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Introduction

This chapter introduces the thermal analysis and design process as it applies to spacecraft projects. After an overview that includes the phases in a typical spacecraft program and the type of thermal-engineering support they require, the chapter provides a detailed discussion of how the analysis is performed, what computer programs are used, and why they are used. A description of the thermal analysis performed for a real program is included.

Spacecraft Project Phases

The phases in a spacecraft project are concept definition, validation, full-scale development, and operation. The actual activities for each vary from program to program, but the following discussion gives a general idea of the thermal engineer's role as a program matures. Throughout the design or program-development process, formal reviews are conducted to verify that the design has reached a particular level of maturity and meets technical requirements. In this discussion, the various reviews are referenced by names that are common in the industry; however, they may also be known by other names.

The Concept Definition Phase

The concept definition phase is normally the first phase of a program for which engineering support is called upon. This phase usually consists of a customersponsored trade study examining the feasibility of various approaches to meet the intent of a concept. At this point, the intent may be little more than a qualitative description of a mission or capability that planners would like to see filled. The concept normally changes during these studies in response to technical and fiscal realities, or competition from alternative systems. Usually the only constraint given to the various specialists is the cutoff year for state-of-the-art technology projections (the date at which all required technologies are projected to be at a flight-ready level of maturity). The basic end products of these concept definition studies are a parametric analysis of viable approaches, a subsystem state-of-the-art assessment, and the definition of a baseline spacecraft-configuration concept.

The thermal engineer's effort in this phase consists of defining and analyzing, parametrically, one or more alternative approaches to thermal control of the spacecraft. The engineer must consider the thermal requirements of all vehicle elements —housekeeping electronics, payload electronics, batteries, sensors, propulsion, antennas, etc.—for all mission phases from prelaunch testing through on-orbit operations. For many of these elements the engineer will quickly see that standard thermal-control techniques involving finishes, small heaters, or multilayer insulation (MLI) will suffice. These elements are of little concern at this point, and a

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very rough estimate of the types of finishes, weights, and heater powers based on the engineer's experience with other programs is usually quite sufficient.

The real effort in the concept phase centers around elements of the thermal-control system that have significant system-level impacts resulting from size, weight, power requirements, or development complexity. Examples include heat-pipe systems for high-capacity batteries, deployable radiators for rejecting unusually large amounts of waste heat, devices for cooling high-power-density electronics, cryogenic coolers, etc. For each significant element a possible thermal-design approach may be identified, and each approach analyzed parametrically to determine its relative merits in terms of performance, weight, volume, radiator area, heater power, etc. For example, an infrared (IR)-sensor design operating at cryogenic temperatures might be analyzed as a function of focal-plane temperature or parasitic heat-leak rates. The bottom line for these analyses is usually thermalsubsystem weight, volume, and power requirements.

The engineer is often tempted at this stage to jump directly to a point design rather than perform parametric analyses. Doing so is generally unwise, however, because the input parameters upon which the thermal design is based usually change quickly, and more important, the parametric analyses allow the design team greater insight into the impact of system requirements on the weight and complexity of the thermal-control subsystem.

From the parametric analyses performed by each of the subsystem engineers, a baseline spacecraft design is synthesized by the study team. Ideally, this design represents an optimum balance between the competing requirements of different subsystems and overall system constraints such as weight, volume, reliability, and cost. The thermal engineer is responsible for specifying a baseline thermal-control system design and providing a preliminary assessment of its performance capabilities and characteristics, weight, power requirements, and any relevant issues or areas of concern.

The final task during this phase is to assess the subsystem state of the art through a review of all required thermal-control technologies. Most of these technologies will be mature and flight-demonstrated. Others, however, may require significant development before they are flight-ready. A critical input at the concept definition phase is identification of any research and development required to bring needed technologies to maturity, along with an assessment of what work is currently in progress in the industry or required in the future to meet program needs. Useful sources of information on the current state of the art for various technologies include The Aerospace Corporation, the Air Force Research Laboratory, NASA, recent conference papers, and the on-line databases available through the Internet and many libraries. A particularly useful tool for characterizing the maturity of a given technology or design is the NASA nine-point scale of technical maturity (Table 15.1).

At the onset of a concept definition study, the technical specialist must plan the depth and breadth of support to be consistent with the study schedule, the amount of funding available, and the level of effort of the other technical disciplines. Understand the overall study milestones and gear your effort accordingly. Be careful that your projected effort does not drive the study schedule. If you anticipate that it may, advise the study leader. If available funding is not sufficient to provide

Development Stage	Corresp. Readiness Levels	Readiness Level Definitions
Basic technology research	1, 2	1. Basic principles observed and reported 2. Technology concept and/or application
Research to prove	2, 3	formulated
Technology	3–5	3. Analytical & experimental critical function and/or characteristic proof-of-concept
development		4. Component and/or breadboard validation in
Technology demonstration	5,6	5. Component and/or breadboard validation in
System/subsystem	68	relevant environment
System test, launch, and operations	8,9	demonstration in a relevant environment (ground or space)
und operations		7. System prototype demonstration in a space environment
		8. Actual system completed and "flight qualified" through test and demonstration (ground or space)
		9. Actual system "flight proven" through successful mission operations

Table 15.1. NASA Civil Space Technology Development Stages and Corresponding Readiness Levels

meaningful support to the study, raise this issue with the leader. Any issues that cannot be adequately addressed because of funding or schedule limitations must be documented.

The scope and detail of supporting thermal analyses should be tailored to fit the schedule and available funding. In general, many parametric analysis cases with a small model are of greater value to a concept study than are detailed analyses with large models. Scaling existing designs from other programs can also be an efficient way of answering study needs without conducting time-consuming analyses and "reinventing the wheel." In other cases, back-of-the-envelope calculations using Earth heating and view-factor tables and/or very simple thermal mathematical models (TMMs) are sufficient to parametrically characterize a design. If, however, more-extensive analyses are absolutely necessary, be sure to budget adequate time to account for machine turnaround time and reanalysis needed because of inadvertent errors.

The type of documentation required will vary. Some team leaders may expect formal weekly presentations to the study team, while others may ask for periodic written reports. You should have a clear understanding of what inputs are expected and the format in which they should be presented.

Finally, be sure to convey to the leader your commitments and their priorities before you join the team. Because many engineers work in matrix organizations and support more than one program, complete or dedicated support during key phases of the study may be out prioritized. Nothing frustrates a program office more than a change in support personnel in the middle of a study, and one way to preclude this is to keep supervisors updated.

The Validation Phase

Once the concept definition studies have been completed, the design concept, supporting trade studies, and predicted system performance are reviewed by officials, and a decision is made on whether to proceed. If a go-ahead is given, the program enters a validation phase, in which the customer team generates a description of the system that they will ask a contractor (or contractors) to build. This is done by refining the concept-phase studies and determining what technologies and capabilities can realistically be achieved, given the cost and schedule constraints of the program. A System Requirements Review (SRR) is then held to reach agreement between customer and contractor personnel as to what the top-level requirements will be. Once these are established, special studies and tests are performed to address any critical technology questions. In many cases, as an effort matures, a System Design Review (SDR) is held to complete the validation of the system design. The end result of this phase will be a Request For Proposal (RFP), which the customer issues to industry.

Contractors who wish to bid on the program will be given an RFP package that includes basic information such as schedules, instructions for submitting a proposal, and government points of contact. Of greater interest to the technical specialists, however, are the Statement of Work (SOW), system and subsystem design specifications, applicable specifications and compliance documents, and the Contract Data Requirements List (CDRL).

For all practical purposes, the SOW is the top-level technical document from the proposal phase onward. The basic contract that is awarded to the winning contractor at the completion of the source-selection phase (to be discussed in the next section) actually takes legal precedence over the SOW, but because it usually contains little technical information it is not of much interest to the technical specialist.

The SOW core document contains numbered paragraphs that define what the contractor shall do and what ground rules and assumptions will be in effect. The SOW is a list of tasks to be performed, such as thermal analyses, trade studies, tests, etc. (Specifications for the design, on the other hand, are usually contained in ancillary SOW documents, which will be discussed later.) Because the SOW is part of the legal contract, wording is extremely important, and all tasks are identified and clearly specified as actions the contractor must perform.

Included in various SOW attachments and appendixes will be the CDRL, a list of applicable specifications and compliance documents, and system/subsystem design specifications, if any.

The CDRL is a list of reports, meeting minutes, Interface Control Documents (ICDs), drawings, and documentation that the contractor must deliver to the customer. The list includes only major items of documentation, not the memos and small reports that are informally transmitted.

Other common specifications and applicable documents that may be cited as requirements or provided as recommendations include:

- MIL-STD-1540D, "Product Verification Requirements for Launch, Upper Stage, and Space Vehicles"
- MIL-HDBK-340A, "Test Requirements for Launch, Upper Stage, and Space Vehicles"
- MIL-STD-24236 (Rev. C), "General Specification for Metallic and Bimetallic Thermostat Switches"

The final area of the SOW is the system/subsystem design specification. Specifications for the design of each subsystem, including thermal control, may or may not be included depending upon the degree of control that the customer wishes to exercise over the contractor's design. In the event that a thermal-control subsystem specification is included, the key items that the engineer should expect to find are:

- A requirement to keep all component temperatures within allowable limits during all mission phases from prelaunch to end of life (EOL), considering worstcase variations in power dissipations, environments, operating modes, and contamination/degradation.
- Emphasis on the simplest, most reliable flight-demonstrated technology with no single-point failures.
- For military programs, guidance on analysis margins and test verification derived from MIL-STD-1540D and MIL-HDBK-340A. For NASA and commercial communication-satellite programs, compliance with the customer's analysis margin and test requirements philosophies, which may not be engendered in a formal specification like those used by the military.
- A design that is predictable by thermal analyses and verifiable by ground test.
- Identification of key trade-off studies required.
- Identification of key development and final verification tests.

Proposal Evaluation

Usually, once an RFP has been issued, any interested contractor may submit a proposal. A typical proposal consists of an executive summary, a technical proposal, a management proposal, and a cost proposal, all limited to a certain length, as specified in the RFP.

To evaluate the technical proposal(s), a team of customer personnel is assembled, representing a wide range of technical disciplines. If you serve on such a team, you will find that the source selection takes priority over all other assignments. Because of the competitive, and sometimes contentious, nature of the contract-award process, no information, notes, or documentation concerning the evaluations or the content of any of the proposals may be taken outside of the proposal-evaluation office area or discussed with anyone who is not a member of the source-selection team. Furthermore, evaluators are expected to provide their own technical evaluations of the proposals free from management concurrence. Your findings are not reported to your managers, even if they are members of the same source-selection team. The objective is to provide a fair, unbiased evaluation.

The proposal-evaluation process usually begins with an overview briefing to all of the evaluators by the customer program managers. From attending this briefing and from reading the actual RFP, each evaluator must become thoroughly familiar with the RFP requirements, the evaluation criteria, and the evaluation procedures.

This familiarity is critical because each proposal must be evaluated against the RFP requirements and evaluation criteria and not against the other proposals. At no time can Contractor A's proposal be compared to Contractor B's; all proposals are evaluated separately for compliance with RFP requirements only.

The actual review of the proposal documents will occur in an office area set aside for that activity, and it will be accessible only to members of the evaluation team. Although you will be asked to evaluate only these areas of the proposals related to your area of expertise (thermal control), you should read the entire technical proposal, and sometimes the executive summary, to ensure that thermal issues affecting other subsystems are properly addressed. If the spacecraft requires large deployable radiators, for example, the impact of that need on vehicle dynamics, sensor fields of view, and launch packaging should be covered in other areas of the proposal. Similarly, the need for large heaters may affect the sizing of the electrical-power subsystem.

Any thermal-control issue affecting other spacecraft subsystems should be discussed with the team members evaluating these subsystems to make sure that they are aware of the impacts. Unfortunately, to maintain propriety and document the evaluation process, a large amount of paperwork is required.

Evaluation of proposals must be conducted in light of the specialists' knowledge of what is required to analyze, design, test, and build a spacecraft thermal-control system. Neither the SOW nor the proposal attempt to specify every detailed task that must be performed by the contractor, but the reviewer must ensure that what the contractor has written demonstrates a sound understanding of what is required to develop the thermal-control system and ensure successful completion of the project, on time and within budget. An overly vague proposal that does not reflect a sound grasp of the requirements and does not commit to specific tasks should be viewed with much concern.

The proposal should discuss the proposed thermal-design approach and testing that the contractor will do to verify the thermal-control system. The proposal should also show that the contractor understands and has made a preliminary evaluation of all environments, operating modes, unusual thermal requirements, and potential areas of concern. Analysis and test schedules and estimated staffing levels should also be presented in enough detail to show that the contractor has made a realistic estimate of the thermal-subsystem cost and development schedule. Particular attention should be paid to any newer or exotic technology items that may require extensive development effort.

The reviewer should also verify that all values presented by the contractor are reasonable and consistent. For instance, if the power subsystem puts out 10,000 W and the thermal design is based on 2000 W of heat dissipation, clearly something is wrong. Similarly, simple hand calculations can determine if estimated radiator sizes are approximately correct. "Sanity checks" like these are useful at this stage to ensure that the contractor understands the thermal-control task.

All findings are documented and are used to create a final briefing that summarizes for customer management the strong and weak points and risk assessment of the entire team. Proposal-evaluation team leaders will return to you for further explanation of significant issues that you have raised. Several conversations with a team leader may be necessary before that person completely understands your concerns, especially if the leader does not have a thermal background. An accurate understanding of this situation must be passed along to ensure a fair and accurate evaluation.

At the conclusion of the source selection, one or more contractors are selected to continue into the full-scale development phase. There may be one prime contractor, two or more prime contractors who will compete until a final selection of one prime contractor is made, or two or more associate contractors developing different parts of the satellite. Generally, the larger the program, the greater the number of contractors.

The Full-Scale Development Phase

Contract Award through PDR

During the period from contract award through Preliminary Design Review (PDR), customer personnel establish working relationships with the contractor and begin engineering development of the spacecraft and related ground systems. The emphasis during this phase is at the system level, and the type of effort is similar to the effort made during the concept definition phase, only with a greater level of detail. The spacecraft design is still fairly flexible at this point, and significant changes in configuration, payloads, and subsystem designs (including thermal) should be expected—hence the importance of continuing the types of tradeoff studies and parametric analyses that were conducted during the concept definition phase. Doing so assists the systems-engineering staff and program management in defining an optimal baseline spacecraft design before the PDR.

At this stage, another important action is identifying potential technical or development problems with the thermal or system designs under consideration. Thinking ahead is crucial, because changing a design now is far easier than later, when program "inertia" makes fundamental changes nearly impossible. You will have the greatest possible leverage on a program at this stage, and forward thinking here can save much frustration later. Don't postpone dealing with any possible problems.

Meetings with specialists in other spacecraft subsystem areas, program managers, and the customer will increase significantly. They may seem like time-consuming affairs that take you away from your "real" work, but they are the medium through which the results of studies and analyses are funneled into the program. You may perform many analyses, but if you do not present your results and press for changes that you think are required, your work may be ignored by program managers and customer personnel who are preoccupied with their own priorities. Sometimes much persistence is required to get your point of view recognized and to effect substantive changes, so don't give up. Walking away and writing a "Pearl Harbor" memo doesn't do much to bring a program to a successful completion.

During this phase, key requirements must be identified and a preliminary thermal design selected and documented. Among the requirements that should be addressed are the following.

- range of mission orbits
- normal attitude(s) of satellite
- launch-phase configurations and attitudes
- ground cooling needs

- autonomy requirements
- attitudes during stressed/failure modes
- · temperature limits and reliability requirements
- equipment power dissipations and operating modes
- thermal-distortion budgets
- launch-system interfaces
- interfaces with other subsystems, such as
 - --payloads
 - -propulsion
 - -attitude control
 - -electrical power
 - -structures
 - -telemetry, tracking, and command
 - -computer and data handling
- contamination control
- special thermal-control requirements for batteries, crystal oscillators, sensors, etc.

To support the selection of a preliminary thermal design, the contractor is expected to provide results of trade-off studies addressing issues such as the degree of active versus passive control, solid-state heater controllers versus thermostats, deployable versus fixed radiators, refrigerators versus stored cryogenics versus cryogenic radiators, hardware or material trades to minimize weight, etc. Clearly not all of these trades are applicable to any given spacecraft, but they are representative of the types of trades that are expected.

Once a preliminary thermal design is established it should be well documented prior to PDR. This documentation should provide a complete description of the design, including key assumptions, radiator areas, insulation requirements, thermal finishes and their assumed optical properties, heater sizes and locations, heat sinks, heat-pipe types, sizes, and locations, a flight instrumentation list, refrigerator descriptions and power requirements, etc. In addition, any thermal development and acceptance/qualification test plans should be addressed, a schedule for all thermal analysis and testing should be presented, and any potential problem areas should be discussed. By the time a preliminary design is identified, analysis should exist to back up the design choices. This analysis, although preliminary, should address all of the issues listed earlier, and it should be well documented.

The PDR itself is usually a large meeting involving many of the contractor's technical specialists, systems engineers, and program managers, as well as a sizable contingent of customer personnel and their technical advisors. During the review, the design is critiqued in a process that may leave uninitiated technical specialists feeling like they are being hounded by an angry mob! Bear in mind that the objective of this process is to identify any weakness in the design are not a criticism of the skill or judgment of the thermal engineer and should not be taken personally. Concerns, recommendations, and action items generated in the PDR should be carefully considered so that the designers can take full advantage of the experience and lessons learned that may be embodied in the reviewer's comments.

Some comments, of course, will reflect a simple misunderstanding of a design that may be difficult to fully appreciate after seeing only a few hours of presentation material. No concern or recommendation, however, should be dismissed without fully considering the implications of the potential problem that has motivated the reviewer's comments.

