initial provision, although this figure is highly dependent upon both mission duration and the type of orbit, with Highly Elliptical Orbits (HEOs) being particularly affected by radiation damage.

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11 THERMAL CONTROL OF SPACECRAFT

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11.1 INTRODUCTION

Spacecraft thermal control—that is the control of spacecraft equipment and structural temperatures—is required for two main reasons: (1) electronic and mechanical equipment usually operate efficiently and reliably only within relatively narrow temperature ranges and (2) most materials have non-zero coefficients of thermal expansion and hence temperature changes imply thermal distortion.

Spacecraft equipment is designed to operate most effectively at or around room temperature. The main reason for this is that most of the components used in spacecraft equipment, whether electronic or mechanical, were originally designed for terrestrial use. It is also much easier and cheaper to perform equipment development and, eventually, qualification and flight acceptance testing at room temperature. Typically, operating electronic equipment requires to be maintained in a temperature range between about $-15\,^\circ$C and $+50\,^\circ$C, rechargeable batteries between about $0\,^\circ$C and $+20\,^\circ$C and mechanisms (solar array drives, momentum wheels, gyroscopes etc.) between about $0\,^\circ$C and $+50\,^\circ$C. There are, of course, exceptions to this—for example, some detectors within astronomical telescopes that need to be cooled to very low temperatures.

Many spacecraft payloads require very high structural stability, and therefore thermally induced distortion must be minimized or strictly controlled. For example, the search for ever-higher resolution from space-based telescopes means that temperature stability of a fraction of one degree is often required within telescope systems several metres in size.

Heat is generated both within the spacecraft and by the environment. Components producing heat include rocket motors, electronic devices and batteries. Initial ascent heating effects are minimized by the launch vehicle's nose fairings or, in the case of launch by the Space Shuttle, by the cargo-bay doors. Heat from the space environment is largely the result of solar radiation. Heat is lost from the spacecraft by radiation, mainly to deep space. The balance between heat gained and heat lost will determine the spacecraft temperatures.

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The configuration of a spacecraft is dictated by many factors and 'thermal control' is only one of them. The task for the thermal control engineer consists, in fact, of three main parts. Firstly, *analysis*—he or she must be able to analyse a given spacecraft configuration and predict equipment and structural temperatures for all phases of the mission. Secondly, *design*—in the rather-likely circumstance that the results of the analysis show temperatures falling outside allowed limits, the engineer must devise suitable solutions, for example, by modifying heat-flow paths or implementing heaters, radiators and so on. Finally, *testing*—the engineer must perform sufficient and appropriate testing to confirm the accuracy of the analysis and of the thermal predictions for the mission.

11.2 THE THERMAL ENVIRONMENT

An important characteristic of the space environment is its high vacuum. Spacecraft are generally launched into orbits where the residual atmospheric pressure, and hence drag, is very small (although often not negligible—the International Space Station (ISS) will require re-boosting a few times per year to compensate for air drag). Fortunately for the thermal control engineer, the very low level of drag implies also the absence of any significant *aerodynamic heating*. For an orbiting spacecraft, aerodynamic heating and indeed any *convective interaction* between spacecraft and environment can be ignored.

The rate at which the Earth's atmospheric pressure falls with altitude [1] is shown in Table 11.1 for conditions of moderate solar activity. Spacecraft in orbit around the Earth usually orbit at altitudes higher than 300 km where the residual atmospheric pressure is typically less than 10^{-7} mb. During the launch phase, the transition from being fully protected within the launch vehicle to autonomous operation in space is the result of a compromise. The sooner the nose fairings can be jettisoned, the more payload a given vehicle can launch. However, if they are jettisoned too early, dynamic pressure and aerodynamic heating will damage the spacecraft. Fairings are normally jettisoned at the point at which residual aerodynamic heating is the same or less than the incident solar heating, which occurs at around 100-km altitude, depending on the characteristics of the launch vehicle and trajectory. Hence, once again, aerodynamic heating effects can be ignored by the spacecraft thermal designer. During re-entry or aerobraking manoeuvres (see Chapter 5), specific protection is provided, which ensures that these phases do not drive the spacecraft thermal design.

