DESIGN AUTOMATION FOR SATELLITE PASSIVE THERMAL CONTROL

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ABSTRACT

We explore the advantage of using genetic algorithms for the automated design of a small satellite passive thermal control. The control law is a layout of heterogeneous material patches covering the satellite surface. We tested our candidate material patch solutions with a finite element method simulation of the satellite thermal conditions during direct sun light orbit. We found that the method has great potential to find interesting solutions although it suffers from premature convergence.

1 INTRODUCTION

The design of a satellite system is a complex task due to constraints on limited space, weight, mission costs as well as the harsh environmental conditions of space operation. Engineers are challenged to face the problem of loading satellites with sophisticate pieces of equipment, each one having particular environmental requirements for successful operation. Amid them, temperature specifications are probably the most difficult characteristics to achieve.

The right selection of materials and the proper placement of components becomes a problem of exponential complexity. Classical engineering methods have been exploited to provide acceptable, yet sub-optimal, solutions to this problem. Approaching optimality in material selection and component distribution is becoming a highly relevant problem, especially as the complexity of systems increase. Moreover, recent advances in digital manufacturing and free form fabrication are opening unseen avenues on the complexity that satellite systems might take.

Genetic Algorithms (GA) have been successfully applied to the problem of component placement in satellite systems or Satellite-Module Layout Design (SMLD) [11, 17]. Due to its complexity, some novel techniques on Evolutionary Computation have been challenged to address this problem as well. Ranging from Artificial Embryogeny [15] to Multi-Objective Optimization [8].

In this study we explore the capabilities of Genetic Algorithms for the problem of thermal satellite control optimization. The idea is to find the right distribution and selection of materials so that all satellite sub-systems operate within their temperature range requirement. We are using a CubeSat currently under development at the Engineering School of the University of Chile (SUCHAI).

The remainder of this paper is organized as follows: In section 2, we describe our satellite mission. In section 3, we describe the thermal control problem under analysis. Section 4 shows the genetic adaptation methods. Section 5 describes the thermal analysis. In section 6, we describe our results. Finally the section 7 shows the conclusions of this study.

2 MISSION DESCRIPTION

The University of Chile's physics, mechanical and electrical engineering departments are carrying a joint satellite development project. The project is called SUCHAI, which stands for Satellite of the University of CHile for Aerospace Investigation. The main goals of the project are to design, build, launch and operate a CubeSat picosatellite [9].

The principal goals of the satellite project are three: 1) to learn the whole process chain required to build a CubeSat that can operate in space and communicate successfully with a ground station, 2) to learn with detail the capabilities and limitations (power, mass, space, etc.) of the current standard to carry out physical experiments (we are preparing four experiments to be carried by the satellite) and 3) to explore the educational impact that this kind of project might have on students.

The CubeSat is a satellite standard that offers a novel view to the conception of satellite missions. Usually the payload defines the vehicle (satellite size, weight, shape and power), using space proven components. Under the CubeSat philosophy, however, the payload should be adapted to a given chassis, mainly using commercial components.

The standard was developed by a team of researchers at California Polytechnic State University and Stanford University. Examples of institutions carrying successful CubeSat projects are the University of Tokyo, Dartmouth College and NetDecide Corporation [5,10]. Before launch, a CubeSat must to be constrained to a 10×10×10 *cm* volume. Its weight should be less than ~1 *kg* (or 1.3 *kg* more precisely).

Our CubeSat should be placed at low near polar orbit. Although the optimal altitude for our experiments is 600 *km*, the communication and experiments can be carried out starting at 300 *km* and up to 700 *km*. At the moment, the launcher has not yet been completely decided. An exploded view of our satellite is shown in Figure 1a. An illustration of the satellite while in constrained launch position is shown in Figure 1b.

The CubeSat standard offers the great potential of democratizing space research. However, due to the limitations on size, mass and power, there are multiple challenges to overcome when industrial applications are desired. Therefore, any form of design optimization is imperative and welcome. The current project is being used to explore optimization alternatives for the material selection process in order to improve the satellite heat transfer and reduce internal temperature.