PDR to CDR

The period from PDR to Critical Design Review (CDR) is the time when most of the design and analysis work takes place. Starting with the PDR itself, the design and supporting analysis and development efforts should be paced against the standards to which they will be judged at program completion. Although a greater deal of work remains to be done and substantial uncertainties may exist, contractor staff members should at all times be able to demonstrate that they are on the right track to deliver a high-quality product. Table 15.2 shows an extensive list of evaluation criteria that the contractor should eventually be able to meet. Progress against this list should be closely monitored during the period from PDR to CDR, with the goal of being able to answer all items by CDR.

During this period, close contact should be maintained between the customer and the contractor's thermal people. Regular Technical Interchange Meetings (TIMs), formal or informal, should be scheduled to discuss progress and any thermal-engineering concerns. Face-to-face meetings at the contractor's office, where data and reports are readily available, are far superior to telephone discussions and should be scheduled on a regular basis, even if the program is going smoothly.

Extensive detailed thermal analysis of the spacecraft and all of its components under worst-case hot and cold conditions must be performed during this period. The description of the Hubble Space Telescope thermal design in Chapter 3 illustrates the number of components that must be analyzed. The report that summarized the results of the thermal analyses conducted for that program is more than 500 pages long—a figure that suggests how much analysis may be required. By CDR, a thermal design must be firmly established and all supporting analyses and development tests of critical components completed.

CDR to Launch

The period following CDR is generally devoted to making any design changes dictated by the outcome of the CDR, conducting subsystem development tests, building the satellite, and testing it. The work during this period becomes less oriented to concepts, more to hardware. Final drawings must be made by the designer and signed off by the technical specialists; thermal-control system hardware must be specified in detail and manufactured or purchased; and thermal-balance and thermal-vacuum tests must be planned and executed (see Chapter 19). Although the level of effort for the thermal engineer is generally less than during the PDR-to-CDR period, a great deal of work remains to be done. Attending to all the low-level details and completing all required documentation and test planning can seem like endless tasks.

The most important activity in this phase, and perhaps the most important single event in the entire program for the thermal engineer, is the thermal-balance test.

_					
	TCS Evaluation Criteria				
	Design Features				
٠	No single-point failures possible				
٠	Reliable				
•	Flight-proven				
٠	Predictable by thermal analyses				
•	Verifiable by ground test				
•	Provides adequate thermal margin				
	- Passively controlled components: 11°C beyond worst-case predictions made by a				
	test-correlated thermal model				
	- Passively controlled components where a temperature margin is not feasible: a				
	rational, well-documented equivalent of 11°C				
	- Actively controlled components: control authority of at least 25%, which can be				
	shown to be equivalent to the 11°C margin specified for passively controlled				
	components				
•	Meets satellite life requirement				
• Insensitive to the space environment					
	– Vacuum				
	 Natural and ultraviolet radiation 				
	– Contamination				
	 Temperature cycling 				
	 Micrometeoroids and manmade debris 				
	 Electrostatic charge accumulation 				
٠	Insensitive to the ground and launch environment				
	– Vibration				
	- Acoustic noise				
	- Venting				
	- Handling and storage				
	- Contamination				
•	Takes into account the maximum range of component power dissipations				
•	Considers the maximum range of orbital thermal environments				
	- Operational hot orbit				
	- Operational cold orbit				
	- Eclipsing orbit (il different than cold)				
	- On-orbit maneuvering				
	- Interplanetary cruise (II applicable)				
	- Failure mode and recovery				
	Takes into account maximum range of other important mission environments, such as:				
-	- Prelaunch				
	- Inclaution				
	- Transfer orbit				
•	Elements of TCS hardware including MLL paints coatings adhesives conductive				
	thermal compounds thermal strans isolators thermal doublers heat nines fasteners				
	tanes, etc., meet or exceed the NASA outgassing criteria.				
	- Weight loss no greater than 1.0%				
	- CVCM (collected volatile condensable materials) less than 0.1%				
•	Autonomous				
•	Fault tolerant				
•	Allows for proper venting and outgassing by well-defined naths for all spacecraft parts				

Table 15.2 Thermal-Control Subsystem (TCS) Evaluation

- Allows for proper venting and outgassing by well-defined paths for all spacecraft parts, subsystems, and payloads Can be readily integrated
- ٠
- Imposes minimum amount of operational restrictions on the satellite and launch vehicle .
- Allows for growth .

Table 15.2. Thermal-Control Subsystem (TCS) Evaluation—Continued

TCS Evaluation Criteria Hardware Development Programs Of sufficient depth and breadth to reduce the risk of not having flight-qualified hardware when needed Heat pipes Phase-change materials (PCMs) High-capacity constant-conductance heat pipes (CCHPs) Loop heat pipes or capillary pumped loops High-capacity variable-conductance heat pipes (VCHPs) High-fin-effectiveness composite radiators Heat plane materials and heat pipes for electronic equipment Completion of Key Trade-Offs Extent of passive versus active thermal control Distributed versus centralized thermal control using an onboard computer Solid-state temperature control versus bimetallic thermostats Selection of location and configuration of radiators Selection of heat pipes Extent of ground cooling required Transfer-orbit battery requirements from upper stage Demonstration that TCS Design Meets Requirements (Analyses) Geometric model of selected configuration **TMMs** Thermal analysis results for orbital worst-case hot and cold environments Thermal analysis results showing sensitivity to assumed beginning-of-life (BOL) and EOL thermal properties Analysis results showing adequate margin for both hot and cold cases Documentation and substantiation of key assumptions Thermal analysis results for worst-case hot and cold environments corresponding to prelaunch, launch, and transfer orbit System-, Subsystem-, and Component-Level Testing Plans and commitment to perform thermal balance (T/B) testing to validate the proposed design Development tests Heat-pipe performance tests System-level qualification T/B test

Despite the sophistication of today's analytical techniques and the maturity of satellite thermal technology, errors occur frequently in the analysis and design of spacecraft thermal-control subsystems. The thermal-balance test is the one chance the designer has to catch these errors before it is too late. In one sense, the test is even more important than all of the work that came before it, in that it may uncover a problem that would have caused the loss of a mission costing hundreds of millions of dollars if the satellite had been launched "as is." Errors large enough to represent fatal flaws in the design of satellites do occur in thermal-control subsystems, although they are not common. A more likely outcome of the test is finding that some adjustments must be made to the design or to the mathematical models. In any event, the thermal-balance test is the critical verification of the thermal design, and a conclusive test is considered mandatory to program success.

The Operation Phase

Once the satellite has been built and successfully tested, it is delivered to the customer for launch and operation. During this phase, the thermal engineer supports launch rehearsals as well as the actual launch, and typically provides an assessment of the performance of the thermal-control subsystem on orbit. If any anomalous performances or failures in the thermal subsystem occur, the thermal engineer typically performs analyses, assessments, or tests to determine what caused the problems and what corrective actions should be taken. If additional spacecraft are to be built in the future, then design changes may need to be investigated.

Assuming that the thermal subsystem performs as expected, the primary activity during this phase will be launch support. The level of support expected from thermal engineers during launch varies tremendously from program to program. Some small programs may have no thermal support, while other programs may have 24hour coverage by several thermal engineers for up to two weeks. Most programs will require a launch-site inspection of the satellite configuration and functional verification of commandable thermal components, such as heaters. Technical assistance is also generally required of the thermal engineer at the satellite control room from launch until the satellite is stabilized in its operational orbit or interplanetary trajectory and initial deployments of solar arrays, antennas, and other appendages are complete. This period generally lasts from a couple of days to a week, and it may involve one thermal engineer who comes in for critical events or several engineers on shifts providing around-the-clock coverage. The principal activities are monitoring temperatures, heater status, and other telemetry to ensure that the thermal design is functioning normally, and also providing recommendations for corrective action if the thermal subsystem or another vehicle subsystem malfunctions.

Thermal Design/Analysis Process Overview

The thermal-design process is a combination of design selection and supporting analysis. The selection of a viable thermal-design approach may become almost intuitive for a thermal engineer who has worked a number of programs. Detailed thermal analyses are, however, always required to verify and refine the design. Experience minimizes the number of time-consuming analysis iterations required to close in on a final design.

A wide range of thermal-control hardware and techniques is available, from simple surface finishes to complex refrigeration systems (Chapters 4 through 14). The spacecraft system requirements to minimize weight, cost, and test complexity while maximizing reliability are usually served best by keeping the thermal design as simple as possible and by avoiding the use of active components. A design that relies only on surface finishes and insulation blankets will be lighter, far less expensive to build, more reliable, and easier to test than a design involving heat pipes, louvers, or refrigerators. Therefore, although active or semi-active components will sometimes be required, they should be avoided wherever possible.

Before starting the design/analysis process, the engineer must plan the overall effort. Remember that the goal is to provide a reliable thermal-control system at minimum cost for the spacecraft or component in question. All of the analysis, design, and testing activities are only tools to be used to reach that goal, and any unnecessary expenditure of time or money should be avoided. Thus the design should be no more complex than is required to do the job, the TMMs should have the minimum number of nodes needed to verify the design, and, if any difficulties are encountered in analysis, design, or testing, the engineer should ask if a simpler route is available. It is easy to get bogged down in an overly complex design, and some experience is required to know the optimum trade between detail/complex-ity and practicality.

The first step in the thermal-design process is to clearly understand the objective(s) and any ground rules or constraints. The objective might be to develop a complete thermal-control system for a new spacecraft, to predict temperatures for an existing satellite in a new attitude, to modify the thermal design of a component in response to changes in component design, etc. Understanding the objective and its requirements may require meetings with program managers and other subsystem specialists. Once an objective is established, project ground rules and constraints must also be considered, since these will affect the thermal-design effort. These factors—such issues as how much staffing is available and what this project's priority is relative to other considerations—will play a major role in structuring the effort.

Once the objectives and ground rules are understood, an approach to problem resolution must be selected. The approach may be to do an analysis, perform tests, do hand calculations, adapt the thermal design from a thermally similar device, or a combination of these activities. Each potential approach must be evaluated to determine whether its elements are meaningful to the solution of the problem. Some problems, for instance, may not lend themselves to detailed analysis, but may find a more meaningful solution in simple hand calculations followed by a good test. To identify the approach, one must also consider schedule, budget, and any risks, such as reliance on new or unproven technologies or analysis software.

Once a technically sound approach to the design effort is established, a preliminary schedule and cost estimate should be made. The engineer develops an outline of tasks required to support the job, which should include major milestones, criteria for determining if objectives are met, staffing levels, and a clear definition of what is expected from whom and when. While this outline may adequately be handled mentally for a simple task, it will quickly grow to memo size for even a small thermal-design effort. This top-level plan is invaluable in keeping the effort

- · identify development-test requirements
- form matrix of required computer runs
- define math model (number and location of nodes)
- obtain thermal property data
- construct math model(s)
- debug model
- make production runs (number of runs, cost)
- conduct development tests
- reduce/review data
- finalize design
- document/present design analysis
- plan, support, and document testing
- evaluate test data and modify design accordingly

If the estimated time and cost for the effort is not consistent with program requirements, the thermal engineer must either find a simpler, lower-cost approach to the thermal design, or renegotiate funding or schedule. Planning up front and negotiating with program management early will avoid headaches later. As the effort unfolds, cost and schedule should be monitored regularly for measuring performance, and any program slips or changes should be incorporated into the thermal plan.

Once a plan that meets technical and program requirements is established, the design analysis begins. The first step is to establish working relationships with all individuals who provide needed inputs or receive results—typically the lead engineers responsible for the other spacecraft subsystems and payloads, such as propulsion, battery, payloads, or attitude control. Coordinating with these individuals to establish objectives, understand requirements of their subsystems, determine impacts on the thermal design, etc., is important. Failure to communicate regularly may result in wasted time analyzing an out-of-date design.

To prepare for the design effort, you must gather a fair amount of data and information about the system. This data typically includes drawings and sketches of the hardware, estimated heat dissipation and weights of components, definition of orbit and attitude, information about thermal environments from prelaunch through EOL, operating modes of the spacecraft, and thermal property data for materials that may be used. This information is needed to identify a preliminary thermal-design approach and to construct the TMMs.

Before the thermal analysis can begin, a thermal-design approach must be identified. This is usually done by a combination of experience and simple hand calculations to determine if a given approach is viable (this process will be discussed in more detail later). Consideration of all factors—including cost, practicality, analyzability, reliability, and testability—is important at this point.

Using the data and design approach discussed above, the analyst constructs the thermal models: a geometric math model (GMM) for calculating radiation interchange factors and a TMM for predicting temperatures. The GMM is a mathematical representation of the physical surfaces of the satellite or component and is used to calculate the radiation couplings between all surfaces in the model, as well as heating rates to each surface from external flux sources such as solar, Earth IR, and albedo radiation. The TMM is usually a lumped-parameter representation of the thermal capacitance of each node and thermal conduction terms between nodes, and it is directly analogous to an electrical RC (resistance-capacitance) network. These models are constructed using a combination of computer-aided design (CAD) technologies and hand calculations, and later they will be discussed in detail.

The completed (and debugged) thermal model is run to predict hardware temperatures under worst-case hot and cold conditions. A number of runs may be required to determine what exactly is the worst-case combination of factors, such as orbit beta angle, operating mode, vehicle attitude, surface properties, etc., and a number of parametric runs may be required to close in on optimum sizing of radiators, heaters, and so on. In addition, many analyses will have to be rerun to reflect design changes or updates to analysis inputs, such as box power dissipations, that will occur as the vehicle design matures. Periodic reviews with management and other program personnel are required to ensure that the analysis reflects the current system design and will provide the results needed for other subsystem design efforts. Peer review is also a good way of uncovering the errors that inevitably occur in any analysis before they can do any harm.

The final and sometimes most-tedious step is documentation. The thermal design analysis report(s) should include a complete description of the final thermal design, an in-depth discussion of all the significant math-model inputs and assumptions, an attachment containing a listing of the thermal models, predicted temperatures and margins for all components and heater powers for worst-case conditions and operating modes, and a discussion of any significant concerns or recommendations. In preparing such a report, one must first review and understand all work performed. A critical appraisal must be made of all results to ensure that they are valid, complete, and consistent. The report itself must be written to meet the needs of those to whom it is addressed, as well as to provide a record for the future reference of the analyst. It should conclude with a concise summary of why each task was done, how it was done, what was found, and what should be done as a result.

Fundamentals of Thermal Modeling^{*}

Thermal Math Modeling as a Cognitive Process

The body of thermal math modeling concepts, principles, and techniques constitutes a valid tool that can be applied to real engineering problems. A brief introduction to the rudimentary techniques of thermal modeling, coupled with a simple understanding of the various basic heat-transfer mechanisms, is the prerequisite for learning thermal math modeling. Succeeding sections of this chapter present the basic principles and techniques of this discipline.

Developing a good lumped-parameter representation of a thermal system requires—in addition to learning the basic concepts, principles, and techniques an elusive mixture of experience (with real systems, both physical and model) and

^{*}This section is taken from the "Thermal Network Modeling Handbook" prepared by TRW under NASA contract 9-10435.

engineering judgment to transfer the end product into an accurate, versatile, and cost-effective TMM. Experience, of course, can only be acquired from hands-on work with real thermal systems and participation in the modeling and analysis thereof. Engineering judgment is a capability gained by abstracting, from the discipline's body of unique, familiar information, a general understanding that can guide the investigation and comprehension of unfamiliar areas. As such, engineering judgment cannot be captured in written form.

Generally, the problems encountered in developing a TMM reduce to an overall object of achieving the greatest accuracy for the least cost. Cost factors are rather well defined and fall into two classes, development and use. Development costs can be based almost solely on the actual engineering staff-hours required to do the job within the constraints of time and budget; however, the potential costs involved in using a model are often not as obvious nor as linear.

The problem of achieving accuracy, while subject to cost constraints, varies greatly from one TMM to another. For example, general accuracy requirements might be stated as straightforwardly as this: "Temperature accuracy shall be compatible with thermocouple A/D converter quantization error." On the other hand, accuracy levels might be indirectly indicated by requiring that a model "be sufficiently detailed to permit meaningful parametric analyses with respect to insulation thickness variations in increments of 0.5 cm." Clearly, a great deal of engineering judgment will be involved in developing a model that is "sufficiently detailed" to be "meaningful."

Network Solution

Two systems are analogous when they are represented by similar equations and boundary conditions, and the equations describing the behavior of one system can be transformed into the equations for the other by simply changing symbols of the variables. Thermal and electrical systems are two such analogous systems, as shown in Table 15.3.

Thermal System	Electrical System		
<i>T</i>	E		
Ż	Ι		
R	R		
G	1/R		
С	С		
$\dot{Q} = GT$	I = E/R		
	Thermal System T \dot{Q} R G C $\dot{Q} = GT$		

Table 15.3. Thermal-Electrical System Analogy

The analogy between thermal and electrical systems allows the thermal engineer to utilize widely known basic electrical laws such as Ohm's Law and Kirchhoff's Laws, which are used for balancing networks. Numerical techniques used to solve the partial differential equations describing such electrical systems have been conveniently adapted to computer solutions of thermal networks, thus enabling the thermal engineer to readily compute temperature distributions and gradients of complex physical thermal networks.

Thermal-analysis computer programs have been developed that require the user to define a system thermal network analogous to an electrical circuit. Once data describing the network components are input, preprogrammed routines calculate the transient or steady-state solutions. This section discusses the development of a thermal network and the numerical techniques for solving it.

Nodes

To develop a thermal network and apply numerical techniques to its solution, one subdivides the thermal system into finite subvolumes called nodes. The thermal properties of each node are considered to be concentrated at the central nodal point of each subvolume. Each node represents two thermal-network elements, a temperature (potential) and a thermal mass (capacitance), as shown in Fig. 15.1.

The temperature, T, assigned to a node represents the average mass temperature of the subvolume. The capacitance, C, assigned to a node is computed from the thermophysical properties of the subvolume material evaluated at the temperature of the node, and it is assumed to be concentrated at the nodal center of the subvolume. Because a node represents a "lumping" or concentration of parameters at a single point in space, the temperature distribution through the subvolume implied by the nodal temperature is linear, as shown in Fig. 15.2(c), and not a step function, as illustrated in Fig. 15.2(b).