Table 11.1 Atmospheric pressure as a function of altitude (moderate solar activity)

Altitude (km)	Pressure (mb)
0	1013
50	7.98×10^{-1}
100	3.20×10^{-4}
150	4.54×10^{-6}
200	8.47×10^{-7}
250	2.55×10^{-7}
300	8.77×10^{-8}
350	3.51×10^{-8}
400	1.45×10^{-8}
450	6.45×10^{-9}
500	3.02×10^{-9}

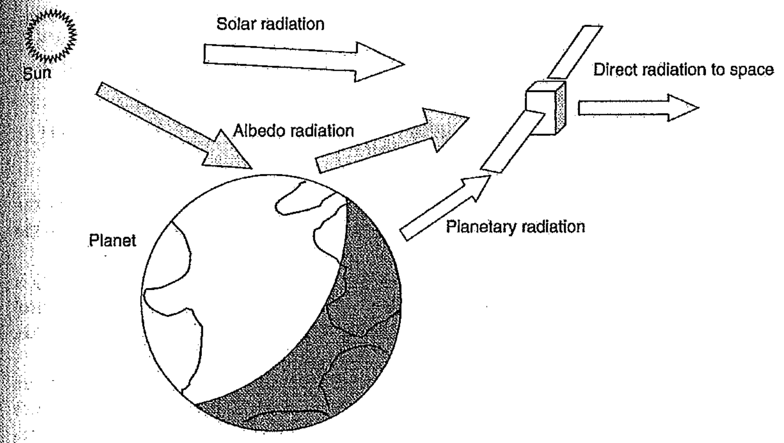

Figure 11.1 Typical spacecraft thermal environment

A spacecraft in space can interact with its environment only by radiation and this interaction is characterized by the exchange of energy by means of the following (see Figure 11.1):

- direct solar radiation;
- solar radiation reflected from nearby planets (albedo radiation);
- thermal energy radiated from nearby planets (planetary radiation);
- radiation from the spacecraft to deep space.

The spacecraft will experience thermal equilibrium when the sum of the radiant energy received from the first three sources listed above, together with any thermal dissipation within the spacecraft, is equal to the energy radiated to deep space. It is this balance that will determine the physical temperature of the spacecraft.

11.2.1 Solar radiation

The solar radiation parameters of interest to the thermal design engineer are (1) spectral distribution, (2) intensity and (3) degree of collimation. The spectral distribution can be considered constant throughout the solar system and the solar irradiance, or spectral energy distribution, resembles a Plank curve with an effective temperature of 5800 K (see Chapter 2). This means that the bulk of the solar energy (99%) lies between 150 nm and 10 μm, with a maximum near 450 nm (in the yellow part of the visible spectrum).

The solar radiation intensity outside the Earth's atmosphere and at the Earth's average distance from the Sun (1 AU) is called the solar constant and is about $1371 \pm 5\,\mathrm{W/m^2}$. The solar radiation intensity J_s at any other distance d from the Sun can be found from the simple relationship

$$J_s = \frac{P}{4\pi d^2} \tag{11.1}$$

where P is the total power output from the Sun, 3.856×10^{26} W. Table 11.2 shows the resulting variation in solar intensity that can be expected at the average distance from the Sun of each of the planets in the solar system.

The angle subtended by the Sun in the vicinity of the Earth (at 1 AU from the Sun) is about 0.5°. This means that the sunlight incident on a spacecraft can, for thermal control purposes, be regarded as a parallel beam emanating from a point source. This is not true, however, for spacecraft whose mission takes them very close to the Sun.

