Heat Load Calculations on an HTS Coil Integrated into a Small Satellite during a Sun-Synchronous Low Earth Orbit

J R Olatunji, C Acheson, M Szmigiel, S C Wimbush, N J Long
Robinson Research Institute, Victoria University of Wellington, New Zealand
jamal.olatunji@vuw.ac.nz, chris.acheson@vuw.ac.nz, stuart.wimbush@vuw.ac.nz, nick.long@vuw.ac.nz

Abstract. High-temperature superconductivity (HTS) has the potential to be a useful technology for space applications, allowing for high current densities and magnetic field generation in compact devices. However, HTS requires cryogenic temperatures and it is not well understood how this can best be achieved in a space environment. Using a modelling approach, the expected heat load on a hypothetical 3U CubeSat with an HTS coil during a sun-synchronous low Earth orbit was predicted. The direction and magnitude of solar, albedo and infrared incident radiation toward the satellite was calculated for each orbital position of a circular 732 km, \( \Omega = 0^\circ \) longitude of ascending node orbit. Using a finite element approach, the surface radiosity and temperature of the CubeSat was predicted and validated. Finally, the instantaneous heat load on the HTS magnet, which generates a 1T magnetic field, was calculated as a function of orbital position. This study provides technical information about the characteristics of the refrigeration device required to maintain cryogenic temperatures for an HTS coil on a space faring vessel, such as a portable cryocooler. Selection and design of a satellite cooling system must be optimised according to the calculated heat load and available solar power.

1. Introduction
High-temperature superconductivity (HTS) is being reconsidered as a viable technology in the aerospace industry, capable of generating large magnetic fields and current densities in lightweight, compact devices—a clear advantage for an industry with extreme weight and size restrictions. Potential applications for HTS in space include electric propulsion [1], energy storage [2] and magnetic shielding [3].

None of the promising applications of HTS in space are possible without a thorough examination of the thermal load on the HTS device in a space environment, as HTS as a phenomenon only occurs at cryogenic temperatures, typically \(<90\,\text{K}\) [4]. The HTS device must be maintained below this temperature constantly while operating or a quench will occur, potentially damaging the spacecraft. Therefore, it is not possible to utilise HTS in space without careful consideration of the refrigeration requirements. The availability of off-the-shelf, space-rated, miniaturised cryocoolers with flight heritage has increased significantly in recent years, making the topic of HTS in space worthy of renewed interest [5], [6].

In this paper, a hypothetical 3U CubeSat [7] was equipped with an HTS coil and was placed into a sun synchronous low Earth orbit in the computational domain. Using a modelling approach, the total heat load on the HTS coil was calculated to give an indication of the size of refrigeration system and power budget required, to enable the previously mentioned applications of HTS in a space environment. First, the
time-dependent orbital position of the satellite was calculated using Keplerian orbital mechanics. Then, the time-dependent magnitude and direction of incident radiation from several sources was calculated. The external radiation vectors were combined with a finite element heat transfer model to calculate the instantaneous spatiotemporal temperature profile of the satellite at a component level. From this, the required cooling power was derived to counteract all parasitic heat loads on the HTS coil, to maintain a coil operating temperature of 77.5 K.

2. Materials and Methods

2.1. 3U CubeSat Design
A hypothetical 3U CubeSat was equipped with an HTS coil for this numerical study, illustrated in Figure 1a. The CubeSat was made of solid aluminium with external dimensions of 0.1×0.3×0.1 m and internal dimensions of 0.0957×0.2871×0.0957 m, so that the chassis of the CubeSat consisted of exactly 1 kg of aluminium. This was done for model validation purposes, which are explained in detail later (see section 3.2). A 3U CubeSat was chosen as it is the smallest standardised satellite size into which a cryocooler or other miniaturised cryogenic cooling system could realistically be incorporated.

Figure 1. A hypothetical 3U CubeSat equipped with an HTS coil: a) the CubeSat, consisting of an aluminium chassis, solar panels and suspended HTS coil; b) the HTS coil, consisting of a double pancake HTS coil sandwiched between two copper thermal busses, a cold connection point for mounting to a refrigeration system, and two copper current leads; c) dimensions of the HTS coil.

