and X_d is the efficiency of the path directly from solar array to loads (~0.80).

T_e, T_d are obtained from the mission sizing section. In order to calculate P_e, P_d it has been assumed that imaging mission is done on day light as well as eclipse time with equal duration per orbit. In addition, the imaging mission is assumed to be done at the same latitudes in daylight and eclipse in order to have a uniform coverage belt parallel to the equatorial. Also, further considerations regarding to the mission operation have been assumed about different subsystems.

With P_{sa} at hand, the solar array properties are determined based on two common solar cells type: Si (silicon) and GaAs (Gallium Arsenide). Table 3 indicates the properties of these two types of solar cells.

Table 3. Properties of two solar cell types

Cell Type	EOL Power delivered by one solar cell in (w)	Mass of one square meter of solar cell (kg)
Si	0.110	1.1
GaAs	0.146	1.9

EOL: End of Life

At the end of this section the mass of the planar solar array will be estimated by the considering the type of solar cell used.

Battery Sizing

Battery sizing is started by choosing the type of the battery cell. Two common types of battery used in microsatellites are NiCd (Nickel Cadmium) and NiH2 (Nickel Hydrogen). The battery capacity required for the satellite can be estimated by means of Eq. (5) from Ref. 6.

$$C_r = \frac{P_e T_e}{X_{b-l} DOD} \qquad (5)$$

In Eq. (5), C_r is the minimum energy capacity requirement, X_{b-l} is power transmission efficiency from battery to eclipse loads (~0.70), and DOD is the depth of discharge characteristics of the battery.

Figure 3. shows how cycle life influences the DOD, the percentage of energy removed from the battery in a discharge cycle, for two common kinds of battery cells.

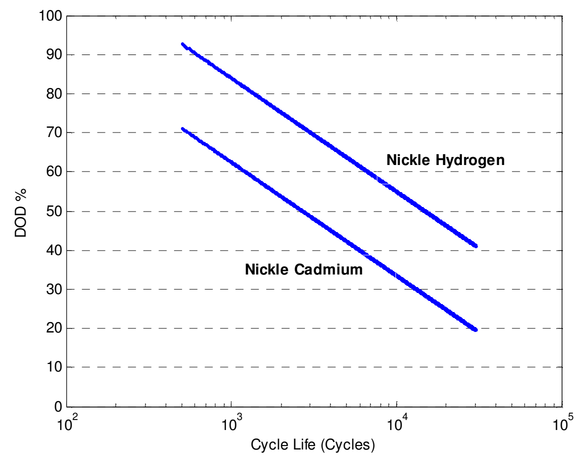

Figure 3. Depth of Discharge vs. Cycle Life for batteries

In mission sizing section it was showed that each SS-Os presents a specific number of revolutions in a specific repeat cycle days. For example according to Fig.2 a SS-O with 810 km altitude and 4 days RT must have 57 revolutions in order to have the global coverage. Regarding to this fact the satellite cycle life can be estimated by Eq. (6).

$$CL = \frac{ML \times 365 \times NR}{RT} \qquad (6)$$

In Eq. (7), CL is Cycle Life, and NR is the number of revolutions for SS-O per RT. By means of Eq. (6) and Fig.3 the DOD is determined according to battery cell and then C_r can be calculated from Eq. (5). The mass of the batteries can be calculated based on required capacity and the properties of the battery type used. Table 4 contains the properties of two common battery cells.

Table 4. Properties of two battery cell types

Cell Type	Operational Specific Energy (w-hr)/kg
NiCd	20
NiH2	40

Finally, the total mass of the EPS is the sum of the solar array mass and the battery plus 30% margin to account for power distribution equipments. Also, the power consumption for the EPS is assumed to be the 10 % of the satellite average power.

D. Sizing Tool Validation

The sizing tool is created by connecting the three above mentioned parts with aid of a computer program written in MATLAB[7] environment to estimate the satellite total mass.

To check out the sizing tool results are with in acceptable tolerance, it was caparisoned with four earth observation microsatellite projects[8] which their major specifications have come in Table 5.

Table 5. Earth observation moicrosatellie projects

Project	Altitude, (km)	Mass, (kg)	Power, (w)
Bird-2	580	77	120
Alsat-1	744	95	240
BILSAT-1	622	100	232
NigeriaSat-1	694	95	240

Table 6. Sizing tool compared to actual projects

Project	Actual mass, (kg)	Sizing tool predicted mass, (kg)	Difference, (%)
Bird-2	77	79.92	3.72
Alsat-1	95	88.46	7.12
BILSAT-1	100	87.90	12.88
NigeriaSat-1	95	88.46	7.12

As indicated in Table 6, the percentage of difference between the sizing tool total mass estimation and the actual mass are obtained. Although there were no detailed data for different parameters of the actual projects, the results seems satisfactory at the conceptual level of the design.