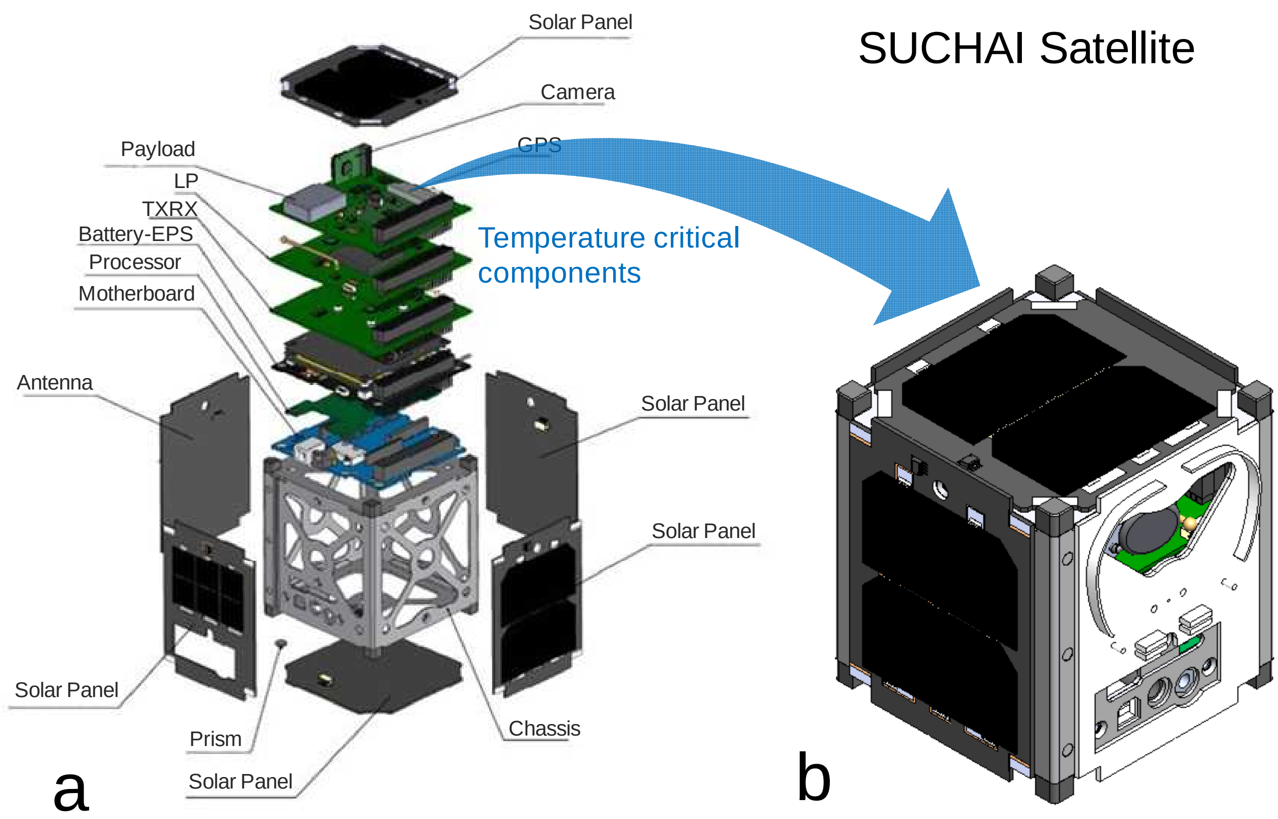

Figure 1. SUCHAI satellite components. a, Exploded view of the satellite showing the internal distribution of PC-104 boards. **b,** The satellite while constrained on a compact launch configuration. A large blue arrow is coming out of a location sensitive to high temperatures.

3 SATELLITE THERMAL CONTROL PROBLEM

Thermal control is important for any satellite, under a small volume we need to integrate many sub-systems that must be kept in operative range. Due to space constraints we cannot implement active control systems such as heaters, louvers and heat pipes. Instead we use passive systems like surface finishes and radiators [2].

