Design of the ISAT-1 Satellite Thermal Control System

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Nomenclature

AA	orbit semi-major axis, nautical miles
Ai	area of surface, m ²
A _o	incident area of solar energy, m ²
A _p	incident area of planetary emission, m ²
A _{sat}	total area of satellite, m ²
Ax	thermal contact area between two surfaces, m ²
Ci	thermal capacitance of node i, J/K
c	specific heat, J/kg·K
EE	orbit eccentricity
J _{i-x}	script-F between surface i and energy source x
J _{i-i}	script-F between surface i and j
G _{i-i}	conductance between surface i and j
Ja	albedo energy flux, W/m ²
J	planetary emission flux W/m ²
k	thermal conductivity, W/m·K
1	distance between mass centers of two surfaces, m
M _x	flux magnitude from arbitrary source x, W/m ²
N	total number of nodes
N+1	deep-space background node
P _i	total number of rays emitted
Q _{i.x}	heat transfer to surface i from source x, W
q _{i-i}	heat transfer from node i to j, W
RÍ	orbit inclination to the equatorial plane, °
R _{i-i}	radiant exchange factor between surface i and j, m ²
Τ̈́	temperature, W
Z _{i-j}	ray tally between surface i and j
α	orbit argument of apogee
α_i	absorptance for surface i
Δτ	time step, s
δ	orbit inclination of the equatorial plane to the ecliptic plane, °
3	surface emittance
σ	Stefan-Boltzmann constant, 5.670×10 ⁻⁸ W/m ² ·K ⁴
Ψ	orbit sun day angle, °
Ω	orbit right ascension, °

Subscripts:

i and j refer to surfaces or nodes i and j

o, p, a and x refer to solar, planetary, albedo, and arbitrary source, respectively sat or s and space refer to satellite and deep-space background, respectively

1.0 Introduction

Satellite technology improves the daily lives of almost all human beings. Satellites aid in communication, weather prediction, and global positioning. Other uses for satellites include conducting scientific experiments in micro gravity and space exploration missions. A major constraint placed on all satellites is the cost of launch. With a larger or heavier satellite, a larger more powerful launch vehicle is need. This larger launch vehicle increases the overall satellite cost. The development of smaller satellites promises lower cost by relaxing the launch vehicle cost constraint. Smaller and lighter satellites are less expensive to launch, and are expected to provide a significant role in the overall space program. This expectation can only be met by densely packing a satellite with needed components. When components are densely packed, the dissipation of internally generated heat becomes more critical. With less space for thermal control devices, substantial care must be used in the design of a satellite thermal control system (TCS). One such satellite is the Iowa Satellite Project (ISAT-1) (Seversike et al., 1994).

The purpose of the ISAT-1 is to develop a small relatively inexpensive satellite that will benefit the State of Iowa. This satellite has a hexagonal prism geometry, as shown in Figure 1, with a height of 0.634 m and a side panel width of 0.170 m.



Figure 1 ISAT-1 Geometry and Payload Configuration

At the current stage of development for the ISAT-1 satellite, the specifics of the orbit and component temperature tolerances are not known, but the general configuration of the payload may be divide into three shelves. The objective of this paper is to

formulate the design methodology of the ISAT-1 TCS and to investigate various internal heat dissipation pertinent to the operation of the ISAT-1

2.0 ISAT-1 Thermal Control Design Process

The thermal design of the ISAT-1 satellite is divided into three main tasks: simulation modeling, selection of the thermal control system (TCS), and thermal balance testing. Simulation modeling is performed in order to determine the effects of the thermal space environment on the satellite. Through this mathematical modeling, the selection or synthesis of the TCS is performed. Once the selection of the TCS is complete, the qualification of the TCS is performed by thermal balance testing.

Work in mathematical modeling of the satellite is being performed using a thermal analysis software package, while preliminary work in the thermal balance testing is currently being conducted on a scaled down model of the ISAT-1 satellite. Selection of the TCS in previous studies has suggested a passive TCS for simplicity and assurance of operation, but at this point no specific TCS has been selected.

The thermal analysis software divides the simulation modeling into two distinct steps. The first step is to determine the radiation heat inputs by creating a preliminary model of the satellite. This preliminary model is made by discretizing the satellite into different sections or surfaces where each surface is isothermal, and then entering both the surface properties and orbital parameters into this discretized mathematical representation of the satellite. From this preliminary model, the radiation exchange factors and the external heat inputs are determined. The second step is to generate an primary model that represents the satellite as a conductance/capacitance network. This network is determined by the thermal and physical properties of the satellite. By adding the radiative exchange factors, the external heat inputs, and the external satellite-deep space radiation heat transfer paths to the primary model, a thermal solver is used to determine both the transient temperature responses and the heat exchanges that the satellite undergoes.