In a homogeneous material, the temperature at a point other than the nodal point may be approximated by interpolation between adjacent nodal points where the temperatures are known.

The error introduced by dividing a system into finite-sized nodes, rather than volume dx^3 where dx approaches zero, is dependent on numerous considerations: material thermal properties, boundary conditions, node size, node-center placement, and time increment of transient calculations. The techniques for proper nodalization to minimize the error will be discussed in a later section.



Fig. 15.1. Nodalization.



Fig. 15.2. Temperature distributions.

Up to this point, only nodes that represent subvolumes with a finite thermal mass (capacitance) have been discussed. In many instances, two other types of nodes are required to define a thermal network. They are nodes having a zero capacitance or an infinite capacitance. Thermal analyzers such as the program SINDA (Systems Improved Numerical Differencing Analyzer) usually name the three types of nodes as follows:

- diffusion (finite thermal mass)
- arithmetic (zero thermal mass)
- boundary (infinite thermal mass)

The diffusion node (finite capacitance) is used to represent normal material, the temperature of which can change as a result of heat flow into or out of the nodes. It is characterized by a gain or loss of potential energy, which depends on the capacitance value, the net heat flow into the node, and the time during which the heat is flowing. Mathematically, a diffusion node is defined by this expression:

$$\Sigma \dot{Q} - \frac{C\Delta T}{t} = 0. \tag{15.1}$$

The arithmetic node (zero capacitance) is a physically unreal quantity; however, its effective use with numerical solutions can often be helpful in interpreting results in such applications as surface temperatures, bondline temperatures, and node-coupling temperatures. It also finds use in representing thermal-system elements that have small capacitance values in comparison to the large majority of the other nodes in the system, which results in computer run-time reduction with minor changes in overall accuracy. These elements could include small components such as bolts, films, or fillets; gaseous contents of small ducts or tubes; and low-mass insulations. The number of arithmetic nodes should be small compared to the total number of nodes in the network. The temperature of an arithmetic node responds instantaneously to its surroundings. Mathematically, an arithmetic node is defined by this expression:

$$\Sigma \dot{Q} = 0. \tag{15.2}$$

The boundary node (infinite capacitance) is used to represent a boundary or sink whose temperature is set and will not change no matter how much heat flows into or out of it from other nodes in the model. Common uses are representation of deep-space sink temperature, recovery temperature, and planet-surface temperature. In addition, boundary nodes may represent thermal-system components that have a very large thermal mass (capacitance) relative to the other nodes, such as the bulk propellant in a large tank. Mathematically, a boundary node is defined as:

T = constant.

The placement of the diffusion-node centers and the choice of node shapes depend on several factors: the points where temperatures are desired, the expected temperature distribution, physical reasonableness, and the ease of computation. The actual size of the node is dependent on other considerations: accuracy desired, structural design, computer storage capabilities, and computer time required. Each factor, however, embodies other considerations. For example, to anticipate the expected temperature distribution, one must draw heavily on engineering judgment as to the effects of the expected boundary conditions and associated material properties.

In general, the shape of a diffusion node is chosen to be a simple geometric figure having areas and volumes that can be easily calculated. Irregularly shaped structural members may be approximated with simple shapes by employing assumptions that are consistent with the desired results. In some cases, nodal divisions are decided first, with the node-center locations thus defined as a consequence. In these cases, nodal edges will usually lie along structural edges, and structural members will be divided in a symmetric and equal fashion. In other cases, output requirements will dictate the locations of node centers, with the nodal edges assigned as a consequence. These two approaches are illustrated in Fig. 15.3. In case (a), the objective is to prepare a general model of the structure, but in (b), the objective is to model the response of two thermocouples located on the bondline between the two members.

The above example suggests that rectangularly shaped nodes are generally desirable. This is true simply because with such nodes, the areas and volumes



(a) General node boundaries (b) Node centers at thermocouple locations

Fig. 15.3. Alternate nodalization methods.

required for the input calculations are easy to compute. The use of such simple nodal shapes is in keeping with current engineering practice. By contrast, Dusinberre^{15.1} suggested that nodalization be performed in such a manner that the paths of heat flow assume a triangular pattern, as shown in Fig. 15.4(a). The only drawback to this theoretically sound approach is that the engineer must compute the volumes of the irregular polygonal nodes that are the consequence of such a tack, as shown in Fig. 15.4(b).

Note how much simpler the rectangular nodalization approach is, as indicated in Fig. 15.4(c). As might be expected, to achieve the same simplicity of calculation, circular structures are nodalized in pie-wedge shapes, annular shapes, or a combination of the two, as shown in Fig. 15.5.

Boundary nodes are used to define points, lines, or surfaces of constant temperature in one-, two-, or three-dimensional models, respectively. The physical location of a boundary node is determined solely by the conduction paths connected to it. A single boundary node may be used to model all boundaries at the same temperature. This point is illustrated in Fig. 15.6, which shows that the indicated boundary node will suffice as a model of the entire constant-temperature edge of the structure (in this case, 30°C).



Fig. 15.4. Polygonal nodalization vs. rectangular nodalization.



Fig. 15.5. Nodalization of circular elements.



Fig. 15.6. Sample boundary node.

Arithmetic nodes have a number of uses that are consequences of the fact that such nodes serve as an engineering model of the proverbial "wafer of thickness dx, where dx approaches zero." A typical application lies in the modeling of exterior surfaces of reentry vehicles, which are often subjected to severe, rapidly changing boundary conditions. In the physical system, the surface temperature remains very close to radiation equilibrium with the surface heating rate, indicating that this system can be accurately simulated by the use of a surface arithmetic node. This application is illustrated in Fig. 15.7.

The case where heat flows from a surface by conduction is usually one in which two structures are bonded together and a bondline temperature is sought. When the structures are homogeneous, a bondline temperature may be established by simple linear interpolation between the nearest node centers. When the materials are dissimilar, a more appropriate technique is to use an arithmetic node at the bondline, leaving to the computer the process of performing a conductanceweighted averaging of the adjoining diffusion-node temperatures, which, in essence, is the result of finding the steady-state (heat in = heat out) temperature for an arithmetic node.

Arithmetic nodes may also be used advantageously in place of diffusion nodes that have a capacitance that is small when compared to the great majority of nodes in the system. This usage often occurs when modeling a small quantity of gas in a tube or other enclosure, or when modeling small structural parts, such as wires,



Fig. 15.7. Use of arithmetic nodes to model surfaces.

bolts, fillets, films, and sheets, where detailed temperatures are desired (which precludes lumping such items along with larger nearby nodes). The correct use of arithmetic nodes in these cases generally results in a considerable saving of computer time when the model is processed.

In the development of a thermal network, computations with respect to nodes are generally limited to calculating the capacitance of diffusion nodes. The following formula is used:

$$C = \rho \cdot V \cdot C_p, \tag{15.3}$$

where C is thermal capacitance (J/°C), ρ is density (kg/cm³), V is volume (cm³), and C_p is specific heat (J/kg.°C).

The specific heat (C_p) and the density (ρ) of materials may vary with temperature. The necessity to utilize temperature-dependent properties for analysis depends on the degree to which the properties vary and the temperature range over which the capacitance of the material will be calculated. Most thermal-analysis computer codes can accommodate temperature-varying thermal properties.

The use of arithmetic nodes may also require some computations. Replacement of small-capacitance diffusion nodes with an arithmetic node must be preceded by computations to verify that the capacitance-conductor effects are such that the node in question will essentially reach steady-state temperatures during the time step required by the larger nodes. The use of an arithmetic node to predict surface temperatures where surface radiation or very high heating rates are involved requires careful analysis to ensure the stability of the arithmetic node. Stability criteria and solution techniques are discussed later. This section shows that solution techniques using linearized "last-pass" temperature values may require the use of analyzer control constants to restrict the maximum node temperature change or computation time step. The engineer must further be cautioned against using coupled arithmetic nodes without a complete understanding of the implications and required analyzer control constants used to ensure a valid solution.

Conductors

Conductors are the thermal math modeling network elements that represent the heat-flow paths through which energy is transferred from one node to another. Figure 15.8 illustrates the element node temperatures (T), capacitances (C), and conductors (G) that comprise a thermal network.

The three processes by which heat flows from a region of higher temperature to a region of lower temperature are conduction, convection, and radiation. Conduction is the process by which heat flows within a medium or between different mediums in direct physical contact. The energy is transmitted by molecular communication. Figure 15.9 illustrates the conduction conductor.

Convection is the process of energy transport by combined action of heat conduction, energy storage, and mixing motion. Heat will flow by conduction from a surface to adjacent particles of fluid; then the fluid particles will move to a region of lower temperature, where they will mix with, and transfer a part of their energy to, other fluid particles. The energy is actually stored in the fluid particles and is carried as a result of their mass motion. Figure 15.10 illustrates the convection conductor.



Fig. 15.8. Thermal network elements.



Fig. 15.9. Conduction conductor.



Fig. 15.10. Convection conductor.

Conductors that represent conduction or convection paths are referred to as linear conductors, because for those paths, the heat-flow rate is a function of the temperature difference between nodal temperatures to the first power.

$$\dot{Q} = G_{ij}(T_i - T_j).$$
 (15.4)

Radiation is the process by which heat flows between two bodies separated in space. Energy is transferred through electromagnetic wave phenomena. Radiation conductors (illustrated in Fig. 15.11) are termed nonlinear, because the heat flow between two surfaces by radiation is a function of the difference of the fourth powers of the surface temperatures:

$$\dot{Q} = G_{ij}(T_i^4 - T_j^4). \tag{15.5}$$



Fig. 15.11. Radiation conductor.

Fluid-flow thermal systems may also be simulated by thermal modeling. Energy stored in the thermal mass (capacitance) of a fluid lump (node) is transferred from one point to another by the movement of the fluid mass. This type of conductor is generally referred to as a one-way or mass-flow conductor, and it is illustrated in Fig. 15.12. The mass-flow conductor is linear and actually asymmetric, because upstream nodes are unaffected by what happens downstream.

$$\dot{Q} = G_{ij}(T_i - T_j)$$
 (15.6)

Conduction

Conduction conductors for rectangular nodes are computed from this equation:

$$G = \frac{kA}{L},\tag{15.7}$$

where G is thermal conductance (W/°C), k is thermal conductivity (W/m·°C), A is cross-sectional area through which heat flows (m²), and L is the distance between adjoining nodes (m). (SI units are shown, but other consistent units could be used.)

The thermal conductivity (k) of materials may vary with temperature or other influencing factors within the system; the cross-sectional area through which the heat flows (A) and distance between node centers (L) are determined by the size and shape of the adjoining nodes. As with the capacitance calculations, necessity



Fig. 15.12. Mass-flow conductors.

to use temperature-dependent properties depends on the degree to which the conductivity changes over the temperature range expected during the analysis.

Rectangular Sections

The length, L, of the heat-flow path, used for conduction-conductance calculations for rectangular nodes, is the distance between node centers, and the area, A, to be used is the area of a node cross-section perpendicular to the line joining the node centers. The convention is depicted in Fig. 15.13.

Cylindrical Nodes

For conductors between nodes that are cylindrical, the conventions shown in Fig. 15.14 should be used.

Parallel Paths

Two or more parallel conduction paths between nodes may be summed to create one conductor value by the following equation:

$$G_T = G_1 + G_2 + \dots G_n. \tag{15.8}$$



Fig. 15.13. Simple conductor representing a heat-flow path through material.



Fig. 15.14. Area and length equivalents for cylindrical nodes.

Equation (15.8) may be helpful in computing an equivalent conductor between two nodes, as illustrated in Fig. 15.15.

Series Paths

Two or more series conduction paths between nodes may be combined to create one conductor value by the following equations:

$$\frac{1}{G_T} = \frac{1}{G_1} + \frac{1}{G_2} + \dots \qquad G_T = \frac{1}{\frac{1}{G_1} + \frac{1}{G_2} + \dots + \frac{1}{G_n}}.$$
(15.9)

These equations may be helpful in computing the conductors between two dissimilarly shaped nodes or two nodes of dissimilar materials, as shown in Fig. 15.16.

Convection

Convection conductors are computed from the expression

$$G = hA$$
,

where G is thermal conductance (W/°C), h is the convective heat-transfer coefficient (W/m².°C), and A is surface area in contact with the fluid (m²). (Again, SI units are used as an example.)

G is the product of the average-unit thermal convective conductance h (convective heat-transfer or film coefficient) and the nodal surface area A in contact with the fluid. However, h is a complicated function of fluid flow, the thermal properties of the fluid medium, and the geometry of the system.

Because the convective process of heat transfer is so closely linked to fluid motion, the first requirement is to establish whether the fluid flow is laminar or turbulent. In laminar flow, the fluid moves in layers and the fluid particles follow a smooth and continuous path. Heat is transferred only by molecular conduction within the fluid as well as at the interface between the fluid and the surface. In turbulent flow, the path of the fluid particles is irregular, and although the general trend of the motion is in one direction, eddies or mixing currents exist. Not only is the conduction mechanism modified, but increased heat transfer also occurs in turbulent flow when energy is carried by fluid particles across flow streamlines and mixes with other fluid particles.

In addition to knowing whether the fluid motion is laminar or turbulent, one must know the process by which the motion was induced. When the heat flows



Fig. 15.15. Parallel conductor flow paths.



Fig. 15.16. Series conductor paths.

between the fluid and the surface as a result of fluid motion caused by differences in fluid density resulting from temperature gradients in the fluid, the heat-transfer mechanism is called free or natural convection. When the motion is caused by some external agent, such as a pump or blower, the heat-transfer mechanism is called forced convection.

Table 15.4 illustrates typical values of average heat-transfer coefficients encountered in engineering practice. The predicted values for h are only approximate. The accuracy of the heat-transfer coefficient calculated from any available equation or graph may be no better than 30%.

Radiation

Most thermal-analysis computer programs linearize the radiation term prior to performing the heat balance at each time-step. This operation simply amounts to the following. First, $(T_i^4 - T_j^4)$ is factored into $(T_i^3 + T_iT_j^2 + T_i^2T_j + T_j^3)$ and $(T_i - T_j)$. Then the term $(T_i^3 + T_iT_j^2 + T_i^2T_j + T_j^3)$ is evaluated by the computer each

	Convective Heat-Transfer Coefficient $h(W/m^2 \cdot °C)$	
Convective Medium		
Air, free convection	1-10	
Air, forced convection	25–300	
Oil, forced convection	50–200	
Water, forced convection	300–12,000	
Water, boiling	3000–60,000	
Steam, condensing	5000-12,000	

Table 15.4. Order of Magnitude of Convective Heat-Transfer Coefficients

time-step using the current values of T_i and T_j . The quantity thus obtained is then multiplied by the input value of the radiation conductor, thus reducing the radiation equation to a linear form. The thermal engineer need only be concerned with the input value of the radiation conductor, which takes the following form:

 $G_{ij} = \sigma \varepsilon_i F_{i,j} A_i$ for radiation to a blackbody, and $G_{ij} = \sigma \Im_{i,j} A_i$ for radiation between gray surfaces,

where G_{ij} is the input value for radiation conductors (W/K⁴); σ is the Stefan-Boltzmann constant, 5.669 × 10⁻⁸ (W/m²·K⁴); ε_i is the emittance of surface *i* (dimensionless); $F_{i\cdot j}$ is the geometric (configuration) factor from surface *i* to surface *j* (dimensionless); A_i is the area of surface *i* (m²); and $\Im_{i\cdot j}$ is the gray-body radiation factor (dimensionless).

The emittance, ε , is a measure of how well a body can radiate energy as compared with a blackbody. Emittance is the ratio of the total emissive power of a real surface at temperature T to the total emissive power of a black surface at the same temperature. The emittance of a surface is a function of the material, the surface condition, and the temperature of the body. The surface of a body, and therefore the emittance, may be altered by polishing, roughing, painting, etc. The values of ε for many common materials and surface conditions have been measured at various temperatures and are presented in Chapter 4, Appendix A, and in many reference manuals. The engineer must determine the value of emittance to be used and whether the variation of ε with temperature is significant over the temperature range expected for the surface.

The geometric (configuration) factor from surface *i* to surface *j*, $F_{i,j}$, is the fraction of total radiated energy from surface *i* that is directly incident on surface *j*; surface *i* is assumed to be emitting energy diffusely. F_{j-i} would be the fraction of total radiant energy from surface *j* that is intercepted by surface *i*. The configuration factors for finite regions of diffuse areas are related by the equation

$$A_i F_{i-j} = A_j F_{j-i} \,. \tag{15.10}$$

The configuration factor, F_{i-j} , is a function of the geometry of the system only. Several computer programs have been developed to compute the shape factors between surfaces with complex geometries, and they will be discussed later. Form factors between some surfaces with simple geometries can be hand-computed. Hand-calculated view factors can be used for preliminary analysis or to check the results of view factors generated by computer programs.

Reference 15.2 presents configuration factors for various simple geometries. The use of these figures and configuration-factor algebra will allow the engineer to determine form factors for many simple radiation problems.

