significant than direct solar flux types but pose a unique problem since the source temperatures are very close to those of the spacecraft. As a result, they cannot be easily reflected away by selecting surface materials with appropriate optical properties since these same types of surfaces would not allow emittance of spacecraft heat.

Energy Generation

Internal energy generation, $\dot{E}_{gen}$ is the result of energy dissipation of components required to operate the satellite. In 2000, Hoeber and Kim found that this power had doubled every five to six years because of increased demand of satellite services. Although recent trends have indicated that the total power has reached a plateau, increased heat fluxes have not. According to Baturkin (2005), the future problem of small satellites is miniaturization of space equipment leading to reduced component sizes and increases in heat flux densities up to 100 W/cm^2. Large satellites have similar problems with high flux space applications including power converters, scientific lasers, microwave solid-state transmitters, MEMS broadband lightsource, directed energy weapons, and space-based radar (Golliher 2002; Ponnappan et al. 2002). Traditional space-based thermal management techniques, both active and passive, are unable to meet these requirements, necessitating the development of new technologies to enable reliable operation in adverse conditions (Maxwell et al. 2008).

Energy Storage

$\dot{E}_{stored}$ is a function of satellite mass and thermal properties and significantly affects the temporal variation in component temperatures that occurs over the orbit of the satellite as it is exposed to time-varying boundary conditions. S/C with low capacitance yield significant spikes in temperature over one orbit while those with high capacitance have a smoother temperature profile. Although high capacitance is desired, this price is typically paid through increased S/C mass, which is unfavorable due to relatively high launch costs. Foust and Smith (2004) provide an exhaustive review of launch vehicle efforts including those of SpaceX and their Falcon 1. This “affordable” unit is capable of launching approximately 450 kg into LEO at a cost of $7.9 million (i.e., $16,667/kg) (SpaceX 2008).

Energy Dissipation

Heat rejection from the satellite, $\dot{E}_{out}$, is traditionally accomplished only through radiation at the S/C boundaries. As a result, $\dot{E}_{out}$ is limited by available surface area and surface optical properties and impacted by interaction with the thermal environment of space. According to Hoeber and Kim (2000), managing $\dot{E}_{out}$ is one of the most challenging design limitations of TCS.

ROBUST S/C THERMAL CONTROL

To satisfy the growing importance and expectations of space while overcoming its inherent challenges, S/C have become exceedingly complex and costly. For example, satellites can take from three to seven years to deploy and cost from millions to billions of dollars (Williams and Palo 2006). Reduced S/C complexity and cost can be realized through robust system design. Whereas traditional approaches focus on designs optimized for a particular mission, robust S/C are developed to handle a broad range of missions, thus reducing long-term design and development costs. In addition, robust S/C allows for drastically reduced timelines, easier technology insertion, and help in withstanding late changes in mission requirements (Birur and O’Donnell 2001). Achieving this requires a significant departure from traditional design approaches, especially for the TCS. Robust S/C TCS must meet a much larger design space of thermal environments, component distributions, and operating states. Consequently, traditional S/C TCS design methodologies must be radically overhauled. This will be realized with technology advances