The purpose of a thermal control system is to maintain the entire satellite component within its design temperature limits over the entire lifespan of a mission. We have followed the satellite thermal control design work flow proposed by J. Wertz [16]. Figure 2 shows this systematic process.

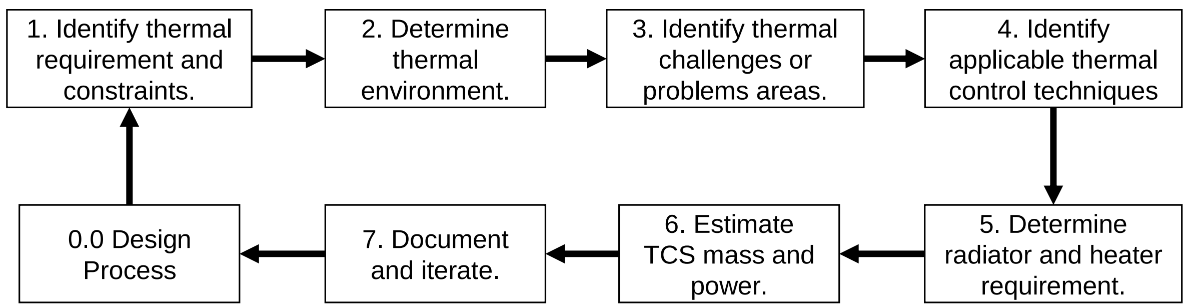

Figure 2. The work flow proposed by J. Wertz [16] allows simplifying the design process of a satellite Thermal Control System (TCS).

While in the space, satellites are exposed to harsh conditions that can lead to catastrophic failures. From the thermal point of view, the main challenge is to cope with the strong temperature variation between the direct sun-light and eclipse phases (from 100 ºC to -130 ºC in a tenth of a second). In case of a LEO-orbit, the impact of this variation is extreme, since the frequency between direct sun-light and eclipse phase is very high (see Figure 3a).

There are two critical phases for this type of orbit: These are the cold phase and the hot phase. During the cold phase, the satellite is in the eclipse zone and dissipates a minimum power. During the hot phase, the satellite is exposed to sun light and dissipates maximum power. In this paper we consider the analysis for the hot phase.

Other heat sources and sinks are important for satellite thermal balance. These are the Albedo, the Earth Infrared Radiation, the internal heat (equipment) and the heat radiated (See Figure 3b).

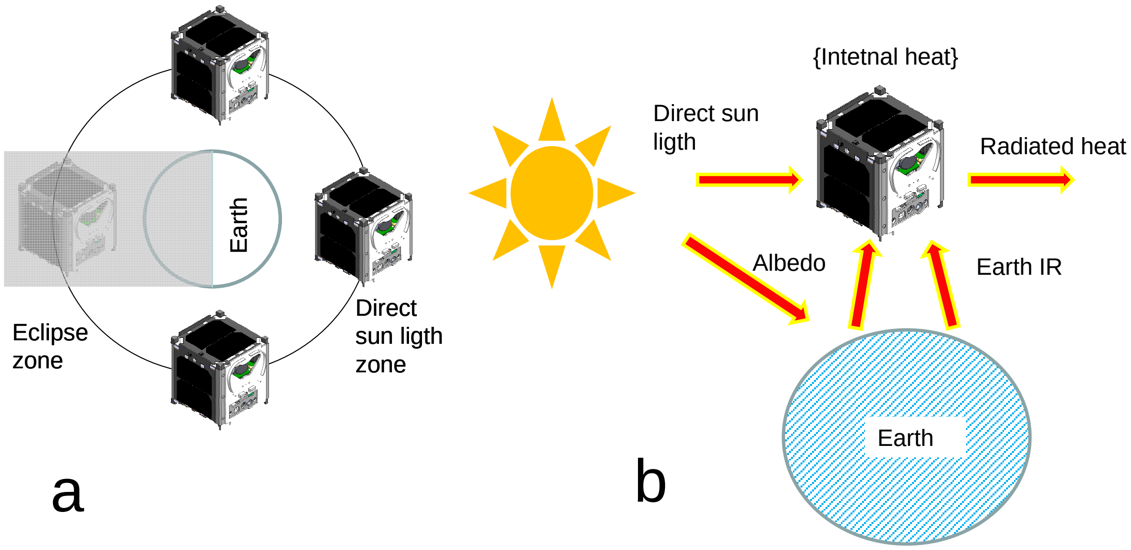

Figure 3. Orbit phases and heat sources. a, Light phases during the satellite orbit. **b**, Heat contributions during operation.