The gray-body shape factor \mathfrak{I}_{i-j} is the product of the geometric shape factor F_{i-j} and a factor that allows for the departure of the surface from blackbody conditions. For radiation enclosures, the \mathfrak{I}_{i-j} factors are generally evaluated with a computer program. The inputs for the program are the $A_i F_{i-j}$ values from every surface of the enclosure to every other surface and the emittance and area for each surface. Simplified equations for \mathfrak{I}_{i-j} exist for two-component gray enclosures. Infinite parallel flat plates: $F_{1-2} = F_{2-1} = 1$.

$$\mathfrak{I}_{i-j} = \frac{1}{\left(\frac{1}{\varepsilon_1} + \frac{1}{\varepsilon_2} - 1\right)}.$$
(15.11)

Concentric cylinders of infinite height or concentric spheres:

$$F_{1-2} = 1, F_{2-1} \neq 0 \tag{15.12}$$

$$\Im_{1-2} = \frac{1}{\frac{1}{\epsilon_1} + \frac{A_1}{A_2} \left(\frac{1}{\epsilon_2} - 1\right)}$$
(15.13)

For nonenclosed surfaces, an effective emittance, ε_{eff} , between the surfaces may be used to compute the gray-body form factor with the following equation:

$$\mathfrak{S}_{i-j} = \mathfrak{e}_{\mathrm{eff}i} F_{i-j} \,. \tag{15.14}$$

The effective emittance is a function of the emittances of the two surfaces and the configuration factors (F) between them. The error induced with use of ε_{eff} is the result of neglecting secondary reflections from surfaces other than the two for which the effective emittance was determined. By reducing Hottel's method for two flat surfaces with emissivities of ε_1 and ε_2 in a black enclosure, one can construct the following equation:

$$\varepsilon_{\text{eff}} = \frac{\varepsilon_1 \varepsilon_2}{1 - F_{1-2} F_{2-1} (1 - \varepsilon_1) (1 - \varepsilon_2)} \,. \tag{15.15}$$

The examples of configuration-factor algebra in Fig. 15.17 should be helpful.

Energy Sources and Sinks

Energy sources and sinks, designated by Q, are modeling elements that allow the impression of positive or negative heating rates on the nodes of a thermal network, independent of conductor paths to the nodes.

$$\begin{split} A_1F_{1\cdot3} &= A_3F_{3\cdot1} \\ A_1F_{1\cdot34} &= A_1F_{1\cdot13} + A_1F_{1\cdot4} \\ A_{12}F_{12\cdot34} &= A_1F_{1\cdot34} + A_2F_{2\cdot34} \\ A_{12}F_{12\cdot34} &= A_1F_{1\cdot3} + A_1F_{1\cdot4} + A_2F_{2\cdot3} + A_2F_{2\cdot4} \\ A_1F_{1\cdot4} &= A_3F_{3\cdot2} \ (symmetrically positioned) \end{split}$$



Fig. 15.17. Configuration-factor algebra.

Common engineering applications of heat sources in thermal models are:

- solar and planetary heating
- aerodynamic heating
- · avionic cold-plate heat loads
- change-of-state latent energy
- thermal-control heaters
 Common applications for heat sinks are:
- change-of-state latent energy
- radiator heat rejection
- aerodynamic cooling

Heating rates may be impressed on diffusion (finite-capacitance) or arithmetic (zero-capacitance) nodes. Most thermal analyzers provide a separate entry block for entering heating or cooling rates. For example, the SINDA computer program uses the SOURCE data block for such entries. In the usual case, heating rates are not considered when computing the time steps for transient analysis, and large heating rates on low-capacitance nodes may create instability in the network solution. Also, the impression of large heat sources on arithmetic nodes with radiation (nonlinear) conductors attached often causes large erroneous temperature oscillations in the arithmetic and adjoining nodes. Both of these difficulties can be avoided with the use of the program-control constants incorporated in most thermal-network analyzers. These control constants are the time-step multiplication factor and the maximum temperature change allowed.

Thermal Design Analysis Example: POAM

The Polar Ozone and Aerosol Monitor (POAM) sensor will be used as an example of a thermal design analysis. This sensor measures the concentrations of ozone and aerosols in the upper atmosphere of Earth's polar regions. The experiment was funded by the U.S. Strategic Defense Initiative Office, administered by the Office of Naval Research, and flown on the French SPOT Earth resources satellite.

The sensor measures the concentrations of ozone and aerosols by observing the attenuation of sunlight as it passes through the atmosphere during sunrise and sunset events while the satellite circles Earth in a polar, sun-synchronous orbit, as shown in Fig. 15.18. The sensor actually contains nine small telescopes, each of which has a filter and a sensor. These telescopes measure the intensity of sunlight in nine very narrow wavebands. Observing the intensity of sunlight as the sun sets or rises enables the measurement of concentrations at different heights in the atmosphere to a resolution of about 1 km. These measurements support research into the depletion of the protective ozone layer in the upper atmosphere, as well as other atmospheric studies.

Physical Configuration

The POAM sensor (Fig. 15.19) consists of a rectangular base with four mounting feet, called the azimuth housing, and a dome-shaped enclosure containing the telescope assembly. The dome is attached to a short shaft that rides on a pair of bearings in the azimuth housing. The only physical connections between these two assemblies are the bearings and a small cable bundle that runs down the center of the



Fig. 15.18. POAM data events.

hollow shaft, which is not shown in the figure. The telescope assembly is similarly connected to the dome only through a pair of bearings and a few small wires. Because heat conduction across ball bearings and along fine wires is relatively weak, the telescope, dome, and azimuth housing are only rather weakly coupled together thermally.

The entire sensor assembly is mounted to an exterior face of the host SPOT satellite (Fig. 15.20). The mounting is accomplished by a bracket, as shown in the figure. The satellite itself is placed in a 822-km, 98.738-deg inclined, circular sunsynchronous orbit with a period of 100 min and a range of beta angles from 14.5



Fig. 15.19. POAM sensor (TTC).



Fig. 15.20. POAM on SPOT spacecraft (TTC, Astrium).

to 29.8 deg. At the sunrise event, the sensor makes one minute of observations and then rotates the dome about 130 deg in azimuth to be in the proper position for the sunset event that occurs about 40 min later. Only small rotations of the telescope on its elevation bearings are required to track the sun during each observation. A typical observation sequence is illustrated in Fig. 15.21.

Thermal-Design Requirements

The thermal-design requirements for POAM, driven by both the instrument and the host spacecraft, are listed in Table 15.5. From the sensor's point of view, the



Fig. 15.21. POAM observation sequence.

Table 15.5. POAM Thermal-Design Requirements

- Optical head case temperature ranges from -10 to 50°C.
- Survival/turn-on limit equals -30°C.
- Spacecraft must survive without power for 3-1/2 hours after launch.
- Spacecraft must survive without power for two orbits returning from safe to nominal operating mode.
- Uncertainty margin of 10°C applied to predicted temperatures, 25% margin on heater power.
- Conduction between optical head and bracket limited to less than 0.07 W/K.

instrument must be maintained between -10 and $+50^{\circ}$ C while operating, and between -30 and $+50^{\circ}$ C while not operating or at turn-on. From the spacecraft's point of view, certain mission-related requirements must be met, including: the ability to go for 3-1/2 hours after launch with no power supplied to the instrument; the ability to survive the spacecraft safe-mode condition in which only survivalheater power is available to the instrument; the ability to withstand a two-orbit (approximately 3-1/2 hour) transition from safe mode to normal operating mode, during which time neither electronics nor survival-heater power will be available; and the ability to limit conduction between the instrument and the spacecraft mounting bracket to less than 0.07 W/K. Furthermore, all organizations involved agreed that an uncertainty margin of 10°C would be applied to all temperature predictions and any heaters would be sized to provide either a 10°C margin to lower temperature limits or a 25% excess capacity at the lower temperature limit.

Conceptual Design

The first step in the design process is to identify the factors that will drive the design. Such factors include the previously discussed design requirements levied by the instrument designers and the satellite, as well as the instrument heat dissipation and range of external environments.

The instrument heat dissipation varies around the orbit because of the operation of motors during telescope slewing. At the conceptual design phase of this program, the electrical-power draw for the instrument (which is all converted to heat because no significant amount of energy is output) was estimated to be no greater than that shown in Fig. 15.22. Because periods could also occur of several orbits or longer during which no observations would take place and the drive motors would not be in operation, the minimum power draw was assumed to be a constant 4.4 W. Most of this heat is dissipated in the azimuth housing, with only a small portion dissipated in the telescope.

The instrument is also exposed to solar, Earth IR, and albedo environmental heating fluxes. Because the satellite is Earth-facing in a sun-synchronous orbit, the sun position relative to the vehicle forms a cone as the satellite goes around Earth, as shown in Fig. 15.23. This cone has an elevation angle that equals the orbit beta angle, 14.5 to 29.8 deg. Eclipse time ranges from 32.5 to 34.6 min, as can be calculated from the equations in Chapter 2. Because the spacecraft is Earth-facing, the



Fig. 15.22. Instrument heat dissipation.



Fig. 15.23. Solar illumination angles.

instrument is always pointing with the dome facing straight down at Earth, and Earth IR loads can therefore be assumed to be constant around the orbit. Albedo loads will, of course, vary around the orbit, but the narrow range of beta angle ensures that orbit-average albedo loads will not change a great deal over time.

Given the estimated heat dissipation, and the requirement that the instrument be conductively isolated from the satellite, some amount of radiator area will be required to reject the instrument waste heat to space. Most of this heat dissipation is in the base, which is largely covered on three sides by its mounting bracket. The side facing away from the spacecraft is the only one with a fairly clear view to space, although it does have a small view to the spacecraft solar array. To determine if this side would have sufficient area to reject the waste heat, a simple calculation can be performed. The maximum orbit-average internal heat plus the maximum orbit-average environmental heat flux must not exceed the energy radiated from the surface at the maximum allowable temperature;

$$\dot{Q}_{\text{ELECTRONIC}} + \dot{Q}_{\text{ENVIRONMENTAL}} = A\varepsilon\sigma T^4$$
. (15.16)

(The small amount of heat backload from the spacecraft solar array may be neglected for this preliminary evaluation. Also, the mass of the instrument, 11 kg, gives it a large heat capacity relative to the heat pulses during motor operations. This ensures that the temperature will not vary too much from the orbit average, making these orbit-average calculations reasonably accurate.)

The maximum orbit-average internal heat dissipation can be calculated in a straightforward manner from Fig. 15.22 to be 4.7 W. The worst-case solar heating for this surface would occur when the sun is at its maximum elevation angle above the surface, 29.8 deg, as shown in Fig. 15.23. The orbit-average solar load is given by:

$$\dot{Q}_{\text{solar}} = (\sin 29.8^\circ) S\alpha(\% \text{ of orbit in sunlight}),$$
 (15.17)

where S is the solar constant and α is the absorptance of the surface.

With a 5-mil silvered Teflon radiator-surface finish, the EOL absorptance would be approximately .18 after 3 years in low Earth orbit. This figure is based on a BOL α of .05, and a degradation of .09. The maximum solar constant is 1414 W/m² (Chapter 2), and the percent sunlight time is

$$\frac{\text{orbit period} - \text{eclipse time}}{\text{orbit period}} = \frac{100 \text{ min} - 32.5 \text{ min}}{100 \text{ min}} = 67.5\%.$$
 (15.18)

Substituting these values in Eq. (15.17) gives a maximum orbit-average absorbed solar load of 84.6 W/m².

The orbit-average Earth IR load can be calculated using Fig. 15.24, which is applicable to flat, unblocked surfaces. Because the satellite's orientation is Earth-pointing, the instrument radiator surface remains perpendicular to Earth all around the orbit, so Earth IR heating will not change. The ρ angle for Fig. 15.23 is therefore 90 deg, and the altitude is 822 km. At the intersection of $\rho = 90$ deg and h = 822 km, find $F_E = .22$, project $F_E = .22$ to the line labeled q_E , then project this point horizontally to the q_E (Earthshine) scale to read $q_E = 47.8$ W/m². The heat absorbed is the incident value times the emissivity of the surface, or (47.8)(.78) = 37.3 W/m² for 5-mil silvered Teflon.

Albedo loads can be calculated in a similar fashion using Fig. 15.25; however, one must calculate the value for several points because albedo changes as the satellite travels the orbit. Using Fig. 15.25, begin with the same altitude and ρ angle as in the Earth-IR calculation, i.e., 822 km and 90 deg. At the intersection of $\rho = 90$ deg and h = 822 km, find $F_R = .22$. Because the satellite is in an orbit with a beta angle of almost 30 deg, shift down to the scale labeled $\beta = 30$ deg and draw a



Fig. 15.24. Incident Earthshine irradiation on a surface element in an Earth orbit (courtesy of Lockheed Martin).

vertical line at $F_R = .22$. The intersections of this line with the family of lines labeled $\theta = x$ deg gives the incident albedo flux for various points around the orbit at position angles of θ deg, measured from the closest approach to the subsolar point. Figure 15.25 gives the values for the half of an orbit, on the sunlit side of Earth. If these values are averaged and then divided by two to account for the dark half of the orbit, which experiences no albedo load, the resulting value is the orbitaverage incident albedo of 33.8 W/m². Multiplying this by the silvered Teflon solar absorptivity of .18 gives an orbit-average absorbed-albedo load of 6.1 W/m². The albedo load is therefore a fairly small contributor compared to the solar (84.6 W/m²) and Earth IR (37.3 W/m²) heat loads.


Fig. 15.25. Incident albedo irradiation on a surface element in an Earth orbit. (Note: Earth reflectance is assumed to be 0.38. Albedo plotted is an approximation, with the largest error near the terminator [θ = 90].) (Courtesy of Lockheed Martin.)

Substituting the orbit-average electronics heat, solar IR, and albedo loads into Eq. (15.19) and solving for the radiator area at a temperature of 40°C (313 K) gives:

$$\dot{Q}_{\text{ELECT}} + \dot{Q}_{\text{ENVIR}} = A\varepsilon\sigma T^4 \tag{15.19}$$

$$16.0 + [84.6 + 37.3 + 6.1]A = A(0.78)(5.669 \times 10^{-8})(313)^4$$
(15.20)

$$A = 0.01571 \text{ m}^2 = 157 \text{ cm}^2. \tag{15.21}$$

Because the face of the azimuth housing has an area of 203 cm², adequate area for a radiator is available. (Note: A radiating temperature of 40°C was selected to allow for the required 10°C margin between analysis and the maximum allowable instrument temperature limit of 50°C.)

The minimum temperature of the instrument under cold-case conditions using the radiator size calculated above was determined in the same manner. Solar, albedo, and Earth IR heating for the cold-case orbit ($\beta = 14.5$ deg, summer) and electronics waste heat without motor operations were summed and, with an area of 157 cm², Eq. (15.19) was solved for *T*. This gave a cold-case temperature of 22°C. These preliminary hot- and cold-case calculations indicated that the entire side of the azimuth housing (203 cm²), rather than the 157 cm² calculated above, could be used as a radiator to bring the average temperature down a little. Lower operating temperatures generally increase the life and reliability of electronic components.

Based on the preliminary radiator sizing and the requirements listed in Table 15.5, the thermal-design concept shown in Fig. 15.26 was identified. The side of the azimuth housing facing away from the spacecraft would be covered with 5-mil silvered Teflon, and it would serve as the primary radiator. All other surfaces of the azimuth housing and dome would be covered with MLI blankets to minimize loss of heat through these surfaces and to essentially eliminate radiative thermal



Fig. 15.26. POAM thermal design.

interactions between the instrument and the spacecraft. A small "window" would be made in the MLI covering the dome to allow the telescopes a view out. The surface of the aluminum dome exposed in the window area, however, would be polished to provide a low absorptance and emittance to minimize both radiative heat loss and energy absorbed from incident environmental heat fluxes. Because of their poor conductive-heat transfer, the bearings between the telescope and the dome and between dome and azimuth housing tend to thermally isolate these components. Therefore, to tie them together radiatively as much as possible, the telescope external surface and the dome internal surface would be given a black high-emittance finish. The bottom of the dome and the top of the azimuth housing would also be painted black to maximize the radiative coupling in the interface. Plastic isolators would be placed under each of the mounting feet to meet the requirement of limiting conductive-heat transfer between the instrument and the spacecraft-supplied support bracket.

Detailed Design Analysis

Once a design concept is identified, a detailed analysis must be conducted to finetune the design and predict instrument temperatures under the entire range of flight conditions. This involves identifying analysis cases to be run and constructing a GMM and a TMM of the instrument. For the POAM program, a thermal analysis of the overall instrument was conducted, with separate additional analyses performed of the individual circuit cards and telescope photo detectors. This discussion will be limited to the instrument-level analysis.

The GMM and TMM serve different purposes. The GMM, a mathematical representation of the physical surfaces of the instrument, is used to calculate greybody radiation couplings between surfaces as well as heating rates resulting from environmental fluxes. The TMM, most often a lumped-parameter network representation of the thermal mass and conduction and radiation couplings of the instrument, is used to predict instrument temperatures. The radiation interchange couplings and environmental heat fluxes calculated by the GMM are used in constructing the TMM. Both the GMM and TMM are constructed and executed using industry-standard computer programs. The most common GMM codes are TRASYS^{15.3} and NEVADA,^{15.4} and the most common TMM code is SINDA.^{15.5,15.6} Other commercially available codes do exist, however, and some large companies use their own internally developed codes. The codes mentioned above will be discussed in detail in later sections of this chapter.

Analysis Cases

Based on the instrument operating modes and thermal-design requirements discussed earlier, four significant thermal-design analysis cases were identified, as shown in Table 15.6. Normal on-orbit operations are bounded by the hot and cold operating cases. The responses of the instrument to launch and a potential spacecraft "safe mode" condition were also analyzed.

The hot operating-case conditions include maximum solar heating, which occurs at the highest beta angle with the winter solar constant, maximum Earth IR and albedo, maximum (EOL) solar absorptance on the external surface finishes, good insulation-blanket performance, maximum motor operations, telescopes

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Hot Operating	Cold Operating	Safe Mode	Launch/Ascent
$\beta = 29.8$ °C, winter	$\beta = 14.5^{\circ}$ C, summer	NA	$\beta = 14.5$ °C, summer
EOL $\alpha = 0.17$	BOL $\alpha = 0.08$	NA	BOL $\alpha = 0.08$
MLI $\varepsilon^* = 0.01$	MLI $\varepsilon^* = 0.05$	MLI $\varepsilon^* = 0.05$	MLI $\varepsilon^* = 0.05$
Telescope sees sun twice per revolution	Telescope does not see sun	Telescope does not see sun	Telescope does not see sun
10 min/rev motor operations (3 W)	No motor operations	No motor operations	No motor operations
Hot spacecraft	Cold spacecraft	Cold spacecraft	Cold spacecraft
Radiator to solar- array view varies around revolution	Radiator to solar- array view varies around revolution	Fixed radiator view to solar array	Radiator to solar- array view varies around revolution
4.4-W electronics heat	4.4-W electronics heat	No power, heaters only	No power, no heaters
Earth IR = 234 W/m^2	Earth IR = 208	No Earth IR	Earth IR $= 208$
Albedo $= 0.42$	Albedo = 0.34	No albedo	Albedo = 0.34

Table 15.6. Design Environments/Assumptions

looking at the sun twice per orbit, maximum spacecraft temperatures, and maximum electronics waste heat. Cold operating-case conditions include minimum solar loads, minimum (BOL) solar absorptances, poor insulation-blanket performance, no motor operations, telescope stopped in a position where it does not see the sun, cold spacecraft temperatures, minimum Earth IR and albedo, and minimum electronics waste heat (which happens to be the same as the maximum heat because it is constant for this instrument).