The selection of the TCS is by synthesizing or continuously adjusting the geometry, materials, and radiation properties until all the temperature tolerances are met. When selecting a TCS, the first step is to determine the temperature tolerances of the various satellite components. Once this is accomplished, the simulation model may be utilized to predict the temperatures that the satellite components will experience. The satellite component temperatures are then compared to the tolerances. For a component whose temperatures do not comply with its tolerances, the heat exchanges may be examined to determine the heat sources that cause noncompliance. Alternatives to rectifying the problem may be examined, such as applying different surface coatings or adding insulation to the satellite model. The heat exchanges are then controlled, thereby, maintaining the satellite component temperatures within the set temperature tolerances. The maintaining of the satellite components with the set temperature tolerances is thermal control.

Once the TCS has been selected, thermal balance testing is performed to identify any errors in the simulation model that may require some modification to the proposed

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TCS or to verify the TCS operation and effectiveness. Thermal balance testing exposes a satellite to a simulated space thermal environment on the ground. The three types of thermal balance testing are: model testing; prototype testing; and qualification testing. Model testing is performed on a scaled down model of the actual satellite, while prototype testing is performed on a near flight configurated satellite, with all subsystems operational. The qualification test is performed on the flight ready spacecraft prior to launch, and exposes the spacecraft to hot, cold, and nominal transient conditions.

3.0 Thermal Design Analysis

The thermal design analysis reported in this study applies to the low-Earth orbit of ISAT-1. With this in mind, the mechanisms of heat transfer are examined to obtain an understanding of the space thermal environment. Once this is accomplished, the simulation of this environment using a simulation modeling is discussed.

3.1 The Space Thermal Environment

Because space thermal environment has no convecting medium, the space thermal environment external to the satellite is dominated by radiation heat transfer. The three external radiation sources that affect the satellite temperatures are the direct solar energy, the solar energy reflected to the satellite from the earth (albedo), and the earth thermal radiation emission.

The magnitudes of these external sources fluctuate due to seasonal and orbital effects. Table 1 shows the approximate flux magnitudes for direct solar S_0 , solar albedo energy J_a , and planetary emission J_D , for a satellite in low earth orbit.

Radiation	Flux
Source	Magnitude
	$[W/m^2]$
S _o	1340 ± 32
J _a	576 ± 362
J _n	199 ± 64

Table 1	Magnitud	les of	external	heat	inputs
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The magnitude of S_0 is dependent on the season or position of the earth with respect to the sun. Additionally, there is no direct solar when the satellite is in the Earth shadow. The magnitude of J_a is dependent on the inclination of the satellite orbit, where the reflectivity of the earth surface is a function of longitude. In the polar regions the reflectivity is high and near the equator the reflectivity is low causing J_a to vary. J_p also varies, but as a function of the Earth surface temperature.

Using these external sources, a lumped capacitance transient energy balance on the satellite may be constructed as

$$(mc)_{sat} \left(\frac{dT}{d\tau}\right)_{sat} = \alpha_s A_0 S_0 + \alpha_s A_a J_a + \varepsilon A_p J_p - \varepsilon A_{sat} \sigma \left(T_{sat}^4 - T_{space}^4\right)$$
(1)

In this equation, mc is the lumped capacitance of the satellite, $(dT/d\tau)_{sat}$ is the change in the satellite temperature with respect to time τ , α_s is the satellite surface absorptance, ε is the satellite surface emittance, σ is the Stefan-Boltzmann constant, T_{sat} is the satellite temperature, and T_{space} is the space background temperature. Also, A_0 , A_a , and A_p are the areas in which solar, albedo and planetary fluxes, respectively, are incident and A_{sat} is the entire satellite surface area. The deep-space emission has been added to this equation, just as the earth emits thermal radiation, so does the satellite. This emission is to a space temperature of 2.7 K.

Although Equation (1) represents accurately the lumped capacitance satellite transient temperature response, it is desirable to determine the heat inputs at discrete positions in the orbit. The external source heat input $Q_{i,x}$ for surface i from source x at a particular instant may be represented by

$$Q_{i,x} = \alpha_i A_i \mathcal{F}_{i-x} M_x \tag{2}$$

where α_i is the absorptance of surface i, A_i is the area of surface i, \mathcal{F}_{i-x} is the script-F between the external source x and surface i, and M_x is the flux magnitude from source x. For the earth emission, which is in the infrared region, the emittance of the surface is used in place of the absorptance.

