Selection of Spacecraft Optical Coatings for Temperature Control

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Nomenclature

A = area, ft^2
B = albedo flux, Btu/hr-ft^2
C_i = specific heat of element i, Btu/lbm-°R
D = $\{\Sigma_m[(\partial T_{ik}/\partial \epsilon_m)^2 + (\partial T_{ik}/\partial \alpha_{sm})^2]\}^{1/2}$
E_{ik} = error in heat-balance equation for element i and environmental condition k [Eq. (3)], Btu/hr
F_{i-a} = radiation view factor from surface i to the earth-reflected solar energy (albedo); F_{i-E}, to the earth flux; F_{i-S}, to the sun
K_{ij} = conductance between elements i and j, Btu/hr-°R
M_{ij} = a term of the inverse of the matrix of conduction and linearized radiation terms
$Q_i^{(k)}$ = internally generated heat in element i and environmental condition k, Btu/hr
$Q_{Ei}^{(k)}$ = total earth-emitted heat flux incident on element i in environment k; $Q_{Si}^{(k)}$ = total solar heat flux, Btu/hr
R_{ij} = radiant interchange coefficient between elements i and j, ft^2
S = solar constant, Btu/hr-ft^2
$\bar{T}_i$ = most desirable temperature of node i, °R
T_{ik} = temperature of element i in environmental state k, °R
W_i = weight of element i, lbm
W = worst-case function defined by Eq. (1)
α_{Si} = solar absorptance of surface i
δ_{im} = the Kronecker delta ($= 1$ if $i = m$; $= 0$ if $i \neq m$)
δT_i = allowable temperature variation of element i, °R
ϵ_i = infrared emittance of element i
θ = time, hr
σ = Stefan-Boltzmann constant, 0.1713×10^{-8} Btu/hr-ft^2-(°R)^4

Introduction

BY properly balancing the external optical coatings, the thermal designer can obtain a wide range of interior space-vehicle temperatures. However, the large numbers of variables, both dependent and independent, make a trial-and-error procedure too costly to be pursued beyond one or two tries. Costello, Harper, and Cline[1] showed in 1963 how the solar absorptance might be optimized under some special circumstances: 1) at only one point in the vehicle was temperature control critical, and 2) all heat transfer was by radiation. This Note summarizes more recent work[2,3] by the present authors in which these restrictions are removed.

In optimizing, we want

$$W \equiv \max_{i,k} \left| \frac{T_{ik} - \bar{T}_i}{\delta T_i} \right| \leq 1 \tag{1}$$

for all k variations in the environmental and internally generated heating rates. W is a measure of the merit of the temperature-control system, which is adequate if W is less than unity and optimum when W is a minimum.

Solution to the Heat-Balance Equations

To determine the T_{ik} and W, the first law of thermodynamics can be applied to each element in the vehicle, yielding equations of the form

$$w_iC_i \frac{dT_i}{d\theta} = \sum_{\substack{j=1 \\ j\neq i}}^{N} [K_{ij}(T_j - T_i) + R_{ij}\sigma(T_j^4 - T_i^4)] + \alpha_{Si}A_i(F_{i-S}S + F_{i-a}B) + \epsilon_iA_i(F_{i-E}E - \sigma T_i^4) + Q_i \tag{2}$$

$$i = 1,2, \ldots, N$$

Two important assumptions can be made to simplify the solution: 1) the F_{i-S}, F_{i-a}, F_{i-E}, and Q_i values, which are usually discontinuous functions of the orbit parameters, will be considered in p discrete sets, $F_{i-S}^{(k)}$, $F_{i-a}^{(k)}$, $F_{i-E}^{(k)}$, and $Q_i^{(k)}$, $k = 1,2, \ldots, p$ (the corresponding temperature will be designated T_{ik}); and 2) the optical properties will be selected on the basis of the time-averaged temperatures. Although optimization of the ϵ_i and α_{Si} on the basis of average temperatures would seem at first to negate the present method, at least two vehicle-design characteristics reinforce the assumption: 1) critical elements of the vehicle are usually buried within the vehicle, so that their temperature fluctuations over one orbital period are small; and 2) the temperature fluctuations of critical externally mounted components can be minimized by increasing the effective mass of the component, insulating it, or keeping both ϵ and α_S for that exposed component small.