During safe mode, the spacecraft turns and points constantly at the sun instead of Earth and the POAM instrument is turned off, although some power is available to run heaters, if required. Because the instrument is off and shadowed from the sun by the spacecraft, this is a cold-case condition. It is therefore also assumed that there is no Earth IR or albedo heating for conservatism.

During the launch phase, POAM is turned off. While sitting on the launchpad, the instrument will be at approximately the same temperature as the purge gas inside the booster fairing $(15^{\circ}C)$, because it has no internal heat dissipation. For the first three minutes after liftoff the booster fairing is in place and experiences a large temperature rise. Because of the large thermal mass of POAM and the brief duration of this phase, the thermal effect on the instrument is negligible. This is followed, however, by a 1/2-hour period during which the spacecraft attitude is not controlled and the POAM radiator may see the sun, Earth, and/or deep space. Because the instrument is not powered, this is a cold case, and the assumption was therefore made that the radiator was facing deep space with no environmental heat

fluxes incident on any surface. After 1/2 hour the spacecraft is stabilized in its normal attitude, but POAM is still not powered. The point of this analysis case is to determine how long the instrument can go after launch with no power without violating its lower survival temperature limit of -30° C.

GMM Construction

The GMM of the POAM mounted on the host spacecraft was constructed using the NEVADA code. The model, shown in Fig. 15.27, consists of a simple representation of the spacecraft, POAM, and the support bracket. It was constructed on a CAD-like system using rectangular, circular, hemispherical, and cylindrical surface elements available in the NEVADA package, and each surface was assigned the appropriate absorptance, emittance, and specularity. Details about how these models are constructed using NEVADA can be found in Ref. 15.4.

The GMM was then run using NEVADA to calculate the radiation interchange factors between all surfaces. NEVADA also outputs a radiation-conductor block that may be merged directly into the SINDA TMM. This block of conductors will be discussed later.

The GMM was then placed mathematically into the proper orbit and attitude using another section of the NEVADA software. Solar, Earth IR, and albedo heat loads absorbed on each surface were calculated for a dozen points around the orbit for both the hot-case (maximum beta angle, winter, maximum absorptance) and the cold-case (minimum beta angle, summer, minimum absorptance) orbits. These heat rates are also output by NEVADA in arrays that can be merged directly into the TMM.

TMM Construction

The TMM consists of nodes representing parts of the instrument, diffusion and radiation conductors between nodes, blocks of arrays and constants for storing inputs such as environmental heating rates calculated by NEVADA, and logic blocks for controlling the execution of the program. A listing of the POAM TMM in SINDA format is shown in Table 15.7.



Fig. 15.27. POAM geometry.



Table 15.7. POAM TMM Listing

Table 15.7. POAM TMM Listing (Continued)

BCD SCONSTANTS DATA TIME0=0., ARLXCA=.1, DRLXCA=.1, NL00P=5000 NUIM-5000, BALENG=.05, CSGFAC=1. PROGRAW CONTROL CONSTANTS 1,0,2,0,3,0,4,0,5,0, 6,0,7,0,8,0,5,0,10,0 11,0,12,0,13,0,4,0,16,0 CONSTANT STORAGE LOCATIONS 16,0.,17,0.,18,0.,19,0. END BCD BARRAY DATA AZIMUTH HOUSING RADIATOR AREA.VIEW TO SPACE VS. TIME FOR ONE ORBIT С 1,0.,.0945,.21,.0917,.42,.115,.63,.108,.84,.111 1.05,.111,1.26,.0917,1.47,.0945,1.68,.0945,END AZIVUTH HOUSING RADIATOR AREA*VIEW TO SOLAR ARRAY VS. TIME FOR ONE ORBIT c 2,0., .0245, .21, .0273, .42, .0027, .63, .0105, .84, .0105, 1.05, .008 1.26, .0273, 1.47, .0245, 1.88, .0745, END SOLAR ARRAY TEMPERATURE VS. TIME FOR ONE ORBIT c 3,0.,176.,.653,133.,.822,-112.,1.212,-126.,1.213,133. 1.68,176.,END ENVIRONMENTAL HEATING ON TELESCOPE BODY VS. TIME FOR ONE ORBIT Ĉ ENVIRONMENTAL HEATING ON FELESCOPE BODY VS. TIME FOR ONE ORBIT 4,0.,1.17,047,79,377,56,374,19,512,19,513,5.26 .56,5.34,606,5.44,.663,5.98,.654,19,1.212,.19 1.213,5.85,1.4,5.02,1.54,3.74,1.68,1.17,FND ENVIRONMENTAL HEATING ON DOME IN WINDOW AREA VS. TIVE FOR ONE ORBIT 5,0.,2.5,04,2.5,186,4.82,42,5.9,513,6.3 .653,4.8,.654,1.5,1.212,1.5,1.213,4.8,1.245,5. 1.32,2.0,1.58,2.6,FND ENVIRONMENTAL HEATING IN GAP BETWEEN DOME AND AZIMUTH HCUSING VS. TIME 6,0.,1.6,.467,1.4,.56,4.3,.653,4.2,.654,1.,1.212,1. 1.213,4.2,:367,4.3,1.4,1.4,1.68,1.6,FND ENVIRONMENTAL HEATING ON OUTER LAYER OF DOME VLT 7,0.,58,.09,48,.19,48,.28,58,.37,70,.47,83. C с С ENVIRUMENTAL HEATING UN UDTER LATER OF DUME %1 7,0.,56.,89.,48.,19,48.,19,45.,37,70.,47,8 .56,89.,.653,95.,654,24.,1.212,24.,1.213,96. 1.26,96.,1.44,83.,1.65,58.,ENU AZIWUTH HOUSING ELECTRONICS WASTE HEAT .47,83. c 2,0.,15.,1.68,15.,END AZIMUTH MOTOR WASTE HEAT С 9,0.,0.,.652,0.,.653,10.23,.736,10.23,.737,0.,1.244,0.,1.245,10.23 1.32,10.23,1.321,0.,1.69,0.,END SOLAR HEAT FLUX PER SQ. IN. INCIDENT ON AZIWUTH HOUSING RADIATOR VS, TIME c 11,0,,1.53, 653,1.53, 654,0,,1.212,0,1.213,1.53,1.69,1.53,END ALBEDO HEAT FLUX PER SQ. IN. INCIDENT ON AZIMUTH HOUSING RADIATOR VS. TIVE С 12,0, 19,187,15,133,0,1,207,0,1493,15,11.88,19,END EARTH IR FLUX PER SQ. IN, INCIDENT ON AZIMUTH HOUSING RADIATOR VS. TIME C 13,0., 107,1.68, 107,END END BCD 3EXECUTION PROGRAM CONTROL CONSTANTS М TINEND=100. OUTPUT=1Ø. v ... в ATSDUF =ND BCD BVARTABLES 1 м IF (TIME0.GT.98.3) OUTPUT=1./60. INTERPOLATES ARRAYS AND PLACES CURRENT VALUES IN CONSTANT LOCATIONS DIICYL(1.68, TIMEO, A1, XK1) D11CTL(1.88, 11MEU,A1,AA1) D11CTL(1.88, TIMEU,A2,XK2) D11CTL(1.68, TIMEU,A3,XK3) D11CTL(1.68, TIMEU,A4,XK4) D11CTL(1.68, TIMEU,A6,XK6) D11CTL(1.68, TIMEU,A6,XK6) D11CTL(1.68, TIMEU,A7,XK7) D11CYL(1.68, 11ME0,A2,XK8) D11CYL(1.68, TIME0,A9,XK8) D11CYL(1.68, TIME0,A9,XK9) D11CYL(1.68, TIME0,A11,XK11) D11CYL(1.68, TIME0,A12,XK12) D11CYL(1.68, TIME0,A13,XK13)

м	T997=XK3		SETS CU	RRENT SOLAR	ARRAY TEMPERATURE
м	G101=(XK1*8.4)*(30.6	•.78) • (.1714E-8)	/144. ADJUST	S RADIATOR R	ELATIVE VIEW TO
м	G102=(XK2+8.4)+(30.6	•.78)*(.1714E-8)	/144. SPACE	AND SULAR AR	KAT AS ARRAT TURIS
2 2 2 2 2 2 2 2 2 2 2 2 2 2 2 2 2 2 2	Q23=32. Q22=XK7 Q101=30.6*(.15*(XK Q3=XK0/4. Q4=Q3 Q7=Q3 Q8=Q3 Q112=XK4 ENV1 Q12=XK5+XK6/2. Q9=XK9/2.+XK6/4.	ENVIRONMENTAL H "" AZIMUTH HOUSING """"" RONMENTAL HEAT OI ENVIRONMENTAL AI ENVIRONMENTAL AI	EATING DN AZ ""DQ 3)" AZ ELECTRUNICS " " N TELESCOPE EATING ON DO ND AZIMUTH D	IMUTH HOUSIN ME MLI OUTER IMUTH HOUSIN WASTE HEAT " " ME RIVE MOTOR "	G MLI OUTER LAYER LAYER G RADIATOR ASTE HEAT
M	Q10=Q9 END BCD 3VARIABLES 2 END BCD 30UTPUT CALLS TPRINT END BCD 3END OF DATA	'n			

 Table 15.7. POAM TMM Listing (Continued)

The nodalization scheme chosen for the TMM is shown in Fig. 15.28. Because the base, dome, and telescope are all constructed of thick (2.5–5.0 mm) aluminum, they can each be assumed to be fairly isothermal, and a minimum number of nodes are required to model them. The entire telescope is therefore modeled as one node, as is the dome assembly. The location of the radiator on one side of the azimuth housing, however, means that some temperature gradient could exist between it and heat-dissipating elements on the opposite face. The azimuth housing was therefore modeled using ten nodes, as shown in Fig. 15.28. The MLI covering the dome and azimuth housing were represented by one node each. Temperatures of the spacecraft solar array, the mounting bracket, and the outer layer of the



Fig. 15.28. POAM TMM nodes.

spacecraft MLI were all supplied by the spacecraft engineers and were therefore put in the POAM TMM as boundary-driver nodes.

Capacitances of each node were hand-calculated either by using a weight found in a mass-properties report for the instrument and multiplying by the specific heat, or by calculating the volume of material and multiplying by the density and specific heat. Arithmetic (zero-capacitance) nodes were used to represent MLI blankets because the blankets are extremely light and respond almost as if they had zero mass. The boundary nodes do not require a capacitance because they are treated by the program as constant-temperature (infinite-capacitance) sinks.

Diffusion conductors were calculated in a straightforward manner using the $k_{\overline{1}}^{A}$

relationship as discussed in the "Fundamentals of Thermal Modeling" section. One exception to this was the conduction between the azimuth housing and its rear cover, which is held in place with screws. The contact conduction between these nodes was based on screw-conduction terms found in Chapter 8. Another exception was the conduction across the azimuth and elevation bearings. Because bearing conduction is so uncertain, two cases were run to bound the problem: zero conductivity at one extreme, and a conduction equal to 315 W/m^2 over the entire area of the bearing race at the other extreme. Both values were used for each of the four design analysis cases for POAM, and the value that resulted in the most extreme temperatures was chosen.

Conductive heat transfer between the instrument and its mounting bracket had to be limited to less than .07 W/°C per spacecraft requirements. To accomplish this, the fiberglass isolators shown in Fig. 15.29 were designed for installation



 $G_{total} = G_{through bolt} + G_1$

Fig. 15.29. POAM mounting isolator.

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under the four mounting feet. The calculations of the conductance across these isolators are shown in the figure.

Radiation conductors generated by NEVADA were merged into the TMM. The conductors through the MLI blankets, however, were calculated manually. The heat leak through MLI can be modeled as an effective emittance, ε^* , as discussed in Chapter 5. A range of effective emittances was used for this analysis, because predicting the exact performance of an insulation blanket before it is built and tested is difficult. For the hot case, a value of .01 was chosen, while .05 was used in the cold case. An ε^* of .05 is rather high, but it is justified in this case because the blankets are small and therefore more susceptible to the heat-leak effects of edges and attachments. For each node covered with MLI, a radiation conductor was calculated as (Area)(ε^*)(σ) between the instrument node and the node representing the outer layer of the MLI blanket. The radiation couplings from the outside face of the MLI blanket to space and to the spacecraft were previously calculated by the NEVADA model and were already merged into the TMM.

The complete hot-case TMM is shown in Table 15.7. The first block contains the node data. Each node is given an integer number, initial temperature, and capacitance. Arithmetic (zero-capacitance) nodes are represented in SINDA by negative capacitance values, and boundary (infinite-capacitance) nodes are represented by negative node numbers, as can be seen in the table. The next block contains the conductor data. Each conductor input contains an integer conductor number, the nodes that the conductor connects together, and a conductor value. Radiation conductors are given negative conductor numbers in SINDA. The next block contains the user and SINDA data constants. In this case, a number of program-control constants are present, as well as ten constant-storage locations, which will be discussed later. The next block contains array data. In this case arrays are here giving time-varying environmental-heat fluxes previously calculated by NEVADA, time-varying electronics-waste-heat rates, and time-varying radiation conductors between the POAM radiator node and the rotating spacecraft solar array (this was also previously calculated by NEVADA and input manually into the TMM).

The next three blocks control the execution of the program. The second of these, "VARIABLES 1," specifies how much heat is on each node as well as what the radiation coupling is from the POAM radiator to the solar array at any given time. This block is accessed before the start of each time step as the program calculates the change of POAM temperatures with time. The final block specifies the data to be output by the program. In this case, temperatures and impressed heat rates for each node are requested.

Predicted Temperatures

The file in Table 15.7 was executed by the SINDA program, and temperatures were calculated. Similar files were constructed for the cold-operating, safe-mode, and launch-ascent cases. Predicted temperatures for these conditions are shown in Figs. 15.30 through 15.32. Comparison of these results to the requirements of Table 15.5 shows that all requirements are met with adequate (10°C or greater) margin. For the safe-mode case a heater was required to maintain the instrument above its lower survival temperature.



Fig. 15.30. Hot and cold operating temperatures



Fig. 15.31. Safe-mode temperatures.



Fig. 15.32. Launch/ascent temperatures.

Thermal-Balance Test

The thermal analysis described above was verified by a thermal-balance test. This was conducted during spacecraft-level thermal-vacuum testing in Toulouse, France. The POAM instrument was installed on the spacecraft in the flight configuration. Hot- and cold-case test phases were planned. Because of limitations associated with the spacecraft, these were not precise representations of the flight hot and cold cases, but they were close, and they provided two good conditions with which the TMMs could be checked and correlated.

The temperature instrumentation used is shown in Fig. 15.33, and the hot- and cold-case steady-state temperatures are shown in Fig. 15.34. As can be seen from this data, the azimuth housing is nearly isothermal with only a 2 to 3°C variation around the box. The dome assembly, however, ran approximately 15°C cooler than the azimuth housing. This temperature difference indicates that little conductive coupling exists between the dome and azimuth housing, as was assumed in the design analysis for conservatism. The dome runs cooler because no electronics



Fig. 15.33. Test thermocouples.



Fig. 15.34. Thermal-balance test temperatures.

waste heat is dissipated in the telescopes and no sunlight is shining into the telescope aperture in the test chamber. Looking at the sun twice per rev on orbit will cause the dome temperature to rise closer to that of the base during flight operations.

The optical-head thermal model was run using the as-run test environment. Comparison of the model predictions to test data showed that the conductance value through the thermal isolators between the optical head and its support bracket was low by approximately 40%. The lower conductance value reflected an earlier isolator design that used titanium rather than stainless-steel bolts. Further comparisons also revealed that heat losses through the gap between the dome and azimuth housing, which had been neglected in the analysis, had a noticeable impact on both the optical-head temperature and the temperature difference between the dome and azimuth housing. After corrections for these two effects, the TMM and test data agreed within $\pm 3^{\circ}$ C, as shown in Table 15.8.

	Cold Case (°C)			Hot Case (°C)		
-	Test	Model	D	Test	Model	D
Dome	11	8	+3	9	8	-1
Azimuth housing	6	8	+2	25	23	-2
Fin	7	9	+2	24	24	0

Table 15.8. TMM Correlation to Test Data

Margins

Even the best thermal analyses are subject to uncertainties. Despite our best efforts and the sophistication of today's analytical codes and computer workstations, flight experience teaches that predicted temperatures are not always precisely accurate. Some inaccuracies result from factors that are known to be uncertain, such as contact conductances and the performance of insulation blankets. Some uncertainties are just the results of the simplifications that are inherent in the analytical techniques. Some are caused by errors. In any event, our understanding of these uncertainties is not yet sufficient to eliminate them from the analysis process.

When one compares temperatures predicted by analysis with those that actually occur in flight, one notes significant dispersions. Figure 15.35 shows such a comparison for two satellites, FLTSATCOM and DSCS II. A study of a number of satellite programs conducted by Stark (Ref. 6 of Chapter 19) concluded that an 11°C margin was required to provide 2- σ (95%) confidence that flight temperatures would be within limits (Table 15.9). This study is the basis of the MIL-STD-1540 analytical uncertainty margin of 11°C. It is important to note that this margin is applied to predictions made by analytical models that have been correlated to thermal-balance test data. For an uncorrelated model, the uncertainty jumps to 17°C. In addition, very large discrepancies (40 or 50°C) do occur now and then. A thermal-balance test is needed to catch these large, potentially mission-threatening, errors before the satellite is launched. Simply using the 17°C margin and forgoing a thermal-balance test could be a costly mistake.