Internally, the satellite conducts heat and exchanges radiation just as if it were on the ground. Conduction is described by Fourier's law of conduction, which, for onedimensional conduction between two surfaces, is

$$q_{i-j} = \frac{kA_x}{l} (T_j - T_i)$$
(3)

In this equation, q_{i-j} is the conductive heat transfer from surface i to surface j, k is the material conductivity, A_x is the transfer area between these surfaces, l is the distance between of the surfaces, and T_i and T_j are the surface temperatures. For, more general applications, the leading terms in Equation (3), are grouped into the conductance G_{i-j} as

$$G_{i-j} = \frac{kA_x}{l}$$
(4)

The conductance between node pairs is determined by the geometry between the nodes and the mean of the thermal conductance between these nodes.

The internal radiation exchange is described by

$$Q_{i-j} = \varepsilon_i A_i \mathcal{F}_{i-j} \sigma \left(T_j^4 - T_i^4 \right)$$
(5)

In this equation, ε_j is the emittance, A_j is the area of surface i, F_{i-j} is the view factor between surface i and j, and T_i and T_j are the temperatures of surfaces i and j, respectively. The first three terms in Equation (5) are consider the radiation exchange factor R_{i-j}

$$\mathbf{R}_{\mathbf{i}-\mathbf{j}} = \boldsymbol{\varepsilon}_{\mathbf{i}} \mathbf{A}_{\mathbf{i}} \boldsymbol{\mathcal{F}}_{\mathbf{i}-\mathbf{j}} \tag{6}$$

The most difficult factor to determine is \mathcal{F}_{i-j} , the fraction of blackbody energy emitted from diffuse surface i absorbed by diffuse surface j taking into account direct transport as well as reflected energy to surface j be reradiating wall or other gray surface.

Both, the G_{i-j} and the R_{i-j} are represented as shown in Figure 2. The thermal connection between any two arbitrary surfaces is defined by a matrix of G_{i-j} 's and R_{i-j} 's.



Figure 2 Radiative and conductance connections

A combined conduction and radiation analysis is performed on the satellite. In this type of analysis, the internal component temperatures of the satellite are determined.

Finite-volume techniques are common to conduction analysis and are modified to account for radiation. The finite-volume technique discretizes the satellite into nodes that represent the average temperature in that region. Each node has a capacitance. The temperature of a node is influenced by the surrounding nodes, where conduction and radiation heat exchange may occur. Using the finite-volume technique, the energy balance on node i of the satellite is expressed as

$$(mc)_{i}\left(\frac{dT}{d\tau}\right)_{i} = \sum_{j=1}^{N} G_{i-j}(T_{j} - T_{i}) + \sigma \sum_{j=1}^{N+1} R_{i-j}(T_{j}^{4} - T_{i}^{4}) + Q_{i}$$
(7)

In this equation, T_i represents the node whose temperature is being sought, T_j is the temperature of node j, Q_i is the external heat inputs and the internal heat generation, and N is the number of nodes. It should be noted that the N+1 node represents the deep-space background.

3.2 Simulation Modeling

Simulation modeling consists of analyzing a conductance/capacitance network and determining both the transient temperature responses and the heat exchanges inside the satellite. This conductance/capacitance network is analyzed by dividing the thermal analysis into two programs (Turner, 1993), specifically, the radiation and the thermal solver programs, as shown in Figure 3.



Figure 3 Mathematical modeling flow chart

These two programs use different solution techniques for which two different types of model construction are needed. In the radiation solver, the statistical Monte Carlo method is used to determine the external heat inputs and internal radiative exchange factors for inclusion into the thermal solver. In the thermal solver, a numerical finite-volume technique is used to determine heat exchanges between nodes and transient temperature responses of these nodes. With the Monte Carlo method, the analysis is performed on surfaces, whereas, the thermal solver uses nodes with specific capacitances and conduction paths between these nodes. Therefore, both programs discretize the satellite but utilize the geometric parameters in two different ways, as indicated in Figure 3.

The radiation program determines both the external heat inputs from the space thermal environment and the internal radiation exchange factors for inclusion in the thermal solver. In the thermal solver, the heat inputs are needed from these external and internal heat sources. Although these magnitudes vary as in Table 1, they are entered as constants into the radiation solver, leaving only the orbital parameters, surface properties and satellite geometry to govern $Q_{i,x}$.

The orbital parameters define the position and orientation of the satellite relative to the earth and sun. The position and orientation parameters are specified by several parameters, as shown in Figure 4.



Figure 4 Orbital parameters (Turner, 1993)

In Figure 4, δ is the inclination of the equatorial plane to the ecliptic plane, Ω is the right ascension, α is the argument of apogee, Ψ is the sun day angle, RI is the orbit inclination to the equatorial plane, θ is the orbital position from perigee, AA is the semi-major axis, and EE is the orbit eccentricity.