The fraction of the solar radiation that is reflected from the surface and/or atmosphere of a planet is known as the *planetary albedo*. Its value is highly dependent on local surface and atmospheric properties. For example, for the Earth, it varies from as high as 0.8 from clouds to as low as 0.05 over surface features such as water and forest [2,3]. Fortunately for the thermal engineer, such changes occur rapidly in relation to the thermal inertia of most spacecraft, and an orbital average value can be used for thermal design purposes. For the Earth, this is in the range 0.31 to 0.39. Table 11.2 lists the albedo values [4] for the planets of the solar system. The reader should be aware that measuring the albedo of the more distant planets is not an easy task and that the quoted figures should be treated with caution. Although the spectral distribution of albedo radiation is not identical to that of the Sun, as is evidenced by the diverse colours of planetary surface features, the differences are insignificant for thermal engineering purposes and can be ignored.

The intensity of the albedo radiation, J_a, incident on a spacecraft is a complex function of planet size and reflective characteristics, spacecraft altitude and the angle β between the local vertical and the Sun's rays. This can be expressed in terms of a *visibility factor* F as follows:

$$J_a = J_s a F \tag{11.2}$$

Table 11.2 Planetary solar constants and albedo values [2,3]

Planet	Solar radiation intensity, J_s (percentage of solar intensity at 1 AU)	Planetary albedo, a
Mercury	667	0.06–0.10
Venus	191	0.60–0.76
Earth	100	0.31–0.39
Moon	100	0.07
Mars	43.1	0.15
Jupiter	3.69	0.41–0.52
Saturn	1.10	0.42–0.76
Uranus	0.27	0.45–0.66
Neptune	0.11	0.35–0.62
Pluto	0.064	0.16–0.40

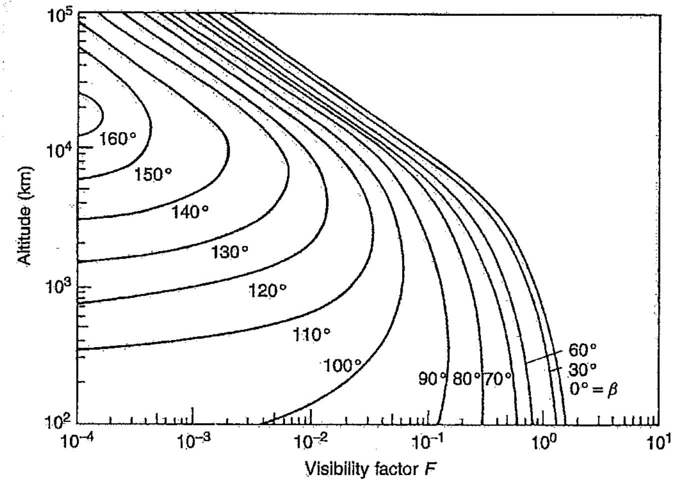

Figure 11.2 Spacecraft albedo irradiation. β is the angle between the local vertical and the Sun's rays

For the purpose of calculating albedo radiation inputs, the Earth can be regarded as a diffuse reflecting sphere, in which case the visibility factor varies approximately as shown in Figure 11.2.

It is emphasized that the above treatment is approximate. For complex spacecraft, particularly in low orbits, accurate calculation of albedo inputs may need to be performed as a function of orbital position for each external surface element. These are complicated calculations for which specific software tools are available.

11.2.2 Planetary radiation

Since the planets of the solar system all have non-zero temperatures, they all radiate heat. Because of its relatively low temperature, the Earth radiates all of its heat at infrared wavelengths, effectively between about 2 and 50 μm with peak intensity around 10 μm. For this reason, the radiation is often referred to as *thermal radiation*. The spectral distribution of the Earth's thermal radiation is shown in Figure 11.3 [2]. The atmosphere is essentially opaque over much of the infrared spectrum, with important transparent windows at around 8 and 13 μm. The radiation that a spacecraft sees is hence composed of radiation from the upper atmosphere, radiating with an effective black-body temperature of 218 K. Superimposed on this is radiation from the Earth's surface passing through the infrared windows. Since terrestrial temperatures vary with time and geographical location, the intensity J_p of the thermal radiation incident on orbiting spacecraft can also be expected to vary with time and position around the orbit. In fact, because of the Earth's large thermal inertia with respect to diurnal and seasonal changes and the spacecraft's