		Temperature Uncertainty (°C)	
Standard Deviation	Percent of Confidence	Unverified Analytical Predictions	Predictions Verified by Testing
1.0	68	8.3	5.6
1.4	85	12.2	7.8
2.0	95	16.7	11.8
3.0	99	25.0	16.7

Table 15.9. Temperature Uncertainty Margin Based on Spaceflight Database



Fig. 15.35. Thermal-model accuracy assessment.

Unlike military programs, NASA and commercial-satellite procurement agencies do not have a specification on uncertainty margins for thermal analysis. An informal survey of NASA and commercial-satellite programs showed that 5°C was the most common margin used, although significantly different margins were used on some programs. A summary of margins typically used on commercial programs is shown in Table 15.10.

	Thermal Analysis Range (°C)	Acceptance Range (°C)	Qualification Range (°C)
Boeing	+5 to 55	0 to 60	-5 to 65
Lockheed Martin	-5 to +50	-10 to +55	-15 to 60
Space Systems Loral	+5 to 55	0 to 60	-5 to 65

Table 15.10. Commercial Satellite Component Temperature Ranges^a

^aTemperature ranges for many commercial programs are self-imposed by the contractor and not contractually required by the customer.

Recommended Margins

For components that have no thermal control or have passive thermal control (PTC) only, an uncertainty margin of at least 11°C should be included in all cases in determining the maximum or minimum expected flight temperature. This 11°C thermal margin is applied to the temperature predictions made by the TMMs after correlation to the thermal-balance test. This usage implies that even larger thermal margins are required at the beginning of a program to accommodate changes that typically evolve from preliminary design to final product. The suggested margin during the design phase is 17°C, which can be reduced to 11°C after the thermal-balance test.

For cryogenic systems operating below approximately -70°C, the heat-load margins shown in Table 15.11 are recommended in lieu of the 11°C temperature margin.

A constant-conductance heat pipe is are considered a PTC element and should use the 11°C margin discussed above. In addition, the heat-transport capability of the pipe should be at least 50% greater than that required for the maximum heat load at the maximum expected flight temperature.

Self-regulating heaters that use resistance elements that exhibit a large variation in resistance with temperature (such as "auto trace" or positive-temperature-coefficient thermistors) are considered passive devices, and they require a margin of 11°C.

For thermal designs in which temperatures are actively controlled by variableconductance heat pipes, louvers, heat pumps, expendable coolant systems, or refrigerators, a heat-load margin of 25% may be used in lieu of the 11°C specified above at the worst-case hot and/or cold extreme design conditions. Similarly, for thermostatically or proportionally controlled heaters, a 25% heater-capacity margin

Milestone	Heat-Load Margin (%)		
Program go-ahead	50		
PDR	45		
CDR	35		
Qualification	30		
FDR/Flight acceptance	25		

Table 15.11. Thermal Uncertainty Margins for Cryogenic Systems

may be used in lieu of the 11°C at the cold end, which translates into a duty cycle of no more than 80% at the minimum expected flight temperature under worst-case cold conditions.

Chapter 19 contains a discussion of how these margins relate to test temperatures of spacecraft components.

TMM Computer Codes

Solving the general heat-transfer equation is the objective of all thermal-analysis codes in the spacecraft industry. The general partial differential equation of heat conduction with source term for a stationary heterogeneous, anisotropic solid is

$$\rho C_p \frac{\partial T}{\partial t} = \nabla \cdot (K \cdot \nabla T) + Q(T, t) \text{ (Energy rate per unit volume)}, \quad (15.22)$$

where ρ is density (kg/m³), C_p is specific heat (J/kg.°C), ∇ is gradient operator (1/m), K is conductivity tensor (W/m.°C), T is temperature (°C), t is time (sec), and Q is the source term (W/m³). Equation (15.22) is a parabolic differential equation in which the Fourier conduction law ($q = -K \cdot \Delta T$) is used. Although temperature (T) is a scalar that can vary with position, i.e., T = (x, y, z, t), the heat flow depends on the temperature gradient in a particular direction and is therefore a vector quantity.

The intent of this section is not to derive heat-transfer equations or provide a detailed discussion of SINDA applications, but to serve as an overview of heat-transfer theory and available software for developing TMMs. For more information on these subjects, consult the textbooks by F. Kreith, ^{15.7} J. P. Holman, ^{15.8} and M. S. Carslaw and J. C. Jaeger, ^{15.9} which provide a good review of heat-transfer theory, and the SINDA manuals, ^{15.5,15.6} which contain more detail about building thermal models.

Most aerospace companies in the spacecraft industry use finite-difference numerical techniques to solve Eq. (15.22) for various heat-transfer problems with appropriate boundary conditions. For this purpose, these companies generally have either SINDA/1987, written by J. Gaski,^{15.5} or SINDA85/FLUINT,^{15.6} developed by Martin Marietta for NASA Johnson Space Center (JSC). Some companies still use the original version of SINDA, developed by Gaski in 1966, known as CINDA (Chrysler Improved Numerical Differencing Analyzer).

SINDA consists of a preprocessor and an execution library. The preprocessor reads a SINDA input file and, following certain rules, constructs a FORTRAN executable. The analyst selects subroutines from the SINDA library to obtain temperatures. SINDA allows the user to include the necessary FORTRAN logic to solve a specific heat-transfer problem. FORTRAN code can be added into any of the SINDA operation blocks. The Gaski SINDA has a one-dimensional incompressible-fluid thermal-analysis capability for evaluating pumped-fluid heat-transfer networks.

SINDA85 represents a significant evolution from the previous SINDA-type codes. It has fluid-network analysis capability for evaluating various types of thermal networks, including incompressible, compressible, two-phase flow, and others, and it also allows the analyst to build a thermal model from separate submodels. Both features are very powerful.

The Finite-Difference Method (FDM)

These codes determine the solution to a finite-difference model that approximates the physical object. The nodes or subvolumes are assumed to be isothermal, and physical properties are assumed to be constant within a node. Some heat-transfer books refer to finite-difference-node meshes as lumped-parameter representations. The nodes are interconnected by conduction and/or radiation. The governing partial differential equation is converted into a system of finite-difference equations by constructing an FDM mesh. The basis for this step is the Taylor series approximation. A three-dimensional Cartesian coordinate system is assumed for this discussion. From Fig. 15.36, which shows typical one- and twodimensional FDM meshes, the Taylor series about x_0 for T(x) is written for the one-dimensional mesh:

$$T(x_o + \Delta_x) = T(x_o) + \frac{\partial T}{\partial x} \Big|_{x = x_o}^{\bullet \Delta x} + \frac{\partial^2 T}{\partial x^2} \Big|_{x = x_o}^{\bullet \Delta x^2/2!} + \frac{\partial^3 T}{\partial x^3} \Big|_{x = x_o}^{\bullet \Delta x^3/3!} + \cdots$$
(15.23)

From this approximation, the first and second derivatives can be derived:

$$\frac{\partial T}{\partial x}\Big|_{x=x_o} = \frac{T(x_o + \Delta x) - T(x_o)}{\Delta x} + 0(\Delta x)$$

$$\frac{\partial^2 T}{\partial x^2}\Big|_{x=x_o} = \frac{\frac{T(x_o + \Delta x) - T(x_o)}{\Delta x} - \frac{T(x_o) - T(x_o - \Delta x)}{\Delta x}}{\Delta x} + 0(\Delta x^2),$$
(15.24)

where $O(\Delta x)$ and $O(\Delta x^2)$ are a means of expressing the order of the truncation error associated with the approximation. Equation (15.22) can be written for a



One-dimensional mesh uniform spacing



Two-dimensional mesh uniform spacing

Fig. 15.36. Finite-difference method.

heterogeneous, anisotropic solid, the conductivity of which in each of the three principal directions is a function of temperature:

$$\rho C_p \frac{\partial T}{\partial t} = \frac{\partial}{\partial x} \left[k_x(T) \frac{\partial T}{\partial x} \right] + \frac{\partial}{\partial y} \left[k_y(T) \frac{\partial T}{\partial y} \right] + \frac{\partial}{\partial z} \left[k_z(T) \frac{\partial T}{\partial z} \right] + Q(T, t).$$
(15.25)

The x-partial derivative, $\frac{\partial}{\partial x} \left[k_x(T) \frac{\partial T}{\partial x} \right]$, can be written as

$$\begin{bmatrix} k_{x}(\delta^{+}) \bullet \left(\frac{T_{n+1}(x + \Delta x, y, z, t) - T_{n}(x, y, z, t)}{\Delta x} \right)$$

$$-k_{x}(\delta^{-}) \bullet \left(\frac{T_{n}(x, y, z, t) - T_{n-1}(x - \Delta x, y, z, t)}{\Delta x} \right) \end{bmatrix} / \Delta x ,$$
(15.26)

where n is the node number about which the Taylor series is applied, and

$$\delta^{\pm} = \frac{[T_n(x, y, z, t) + T_j(x \pm \Delta x, y, z, t)]}{2}, \quad j = n + 1 \text{ or } n - 1$$
(15.27)

or $\delta = T_n(x, y, z, t)$, (15.28)

where j is the adjacent node, x, y, and z are the spatial coordinates of n, and t is time. Multiplying Eq. (15.26) by the volume $\Delta x \cdot A$, where $A = \Delta y \cdot \Delta z$, one obtains

$$A \cdot k_x(\delta^+) \cdot \frac{(T_{n+1} - T_n)}{\Delta x} - A \cdot k_x(\delta^-) \cdot \frac{(T_n - T_{n-1})}{\Delta x}, \qquad (15.29)$$

where T_i is shorthand for $T_i(x, y, z, t)$ and i = n.

Let the coefficient $A \cdot k_x(\delta^{\pm}) / \Delta x$ be defined as the parameter G, the conductance. Hence, Eq. (15.29) becomes

$$G_{n+1} \cdot (T_{n+1} - T_n) - G_{n,n-1} \cdot (T_n - T_{n-1}), \qquad (15.30)$$

where $G_{n+1,n} = \frac{k(\delta^+) \cdot A}{\Delta x}$ and $G_{(n,n-1)} = \frac{k(\delta^-) \cdot A}{\Delta x}$. Similar expressions can be written for the other terms, $\frac{\partial}{\partial y} \left[k_y(T) \frac{\partial T}{\partial y} \right]$ and $\frac{\partial}{\partial z} \left[k_z(T) \frac{\partial T}{\partial z} \right]$, in Eq. (15.25).

The conductance, G, is placed in the conduction block of SINDA. Hence, through the Taylor series approximation, a partial differential equation has been converted into a set of finite-difference equations that can now be solved numerically. The source term in Eq. (15.25), Q(t), is the means by which external and internal radiation, convection, and heat sources are added to the difference equation. The radiation term is typically written as

$$\sigma A \mathfrak{I}_{n,n+1} (T_n^4 - T_{n+1}^4), \qquad (15.31)$$

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where σ is the Stefan-Boltzmann constant, A is the surface area of the radiating surface, and $\Im_{n,n+1}$ represents the net radiation exchanged between two real surfaces, including all possible reflection paths. From the SINDA perspective $A\Im_{n,n+1}$ is just another conductance, except it is a radiation coefficient. In SINDA, radiation conductances are distinguished from convection and conduction coefficients by a minus sign; e.g., -G denotes a radiation conductor and G indicates a normal (conduction or convection) conductor.

In building a thermal model the analyst decides how many nodes to use, how to distribute them, and how to connect them by radiation, conduction, or convection. The resulting model network produces a system of finite-difference equations with either constant or variable coefficients. The number of equations to be solved depends on the number of nodes selected by the user in the thermal model minus any boundary nodes, which have a prescribed temperature history. For example, space is considered a boundary node and is set at 0 K ($-273^{\circ}C$).

To convert the finite-difference equations to a set of algebraic equations that are then solved within SINDA, you must approximate the time derivative, just as the spatial derivatives are approximated. The $\frac{\partial T}{\partial t}$ in Eq. (15.25) can be approximated as follows :

$$T(t^* + \Delta t) = T(t^*) + \theta \cdot \frac{\partial T}{\partial t}\Big|_{t^* + \Delta t} \cdot \Delta t + (1 - \theta) \cdot \frac{\partial T}{\partial t}\Big|_{t^*} \frac{\cdot \Delta t}{t^*}, \quad (15.32)$$

where θ is a variable-weighted implicit factor. Multiplying Eq. (15.25) by the volume $(\Delta x \cdot A)$, one observes that the coefficient for $\frac{\partial T}{\partial t}$ becomes

$$C_n = \rho \cdot C_p \cdot \Delta x \cdot A , \qquad (15.33)$$

where C_n denotes the capacitance of node *n* and *A* is the cross-sectional area $\Delta y \cdot \Delta z$. Combining Eqs. (15.30), (15.31), and (15.32), one finds that Eq. (15.26) becomes

$$C_{n} \frac{[T_{n}(t + \Delta t) - T_{n}(t)]}{\Delta t}$$

$$= \theta \cdot \left[\sum_{j=1}^{N} G_{jn}(T_{j} - T_{n}) + \sigma \cdot \sum_{j=1}^{N} \Im_{jn} A_{n}(T_{j}^{4} - T_{n}^{4}) + Q_{n}(T_{n}, t) \right]_{t^{*} + \Delta t}$$

$$+ (1 - \theta) \cdot \left[\sum_{j=1}^{N} G_{jn}(T_{j} - T_{n}) + \sigma \cdot \sum_{j=1}^{N} \Im_{jn} A_{n}(T_{j}^{4} - T_{n}^{4}) + Q_{n}(T_{n}, t) \right]_{t^{*}} .$$
(15.34)

This equation contains the parameter θ , which can be adjusted along with the FDM mesh size and time step to yield various finite-difference approximations with different local truncation errors. The values $\theta = 0$, 1/2, and 1 yield the forward-explicit, Crank-Nicolson, and backward-implicit approximations.

Selecting a particular FDM mesh scheme and evaluating the coefficients in Eq. (15.34) yields a system of *n* algebraic equations where *n* is the number of finitedifference nodes. Note that *n* does not include boundary nodes. If $\theta = 0$, each equation is explicit and has only one unknown temperature, T_n . If $\theta > 0$, a system of algebraic equations exists and must be solved by either iterative techniques, matrix-inversion schemes, or decomposition procedures. Typically the system of equations is written as

$$T_{\text{new}} = [A] \cdot T_{\text{old}}, \tag{15.35}$$

where [A] is an $n \times n$ matrix and T is an $n \times 1$ or column matrix. For thermal models of ten or more finite-difference nodes, [A] is typically a sparse matrix because each node is normally connected to a small subset of the total number of nodes in the model. For most heat-transfer problems, [A] is not banded because of radiation interchange between the nodes. Consequently the efficient solvers for tridiagonal matrices are not generally useful.

FDM Errors

Three types of errors can occur with the application of the FDM to heat-transfer problems. The first is the truncation error, which is the difference between the differential equation and the approximating difference equations.^{15.10–15.12} This type of error can be illustrated for the one-dimensional heat-transfer equation with constant conductivity. Let

$$F_{\text{pde}}(T) = \left(\frac{\partial T}{\partial t} - k \cdot \frac{\partial^2 T}{\partial x^2}\right)$$
 (partial differential equation) (15.36)

and

$$F_{\rm fd}(T_i) = \left(\frac{T_{t^* + \Delta t, x^*} - T_{t^*, x^*}}{\Delta t}\right)$$

$$-k\left(\frac{T_{x^* + \Delta x, t^*} - 2T_{x^*, t^*}T_{x^* - \Delta x, t^*}}{\Delta x^2}\right)$$
(finite-difference equation), (15.37)

then $[F_{fd}(T_i) - F_{pde}(T)]$ represents the truncation error at each node. T_i refers to the temperature at three successive nodes, $x^* + \Delta x$, x^* , and $x^* - \Delta x$, and t^* designates a discrete time. The temperature T in the analytical solution is a continuous function. The truncation error is determined from the finite-difference node spacing (mesh size) and the size of the time step. As the number of finite-difference nodes is increased and the time step decreased, the error associated with the Taylor series approximation (truncation) decreases and approaches zero in the limit. In this case the truncation errors approach zero and the difference equation is said to be consistent with the partial differential equation. However, as the number of nodes in the network expands, the corresponding number of difference equations to be solved increases. This, in turn, increases execution time.

From the viewpoint of algebraic simplicity, an analyst prefers the coarsest network possible. The best thermal model is a compromise between node size and

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computational cost. No specific rules are available for selecting the optimal network size; one must rely on insight and experience. One way to judge the truncation errors introduced by too coarse a network is to estimate the truncation error as the calculation proceeds.

The second type of error is related to the stability of the numerical solution. If the effect of errors tends to diminish as the numerical solution progresses, then the solution is stable and converges. However, if the errors tend to grow with time, then the solution becomes unstable and diverges.

The third type of error is the computer rounding error made during numerical calculations. This is the difference between the exact numerical answer and the actual numerical answer (i.e., the truncated numerical answer generated by the computer). Rounding error is a significant problem with 16-bit computers, somewhat of a problem with 32-bit machines, and generally not a problem with 64-bit computers. Hence, the numerical temperature, T_{num} , is given as

$$T_{\rm num} = T_{\rm ex} + (T_{\rm num} - T_{\rm exn}) + (T_{\rm exn} - T_{\rm ex}), \qquad (15.38)$$

where T_{ex} is the analytical solution and T_{exn} is the exact numerical solution. The discrete error is the combination of the truncation and stability errors. As discussed previously, these errors are directly coupled to the mesh size and time step assumed by the analyst. The truncation error for a uniform mesh (Fig. 15.36) is typically $O(\Delta x^2)$, i.e., second-order. However, for a nonuniform mesh (see Fig. 15.37), the truncation error becomes $O(\Delta x)$, i.e., first-order. Hence, a nonuniform FDM mesh reduces the order of the truncation error and decreases the accuracy of the approximation. Most spacecraft thermal models are not uniform; however, if sufficient thermal nodes are used, the numerical answers will be reasonably accurate.