In time-averaged form, Eq. (2) can be rewritten as

$$E_{ik} = \sum_{\substack{j=1 \\ j\neq i}}^{N} [K_{ij}(T_{ik} - T_{jk}) + R_{ij}\sigma(T_{ik}^4 - T_{jk}^4)] + A_i\epsilon_i\sigma T_{ik}^4 - [\alpha_{Si}Q_{Si}^{(k)} + \epsilon_iQ_{Ei}^{(k)} + Q_i^{(k)}] = 0 \tag{3}$$

$$i = 1,2, \ldots, N; \quad k = 1,2, \ldots, p$$

provided we accept the small error in letting the time averaged T_i^4 equal the fourth power of the time averaged T_i. Equation (3) is solved by linearizing it in terms of the $\bar{T}_i$ and an initial estimate of ϵ_i.[4] An iterative scheme of the form

$$T_{ik}^{(n+1)} = T_{ik}^{(n)} - \textstyle\sum_j M_{ij}E_{jk}^{(n)} \tag{4}$$

can then be used, where $\{M_{ij}\}$ is the inverse of a matrix of conduction and linearized radiation terms. Since $\{M_{ij}\}$ is independent of n and k, the inverse need be found just once for each optimization problem.

Optimization Scheme

The problem is to minimize W in Eq. (1) with respect to the α_{Si} and ϵ_i, where the T_{ik} are determined from Eqs. (3) and (4). The method used to minimize W was basically the maximum-rate-of-descent (MRD) method; an acceleration procedure was sometimes used along ridges. For MRD to be usable, the derivatives, $\partial W/\partial \epsilon_i$ and $\partial W/\partial \alpha_{Si}$, must be evaluated. Equation (3) can be differentiated with respect to both ϵ_m and α_{Sm} to obtain two systems of linear equations for $\partial T_{ik}/\partial \epsilon_m$ and $\partial T_{ik}/\partial \alpha_{Sm}$. The systems can be solved explicitly for the derivatives to yield, for example,

$$\partial T_{ik}/\partial \epsilon_m = [Q_{Em}^{(k)} - A_m\sigma T_{mk}^4] \cdot J_{im}^{(k)} \text{ (no sum)} \tag{5}$$

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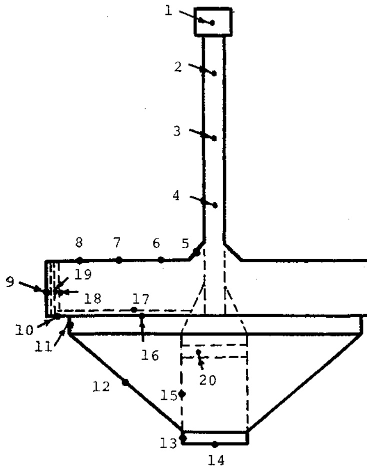


Fig. 1 Nodal breakdown of EPE-D spacecraft.

where $\{J_{im}^{(k)}\}$ is the inverse of the coefficient matrix in the linear systems. Then according to the MRD scheme,

$$\epsilon_m^{(\text{new})} = \epsilon_m^{(\text{old})} - |\partial W/\partial T_{ik}|/(\partial W/\partial T_{ik})\cdot(\partial T_{ik}/\partial \epsilon_m)\Delta S/D \tag{6}$$

where (ΔS) is a suitable and somewhat arbitrary step size that must be chosen small enough for the MRD method to converge to the minimum point. The new α's are found in the same way.

The optimization process can now be executed as follows: 1) make initial estimates of the ϵ_i and α_{Si} $(i = 1,2, \ldots, N)$; 2) calculate the temperatures for every element i and every heating condition k from Eqs. (3) and (4); 3) calculate the worst-case function by Eq. (1); 4) calculate partial derivatives of W by Eqs. (5) and (1); and 5) determine new values of the ϵ_i and α_{Si} from Eq. (6).