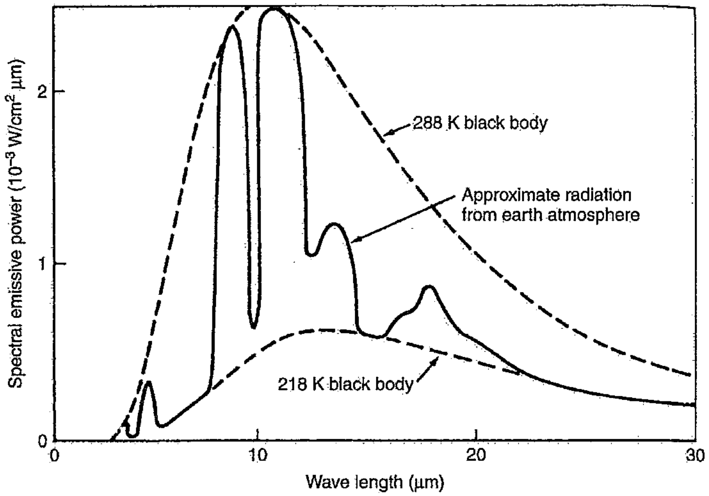

Figure 11.3 Typical spectral emissive power for the thermal radiation from Earth (*Note*: The 288-K black-body curve approximates the radiation from the Earth's surface, and the 218-K black-body curve approximates the radiation from the atmosphere in those spectral regions where the atmosphere is opaque)

large thermal inertia with respect to its orbital period, only very small errors occur if averaged values are used. For most practical purposes, the thermal engineer can assume that the Earth radiates with an intensity of 237 W/m² and that the thermal radiation emanates uniformly from the whole cross-sectional area of the Earth.

Since the intensity falls with altitude according to the inverse-square law, the approximate value of J_p in W/m² at a given altitude can be found from the following expression

$$J_p = 237\left(\frac{R_{rad}}{R_{orbit}}\right)^2$$

where R_{rad} is the radius of the Earth's effective radiating surface and R_{orbit} is the orbit radius. The precise value of R_{rad} is not easy to determine and for most practical purposes it can be assumed equal to the radius of the Earth's surface, R_E. For other planets, care needs to be taken to verify the validity of these assumptions on a case-by-case basis. For example, Mercury, with a sidereal day of the same order of magnitude as its year (59 and 88 Earth days, respectively), sustains temperature differences of hundreds of degrees between sunlit and shadowed sides, with a slow-moving terminator. Its orbit is also sufficiently eccentric for its solar constant to more than double between apogee and perigee, giving rise to large seasonal variations.

11.2.3 Spacecraft heat emission

The spacecraft itself has a finite temperature, so it will also radiate heat to space. Since the spacecraft temperature will be similar to that of the Earth (if the thermal engineers have done their job properly), it too will radiate all its heat in the infrared region of the spectrum.

11.3 THERMAL BALANCE

As already noted, the temperature of a spacecraft depends on the balance between the heat received from external and internal sources, and the heat radiated to space. In order to control spacecraft temperatures, it is necessary to control the heat absorbed, the heat radiated or (usually) both.

If spacecraft were *black bodies*, that is, radiated as black bodies and absorbed all the radiation that fell on them, they would acquire a certain temperature and that would be the end of the story. Thermal control would be impossible, except perhaps by varying the internal heat dissipation. However, spacecraft are not black bodies but absorb only a fraction α of incident energy. They also emit as a *grey body*, radiating a fraction ε of the radiation of a black body at the same temperature. This may be expressed as

$$J_{absorbed} = \alpha J_{incident} \tag{11.3}$$

$$J_{radiated} = \varepsilon\sigma T^4 \tag{11.4}$$

where α and ε are known as the *absorptance* and the *emittance*, respectively, and σ is the *Stefan–Boltzmann constant* equal to 5.67×10^{-8} W/m² K⁴.