Fig. 15.37. FDM two-dimensional nonuniform mesh.

Forward-Differencing Approach to Heat-Transfer Equations

The forward-differencing expression is obtained from Eq. (15.32) by setting $\theta = 0$; hence,

$$T(t^* + \Delta t) = T(t^*) + \frac{\partial T}{\partial t} \Big|_{t^*}^{\bullet \Delta t} , \qquad (15.39)$$

and the expression is shown in Fig. 15.38. This method requires that the calculation of T_i at $t^* + \Delta t$ be based on values of T_i that are known at t^* , the previous time. This is illustrated by setting $\theta = 0$ in Eq. (15.34). The forward-differencing assumption is explicit, and the solution can be unstable if the time step, Δt , is too large. The criteria for stability are determined by calculating the minimum value

$$\tau_n = \frac{C_n}{\sum_j G_{nj}} \tag{15.40}$$

for each finite-difference node, where τ is the stability factor and *j* is the sum of all conductors connecting other nodes to *n* by conduction or radiation. (Convection would be included in a prelaunch environment.) The thermal capacitance of the node is C_n , and the values of G_{nj} are the conductance values between adjacent nodes. If radiation occurs between two nodes, the value is linearized to obtain

$$G_{nj} = \sigma \Im_{nj} A_j (T_n^2 - T_j^2) (T_n + T_j) . \qquad (15.41)$$

In SINDA, τ is called CSGMIN. CSGMIN represents the smallest time constant in the thermal network at each time step. It can change from time step to time step. CSGMIN includes the effect of boundary conditions if the node that has the smallest τ is connected to any boundary nodes. The solution process will remain stable if the time step, Δt , is always less than CSGMIN. In SINDA $\Delta t = 0.95^*$ CSGMIN/ CSGFAC is always used, with CSGFAC defaulted to 1.0.



Fig. 15.38. Explicit and implicit derivatives.

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The forward-differencing equation has one unknown node temperature at $t^* + \Delta t$, with all the other temperatures known at t^* (see Eq. [15.34] with $\theta = 0$). Any radiation terms are approximated by Eq. (15.41). Although this explicit equation is simple to solve, the time step, Δt , is limited by the stability criteria for the node with the smallest time constant. Hence, in using this technique the analyst is trading simplicity for potentially many small time steps, a situation that can cause excessive execution time and completely consume the CPU on a local workstation. In applying the forward-differencing equations, the analyst does not have to specify the convergence criteria and a time step, since these can be conveniently computed from the specified thermal data.

Backward-Differencing Approach to Heat-Transfer Equations

Another technique used to solve heat-transfer equations is backward differencing. In this case the heat balance is written in terms of the unknown temperatures at $t^* + \Delta t$,

$$T(t^* + \Delta t) = T(t^*) + \frac{\partial T}{\partial t} \Big|_{t^* + \Delta t}^{\bullet \Delta t} .$$
(15.42)

This equation is obtained by setting $\theta = 1$ in Eq. (15.32). Figure 15.38 illustrates the backward slope $\frac{\partial T}{\partial t}$. This approach yields a system of *n* equations, where *n* is the total number of finite-difference nodes whose temperatures are calculated at each time step. Boundary nodes are excluded. This formulation is called implicit. The minimum time constant CSGMIN is still calculated in SINDA for implicit methods. Since implicit methods are unconditionally stable, the time step Δt can exceed CSGMIN. However, if the time step selected is too large, although stable, the truncation error can become significant. When using an implicit method, the analyst must specify the time step. The user should always compare the specific time step to CSGMIN. If the selected time step is five to ten times CSGMIN, it is probably too large. Obviously, this judgment depends on the problem being solved.

Iterative schemes are typically used to solve systems of equations. Such techniques require a convergence criterion. For transient problems, the SINDA constants DRLXCA and ARLXCA must be specified to use the implicit schemes. Two constants are required because SINDA allows both diffusion and arithmetic nodes. Diffusion nodes have mass, whereas arithmetic nodes do not. DRLXCA is the convergence criterion for diffusion nodes, and ARLXCA is the convergence criterion for arithmetic nodes.

The advantage of backward differencing rests with the ability to vary the time step. During periods of rapidly varying boundary conditions, the time step can be reduced. Similarly, during periods of slowly changing boundary conditions, the time step can be increased. Typically, implicit numerical schemes are faster than the explicit-forward method because of the large time steps allowed. However, the larger the time step, the more iterations required to achieve a solution. Each iteration is essentially equivalent to a time step. Hence, the actual implicit time step is approximately the specified Δt divided by the number of iterations required to achieve a solution. The user needs to compare this modified time step to CSGMIN

to verify that the specified time step is providing the increased computational speed expected over the explicit method. For some problems the implicit scheme may not be any faster than the explicit method.

The following stability criteria are associated with Eq. (15.34):

 $\begin{array}{ll} \theta = 0, & \Delta t < \text{CSGMIN.} \\ \theta < 1/2, & \text{conditionally stable, i.e., CSGMIN} < \Delta t < \text{CSGMIN} (1+\Delta) \\ & \text{where } \Delta \rightarrow \infty \text{ as } \theta \rightarrow 1/2. \\ 1/2 \le \theta \le 1, \text{ unconditionally stable for any } \Delta t. \end{array}$ (15.43)

All the SINDA codes allow three types of nodes: diffusion (with mass), arithmetic (no mass), and boundary (specified temperature). These definitions are particularly useful when solving equations whose time constants vary by several orders of magnitude or higher values. If arithmetic nodes were not allowed, the algebraic system of equations would be very stiff. This class of problem (stiff equations) can only be solved with implicit techniques. However, when CSGMIN is very small for some diffusion nodes, they can be converted to arithmetic nodes. This helps make the equations less stiff and improves the computational efficiency without sacrificing accuracy. The temperature of an arithmetic node is obtained by noting that the total heat flow into the node is zero. An example of the use of arithmetic nodes is the inclusion of MLI blankets in a design; typically, they are modeled as arithmetic nodes because they are light and respond instantaneously to the environment. Another example is the analyst's placement of arithmetic nodes at the interface of two surfaces to obtain the interface temperature.

Limiting ΔV and Δt

In summary, the finite-difference approximations to the partial differential heattransfer equation discussed in the preceding sections require that continuous variables be quantized. Spatial variables are quantized as nodes and connected by conductors, while time is divided into discrete steps, denoted by Δt . For finitedifference nodes of size $\Delta V = \Delta x \cdot \Delta y \cdot \Delta z$, the time step and spatial dimensions are related in SINDA through the CSGMIN definition.

Since the finite-difference solution approaches the exact solution as ΔV and Δt approach zero, the logical question is, what limits ΔV or Δt ? The answer is cost and computer storage (model size). Together these constraints limit ΔV to a non-zero minimum value. A small ΔV means a large number of nodes and conductors, and the computer memory must contain enough space to hold all parameters (capacitance, temperature, conductance, etc.) associated with the model. A large model takes a long time to execute on a computer and many engineering staffhours to develop. The analyst has to use engineering judgment to decide how much detail is sufficient to determine the thermal response of the physical system being analyzed.

The time step is chosen consistent with CSGMIN for the forward-explicit method. An excessively small value of CSGMIN can greatly increase the computation time. The implicit methods allow time steps greater than CSGMIN. In these cases the analyst needs to determine the largest acceptable Δt that will not impact the accuracy of the calculated temperatures. For each thermal problem the analyst is faced with the task of developing a model and selecting a technique for solving it that will yield the most accurate, stable temperatures for the least cost.

Other Approaches to Finite-Difference Equations

Numerous other approaches to formulating and solving finite-difference equations are available. The Gaski SINDA/1987 has several transient and steady-state solutions. This approach provides the analyst with the flexibility to select a solution subroutine that will work. As noted previously, the forward-explicit method is limited to time steps less than CSGMIN. However, two other explicit schemes, one by Saul'yev and the other by Dufort-Frankel, allow time steps greater than CSGMIN because these methods are unconditionally stable. The Saul'yev alternating-direction explicit method is implemented in the Gaski SINDA/1987 as the execution subroutine SNADE, and the modified Dufort-Frankel technique is called SNDUFR in the Gaski SINDA/1987.^{15.4} The original Dufort-Frankel method used the following approximations: in time.

$$\frac{\partial T}{\partial t} = \frac{1}{2} (T_n \big|_{t^* + \Delta t} - T_n \big|_{t^* - \Delta t}), \qquad (15.44)$$

and in space,

$$T_{n}|t^{*} = \frac{1}{2} \left(T_{n+1} |t^{*} - T_{n-1}|_{t^{*}} \right).$$
(15.45)

These approximations produced a finite-difference technique whose truncation error was $0(\Delta t^2, \Delta x^2)$ accurate as compared to the forward-differencing scheme, which is $0(\Delta t + \Delta x^2)$ accurate. The central-difference approximation in Eq. (15.44), which is $0(\Delta t^2)$ accurate, causes a term

$$\frac{\partial^2 T}{\partial t^2} \cdot \frac{\Delta t^2}{\Delta x^2} \tag{15.46}$$

to appear in the truncation error. Depending on the relationship between the spacing of the nodes and the time step, the resulting equation can be hyperbolic instead of parabolic. The old SINDA codes used the original Dufort-Frankel approximation, which generated temperatures that were off by 5 to 10°C. This was a direct result of the term

$$\frac{\partial^2 T}{\partial t^2} \cdot \frac{\Delta t^2}{\Delta x^2} \ . \tag{15.47}$$

By using the Euler approximation in time, which is normally used, we have

$$\frac{\partial T}{\partial 2} = T_n \big|_{t^* + \Delta t} - T_n \big|_{t^*} \quad . \tag{15.48}$$

The modified Dufort-Frankel method uses Eqs. (15.45) and (15.48), and it produces accurate solutions for time steps greater than CSGMIN. The truncation

error is $0(\Delta t, \Delta x^2)$. The $\frac{\partial^2 T}{\partial t^2} \cdot \frac{\Delta t^2}{\Delta x^2}$ term is higher-order and is not a factor with this

approximation. One final note; the Richardson explicit method used the time approximation in Eq. (15.44) with the standard approximation in the spatial derivative. This method was unconditionally unstable. The application of Eq. (15.44) as an approximation in time has generally not been successful.

The most common implicit scheme is the Crank-Nicolson (CN) method. Its approximation is obtained from Eq. (15.34) with $\theta = 1/2$. The Crank-Nicolson technique is simply an application of the trapezoid rule. The method has a truncation error $0(\Delta t^2, \Delta x^2)$. The basic CN equation for one dimension can be derived from adding:

$$\frac{T_{t^* + \Delta t/2} - T_{t^*}}{\Delta t/2} = \left[\frac{T_{n+1} - 2T_n + T_{n-1}}{(\Delta x^2)}\right]_{t^*} \text{ Explicit forward,}$$
(15.49)

and

$$\frac{T_{t^* + \Delta t} - T_{t^* + \Delta t/2}}{\Delta t/2} = \left[\frac{T_{n+1} - 2T_n + T_{n-1}}{(\Delta x)^2}\right]_{t^* + \Delta t}$$
 Implicit backward, (15.50)

to yield

$$T_{t^{*}+\Delta t} - T_{t^{*}} = \frac{\Delta t}{2\Delta x^{2}} \cdot \left[\left[T_{n+1} - 2T_{n} + T_{n-1} \right]_{t^{*}+\Delta t} + \left[T_{n+1} - 2T_{n} + T_{n-1} \right]_{t^{*}} \right].$$
(15.51)

This approximation is not limited to one dimension, which was selected to illustrate the method. The CN approximation can also be obtained by simply adding the classic explicit and implicit methods. Centering the approximation about $t^* + \Delta t/2$ accounts for the increased accuracy in time over the classic-explicit and backward-implicit methods, which are centered at t^* and $t^* + \Delta t$, respectively. FWD-BCK in the Gaski SINDA/1987 uses the CN method. The SNTSM method in SINDA/1987^{15.5} is a Taylor series method with the weighted average approximation [Eq. (15.33)] and automatic time-step selection. Table 15.12 lists the various transient and steady-state execution subroutines used in SINDA/1987. SINDA/ FLUINT uses only four execution subroutines.

Several steady-state subroutines are used in SINDA/1987. Table 15.12 contains the available subroutines. For most reasonably sized thermal models, SCROUT, which is based on the Choleski method, is the best selection for steady-state solutions. The steady-state convergence criterion for the iterative methods is based on first meeting a global-temperature relaxation error, next a system-energy balance criterion, and finally a nodal-energy balance error. The user specifies the minimum acceptable criteria. Typically, a $T_{i+1} - T_i$ difference of less than .005°C (if °C is the unit being used), a system-energy balance error of 1% and a nodal energy balance error of .5% are used. Note *i* is the iteration count. The analyst can adjust the error

Subroutine		Description		
Transient				
SNFRDL		Explicit forward differencing		
FWDBKL	(FWDBCK, FDBKCD)	Quadratic or linear equation, implicit finite differencing, successive point iteration		
SNADE		Alternating direction explicit finite differencing		
SNDUFR		Modified Dufort-Frankel explicit finite differencing		
ATSDUF		Automatic time-step selection, like SNDUFR otherwise		
ATSFBK		Automatic time-step selection, like FWDBCK otherwise		
SNTSM	(SNTSM1) (SNTSM3) (SNTSM4)	Taylor series with weighted average, automatic time-step selection		
Steady-state				
STDSTL		Quadratic or linear equation with successive point iteration		
SNHOS	(SNHOSD)	Taylor series, explicit		
SNSOSS		Taylor series, implicit		
SNDSNR		Newton-Rhapson with Gauss-Jordan reduction		
SCROUT		Matrix decomposition, elimination method		
SNSOR	(SNSOR1)	Successive over-relaxation		

Table 15.12. Typical SINDA Execution Subroutines for FDM

constants in SINDA by specifying values for specific user constants. Three steadystate criteria are used because temperature relaxation can signal false convergence for some problems; thus, by specifying system- and nodal-energy balance criteria, convergence to the correct answer is assured even if the temperature relaxation is misleading.

The Finite-Element Method (FEM)

The FEM provides a second approach to the numerical solution of heat-transfer problems.^{15,13} The FEM-mesh schemes are the real strength of this technique. Each finite-element model normally has hundreds of elements. Two approaches are used to develop a solution with the FEM: the methods of weighted residuals (MWR) and the Ritz variational method. The most widely used finite-element technique is the Galerkin approach, which is one of four MWRs. The FEM seeks an explicit expression for the temperatures, \tilde{T} , in terms of known functions that, on average, satisfy the governing differential equations and the boundary conditions

exactly on an element. The \tilde{T} is the finite-element approximation to the actual temperature, T. The form used for \tilde{T} is

$$\tilde{T}(t, a_i) = \sum_{i=0}^{N} a_i \emptyset_i(t) , \qquad (15.52)$$

where the a_i are referred to as degrees of freedom (DOF), N is the total number of DOF, and the $\mathcal{O}_i(t)$ are called by various authors as trial, basis, shape, interpolation, or coordinate functions. Typically the \mathcal{O}_i are assumed to be powers of x, sine, cosine, etc., on the element. This approach parallels the analytical technique of finding a function or set of functions that solves the differential equation and also satisfies the prescribed boundary conditions. An example of a basis function is

$$\begin{aligned} & \varnothing_i = 0, t = a \\ & \varnothing_i = \frac{a-t}{a-b}, a < t < b \\ & \varnothing_i = 1, t = b \end{aligned}$$
 (15.53)

Hence \emptyset_i is a linear function whose value varies from 0 to 1. A bar element has an element node at each end, and a triangle element has a node at each corner (Fig. 15.39). The a_i are specified at each element node. For a thermal problem, a_i equals T_i , where the T_i are the element-node temperatures. The essence of the method is to obtain a set of algebraic equations for the element-node temperatures T_i that form a column vector called \tilde{T} . The temperatures between element nodes are found by applying the basis function between those nodes. For example Eq. (15.53) for \emptyset_i would be used to find the temperature between element nodes a and b. For a reasonable FEM mesh, \tilde{T} should approach T, the exact temperature solution. Basis functions used in Eq. (15.52) can be linear [Eq. (15.53)], quadratic, cubic, or quartic. Examples of element shapes are:

- bar (one-dimensional)
- triangular, rectangular, quadrilateral (two-dimensional)
- hexahedral, pentahedral, tetrahedral (three-dimensional)

To improve the accuracy of the finite-element method, either a smaller mesh (more elements) is used or higher-order basis functions (increased DOF) on the elements are needed. This FEM-mesh technique does not provide the analyst with an error estimate like finite-difference, which is based on Taylor series expansions. Hence, the analyst either repeats the problem with a smaller mesh or, based on experience, develops an FEM mesh that appears to provide an acceptable solution. This process leads to detailed-mesh structures, since the associated errors are not easily calculated and the analyst does not want to solve the same problem twice for two different mesh sizes to establish a convergence criterion.

Many finite-element codes that can be used for thermal analysis are available in the aerospace industry. These include NASTRAN, ADINAT, ABAQUS, ANSYS, COSMOS, and TOPAZ. All were developed to perform structural analysis. Every major aerospace company has at least one.

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Fig. 15.39. FEM-mesh subdivision.