V. Problem statement with MDO framework

At this step, the problem of conceptual sizing of a LEO earth observation microsatellite must be defined in a proper way for being used in the MDO framework suggested in section III.

A. Design Variables

Seven design parameters which have major influence in determining the guidelines for the conceptual design are chosen. They are also dependent to one specific discipline and this characteristic creates correlation between mission, payload and EPS as well as other subsystems.

Although the selected seven variables are interrelated through the whole sizing process, it can be said that the four variables: ML, RT, altitude (h), and Imaging duration per orbit (t-Img) are in the mission sizing block, the required accessible GSD is due payload sizing and solar cell type and battery cell type are considered in bus sizing block.

To obtain a practical design space there have been assumed five bound constraints: ML variation range is between 3 and 5 years, RT is between 3 to 26 days, Altitude range is between 500-1000(km), t-Img is from 5 to 20(min), and accessible GSD is between 30-65(m).

B. Satellite Sizing Tool and Objective

The design space created by satellite sizing tool is a discrete design space, each point represents a specific SS-O with a practical RT as well as other design parameters.

Owing to the fact that the satellite mass is known as an important parameter in microsatellite projects, it is selected as the objective to be minimized by the optimizer.

C. GA as Optimization Algorithm

The GA is a stochastic global search method that mimics the observed behavior of natural biological evolution. GA operates on a population of potential solutions applying the principle of survival of the fittest to produce (hopefully) better and better approximations to a solution. At each generation, a new set of approximations is created by the process of selecting individuals according to their level of fitness in the problem domain and breeding them together using operators borrowed from natural genetics. This process leads to the evolution of populations of individuals that are better suited to their environment than the individuals that they were created from, just as in natural adaptation[9].

The GA has some advantages compared to conventional optimization methods: GA can be used with a code consisting both discrete and continues design variables, GA is a population-based method and it gives the capability of having a better design space by its reproduction in each generation, and finally because GA employ probabilistic choices rather than deterministic rules, it is likely to search the entire design space and is not trapped in local minima. Based on these considerations, for this problem a GA has been used as an optimizer.

VI. Results and Discussion

The GA was jointed with developed Satellite Sizing Tool according to the MDO framework. There were executed 9 runs of the GA where each three set of runs were done with a specific initial population vector i.e. three different initial populations were assumed for 9 runs. Figure 4, shows the process of optimization for three sample runs with three different initial populations, and Table7, contains the final results of 9 GA runs.

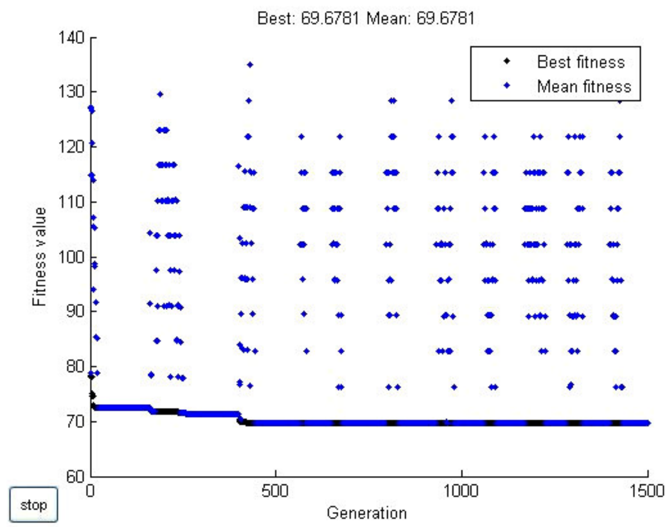

Figure 4. a) Fitness history in each generation for 1st sample initial population

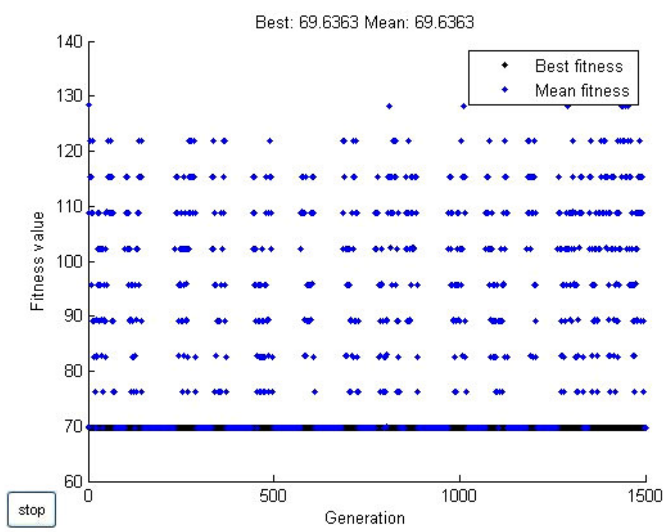

Figure 4. b) Fitness history in each generation for 2nd sample initial population

