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Engineering Notes

Designing for the Space Environment via Trade Space Exploration

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I. Introduction

SPACECRAFT design is an inherently complex problem. Even the most basic designs involve highly coupled systems. As a result, optimizing the design of spacecraft is especially challenging given the tradeoffs that occur between multiple competing objectives. Traditional optimization methods involve black box algorithms into which designers feed well-defined sets of criteria before beginning the process in order to determine the most preferred designs. Trade space exploration provides an alternative approach to solve this problem. The designer is kept in the loop and is allowed to make changes to the objectives, constraints, and preferences on the fly, based on the trends and insights that arise during the optimization process. The complexity of spacecraft systems makes it difficult to characterize the overall design space a priori, so this a posteriori approach lends itself well to the problem.

This work applies the trade space exploration process to the specific problem of designing for the space environment. A model is developed, and the estimated cumulative effects of the space environment on the lifetime and performance of the spacecraft are quantified. This model is then driven with a trade space exploration tool, and the most preferred designs are characterized.

II. Background

Operating in the space environment presents many unique challenges. A brief summary of the different effects the space environment has on spacecraft, as well as background information on trade space exploration, are now presented.

A. Space Environment Effects

The fact that a spacecraft is operating in a vacuum means that the only possible heat transfer mechanisms are either radiation or direct contact (between components, for example). The spacecraft as a whole absorbs energy from direct solar exposure, Earth albedo, and infrared emissions from the Earth while also producing internal heat as a result of power dissipation [1]. The main mechanism for heat rejection is radiation to space. The radiative properties of the spacecraft, however, can be altered by the presence of contamination.

Contamination occurs when molecules or atoms attach to the spacecraft surfaces. This material can originate from several sources (e.g., the plume of an onboard thruster, the space environment itself, or outgassing of spacecraft materials). Generally, spacecraft are designed in such a way so as to avoid direct impingement of thruster flow on a surface, although many thrusters produce some amount of backflow [2].

The presence of atomic oxygen in low Earth orbit [2] also poses a threat to spacecraft survivability. Because of its high reactivity and impact energy, atomic oxygen causes certain spacecraft materials, especially polymers, to erode in orbit [1,3]. The rate of erosion is quantified by the material's erosion yield. This value is determined for each material through testing, a large amount of which occurred on the long-duration exposure facility spacecraft [3].

Modeling and simulation of spacecraft glow in the infrared [4] through the visible [5] spectral regions from the Atmospheric Explorer satellites and by space shuttle have been studied. The studies show that the glow background radiation is primarily due to atomic oxygen reactions with molecular nitrogen to form nitric oxide in the ram portion of the spacecraft. The nitric oxide formed in the gas phase collides with the spacecraft surface and remains absorbed with a temporary residence time, which enables a chemiluminescent reaction due to subsequent atomic oxygen bombardment. As a result, this background glow could have an effect on the performance of onboard sensors. Solar arrays tend to lose efficiency over their lifetime as a result of the radiation environment in orbit. Because this is difficult to protect against, the effect is generally included in spacecraft design. These losses can be significant: up to a 19% power loss in the first year alone [6].

B. Trade Space Visualization

Recently, Balling has introduced the design by shopping paradigm whereby designers could shop for the best design [7]. In particular, designers are allowed to change the importance of each objective in a design problem after the trade space has been populated with design alternatives. This means that designers no longer have to commit to an ideal design early in the process, because they can now formulate a set of optimal design criteria a posteriori, after exploring and visualizing the trade space, as opposed to a priori [8]. This research was the motivation that led to the development of the Applied Research Lab Trade Space Visualizer (ATSV) [9], which is used in this research.

ATSV is an engineering decision-support tool that allows users to search for an optimal design with visualization tools and the ability to quickly identify nondominated designs and Pareto sets. Pareto sets (more precisely, Pareto-optimal sets) contain the optimal points in a trade space. Analysis models are developed separately (e.g., in MATLAB or Microsoft Excel) and then linked with the Java-based ATSV. Once linked, ATSV samples the trade space by driving the model using one of several samplers [10]. The designer can then explore the trade space using a variety of multidimensional data visualization tools. ATSV allows designers to easily explore and visualize highly coupled systems inherent in spacecraft design, as well as avoid unintended consequences of particular combinations of variables, such as higher-than-expected levels of contamination. In addition, the flexibility of ATSV makes it suitable for both component-level and system-level design visualization and optimization.

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III. Model Development

The traditional focus in spacecraft design has generally been to identify and mitigate the effects of the space environment rather than design to minimize these effects [11]. However, there is a host of spacecraft anomalies that suggest that the guidelines that are currently in place are not given the level of priority required [12,13]. The recent push toward small, low-cost, and high-reliability satellites [14] suggests that space environment effects need to be given a higher priority [15,16]. The modeling efforts introduced in this Note feature high-level spacecraft design properties and how they can minimize space environment effects.