For a spacecraft with no internal heat dissipation, an effective absorbing area (projected area facing the Sun) A_α and emitting area A_ε, the equilibrium temperature, T, is given by

$$A_\alpha J_{absorbed} = A_\varepsilon J_{radiated} \tag{11.5}$$

which from equations (11.3) and (11.4) gives

$$A_\alpha \alpha J_{incident} = A_\varepsilon \varepsilon\sigma T^4$$

so that

$$T^4 = \frac{A_\alpha}{A_\varepsilon}\frac{J_{incident}}{\sigma}\left(\frac{\alpha}{\varepsilon}\right) \tag{11.6}$$

Since A_α, A_ε and σ are constants, and for a given value of $J_{incident}$, the value of T can be controlled by varying the value of α/ε.

In fact, things are not quite as simple as that, as α and ε are not independent variables. Over any given wavelength range, the laws of thermodynamics require that $\alpha = \varepsilon$ (Kirchhoff's law). Thus for any surface, its *absorptivity* at a given wavelength is equal to its *emissivity* at the same wavelength.

Fortunately for the thermal control engineer, absorptivity and emissivity generally vary with wavelength and we have already learnt that the radiation environment of a spacecraft is basically composed of radiation either at 'visible' wavelengths or in the infrared. It is this feature that makes spacecraft thermal control possible. For the spacecraft thermal control engineer-

- α means the absorptance of a surface to solar radiation (peak intensity at about 0.45 µm)—it is therefore often referred to as the '*solar absorptance*'.
- ε means the emittance of a surface radiating in the infrared region (peak intensity at about 10 µm)—it is therefore often referred to as the '*infrared emittance*'.

Note that, according to Kirchhoff's law, the absorptance of a spacecraft to planetary radiation is equal to its infrared emittance, ε.

By way of an example, let us consider a simple spacecraft in Low Earth Orbit (LEO). For the sake of convenience, let us assume a polar orbit that does not suffer from eclipses (a so-called dawn–dusk orbit—see Chapter 5) and let us furthermore assume that our spacecraft has a high thermal inertia and is isothermal. We have

heat received directly from the Sun $= J_s \alpha A_{solar}$
albedo contribution $= J_a \alpha A_{albedo}$
planetary radiation contribution $= J_p \varepsilon A_{planetary}$
heat radiated to space $= \sigma T^4 \varepsilon A_{surface}$
internally dissipated power $= Q$

where A_{solar}, A_{albedo} and $A_{planetary}$ are the projected areas receiving, respectively, solar, albedo and planetary radiation, and $A_{surface}$ is the spacecraft total surface area. If we assume that J_s, J_a, J_p and Q remain constant, our spacecraft will acquire an equilibrium temperature T given by

$$(A_{solar} J_s + A_{albedo} J_a)\alpha + A_{planetary} J_p \varepsilon + Q = A_{surface} \sigma T^4 \varepsilon$$

Hence

$$T^4 = \frac{A_{planetary} J_p}{A_{surface}\sigma} + \frac{Q}{A_{surface}\sigma\varepsilon} + \frac{(A_{solar} J_s + A_{albedo} J_a)}{A_{surface}\sigma}\left(\frac{\alpha}{\varepsilon}\right) \qquad (11.7)$$

So once again we see that the spacecraft temperature is dependent on the ratio α/ε, particularly for spacecraft for which Q is small. For simplicity, let us assume a spherical spacecraft, radius r, for which

$$A_{surface} = 4\pi r^2,\ A_{solar} = A_{albedo} = A_{planetary} = \pi r^2,\ Q = 0$$

$$J_a = 0.33F\ J_s (\text{Earth albedo } a = 0.33),$$

$$J_p = 220\ \text{W/m}^2 (\text{corresponding to an orbit altitude of around 240 km})$$