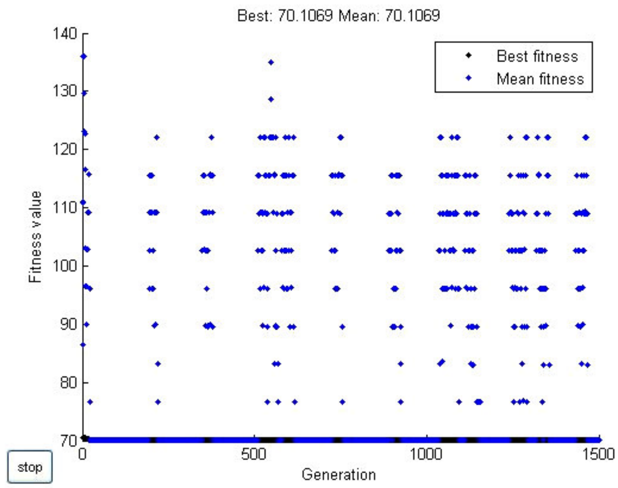

Figure 4. c) Fitness history in each generation for 3rd sample initial population

According to Fig. 4, although the fitness history is different when the initial populations are different, the fitness value will remain almost the same, and this is what expects from the GA.

The existence of scattered points away from best fitness line in the fitness history figures can be the result of a large discrete design space. During the search through the design space, GA may generate acceptable but suboptimal generations. Also, these design point specifications can not be determined because of GA limitations.

Table 7. Results for 9 runs of the GA

Run #	ML (years)	RT (days)	h (km)	t_Img (min)	Accessible GSD, (m)	Solar cell	Battery cell	Objective, Satellite mass (kg)
1	3.0	8	937.52	5.0	62	Si	NiH2	69.80
2	3.0	15	987.65	5.0	65	Si	NiH2	69.64
3	3.0	10	928.64	5.0	61	Si	NiH2	69.74
4	3.0	15	987.60	5.0	65	Si	NiH2	69.64
5	3.0	15	987.60	5.0	65	Si	NiH2	69.64
6	3.0	15	987.60	5.0	65	Si	NiH2	69.64
7	3.0	18	689.60	5.0	45	Si	NiH2	69.93
8	3.0	26	508.17	5.0	34	Si	NiH2	70.11
9	3.0	26	508.17	5.0	34	Si	NiH2	70.11

For the 9 runs the obtained satellite mass is almost the same. In addition 4 of 7 design parameters: the ML, the imaging duration per orbit (t_Img), the type of solar cell and the type of battery cell are the same for 9 design points due to the fact that they result the lowest possible mass to their count. Owing to this fact, it can be said that these parameters have the greatest effect on the design objective i.e. their influence on satellite total mass is considerable.

The other 3 design parameters: RT, orbital altitude and accessible GSD are different. Based on this difference the 9 design points can be categorized in five different design points as indicated in Table 8.

Table 8. Optimum but different design point

No.	ML (years)	RT (days)	h (km)	t_Img (min)	Accessible GSD, (m)	Solar cell	Battery cell	Objective, Satellite mass (kg)
1	3.0	8	937.52	5.0	62	Si	NiH2	69.80
2	3.0	10	928.64	5.0	61	Si	NiH2	69.74
3	3.0	15	987.65	5.0	65	Si	NiH2	69.64
4	3.0	18	689.60	5.0	45	Si	NiH2	69.93
5	3.0	26	508.17	5.0	34	Si	NiH2	70.11

According to Table 8, design point No.1, have the lowest RT. This means that during a constant ML, 3 years, the number of imaging access will be the maximum in comparison with others. In addition, design point No.5 has the best accessible GSD so it can be used for more critical remote sensing missions that need better resolution. By this comparison it can be found that some parameters like: RT and accessible GSD influence the mission and performance of greater importance than the satellite total mass. This fact may lead us to the value of application multi-objective optimization to the satellite design problem.

VII. Conclusion

Satellite design at any class is a complex, iterative process that involves multi disciplinary engineering expertise. The basic MDO approach introduced here seems to be successful in application to LEO earth observation microsatellite conceptual design. This approach will give valuable insight to the system parameters and can be used as an effective design methodology from the early steps of satellite design projects. Also, the GA seems to be a good optimizer for being jointed with MDO framework. In fact, its capability to search the global large scale discrete design space with saving a valuable amount of computational time is an advantageous.

In conclusion, the MDO methodology seems having the capability to be used widely in complex satellite design projects even at early conceptual levels. It saves time to implement preliminary trade offs and also presents the road map of optimal design destination.

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