We have considered as thermal boundary conditions the reference values presented in Table 1. These values are affected by the view factor due to the earth curvature. This view factor was computed by the algorithm proposed by J. Richmond [12]. For the remainder of this paper we have considered data of two CubeSat missions: The InKlajn-1 [14] and SUCHAI.

Table 1. Reference values for the thermal environment.

Source/Skin	Reference value	Description
Sun (solar constant)	1367 W/m^2	From [14].
Albedo	479 W/m^2	Fraction of solar constant, function of the view factor.
Earth IR	221 W/m^2	Function of Earth temperature and view factor.
Internal heat	20 W/m^2	Heat generated by the internal equipment.
Radiated heat	To be computed depending on the design.	

4 GENETIC ADAPTATION

In this section we describe the process for automatically deriving the materials to be used on the satellite surface, so as to meet thermal mission requirements. The method allows choosing the right material combination as well as the geometric distribution of surface material patches. The result is interpreted as the passive thermal control law to be applied during the entire satellite lifespan.

A simple Genetic Algorithm [4] is used to search the space of possible material solutions. We implemented a three unit (3U) CubeSat thermal simulation using the finite element method (FEM). The simulation was used to test the quality of material solutions so as to meet the desired thermal requirements.

A solution is represented by a 32 elements length real-valued genome. Two elements are used to represent the pair of selected materials and the remaining 30 elements are used to represent the geometrical distribution of material patches over the surface. As Figure 4 illustrates, there is a total of 10 satellite faces (five on top and five at the bottom) under genetic optimization, since the central portion of the satellite consists of fixed solar panels.

Solution(i) = [mat1, mat2, SF1, SF2,…,SFn,…, SF30]