Then equation (11.7) reduces to

$$T^4 = 9.70 \times 10^8 + 4.41 \times 10^6 (1 + 0.33F) J_s \left(\frac{\alpha}{\varepsilon}\right)$$

For $J_s = 1371$ W/m², F = 0.15 (from Figure 11.2) and a black paint finish for which $\alpha/\varepsilon = 1$, our spacecraft equilibrium temperature is about 293 K or 20 °C.

If we now turn the orbit plane until the spacecraft passes through the Earth's shadow, the heat absorbed from the Sun (directly and as albedo) will be reduced. Assuming a sufficiently high thermal inertia, a new equilibrium temperature will be obtained. Let us consider the case in which the Earth–Sun vector lies in the plane of the orbit. This will evidently give the minimum time in sunlight, which, for a LEO spacecraft at an altitude of 240 km, is about 59% of its orbit period. Under these conditions, an average albedo visibility factor can be estimated from Figure 11.2 for the illuminated part of the orbit, F ~ 0.7. Note that the albedo radiation is zero during eclipse. The new equilibrium temperature will then be obtained from

$$T^4 = 9.70 \times 10^8 + 5.43 \times 10^6 J_s \left(\frac{\alpha}{\varepsilon}\right) f$$

Table 11.3 Equilibrium temperatures for a simple spacecraft in LEO

Surface finish	White paint $\alpha = 0.15$ $\varepsilon = 0.9$	Black paint $\alpha = 0.9$ $\varepsilon = 0.9$	Electroplated gold $\alpha = 0.25$ $\varepsilon = 0.04$
No eclipse	−61 °C	+20 °C	+176 °C
Maximum eclipse	−70 °C	−2 °C	+138 °C

Table 11.4 α and ε values for several surfaces and finishes [5,6,7]

Surface	Absorptance (α)	Emittance (ε)	α/ε
Polished beryllium	0.44	0.01	44.00
Goldized kapton (gold outside)	0.25	0.02	12.5
Gold	0.25	0.04	6.25
Aluminium tape	0.21	0.04	5.25
Polished aluminium	0.24	0.08	3.00
Aluminized kapton (aluminium outside)	0.14	0.05	2.80
Polished titanium	0.60	0.60	1.00
Black paint (epoxy)	0.95	0.85	1.12
Black paint (polyurethane)	0.95	0.90	1.06
—electrically conducting	0.95	0.80–0.85	1.12–1.19
Silver paint (electrically conducting)	0.37	0.44	0.84
White paint (silicone)	0.26	0.83	0.31
—after 1000 hours UV radiation	0.29	0.83	0.35
White paint (silicate)	0.12	0.90	0.13
—after 1000 hours UV radiation	0.14	0.90	0.16
Solar cells, GaAs (typical values)	0.88	0.80	1.10
Solar cells, Silicon (typical values)	0.75	0.82	0.91
Aluminized kapton (kapton outside)	0.40	0.63	0.63
Aluminized FEP	0.16	0.47	0.34
Silver coated FEP (SSM)	0.08	0.78	0.10
(OSR)	0.07	0.74	0.09

Note: SSM, Second Surface Mirror.
OSR, Optical Solar Reflector.

where f is equal to the fraction of the orbit that is illuminated by the Sun, 0.59 in this case. It will be seen that our black-painted spacecraft has fallen in temperature to about −2 °C. Table 11.3 shows the results of similar calculations for a white paint finish (low α/ε) and electroplated gold (high α/ε). It is evident that, by using different surface finishes in different ratios, spacecraft temperatures can be controlled over quite large ranges. Table 11.4 lists α and ε values for a number of common spacecraft surface finishes [5,6,7]. It should be realized that, particularly for metal surfaces, the values of α and ε may be very dependent on preparation and surface treatment.