These orbital parameters are entered into the radiation solver program to orient the satellite for determining the F_{i-x} 's for the external heat inputs. These F_{i-x} 's define the view factor between the external source x and the surface i. The orbital period and the number of discrete orbital positions are also entered into the model that determines the time array that accompanies the external heat inputs $Q_{i,x}$. The radiation solver produces an array for $Q_{i,x}$ for each progressive time step in a single orbit as the satellite orbits the earth.

The radiation solver program also calculates internal radiation exchange factors R_{i-j} using the Monte Carlo method. This method approach selects a random position on, and a random direction from the current surface i and emits a ray. The vector created by this ray emitted either encounters another surface j or is emitted to space. If this ray encounters arbitrary surface j, the ray is either absorbed by surface j or reflected. If the ray is absorbed, the tally Z_{i-j} , which counts the number of rays emitted from surface i to surface j, is increased by one. Otherwise the ray is reflected, again in a random direction, but from surface j. When this re-emitted ray is absorbed by another arbitrary surface k, the tally Z_{i-k} is increased. This tally always refers to the current surface i under investigation no matter how many reflections a ray has gone through. The script-F between surface i and j is determined from

$$\mathcal{F}_{i-j} = Z_{i-j} / P_i \tag{8}$$

where P_i is the total number of rays emitted from surface i. Typical values of P_i range from 10,000 to 50,000 rays, depending on the accuracy needed. Higher accuracy is obtained with a higher number of rays at the expense of computational time.

Once the radiation solver has created both the R_{i-j} and $Q_{i,x}$ arrays for each time step, the thermal solver combines these arrays into the conductance/ capacitance network. At this point in the analysis, the use of the radiation solver has been completed.

The numerical solution used by the thermal solver for the finite difference technique is the implicit Crank-Nicolson method. The governing equation being solved accounts for conduction between nodes, radiation between nodes and additional heat inputs from external sources, and is the finite difference form of Equation (7).

$$\frac{2C_{i}}{\Delta \tau}(T_{i}'-T_{i}) = \sum_{j=1}^{N} G_{i-j}(T_{j}-T_{i}) + \sum_{j=1}^{N} G_{i-j}(T_{j}'-T_{i}') + \sigma \sum_{j=1}^{N+1} R_{i-j}(T_{j}'^{4}-T_{i}'^{4}) + \sigma \sum_{j=1}^{N+1} R_{i-j}(T_{j}'^{4}-T_{i}'^{4}) + 2Q_{i}$$
(9)

where the primes denote a temperature at one time step in the future. In this equation, the thermal solver solves for the unknown future temperature T_i' , T is the known current temperatures, C_i is the thermal capacitance, and $\Delta \tau$ is the time step. This solution method is implicit and is, therefore, unconditionally stable (Gerald and Wheatley, 1985).

The R_{i-j} , Q_i , and G_{i-j} arrays must be entered for the thermal solver to solve for the unknown temperatures. In addition to these arrays, the output required must be specified. Typical outputs consist of temperatures and heat exchanges, both per node per time step.

4.0 Thermal Design Synthesis

The design synthesis of ISAT-1 TCS is being performed using the simulation model. The results of dissipating the electrical power directly from the solar panels to the internal components and dissipating the power constantly over the orbit time are presented. The constant dissipation of energy over the orbit may be considered normal operation. Failure of the batteries to store energy would force the satellite components to operate directly from the solar panels.

Power dissipation per shelf is shown in Table 2.

	Percent of total		
Shelf number	power		
	dissipation, [%]		
1	20		
2	30		
3	50		

Table 2 Power dissipation per shelf

The orbital parameters are listed in Table 3.

Orbital Parameter	Value		
	·		
Altitude [nautical miles]	520		
Inclination [°]	98		
Orbital period [s]	5282		
Orbital position increments [°]	10		
Number of orbit positions	36		
S _o [W/m ²]	1309		
$J_a [W/m^2]$	243		
J _p [W/m ²]	360		
Spin rate [rev/min]	0		

Table 3	ISAT-1	orbital	parameters
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Table 4 shows the various capacitance values for the nodal network of ISAT-1.