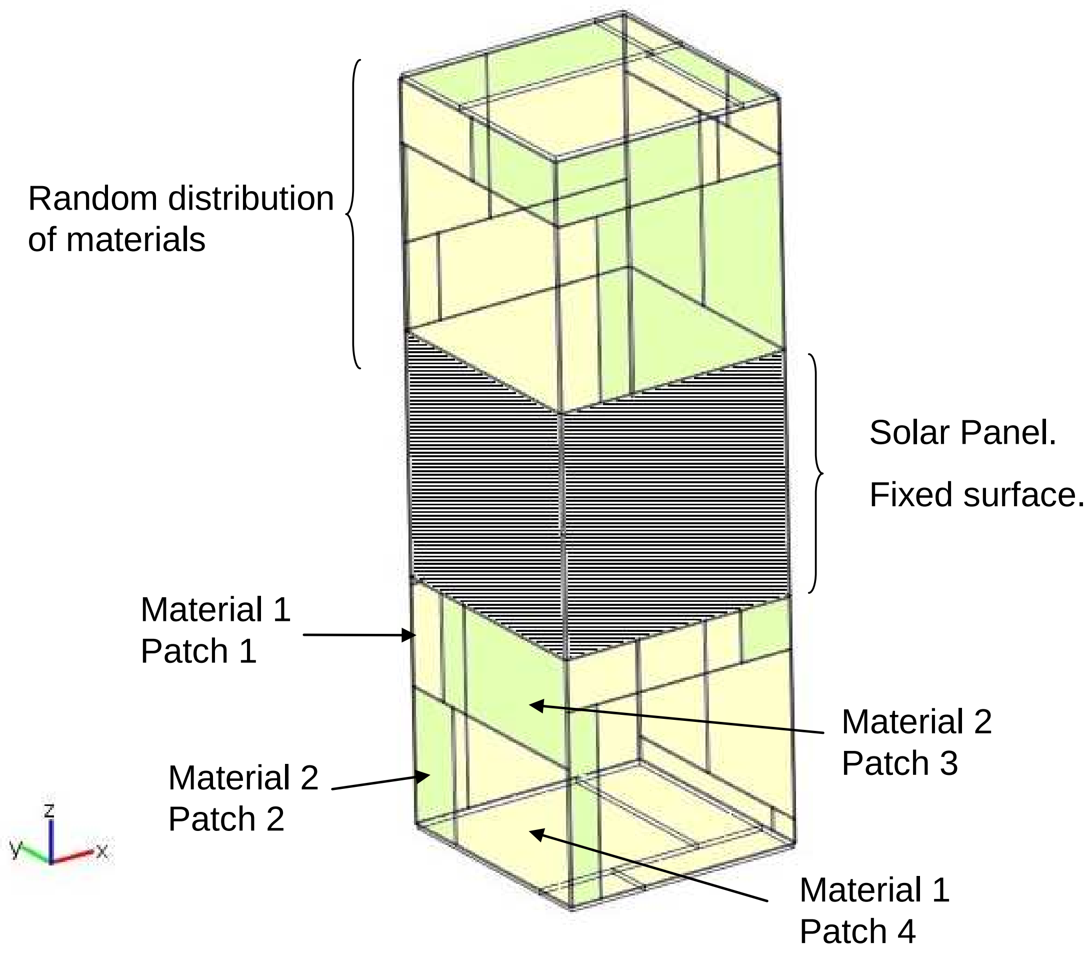

Figure 4. Example of thermal control material solution. The distribution of material patches over the ten faces under optimization is shown. Each face contains four patches of two different materials.

A material patch is a rectangular region occupying a percentage of a face area. The geometry is defined by three surface factor coefficients in the range [0,1]. Figure 5 illustrates a solution with two different materials shown with varying shades of green. An example of patch distribution is shown at the bottom left portion of this satellite. Table 4 shows the properties of the four materials that were considered for optimization.

An initial population of 12 candidate solutions was first generated as random. Then our algorithm proceeds as follows: If the thermal condition is satisfied the algorithm stops. If not, each solution is evaluated using the thermal simulation. Parents for a new generation are selected using stochastic universal sampling (SUS) [1,3]. Crossover is performed with a probability $P_c = 0.8$ and mutation with a probability $P_m = 0.1$. The new generation is again evaluated and the algorithm proceeds to a new iteration.

The quality of each candidate solution is evaluated using a FEM based thermal simulation. The simulation takes into consideration the external radiation (ambient) as well as the internal heat dissipated by the electronic components inside the satellite. Each simulation is executed automatically by taking the material solution genome as a simulation input.

The simulation results in a measure of the fitness of the material solution being tested. The fitness is a measure of the total product between patch emissivity and local temperature, the idea behind this approach is to maximize the amount of heat dissipated by the satellite.

$$Fitness = \sigma \cdot \sum_{k=1}^{N} \left(\varepsilon_k \cdot \int_{A\_k} t \cdot dA \right) \qquad (1)$$

Where $N = 40$ is the total number of patches under optimization, A_k is the area of a radiating patch k, ε_k is the emissivity factor of patch k and σ is the Stefan-Boltzmann constant.

5 THERMAL ANALYSIS

The satellite thermal behavior can be analyzed by looking at the balance between the heat being absorbed and the heat being emitted by the satellite. The heat contributions are shown in Figure 3 and summarized in Eq. 1. A thorough description of these concepts can be found in [14, 16]:

$$Q_{absorbed} - Q_{emited} + Q_{power-generated} = 0\,, \qquad (1)$$

where

$$Q_{absorbed} = G_{DS} \cdot A_{DS} \cdot \alpha \cdot \cos(\gamma) + G_{DS} \cdot A_E \cdot a \cdot \alpha \cdot vf + G_{IR} \cdot A_E \cdot vf \qquad (2)$$

$$Q_{emited} = A_{out} \cdot \varepsilon \cdot \sigma \cdot T^4\,, \qquad (3)$$

The parameters included in Eqs. 2 and 3 are summarized in Table 2.