Real spacecraft are, of course, far more complicated than the one discussed above. They are certainly not isothermal and often contain components (e.g. the solar arrays) with a relatively low thermal inertia, which will change temperature significantly around an orbit (particularly when entering or leaving an eclipse). Whilst the overall spacecraft thermal balance is determined by its external surface characteristics and the radiative environment, the internal thermal balance determines equipment temperatures and is hence of crucial importance to the thermal engineer. Calculation of the internal thermal balance, involving radiative and conductive exchanges between all the spacecraft components, is complex and is covered in some detail in the next section.

11.4 THERMAL ANALYSIS

11.4.1 Thermal mathematical model (TMM)

Spacecraft are generally very complex structures within which temperatures are varying continuously as a function of location and time. Calculating these temperature fields in rigorous detail is, for all practical purposes, impossible. In order to progress further, it is first necessary to simplify the problem. This is done by generating an approximate representation of the spacecraft that is amenable to mathematical treatment. Such a representation is known as a *thermal mathematical model* (TMM).

In order to construct a TMM, the spacecraft is considered as being composed of a number of discrete regions within which temperature gradients can be neglected. These regions are known as *isothermal nodes*. Each node is characterized by a temperature, thermal capacity, heat dissipation (if any) and radiative and conductive interfaces with the surrounding nodes. Nodes that can 'see' space directly will also have radiative interfaces with the external environment.

11.4.2 Conductive heat exchange

It will be recalled that the conductive heat flow rate is given by

$$Q_c = \frac{\lambda A}{l}\Delta T \qquad (11.8)$$

where λ is the thermal conductivity, A the cross-sectional area, l the conductive path length and ΔT the temperature difference. The term $\lambda A/l$ is known as the *thermal conductance*, h_c, and hence the temperature difference can be written as

$$\Delta T = Q_c\frac{1}{h_c} \qquad (11.9)$$

In most engineering applications, A and possibly λ may vary significantly along the path length. If the conductive path is considered as a number of discrete conductive paths connected in series, the temperature difference can be rewritten as

$$\Delta T = Q_c\left(\frac{1}{h_1}+\frac{1}{h_2}+\frac{1}{h_3}+\dots\right) = Q_c\frac{1}{h_c}$$

and hence the effective thermal conductance, h_c, for the path can be found from

$$\frac{1}{h_c} = \frac{1}{h_1}+\frac{1}{h_2}+\frac{1}{h_3}+\dots \qquad (11.10)$$

For a spacecraft composed of n isothermal nodes, the heat conducted from the ith to the jth node is given by

$$Q_{c_{ij}} = h_{ij}(T_i - T_j) \qquad (11.11)$$

where h_{ij} is the effective conductance between nodes i and j and T_i and T_j are the temperatures of the ith and jth nodes, respectively.

11.4.3 Radiative heat exchange

Radiative heat exchange between two surfaces is determined by three important parameters—the surface temperatures, the radiative view factors and the surface properties. For diffuse surfaces, the amount of radiation leaving a surface i and absorbed by a surface j can be shown [8] to be of the form

$$Q_{r_{ij}} = A_i F_{ij}\varepsilon_{ij}\sigma(T_i^4 - T_j^4) \qquad (11.12)$$

where A_i is the area of the surface i, F_{ij} is the view factor of surface j as seen from surface i and ε_{ij} is a parameter known as the *effective emittance*. Note that it is tacitly assumed in the above that the value of the view factor F_{ij} remains constant over the surface i.

View factors

The *radiative view factor* F_{ij} is defined as the fraction of the radiation leaving one surface that is intercepted by another. It follows that from any node i inside a spacecraft, the sum of the view factors to surrounding equipment must be unity,

$$\sum_{j=1}^{k} F_{ij} = 1 \qquad (11.13)$$

where k is the number of surrounding surfaces.