A simple example with simple models is used to demonstrate the capability of trade space visualization. While these models are not as complex as models used in a detailed spacecraft design process, they are shown simply as examples. A user can easily insert the model of their choice into the trade space visualizer to obtain the fidelity of the design desired. Users can also customize their models, depending upon the regime of the orbits under consideration. Here, an Earth-sensing satellite is used as an example. The main priorities for its design are low cost, high reliability and lifetime, and good sensor performance. The representative sensor is a wideband optical sensor, which tends to be very sensitive to contamination due to the cryogenic temperatures at which they are maintained (in order to reduce noise).

The simple models that have been developed are not meant to be all-inclusive complete spacecraft design models; instead, they focus on specific aspects of the design and are meant to provide estimations and comparisons to help guide the spacecraft design process. All calculations are made assuming worst-case conditions, so the proposed results are not underdesigned. The calculations are done entirely in Microsoft Excel, which is later linked to ATSV for trade space exploration.

The main input variables in the model are the dimensions of the spacecraft, the thermal control material (chosen from a database of options [17]), and the thruster selection (chosen from a database of options [18]). The model also has a variable orbit altitude (225–650 km) that is chosen at the beginning of the optimization process and kept constant throughout. The constraints on the satellite dimension inputs are dictated by the launch vehicle fairing dimensions (approximately, a cylinder 3 m in diameter and 8.9 m long [19]).

A. Thermal Balance

A simple thermal rate balance equation is used, accounting for direct solar exposure, Earth infrared emission, Earth albedo, heat radiated to space, and heat generated within the spacecraft. One of the main variables in the thermal balance is the thermal control material absorption coefficient, which changes with the level of contamination. The thermal portion of the model checks that the modified, contaminated value for absorption satisfies the balance equations. The dimensions of the spacecraft are linked both to the radiative heat transfer rate as well as the amount of internal heat generation.

B. Contamination

The amount of propellant expelled in a given time frame is estimated, assuming the only use for the thrusters is orbit stationkeeping, specifically drag compensation. A value of 1% of the total propellant mass is assumed to come into contact with sensitive surfaces on the spacecraft [2]. This is a high estimate, representing the worst-case scenario for spacecraft self contamination.

Contaminants will remain on a surface for a finite time, referred to as residence time. This value depends on the temperature of the surface T and the desorption activation energy ΔH for the contaminant. A conservative value of 66 days is given by Eq. (1) and is used as an estimate for the residence time on the spacecraft surface:

$$\tau = \tau_0 \exp\left(\frac{\Delta H}{kT}\right) \tag{1}$$

where τ_0 is the residence time at a nominal temperature, and k is Boltzmann's constant. Note also that solar arrays will have a much lower residence time (on the order of hours, a result of the higher operating temperature).

Finally, the distribution of the contaminant film on the surface of the spacecraft is estimated to be evenly distributed across the entire surface. This is clearly an oversimplification; actual contaminant deposition patterns are much more complex (and as a result, difficult to model). However, this approach suits the high-level nature of this study well, and a more refined deposition method could be added later as the user further narrows the design space. This method yields a thickness, which translates to a change in absorptance for the material and a loss of efficiency for the solar arrays.

C. Atomic Oxygen Erosion

This portion of the model comes from a relatively simple concept. For a given lifetime estimate, the end of life is considered to be when the thermal control material has completely eroded away. Using the chosen material's reaction efficiency (RE) and the atomic oxygen density at a specific orbital altitude, this is a simple calculation [1]:

$$\frac{dx}{dt} = (\mathrm{RE})\phi \tag{2}$$

where φ is the atomic oxygen fluence, which is the product of the ambient concentration of atomic oxygen and the orbital velocity of the spacecraft. The rate of erosion is given by dx/dt. A typical value of RE is on the order of 10^{-24} $\mathrm{cm^3/atom}$, which yields an erosion rate of less than 0.01 mm/year. However, often the coating is applied in a very thin layer, so this can become an issue, particularly on long-life missions.

D. Solar Array Sizing

Solar array sizing is similar to other methods of estimating required solar array area, except the inputs and outputs are reversed. Rather than designing a solar array to meet a given lifetime requirement, the maximum lifetime is estimated given the design parameters of the array. The required end of life power is given as a function of spacecraft size. The degradation rate due to radiation is chosen to be a constant 4% per year, based on past missions [20]. This is a simplification, as actual degradation rates due to radiation will vary by a host of factors: orbit, attitude, and launch time among them. However, the purpose of this study is to investigate designs relative to one another, with the planned orbit remaining constant across the design space. Therefore, the constant rate serves as a good method for an early trade space exploration.