Table 2. Parameters used for the thermal analysis [14, 16].

Parameter	Description	Value
G_{DS}	Solar constant.	1367 W/m^2
G_{IR}	Infrared radiation, which is a function of the Earth temperature and view factor.	221 W/m^2
α	Absorption of each material.	See Table 4
ε	Emission coefficient of each material.	See Table 4
γ	Angle between an imaginary ray linking the sun and the satellite, versus a vector normal to the face receiving direct sun light. One value is required for each face.	$\left[\frac{\pi}{2}, \frac{3\pi}{4}, \frac{3\pi}{4}, \frac{\pi}{4}, \frac{\pi}{4}, \frac{\pi}{2}\right]$
a	Albedo factor.	35%
vf	View factor [12]. Six values were used.	[0.24,0.6,0.6,0.02,0.02,0.24]
σ	Stephan-Boltzmann constant.	5,67e-8 W/m^2K^4
A_{SD}, A_E, A_{out}	Areas pointing toward the sun, earth and space respectively.	Defined by patch parameters.

The power $Q_{power\text{-}generated}$ is generated by the onboard electronics, such as batteries, the GPS, the transceiver, the camera, and the payloads. Although, these values are different for each satellite, the standard values used in this paper are summarized in Table 3. We assume that the thermal loads are homogenously distributed on the internal walls of the satellite.

Table 3. Nominal power generated by onboard electronic components [14].

Component	Day (W)	Nigth (W)
Alinco	1,92	1,92
PicoPacket	6,00E-02	6,00E-02
ATOMIC CLOCK	7,5	0
GPS	1,2	0
ABSL BATERY	0,1	0,3
MGT	0,3	1,5
MCU-2	1,00E-01	1,00E-01
ISIS TXRX	3,24	3,24
PDU	0,25	0,25
EPS-2	1,63	0,37
EPS-1	1,63	0,37
MCU-1	1,00E-01	1,00E-01
ARAZIM MGM	0,132	0,0055
GYRO	0,5	0,5
Total	**18,662**	**8,7155**

Although the materials to be used in the optimization can be selected from a large list of materials [2, 6], for this work the selection of the two materials utilized for the satellite is obtained from a pool composed by four materials which are summarized in Table 4.

Table 4. Materials used during simulations.

ID	Material	α	ε
1	BrillantAluminiumPaint	0.3	0.31
2	EpoxyAluminiumPaint	0.77	0.81
3	FinchAluminiumPaint	0.22	0.23
4	LeafingAluminiumPaint	0.37	0.36

6 RESULTS

Figure 5 shows results of applying our thermal optimization method on a 3U CubeSat. The illustration shows a solution obtained using our method, a human-made solution and a trivial naïve solution. The temperature color bar is shown on top, at the left side of corresponding solutions. Resulting surface heat pattern is shown with shades of gray at the central part of the figure. Corresponding values of fitness are presented at the bottom part of the figure. The naïve solution consisted on patches of equal size. The human-made solution was obtained by manual tuning.