Suspended at the centre of the CubeSat was an HTS coil (Figure 1b), consisting of an 80 mm diameter double pancake coil made of 4 mm wide REBCO HTS tape [4]. The HTS coil was sandwiched between two 2 mm thick copper thermal busses 90 mm in diameter, connected to each other by four 2 mm diameter copper pins (Figure 1c), similar to other conductively cooled HTS devices in other applications [8]. The refrigeration device is not modelled explicitly and is instead represented by a 20 mm diameter cold connection point where for example the cold head of a cryocooler would be mounted.

Attached near the cold connection point are two copper current leads, which are attached at the other end to the chassis of the CubeSat. This model does not explicitly model the battery or power supply (or other equipment a CubeSat requires, [7]) where these would realistically be connected. Therefore, the current leads as shown in Figure 1b represents a thermal connection between the CubeSat and HTS coil, rather than a realistic electrical connection.

Mechanical details of how the HTS coil was suspended inside the CubeSat are not considered. The outcome of this suspended design is zero contact between the HTS coil and the CubeSat, restricting
conduction heat transfer to through the copper current leads only; and a significant amount of thermal radiative heat transfer between the CubeSat interior and the HTS coil. The coil is intended to be energised with direct current, so AC losses can be ignored.

2.2. Orbital Mechanics

The velocity and position of the 3U CubeSat in orbit around the Earth were determined using well-known Keplerian orbital mechanics [9]. In short, the Keplerian orbital elements describe an orbit as an ellipse in a specific orientation with respect to a reference plane. The five Keplerian orbital elements utilised in this study are: eccentricity, $e$, semi-major axis, $a$, inclination, $i$, argument of periapsis, $\omega$, and longitude of ascending node, $\Omega$. The sixth Keplerian orbital element, true anomaly, $\nu$, was incorporated in the temporal mesh and therefore was not used explicitly. Calculations were performed for a time of year close to the Spring Equinox. Using the Earth’s equatorial plane as a reference, and the reference vector of the First Point of Aries [9], the velocity and position of a satellite in the following sun-synchronous orbit was calculated: $e = 0$, $a = 7103$ km, $\omega = 0^\circ$, $i = 98^\circ$ and $\Omega = 0^\circ$ (12 noon). This is a circular orbit 732 km above the surface of Earth, where $\Omega = 0^\circ$ was chosen specifically for this investigation as the satellite spends approximately half of its orbital period in Earth day, and approximately half in Earth night. This orbit should generate a large temperature fluctuation over an orbital period and allows the predicted satellite temperature to be compared to a literature source [10].

2.3. Incident Radiation

In this study, three sources of incident radiation were considered: direct solar radiation; Earth albedo (sunlight reflected by the Earth’s surface); and the Earth’s blackbody radiation. Each of these contributions was determined numerically by first considering the position of the satellite at each point in time, and later considering the spacecrafts geometry, surface properties and orientation. Splitting the calculation into these two separate parts allows different satellite geometries to be examined without recalculation of the incident radiation. The incident radiation is represented by many vectors whose direction and magnitude represents the intensity of radiation from that source. In practical terms, it is easiest to represent these vectors in the reference frame of the satellite. A vector basis was created with unit vectors:

1. From the centre of the Earth to the centre of the satellite (position);
2. In the direction of the velocity component perpendicular to position, and;
3. In the direction of angular velocity, i.e. the direction normal to the orbital plane.

This vector basis forms a non-inertial frame of reference. The direct solar radiation incident upon the satellite can be described by a single vector whose direction is equal to the solar vector expressed in this basis. It has a magnitude of 0 when the satellite is in the Earth’s shadow, or a magnitude equal to the solar irradiance when the satellite is illuminated by the sun. The solar heat load on the $i^{th}$ satellite face can be determined through:

$$Q_{s,i} = \alpha_i A_i \cos \left( b_i \right) I_s Bool_s$$

where $Q_{s,i}$ is the heat flux transferred from the sun, $\alpha_i$ is the satellite absorptivity, $A_i$ is the unshaded surface area of the satellite, $b_i$ is the angle between the normal to the satellite and the sun, $I_s$ is the intensity of solar radiation (equal to 1367 W/m², [10], [11]) and $Bool_s$ is a Boolean that is equal to 1 when the satellite is in the sun, and is equal to 0 when the satellite is in the Earth’s shadow. The subscript $i$ refers to a specific face of the satellite, and the total direct solar heat load on the satellite is determined by summing over all $i$ faces.