Common values for the various efficiency loss terms are used, except in the case of loss due to contamination, which is taken from the contamination portion of this model. However, instead of giving a desired lifetime and outputting the required size, the maximum size is given as an input, and the resulting lifetime is calculated.

E. Other Effects and Objective Evaluation

The main outputs of the model correspond to the three objectives: 1) a lifetime estimate, 2) a launch cost estimate, and 3) an optical payload performance estimate. Different spacecraft lifetime estimates are calculated from solar array degradation, atomic oxygen attack, and propellant budget, and the lowest lifetime is chosen.

Cost is a function of spacecraft mass in this study, although a more refined cost model should be added as the fidelity of other subsections is increased. The optical performance metric consists of a weighted objective function combining estimates of the intensity of luminescence from spacecraft glow and the estimated thickness of condensed contaminants. This objective is called glow in the model, and it is desirable for it to be minimized (in order to maximize optical sensor performance).

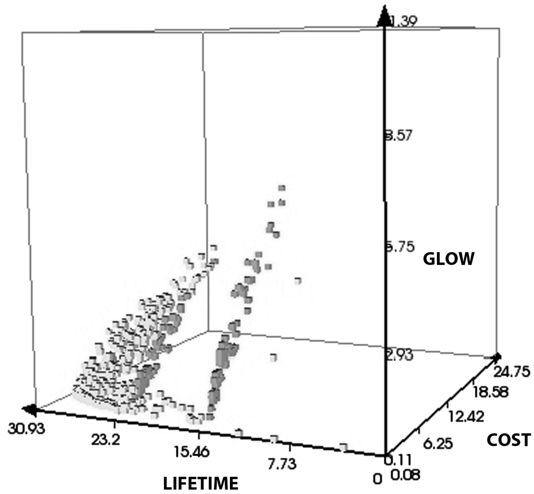

Fig. 1 All feasible designs generated.

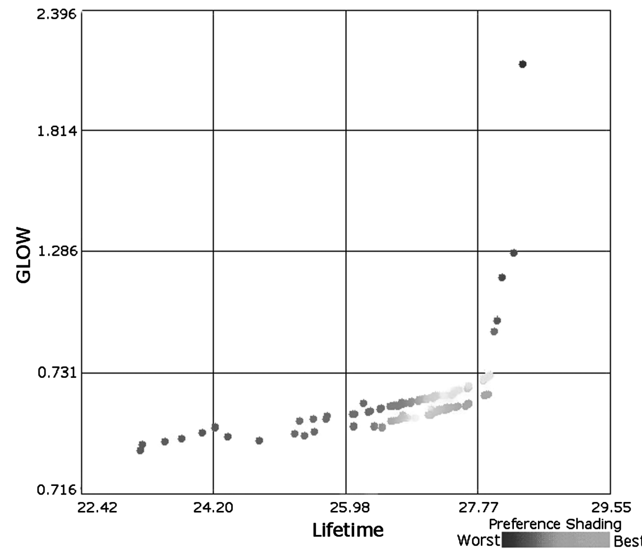

Fig. 3 Developed Pareto designs: glow vs lifetime.

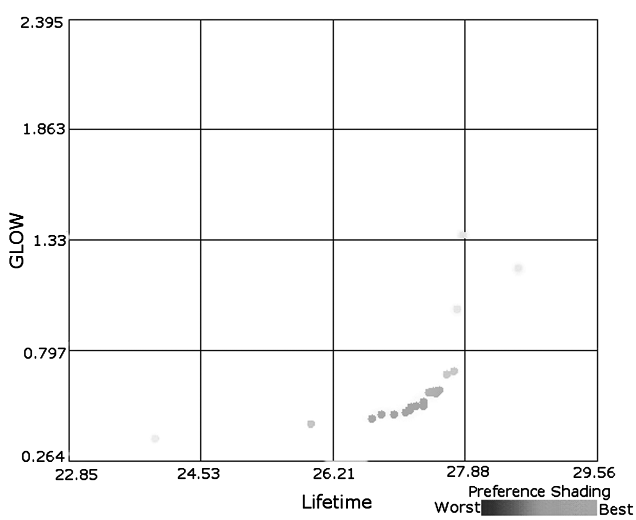

Fig. 2 Initial Pareto designs: glow vs lifetime.

IV. Results and Discussion

A. Design Space

For the purposes of demonstration, only the objective set that best optimizes all three objectives is presented (other cases are presented in [21]). The total number of designs generated for this particular problem was 6650. Of these, 2437 were considered feasible. Because the calculations were done in Excel, the designs were generated very quickly, and computer processing time was not a prohibitive issue. An overview of the entire design space (excluding infeasible designs) is shown in Fig. 1. Each axis represents an objective: namely, cost, lifetime, and glow. Lighter points represent designs limited by the solar panel lifetime, and darker points are designs limited by the propellant budget.