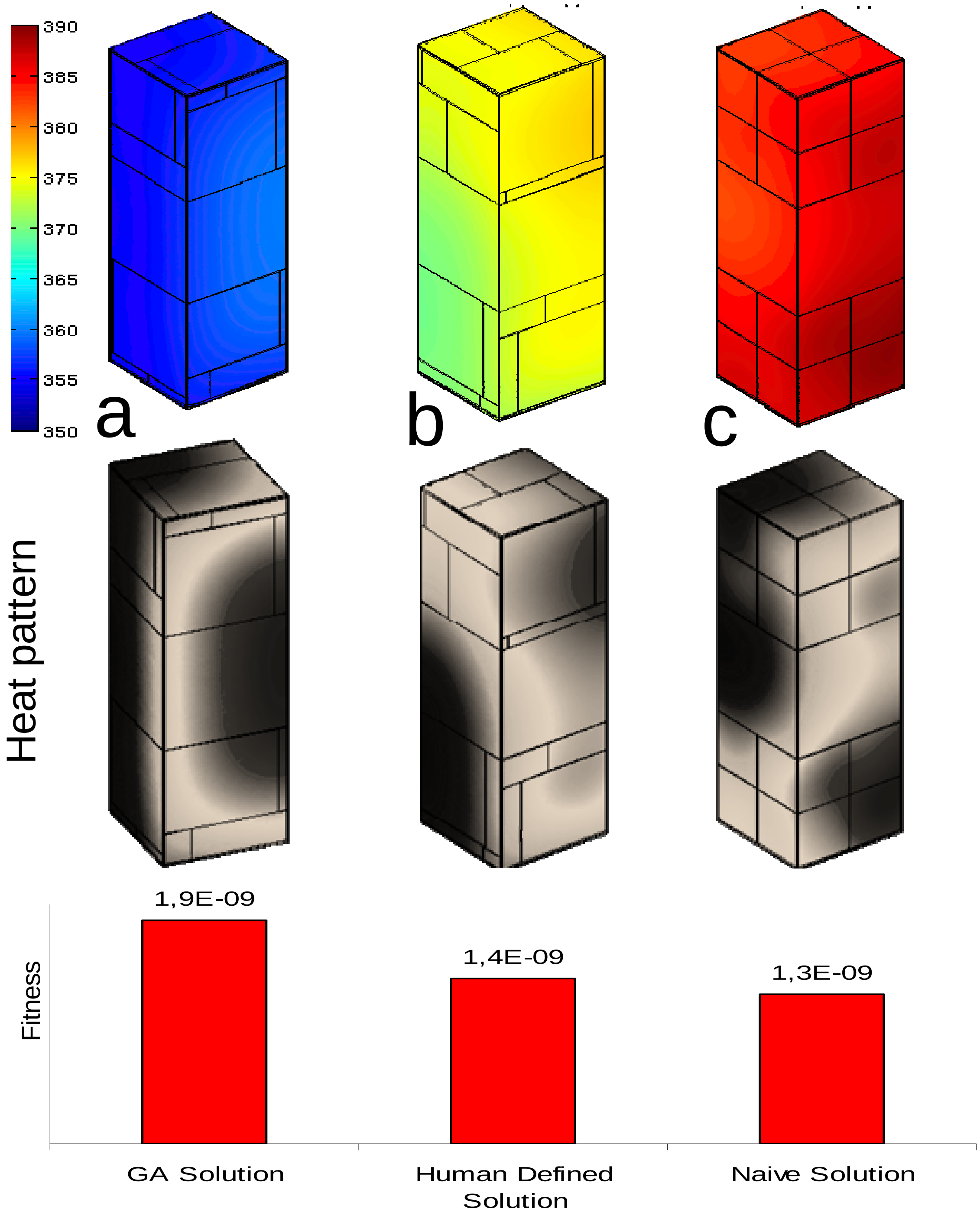

Figure 5. Results of applying the thermal optimization method on a 3U CubeSat. a, Solution obtained using the GA method. **b**, Human-made solution. **c**, Trivial solution. A temperature color bar is shown on top together with corresponding solutions. Resulting heat pattern is shown with varying shades of gray. Fitness values are shown at the bottom.

It was found, using GA, that homogeneous material patches, all made of EpoxyAluminumPaint (ID 2, Table 4), were better than using a heterogeneous material distribution. This is also natural since this material also shows higher emission coefficient.

7 CONCLUSION

In this study we have explored a technique for the automatic design of a satellite passive thermal control system. Due to the complexity of these type of systems, optimization is important on every state of development.

It appears convenient to use GA optimization when designing the material layout for a CubeSat passive thermal control. The technique allows the evaluation of many design options over a small period of time.

One limitation we found is the premature convergence to some mediocre solution [13]. One solution to this problem is to restart the algorithm. We expect to use the Age-Layered Population Structure (ALPS) [7] method in a forthcoming work.

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