Finite-Difference vs. Finite-Element

In summary, either method, finite-difference or finite-element, can be used to solve heat-transfer problems. The FDM is based on Taylor series approximations to develop the algebraic equations that are solved numerically to find a set of temperatures. Each finite-difference node is located at the center of mass and is assumed isothermal throughout the volume occupied. The error associated with the calculation can be estimated. The finite-element method is based on using elements that are one-, two-, or three-dimensional, depending on the problem being solved. Combinations of elements can also be used. Each element has element nodes at its corners. Parameter values, for example temperatures, are usually specified or calculated at element nodes. Variations within the element are calculated by using interpolation (basis) functions within the element. Hence the properties and temperature can vary across the element. The Galerkin MWR is normally used to develop the algebraic equations that determine the element-node temperatures.

The FDM is excellent for building spacecraft-system models. It is compatible with the basic surface primitives (e.g., cones, cylinders, spheres) used to describe spacecraft surfaces in the radiation codes. Heat-transfer problems that are primarily driven by radiation can be easily solved with this method. The FDM mesh does not have to be uniform; however, the truncation error decreases from $0(\Delta x^2)$

to $0(\Delta x)$. The accuracy of the method is judged by the truncation error from the Taylor series expansions. This assumes that the analyst is using an inherently stable integration scheme and that rounding error is small, which may not always be the case. The truncation error can be reduced with a smaller FDM mesh and smaller time steps.

Comparing the accuracy of finite-difference and finite-element methods is very difficult unless an exact solution is available. This is never the case for nonlinear problems, which is typical for radiation-dominated thermal analyses; the dominant mode of heat transfer for spacecraft is radiation. Thermal models developed with this method can have three types of isothermal nodes: diffusion, arithmetic, and boundary. The arithmetic node, which is a zero-mass node, can be used to avoid stiff equations that always have a large spread in the time constants. In many spacecraft models one thermal node represents an electronics box. This is easily accommodated by finite-difference schemes, but not by finite-element schemes. To overcome the mesh-generation problem for finite-difference techniques, several aerospace companies have used FEM-mesh generators like PATRAN to build a mesh and then convert it to finite-difference for the finite-difference analysis codes like SINDA. The resulting temperatures are then returned to the finite-element mesh-generation code for display. Without a finite-difference pre- and postprocessor, FDM has a serious disadvantage in building thermal models and displaying the results.

The FEM is extensively used in structural analysis. The method is excellent for solving thermal/stress problems. Normally the structural model requires significantly more detail than the equivalent thermal model. Hence the structural characteristics will typically drive the size of the thermal analysis for a combined thermal-stress analysis. Applications of finite-element techniques to the thermal analysis of circuit boards, traveling wave tubes, and rocket nozzles are common. The real strengths of finite-element techniques are the mesh-generation schemes. These techniques can easily handle irregular surface shapes and the interface between two different mesh schemes.

Because of the longtime application of FEMs to structural problems, several excellent commercial mesh-generation packages are available. These include PATRAN from MSC, GEOMOD (IDEAS-SDRC), and ANVIL from MCS. The pre- and postprocessing capabilities of these codes are excellent. Currently, some commercial FDM-mesh-generation/postprocessing packages of comparable capability are also available.

FEM-mesh-generating schemes are still used in most thermal software packages to develop and postprocess finite-difference temperature results. The finite-element codes have the equivalent to diffusion (nodes with mass) and boundary nodes. They do not allow arithmetic (zero-mass) nodes. Because of this the resulting algebraic equations can be very stiff and lead to excessive computational costs. Also, the finite-element codes cannot use just one node for an electronicsbox simulation, as finite-difference codes can. For typical thermal analysis, finiteelement models will always be larger than necessary. This condition is driven by the requirement that each element face must share a complete interface with another element, and it is also driven by the lack of information about the error associated with the calculations. Hence, the analyst tends to construct smaller meshes than may be necessary. Typically, curved surfaces like cones and cylinders require far more finite-element surfaces to describe the shape than are needed for finite-difference codes. One node of 360 deg may be all the analyst really needs. Such a representation is not possible with finite-element codes. The Monte Carlo radiation codes recognize and use the actual surface description for a cone and cylinder. This provides radiation-interchange factors that are correct. Approximation of these surfaces by flat surfaces or polynomial fits can impact the accuracy of the interchange factors and unnecessarily increase the cost and complexity of obtaining them. The combination of increased surface numbers and resulting interchange factors can significantly impact the execution time of the thermal model. This increased detail forced by the method of solution will normally not add any additional real information.

Most finite-difference codes, such as SINDA, allow the analyst to include extensive user logic (e.g., FORTRAN subroutines) in the thermal model. Finite-element codes, like NASTRAN, are far more restrictive in this area.

Implicit-solution schemes are best for transient finite-element analysis. This is mainly driven by the fact that the algebraic equations being solved can be very stiff. Many finite-element solution schemes are most efficient with banded matrices; however, with radiation the matrices are not conveniently banded.

The only way to check the accuracy of the finite-element codes is to run the problem again with a smaller mesh size or high-order elements. This is obviously not an inexpensive procedure for determining the error. Typically, error calculations are not made within the finite-element codes.

An approach used to construct a spacecraft thermal model is shown in Fig. 15.40. The various codes used to complete this process at The Aerospace Corporation are listed in Table 15.13. This software was first developed between 1981 and 1985. Whereas in the 1960s and 1970s most analysts developed thermal models by hand with the time-consuming use of punch-cards, in the 1980s, with the development of minicomputers (e.g., the DEC VAX) and workstations (e.g., those from companies such as Sun Microsystems, Inc., and Hewlett-Packard Co.), the time required to build a thermal model could be greatly reduced through the interactive use of software codes that aid the analyst in model construction, debugging, and execution.

In the 1990s several integrated thermal-analysis programs were developed that were similar to the Aerospace software. These programs allow the analyst to generate complete TMMs and GMMs, execute them, and display the results in a user-friendly, menu-driven environment on a workstation or PC. These newer systems typically have a model builder, an orbital display capability, a radiation analyzer, a thermal analyzer, and postprocessing software to display temperature distributions and temperature heat-flux plots. Most have a limited capability to read in models built in other CAD systems. For example, a NASTRAN model can be read into IDEAS. The NASTRAN model could have been built by PATRAN. Another example is a TRASYS geometric model that can be read into TSS. The thermal analyzers are mostly finite-difference (e.g., SINDA). The radiation codes are based either on the Monte Carlo technique or the gray-diffuse assumption (these are discussed in the next section).



Fig. 15.40. Thermal-analysis system flow used by the integrated thermal analysis platform (ITAP) at The Aerospace Corporation.

Component	Function/Type of Software
TIDES	Converts surface geometric models between radiation codes
SURTRAN	Constructs, verifies surface geometric radiation model
SOAP	Verifies surface model orientation for specified orbit parameters
SOTRAN	Constructs solid-conduction model
SOSURF	Converts solid model to surface model
ORBITDEF	Defines orbit parameters
ATRIUM	Calculates radiation interchange factors and absorbed fluxes
SINTAB	Formulates ATRIUM output to SINDA tables
SINDA	Finite-difference thermal-analysis code
THERMODATA	Database containing thermophysical properties
CASCADE	Graphical postprocessing for temperatures
EXCEL	Spreadsheet program

Table 15.13. Software Used by ITAP at The Aerospace Corporation

The commercially available thermal-analysis software packages are the following.

- Thermal Synthesizer System (TSS) by SPACEDESIGN under license to NASA/JSC
- Thermal Desktop (TD) by Cullimore and Ring Technologies; TD uses AUTOCAD
- THERMICA by Network Analysis Inc. under license to ASTRIUM
- FEMAP/SINDAG Modeling System by Network Analysis Inc.
- IDEAS TMG Thermal Modeling System by MAYA; FEMAP can also be used instead of IDEAS
- ITAS by Analytix Corporation
- Thermal Analysis System by Harvard Thermal

The model builders for these systems are based on either surface primitives (shapes) or elements, e.g., patches. The commercially available shape-based systems are TSS, THERMICA, and ITAS. TD can be shape or element based. The others are element-based systems. ITAP, used by The Aerospace Corporation, is based on shapes. The thrust of these systems is to facilitate the analyst's ability to build thermal models in a fast, efficient manner. The goal is to let the computer perform as many of the calculations as possible, so that the analyst can think more about the physics of the problem. The workstations and PCs presently available allow the incorporation of all aspects of the thermal-model building process (Fig. 15.41). This relieves the user of the need to interface with different computer systems to build, analyze, and postprocess the results of a thermal model. These workstations and PCs provide the analyst with enough computer power to support graphics, analysis packages, and window-display systems, all integrated into powerful analysis platforms. Because platforms are constantly being improved, the software systems developed for specific workstation applications need to be portable to reduce redevelopment costs because of hardware obsolescence.

In addition to these commercially available thermal-analysis systems, several useful commercial codes can assist an analyst in either building a thermal model or analyzing results:

- SINAPS (Cullimore and Ring Technologies, Inc.)
- Pre-SINDA (VERIDIAN)
- SSPTA (Swales and Associates, Inc.)

Radiation Analysis Codes

Radiation interchange factors between surfaces and energy absorbed on surfaces of spacecraft are calculated by radiation codes. The book *Thermal Radiation Heat Transfer* by Siegel and Howell^{15.14} provides an excellent summary of assumptions made by these radiation codes and the techniques that they use. The codes use either the gray-diffuse assumption or the Monte Carlo approach.

The Gray-Diffuse Assumption

Codes like TRASYS,^{15.3} an industry standard for many years, assume a gray-diffuse surface to calculate the emission and absorption of radiation on a surface. This assumption implies: 1. The temperature is uniform over that surface.

2. The emittance, absorbance, and transmittance of a surface are independent of wavelength and direction.

3. All energy from a surface is emitted and reflected diffusely.

4. The incident and reflected energy flux is uniform over each surface.

With these assumptions a set of blackbody geometric configuration factors or view factors are calculated. A blackbody is a surface that completely absorbs all incident radiation of all wavelengths and from all directions. The view factor, F_{ij} , is simply the fraction of energy leaving black surface *i* that arrives at black surface *j*. The view factor can be calculated from a double integral sum, the unit-sphere method, or the contour integration method. TRASYS uses one of the first two techniques.

The energy per unit time transferred from black element dA_1 to dA_2 over the distance S is given as:

$$dQ_{dA1 \to dA2} = F_{12} \sigma T_1^4 dA_1, \tag{15.54}$$

where

$$F_{12} = \frac{\cos G_1 \cos G_2}{\Pi S^2} dA_2 \tag{15.55}$$

is the configuration or view factor (Fig. 15.41). This assumes the blackbody total intensity, i_b , is related to the blackbody total hemispherical emissive power, e_b , by the equation

$$i_{b_1} = \frac{e_{b_1}}{\Pi} = \frac{\sigma T_1^4}{\Pi},\tag{15.56}$$

where σ is the Stefan-Boltzmann constant, π is 3.14159265, and T is the temperature of the surface.

Once the values of F_{ij} are known, the values of B_{ij} can be calculated. B_{ij} is the fraction of energy emitted by black surface *i* that is absorbed by real surface *j*, including all intervening reflections from other real surfaces, including *i*. For the assumptions in TRASYS the real surface is assumed to be gray, a diffuse emitter and a diffuse reflector. A gray surface has radiative properties that do not vary with wavelength. A gray surface emits and absorbs a fraction of what a black surface does. For these assumptions, the Gebhart Method^{15.14} can be used to calculate the B_{ij} values from the F_{ij} values and the specified surface emittances.

TRASYS can also calculate energy absorbed by a surface. To perform this calculation TRASYS determines the shadow factors, i.e., how one surface shadows another. This information is obviously dependent on the direction of the incident solar energy. The accuracy of the shadow factor depends on how a surface is subdivided into a mesh. The shadow factor requires each small mesh element to be either illuminated or shadowed.



Fig. 15.41. Radiative interchange between two black differential area elements.

To calculate absorbed energy on a surface, whether shadowed or not, the code must be able to specify the orientation of a surface in an orbit with respect to the sun and Earth (these are the external-environment sources of heat for a spacecraft surface). The parameters to specify an orbit are discussed in Chapter 2. After defining the orbit, the code specifies a set of reference axes, e.g., solar-inertial, planet-centered (one axis always pointed at the center of Earth), etc. The coordinate system for each surface is then specified in relation to the spacecraft reference axes. This reference axis is then oriented with respect to the orbital reference axes. Hence, the orientation of any surface with respect to the sun or Earth can be specified and its illumination calculated.

TRASYS^{15.3} was developed in the 1960s by Lockheed Martin and is still used by many aerospace companies. A limited ray-tracing capability was added to TRASYS in the early 1980s.

The Monte Carlo Approach

The Monte Carlo method was first used in the aerospace industry in the 1970s. In the late 1970s NEVADA,^{15.4} a Monte Carlo code, became commercially available. Eventually most aerospace companies licensed and used this stand-alone software despite the limitations of its very basic two-dimensional graphics display package. Most of the commercially available thermal-analysis systems that proliferated in the late 1990s included a Monte Carlo radiation code. TSS, TD, THER-MICA, IDEAS, and ITAS provide examples. NEVADA is used with the SINDA/G system.

Most of the Monte Carlo codes use surface primitives, i.e., they are shape based. Those thermal-analysis systems that use finite-element mesh-generating schemes (e.g., IDEAS/MAYA) cannot construct surface primitives with one element. For example, several elements are needed to create a cylinder or cone. This requirement can add unnecessary surfaces to a geometric model. As noted previously,
those extra surfaces can impact the execution time of a Monte Carlo code if many curved surfaces have to be constructed from smaller elements.

With the tendency to build large detailed thermal models of subsystems and spacecraft, the geometric models can become very large. This can greatly increase the execution time of the Monte Carlo software.

The amount of time spent in finding a ray/surface interaction can be excessive in Monte Carlo codes. Techniques for speeding up the ray tracing within the code by reducing the number of time-consuming ray/surface intersection calculations have been developed. One of these techniques, the OCTREE method, subdivides a three-dimensional surface geometric model into cells or compartments. Typically only a small number of surfaces are in each cell. Some surfaces may be split between cells. When a ray is emitted from a surface in one cell, the code checks to determine if any surface in that cell is hit. If not, the code moves to the next or adjacent cell in the direction the ray is moving. The method greatly reduces the search time to find the surface the emitted ray intersects or hits. The key to this method is breaking the surface geometric model into a reasonable number of three-dimensional cells. If too many cells are used, then the Monte Carlo calculation time can become excessive.

The Monte Carlo codes, like TRASYS, can calculate the energy absorbed on a surface. The shadowing of a surface by another surface automatically falls out from the ray/intersection calculations, i.e., a ray either hits the targeted surface or the shadowing surface. To calculate energy absorbed on a surface whether shadowed or not, the code, like TRASYS, must be able to specify the orientation of a surface in an orbit with respect to the sun or Earth. As with TRASYS, after a set of reference axes is specified in the orbit, e.g., solar-inertial, planet-centered (one axis always pointed at the center of Earth), etc., the coordinate system for each surface is then specified in relationship to the spacecraft's reference axes. These reference axes are then oriented with respect to the sun or Earth can be specified, and its illumination can be calculated.

All the radiation codes use the semigray approximation, a solution method that assumes that radiant interchange can be treated in two independent spectral regions, one solar and the other IR.

As noted previously, TRASYS assumes all surfaces are gray diffuse.

The Monte Carlo codes are not limited to this assumption. The surfaces can be gray diffuse, specular, or some combination of the two reflectances. In addition, transmittance can be allowed. Direction-dependent surface properties can also be used. For example, the NEVADA software allows the analyst to enter the directional dependence in the form of tables. The directionally dependent data would come from bidirectional surface-property measurements made at certain wavelengths and at certain angles. The Aerospace Corporation has an in-house project to analytically calculate directional properties using surface fractals. Directional dependence could impact Monte Carlo code execution time. The conservation of energy yields, for incident energy on a surface,

$$\alpha \text{ or } (\varepsilon) + \rho + \tau = 1 \tag{15.57}$$

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where α is the absorptance of the surface at solar wavelengths, ε is the fraction of energy emitted or absorbed by a surface in the IR wavelengths, ρ is the fraction of energy reflected by a surface at solar or IR wavelengths (the reflectance could be diffuse, specular, or directional), and τ is the fraction of incident energy transmitted through a surface. (IR refers to that part of the electromagnetic spectrum in the wavelength region of 0.7–1000 μ m. The solar part of the electromagnetic spectrum encompasses 0.3–0.7 μ m.)

As discussed previously, the OCTREE method is useful for speeding up the determination of a ray/surface intersection in a single processor. Another powerful method is to develop a distributed-processing system for a Monte Carlo code. In this case tens of processors can be used. Such a system can significantly reduce execution times for problems with hundreds of surfaces. The efficiency of any of the commercially available Monte Carlo code on a single processor is important. However, running a Monte Carlo code on multiple processors and on numerous computers has a significantly greater impact on reducing execution times for a given problem.

The Aerospace Corporation's ATRIUM and one commercially available thermalanalysis system, TSS by SpaceDesign, are the only codes that have a distributedprocessing capability at this time.

Nomenclature

Α	area
С	thermal capacitance
C _p	specific heat
D	diameter
D_H	hydraulic diameter
Ε	voltage
F	radiation configuration (form) factor
G	thermal conductance
h	convective heat-transfer coefficient
K	conductivity tensor
L	length or running length
Ι	current
k	thermal conductivity
ṁ	mass flow rate
n	arbitrary exponent
Ν	number of iterations
<i>Q</i>	heat rate
r	radius
R	resistance

Nomenclature—Continued

t	time
Τ	temperature
T_{∞}	surrounding media or free stream temperature
U	velocity
U_{∞}	free stream velocity
V	volume
ŵ	flow rate
ws	sampling frequency
wc	maximum frequency
x	arbitrary distance
dT	temperature difference
I	radiation interchange factor for real surfaces (script "F")
α	thermal diffusivity
β	coefficient of volumetric expansion
δ	convergence criterion (relaxation criterion)
ε	emittance
ζ	damping factor
θ	angle
λ	radiation linearization factor
ρ	density
σ	Stefan-Boltzmann constant
τ	stability factor

Symbols, subscripts, and units not specifically mentioned in the nomenclature are explained at the point of usage within the text.

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