The calculation of the albedo and Earth blackbody radiation cannot be simplified to a single vector, and therefore relies upon the approximation of the Earth’s surface as multiple flat surfaces, each of which is a source of radiation whose magnitude and direction must be accounted for. The Earth’s blackbody
radiation lies in the infrared range and therefore the emissivity of satellite surfaces must be used to determine the resultant heat load. With the direct solar radiation and albedo lying in the visible light band and the resultant heat being transferred to the satellite according to the surface absorptivity, the Earth’s blackbody radiation had to be separated entirely from the other contributions. The heat load on the satellite’s $i^{th}$ face from the Earth’s blackbody radiation is given by integrating over Lambert’s cosine law for the surface of the Earth:

$$Q_{rad,i} = e_i A_i I_{Earth} \int \frac{\cos(a_{E}) \cos(a_i)}{\pi \rho^2} dS$$

where $e_i$ is the emissivity of the satellite, $\rho$ is the magnitude of $\vec{\rho}$ (the vector from the centre of the Earth’s surface element to the centre of the satellite), $I_{Earth}$ is the mean intensity of the Earth’s blackbody radiation (213 W/m$^2$, [10], [11]), $a_{E}$ is the angle between the Earth surface element normal and $\vec{\rho}$, $a_i$ is the angle between $\vec{\rho}$ and the vector normal to the satellite, $dS$ is the surface element of the Earth, and the integral limits are set such that the integral is over the surface of the Earth within the field of view of the satellite. An icosphere was chosen to represent the Earth’s surface, allowing the above equation to be solved numerically by discretising the surface of the Earth in the satellite’s field of view. This shape demonstrates a good trade-off between the sphere’s constituent surfaces having similar area and its central surface points having a similar angular spacing. This can be seen in Figure 2.

![Earth Surface Divided into Surface Elements](image)

**Figure 2.** Incident radiation calculations for infrared and albedo. Infrared radiation is emitted by the $k^{th}$ surface element in the direction of the satellite according to Lambert’s cosine law applied to angle $a_{Ek}$. Similarly, for the albedo, Lambertian reflectance of the sun from the $k^{th}$ surface element in the direction of the satellite depends on $b_{Ek}$ and $a_{Ek}$.

To allow different satellite geometries to be investigated, a new vector $\vec{H}$ was introduced. The direction of $\vec{H}$ indicated the direction to a point source of radiation infinitely far away (such that the rays
are parallel) and the magnitude of $\vec{H}$ corresponded to the intensity of the radiation from that direction. $\vec{H}_{rad,k}$ was calculated for each surface element on the icosphere representing the Earth’s surface, where subscript $k$ represents an individual surface element:

$$
\vec{H}_{rad,k} = I_{Earth}Bool_{rad,k}S_k \left( \frac{\cos(a_{E,k})}{\pi p_k^2} \right) \left( \frac{-\hat{p}_k}{|\hat{p}_k|} \right)
$$

(3)

where $Bool_{rad,k}$ is a Boolean value that incorporates the satellites field of view, and $S_k$ is the surface area of the $k^{th}$ Earth surface element. This enabled the heat load on the satellite to be determined through:

$$
Q_{rad,i} = \varepsilon_i A_i \sum_{k=1}^{n} \vec{H}_{rad,k} \cdot \vec{N}_i
$$

(4)

summing over all Earth surface elements, where $\vec{N}_i$ is the unit normal of the $i^{th}$ satellite face, and the vector inner product is represented by the dot.

Similarly, the heat load from the Earth albedo was given by:

$$
Q_{alb,i} = \alpha_i A_i \gamma I_s \int \frac{\cos(a_E) \cos(a_i) \cos(b_E)}{\pi p^2} dS
$$

(5)

where $\gamma$ is the Earth diffuse reflectance (approximated as an average value of 0.273, [10], [11]), $b_E$ is the angle between the Earth surface element normal and the solar vector and the limits of the integral are set to incorporate the surface of the Earth that is both in the sun and in the satellites field of view. An $\vec{H}$ vector for albedo was calculated as:

$$
\vec{H}_{alb,k} = \gamma I_s Bool_{alb,k}S_k \left( \frac{\cos(a_{Ek}) \cos(b_{Ek})}{\pi p_k^2} \right) \left( \frac{-\hat{p}_k}{|\hat{p}_k|} \right)
$$

(6)

where $Bool_{alb,k}$ is a Boolean value incorporating the satellite’s field of view and the sunny side of the Earth. The heat load on the satellite from the albedo was calculated as:

$$
Q_{alb,i} = \alpha_i A_i \gamma I_s \sum_{k=1}^{n} \vec{H}_{alb,k} \cdot \vec{N}_i
$$

(7)

The result of this calculation method is shown in Figure 3b.