Atomic oxygen effects rarely drive the lifetime estimate, most likely because only certain thermal materials are subject to erosion. Figures 2 and 3 show an evolution of the Pareto sets for this space in terms of the performance objective (glow) versus lifetime. In both, shading represents preference: more preferred designs appear lighter. Figure 2 is early on in the design process. ATSV is then used to generate samples near this front, resulting in the more developed Pareto space shown in Fig. 3. This Pareto space is fairly straightforward, and the fact that two fronts appear indicates that the

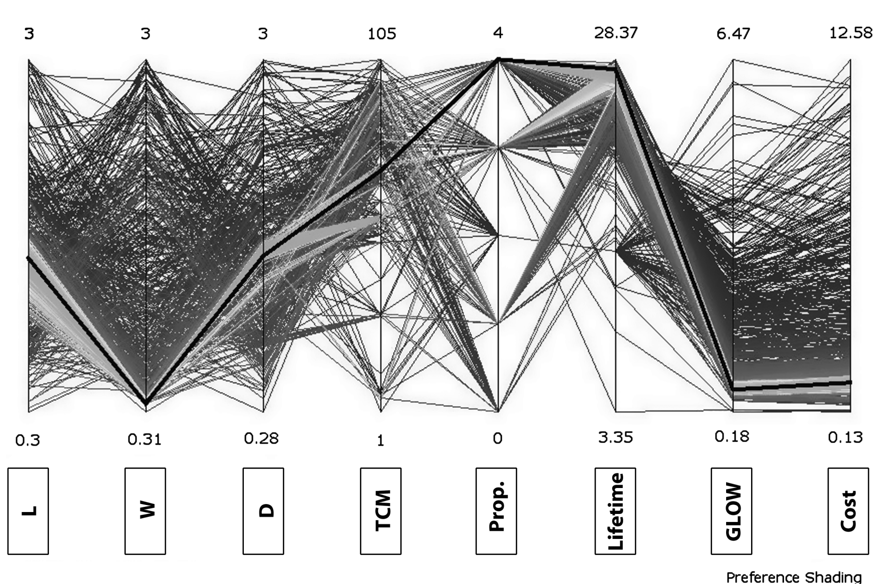

Fig. 4 Parallel coordinate plot of entire trade space (L, W, and D denote length, width, and depth, respectively, and TCM denotes thermal control material).

three-dimensional Pareto space is an envelope around the outer edge of the trade space.

The parallel coordinates plot presented in Fig. 4 serves to show the diversity of the trade space. Each design is represented as a line, and each vertical axis is either a design variable or objective. Again, designs are shaded according to preference, and the best single design, as chosen by ATSV, is in bold. Feasible designs are found across nearly the entire range of each design variable, including every propellant option.

B. Trends

Although all of the results are not included here, several sets of objective values were studied. While acceptable designs are generated across the entire range of thermal control materials, most black materials do not yield feasible designs, which is simply a result of the specific heat rates for this example spacecraft. Most runs, in fact, converged on white paint thermal materials. Similarly, while all propellant options yield feasible results, the majority of runs converge on either hydroxylammonium-nitrate-based, or O_2/RP-1 propellants. It is interesting to note that neither propellant is an extrema in terms of bulk density or specific impulse, which is indicative of the tradeoffs made within the model that was developed for this study.

The industry trend toward smaller spacecraft is reflected in this analysis. Although the maximum volume of the satellite in this problem is restricted to 9 m^3, all preferred designs converge to a fraction of that, generally less than 30%. In addition, most preferred designs have a combination of two large dimensions and one small, as can be seen in Fig. 4. This allows the spacecraft to have a maximum solar panel size (on the two largest faces) while having a minimum projected area (the smallest face), which reduces spacecraft glow.

V. Conclusions

This work demonstrates the utility of applying trade space exploration to the problem of spacecraft design. The complexities and coupled systems can be easily characterized using a variety of multidimensional data visualization tools. Most important, this technique can be scaled to produce much more useful results and insights into the underlying tradeoffs in the system.

While the resolution of the model developed here is relatively low, it could be combined with a series of more refined and higher-fidelity models to produce a workable set of detailed designs. Rather than using ATSV to choose a single best design with this model, the most preferred set of designs can be more useful. These could then be exported and fed into a model with higher fidelity, repeating the optimization process. The flexibility of ATSV means that the level of detail could range from a higher mission-level design all the way down to component-level detail. The utility of trade space exploration is limited only by the difficulty in creating such detailed models.

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