Surface/node	Material	Capacitance,	External radiation		Internal radiation	
		C _i [J/K]	α	3	α	3
top & bottom	aluminum	192.6	0,16	0.03	0.16	0.03
side panels	aluminum	275.5	0.67	0.81	0.16	0.03
boom	stainless steel	1118.0	0.47	0.14	NA	NA
tip mass	brass	1185.0	0.47	0.14	NA	NA
shelf 1	Al/other mat.	10758.0	NA	NA	0.16	0.03
shelf 2	Al/other mat.	4221.0	NA	NA	0.16	0.03
shelf 3	Al/other mat.	5357.0	NA	NA	0.16	0.03

 Table 4 ISAT-1 Simulation model capacitance and surface properties

Table 5 shows selected conductance values for the nodal network of ISAT-1.

Conduction path	Conductance, Gi-j [W/K]
shelf 1 to side panel	0.474
side panel to side panel	0.878
top panel to side panel	0.087

Table 5 ISAT-1 Simulation model selected conductances

4.1 Constant Power Dissipation

The normal operation of the ISAT-1 electrical system is collecting solar energy through the solar cells mounted on the side panels. Storing this energy in the batteries and operating components from a constant power supply, thereby, dissipating heat from these components at a constant rate. If the total energy collected is dissipated equally throughout the orbit, the heat dissipation for all shelves would be a continuous 15.5 W.

Figure 5 shows the transient temperature response of each shelf for one orbit.



Figure 5 Transient temperature response of satellite payload with constant power dissipation

In this figure, the shelves transient temperature responses repeat every orbit as a periodic. All shelf temperature responses vary with time. The temperature variation of shelf 1 is $1.4 \,^{\circ}$ C that is smaller than the variation in temperature experienced by shelf 2 and 3, 7.4 $\,^{\circ}$ C and 10.4 $\,^{\circ}$ C, respectively. This smaller variation is due to the higher shelf 1 capacitance. The variations in the shelf temperatures are due to the conduction and radiation heat transfer to the satellite side panels. At 0 min, the satellite is exposed to all three external heat inputs, solar, albedo and Earth emission. At 2.5 min, the satellite begins to decrease. This decrease in temperature is caused by the loss of heat input from the solar and albedo sources. The satellite exits the eclipse at 27.5 min, where the satellite temperatures begin to increase. At 55 min, the satellite is oriented with only the top surface exposed to the Sun. This lowers the solar heat input and the satellite temperatures experience a slight decrease. At 70 min, the satellite has progressed through enough of the orbit to once again have the sides of the satellite exposed to the solar heat input, where the temperatures continue to rise.

4.2 Directly Coupled Power Dissipation

If the satellite batteries should fail during the life of ISAT-1, the electrical system could be altered to directly couple the power from the solar cells to the satellite components bypassing the batteries. This would not allow the power to the components to be continuous and constant throughout an orbit. The satellite power dissipation would be a function of the total solar and albedo energy incident on the solar panels.

Figure 6 shows the transient temperature responses for the directly coupled power dissipation.



Time [min]

Figure 6 Transient temperature response of satellite payload with directly coupled power dissipation

The temperature variations for shelf 1, 2, and 3, are 1.86 °C, 8.97 °C, and 12.45 °C, respectively. These temperature variations although larger have not increased appreciably. The loss of the batteries although have a critical effect on the operation of the satellite would have little effect on the thermal control of the satellite.

5.0 Thermal Design Validation

Once the TCS design of ISAT-1 is complete, the TCS will be validated by a thermal balance qualification test. Currently, a thermal balance model test on the scaled model of ISAT-1 is being performing. This test is to understand the effects of radiation inside a vacuum chamber, and will not directly affect the TCS of ISAT-1.

6.0 Conclusions and Recommendations

The formulation of the design methodology for the ISAT-1 TCS is complete. The TCS of ISAT-1 will be designed using the following strategy:

- 1. Thermal Simulation
- 2. Synthesis of the satellite geometry, materials, and radiative properties
- 3. Validation of the TCS by thermal balance testing.

The thermal simulation may be divided into two programs, radiation solver and thermal solver programs. The radiation solver determines heat inputs from external sources and the internal radiation exchange factors. The thermal solver is used to determine transient temperature responses. These temperatures are maintained within tolerances by implementing a TCS that assure proper operation of the satellite.

With heat dissipation crucial in small satellites, the preliminary study of the ISAT-1 showed little affect on temperature due to internal heat dissipation. The two scenarios studied are constant and directly coupled power dissipation. In both scenarios the fluctuation of the shelf temperatures are influenced by the capacitance of the shelf and the percentage of total heat dissipated by each shelf. The solar and albedo flux magnitudes in this study are conservative estimates, and a worst case study should be performed. The preliminary results from this study indicate that the satellite TCS need only consist of surface coatings and material selection. To further the design of the ISAT-1 TCS, both the specific orbit of the satellite and formal temperature tolerances of components need to be established.

7.0 References

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