2.4. Finite Element Heat Transfer Model Formulation

Heat transfer calculations were performed using a finite element modelling scheme in COMSOL Multiphysics. Incident radiation was divided into two spectral bands, visible and infrared light, and were modelled as an array of infinite distance external radiation sources with the same direction and time dependent magnitude as the previously calculated $\vec{H}$ vectors (section 2.3). Radiation inside the satellite was also modelled using the ‘Hemicube’ surface-to-surface radiation method with a resolution of 32. All surfaces were given the emissivity and absorptivity of shiny metal, $\varepsilon_{metal} = 0.05$ and $\alpha_{metal} = 0.5$ respectively; the exception were the photovoltaic panels that were given $\varepsilon_{photovoltaic} = 0.89$ and $\alpha_{photovoltaic} = 0.91$. Because the photovoltaic panels converted some energy into electricity at an efficiency
of 28%, the absorptivity was in practice $\alpha_{\text{photovoltaic}} = 0.63$. All surfaces were modelled as opaque and diffuse surfaces, and emitted radiation into an ambient temperature of 0 K.

The satellite chassis was given the thermal properties of aluminium ($C_{p,Al} = 900$ J/kg·K, $\lambda_{Al} = 238$ W/m·K, $\rho_{Al} = 2700$ kg/m$^3$) and the HTS coil the thermal properties of stainless steel ($C_{p,SST} = 475$ J/kg·K, $\lambda_{SST} = 44.5$ W/m·K, $\rho_{SST} = 7850$ kg/m$^3$). The copper was modelled as RRR = 100 and was given temperature dependent thermal and electrical properties so that Joule heating and conduction heat transfer could be modelled accurately. These properties were taken from [12]. Space was assumed to be a perfect vacuum, so there was no air inside the satellite for heat to conduct through. Conduction heat transfer was modelled using Fourier’s law throughout the entire geometry.

As is common for HTS devices powered by current leads, there is an optimum length to area ratio that minimises the sum of conduction and Joule heating heat transfer while the coil is being fed with current. This was determined using the method outlined in [13]. It was decided that the coil would be fed with 60 A of current to produce a 1 T central field, so that the optimum length to area ratio for the current leads with hot and cold end temperatures of 270 K and 77.5 K was determined to be 60000 m$^{-1}$. The length of the current leads was 141 mm, so the width of the square cross-sectional wire was 1.5 mm.

The chassis of the satellite was given an initial temperature of 273.15 K to match the validation data used [10], while the HTS coil and current leads had an initial temperature of 77.5 K. The cold connection point (green circle in Figure 1b) was set at a constant temperature of 77.5 K. The heat flux across the cold connection surface is equal to $Q_c$, the required cooling power of the refrigeration system to maintain the 77.5 K setpoint. To maintain an energy balance, $-Q_c$ was applied to the chassis of the satellite as a heat source, similar to how the hot end of a cryocooler would be mounted to the chassis of the satellite to dissipate heat.

The geometry was discretised into 7667 tetrahedral elements. A transient heat transfer simulation was then performed over five orbits (29742 s) using the PARDISO solver (i7-4790 CPU, 16GB RAM).

3. Results and Discussion

3.1. Orbital Mechanics and Incident Radiation

The orbit discussed in section 2.2 is visualised in Figure 3a. One orbital period was 5948.4 s.

Results of the calculation of incident radiation (section 2.3) are visualised in Figure 3b for four orbital positions, when the CubeSat was: i.) in direct sunlight; ii.) entering the shadow of the Earth; iii.) in the Earth’s shadow, and; iv.) exiting the Earth’s shadow. Arrow direction, length and colour represent the direction and magnitude of direct solar (yellow = Earth day, black = Earth night), albedo (blue) and infrared radiation (red). When the CubeSat is in direct sunlight, it receives direct solar radiation on the top surface, and albedo and infrared radiation on the bottom surface. As it enters the shadow of the Earth, the magnitude of albedo radiation declines, and the direction of visible light shifts to strike the back of the CubeSat. In the shadow of the Earth, the CubeSat only receives infrared light on the bottom surface, so that the solar panels in this period of the orbit act as radiators. As the CubeSat exits Earth’s shadow, direct solar and albedo radiation return, but instead strike the front of the satellite. This should result in the satellite cooling significantly while in the shadow of the Earth and warming while in direct sunlight, creating a temperature fluctuation over the orbital period.
Figure 3. a) Orbital pathway for a satellite in a sun-synchronous low Earth orbit with a longitude of ascending node of $\Omega = 0^\circ$; b) incident radiation magnitude and direction when the CubeSat is in four orbital positions: i) in direct sunlight; ii) entering the shadow of the Earth; iii) in Earth shadow, and; iv) exiting the shadow of the Earth.