This process is repeated until the method fails to improve W for an arbitrarily small ΔS.

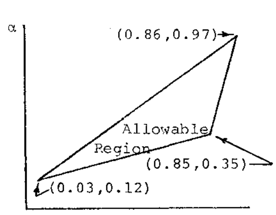
Example Application

The application considered is the NASA Explorer XXVI (EPE-D) spacecraft. (Reference 3 also presents an example of a hypothetical cylindrical satellite.) A schematic of the vehicle is presented in Fig. 1 showing the element numbers as used in the analytical model developed at NASA Goddard. Approximately 20 man-days and 4 hr of IBM 7094 computer time were required to "optimize" the coating pattern by the former trial-and-error method. The present method, applied with no foreknowledge of the values obtained at NASA, yielded the results (shown on Table 1) after only 2 man-days and $\frac{1}{2}$ hr of IBM 7094 time. The respective values of W are also shown on this table. The advantages of cost saving and better performance are evident.

Table 1 Summary of EPE-D designs

Temperature Ranges - °R			
Node	Allowable Limits	Predicted Limits	
		Using NASA Coatings	Using Present Method
1	420 - 564	468 - 537	478 - 542
6	456 - 600	471 - 600	477 - 572
7	456 - 600	468 - 599	475 - 574
8	456 - 600	470 - 578	482 - 566
17	492 - 546	495 - 535	499 - 540
18	492 - 546	509 - 524	501 - 540
20	474 - 582	488 - 564	517 - 572
Value of "W"		1.0	0.78

Coatings					
Node	Allowable Limits	NASA		Present Method	
		α	ε	α	ε
1		.278	.260	.152	.118
2		.170	.112	.149	.115
3		.170	.112	.153	.116
4		.170	.112	.155	.118
5		.320	.260	.150	.119
6		.310	.240	.167	.196
7		.310	.240	.168	.199
8		.310	.240	.168	.198
9		.970	.850	.207	.166
10		.970	.850	.146	.123
11		.970	.850	.170	.147
12		.160	.110	.163	.183
13		.310	.240	.153	.123
14		.350	.850	.144	.116



Conclusions

The optimization process outlined above has proven to be a significant improvement over former trial-and-error methods, both in cost and performance. Because of the general nature of the temperature equations used, the method has a wide range of applicability in spacecraft design.

References

[1] Costello, F. A., Harper, T., and Cline, P., "A Rational Approach to the Selection of Satellite Optical Coating Patterns for Temperature Control," preprint 63-HT-41, Aug. 1964, American Society of Mechanical Engineers.

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[4] Costello, F. A. and Schrenk, G. L., "Numerical Solution to the Heat Transfer Equations with Combined Conduction and Radiation," *Journal of Computational Physics*, Vol. 1, No. 4, 1967, p. 541.

Effect of Debris Shielding on Energy Partition

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In a recent paper Kuby and Lewis[1] reported on the damage produced by impingement of a cloud of micron-sized aluminum-oxide particles on aluminum targets. Their results indicated that the impingement damage was considerably less than that expected on the basis of single impact data but, although several possible reasons were advanced, no clear explanation was found. Subsequently[2,3] it has been shown that for a wide range of incident particle mass flux, a debris layer is formed immediately ahead of the target, partially shielding it from subsequent impingement by oncoming particles. In the presence of shielding, relative damage, i.e., damage per incident particle, was observed to increase with decreasing particle mass flux, exceeding even that predicted by direct extension of single impact data, while

Table 1 Test results of particle impingement heating

Number of tests	Target material	$\dot{m}_p$, g/cm²-sec	T_0, °K	q_0, cal/cm²-sec	$\dot{T}$, °K/sec	α
3	1100-F	1.3	300	560	140–165	0.080–0.095
3	6061-T6	1.3	300	560	150–170	0.086–0.10
5	1100-F	1.3–1.5	550	1000–1100	230–315	0.065–0.105
3	2024-T3	1.2–1.4	550	900–1000	240–270	0.085–0.095
3	6061-T6	1.2–1.4	550	900–1000	160–215	0.05–0.075

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