3.2. Satellite Temperature and Heat Load Prediction

The model was validated through a comparison of the satellite temperature with a published literature source [10]. Garzon et al. published results for a 3U CubeSat in a 732 km circular orbit with a longitude of ascending node of $\Omega = 0^\circ$, modelled as a single node comprising 1 kg of solid material with the internal thermal properties of aluminium and the surface properties described in section 2.4. Results are shown in Figure 4a. The present modelling approach is highly comparable to Garzon et al., but predicts a slightly higher average temperature. This is partly due to the addition of Joule heating in the present model, of which the heat generated is removed at the cold connection point and added to the chassis of the satellite to maintain an energy balance. Furthermore, this model explicitly computes the temperature at many node points on the satellite, seen in Figure 4e and f, whereas the comparison model treats the satellite as a single node with a homogenous temperature.

The satellite’s mean temperature is considerably greater than the critical temperature of the HTS coil at all times, and hence an active cooling device such as a cryocooler is required to prevent the HTS coil from quenching. The maximum temperature of the coil oscillates between 77.85 K and 77.90 K, indicating that a cold head temperature of 77.5 K is sufficient to adequately cool the coil. However, the cryocooler must produce 4W of cooling power to achieve a cold head temperature of 77.5 K. To extract this at 20% Carnot efficiency, ~52 W of input power is required. However, most of this heat load stems from the copper current leads, seen generating hot spots on the HTS coil in Figure 4d. Radiative flux accounts for just 5% of the heat load and this can be vastly reduced with the incorporation of multi-layer insulation (MLI). Therefore it is suggested that a HTS flux pump [14] be utilised for in-orbit operation of an HTS coil. This non-contact method of generating current means that the coil can be thermally isolated from the satellite chassis. Further work would include the characterisation of the resultant heat load from such a flux pump.
Figure 4. Simulation results over five orbits: a) average satellite temperature (solid black line) compared with a literature source [10] (dashed red line); b) maximum HTS coil temperature; c) power and heat fluxes on the HTS coil: required cooling power to maintain 77.5 K (black), heat flux from current leads (orange), heat flux from infrared radiation (red), instantaneous solar power generated by the solar array (yellow). Temperature distribution of d) the HTS coil at 1 orbit; e) the CubeSat at 0.5 orbits, and f) the CubeSat at 1 orbit.

The instantaneous solar power peaks at 8 W, and the time average is 2.8 W. If the heat load on the coil were reduced to 0.5 W, a cryocooler with 20% of Carnot efficiency would require approximately 6.5 W of continuous input power. Therefore it is clear that there is a power deficit, to which potential solutions are: to choose a more favourable orbit in terms of solar power; to increase the size and therefore solar panel surface area of the satellite; or to equip the satellite with deployable solar panels, such as a “turkey tail” solar panel array [15].

4. Conclusions
A digital test environment for thermal management of satellites in low Earth orbit was created. This consisted of an orbital mechanics solver incorporating an incident radiation ray-tracing algorithm and a finite element model which calculated local heat transfers and absolute temperatures. A 3U CubeSat equipped with an HTS coil was tested digitally in a low Earth, sun-synchronous, partially shaded orbit. The heat load on the cryocooler cold head at 77.5 K varied between 3.4 W and 4.2 W, which is too high to be extracted with a cryocooler powered by non-deployable solar panels on a 3U CubeSat. However, 95% of the heat load on the coil was from Joule heating and thermal transfer in the copper current leads. This heat source could potentially be eliminated through the inclusion of a space qualified HTS flux pump, but further work is required to understand the resultant heat load from such a device. The radiation heat load on the coil could be reduced by the inclusion of MLI. Even with a reduced heat load on the coil, the satellite will
be underpowered. Suggested solutions are to increase the size of the satellite, or to include deployable solar panels in the design.

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