Fox-1 Thermal Design - Update

Presented by

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Abstract

The very dramatic reduction on the size of electronics and their concurrent increase in complexity and capability are making possible multi-function satellites in very small volumes. The CubeSat program was born as this complex electronic environment was birthing. CubeSat is a small, 10cm cube 1/12 the volume of the previous “small” AMSAT Microsat program of 1990. Owing to the small size and mass it has become financially possible to launch these satellites on small budgets. AMSAT is now in the process of exploring this field and size of satellites as finding launches for our previous designs has become nearly impossible.

While the size of these satellites has been reduced, the complexity of the thermal design and the supporting analyses has not really diminished at all. This paper will document the thermal design and analyses of the AMSAT Fox-1 program. This paper is an update of the one presented last year at the Space Symposium.

About the Author

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Dick has been participating in the thermal and mechanical designs of AMSAT satellites since 1978 starting with the late lamented Phase 3A; then to Phase 3B – AO-10; and to Phase 3C – AO-13. He conducted a study effort in the Geosynchronous Phase 4 effort before working feverishly on the 27 month Microsat Program culminating in 1990 with the quad launch of AO-16, DO-17, LO-18, and WO-19. Attention was almost immediately then turned to the ten year Phase 3D program resulting in the AO-40 satellite. In 2000 a study period commenced for a follow on in the Eagle study program which was subsequently abandoned. Other recent efforts have been dealing with ARISSat and now the CubeSat, Fox-1.

1 Introduction

A CubeSat is a type of miniaturized satellite for space research that usually has a volume of one liter, standardized as a nominal 10x10x10cm cube, weighs no more than 1.33 kilograms, and typically uses commercial off-the-shelf (COTS) electronics components. Beginning as the brainchild of Prof. Bob Twiggs, KE6QMD, AMSAT member, of Stanford University in 1999 and promulgated in conjunction with California Polytechnic State University (Cal Poly) the CubeSat specification was developed to help universities worldwide to perform space science and exploration.

The term "CubeSat" was coined to denote nano-satellites that adhere to the standards described in the CubeSat design specification. Cal Poly published the standard in an effort led by aerospace engineering professor Jordi Puig-Suari. The specification does not apply to other cube-like nano-satellites such as the NASA "MEPSI" nano-satellite, which is slightly larger than a CubeSat. AMSAT has evolved its own initial specification based on the Cal Poly document.[i] This AMSAT document is ITAR controlled and is not a generic CubeSat document.

With their relatively small size, CubeSats could each be made and launched for a 2013 estimated cost of under $100,000 to LEO for a 1U model. This price tag, far lower than most satellite launches, has made CubeSat a viable option for schools and universities across the world. Because of this, a large number of universities and some companies and government organizations around the world are developing CubeSats — between 40 and 50 universities were developing CubeSats in 2004, Cal Poly reported.

This paper will examine the thermal environment of a CubeSat and its thermal design to insure workable temperatures for the spacecraft electronics.

2 Mechanical Arrangement

From a thermal standpoint the only part of the AMSAT specification that is of thermal design interest is that of the operating temperature: -40°C to +50°C. While this seems a bit stringent, one of the goals of this thermal design effort is to provide more suitable “arm-chair” conditions for the installed equipment in Fox-1.

The Fox-1 external mechanical arrangements are shown in Figure 1. This structure is covered on all six sides with solar panels, each composed of two large UTJ solar cells of ≈28% efficiency. The operational electronics Printed Circuit Boards [PCB] are arranged in a stack, Figure 2 that is located in the central portion of the spaceframe, Figure 3, providing a centralized placement for the battery mass. This arrangement is to comply with the requirements that the Center Of Mass [COM] is within 2cm of the geometric center of the allocated volume for the spacecraft.

Panels on the sides of a CubeSat are allowed to protrude up to 6.3mm in certain proscribed areas of each side. The outer 8.5mm of each side of the spaceframe is reserved for a ***Rail***, running in the Z direction, reserved for the launch system, with proscribed finish and aluminum oxide coating. The AMSAT design sets aside 9.5mm for each rail. That, then, allows for an 81x108mm PCB-based solar panels on the X and Y sides and a nominal 100x100mm solar panel PCB on the Z sides. These Z-side dimensions seemingly imply space left over on the Z panels but there are other functions that must be accommodated in that space, such as the separation switch, separation pads and their push-off springs, and U and V band antennas. With all of these functions using spacecraft external area, there is just no room left over for any kind of thermal control surfaces. Essentially the entire spacecraft surface is spoken for and with materials with preset radiative properties. We may be able to sneak some solar-absorbing gold plating on spare areas of the solar panels.

While it appears that there is room left over on the solar panel PCB, the high-efficiency triple-junction solar cells shown are the largest practical at 68.9x39.5mm. Manufacturer's catalogs show some larger cells that would more adequate fill this PCB area, but those cells are not really a standard product and would be <u>very</u> costly to purchase and implement in this design.

3 Heat Sources

By far the single most profound heat source for a spacecraft is the Sun, providing a 6000K radiant heat source of from a distance of 149Mkm (1 AU). As such it provides heating in the visible and near infrared spectrum, 0.3μm to (about) 4 μm. The response of materials to this radiant source is measured in terms of their non-dimensional Solar Absorptance, α_S, property. The α_S is in deference to a material's radiation response to radiation in the far infrared, in the range of 4.0 μm to 25 μm, called the Emittance, or ε_{IR}, property. Compounding the solar heating issue is the fact that there are a range of values of the solar "constant", going from a "cool" sun, at 0.13081W/cm^2, to a "warm" sun, at 0.13995W/cm^2. A nominal value of solar heating of 0.13660W/cm^2 is often used.

A second heat source to a spacecraft is that of the reflected solar radiation from the illuminated side of the earth, called Albedo with maximum values ranging from 0.05516W/cm^2 to 0.03530W/cm^2. Spacecraft surfaces respond to lower heating values of albedo radiation in the same manner as to solar radiation. Albedo radiation is a property that must not be ignored as it can be very useful to a spacecraft. Albedo radiation is an orbital geometric heating property that depends where a spacecraft is located over the solar illuminated side of the earth and the orbital altitude.

The third radiant heat source for a satellite, especially for those in Low Earth Orbit, or LEO, is that of the planetary far infrared emission. The earth is a heat source of approximately 290K providing far infrared heating values of 0.01891W/cm^2 to 0.02616W/cm^2. The amount of this radiant source is pretty much a constant value for a satellite in circular orbit but is a changing value for satellites in elliptical orbit owing to the pure geometry of how much of the 4π steradian, sr, of the solid spherical angle of space sphere around a satellite is filled by the earth. For satellites in High Elliptical Orbit, HEO, the planetary emission heating value is generally so small that is ignored.

The fourth heat source for a satellite is that of internal dissipation of the electronics; see **Table 1** for the cold case and **Table 2** for the hot case. This depends upon how much of the incident solar, and albedo radiation is converted to electrical power then consumed (dissipated), by the electronics and thence converted to heat. In many satellites the electrical dissipation is often considered a nuisance but in Fox-1 it turns out to be a beneficial property. The battery dissipation is that of specific thermostatically controlled heaters on the cells to keep them warm enough to be able to accept charging.

Table 1 Cold Case, Tx Off, 33% Eclipse PCB Power Dissipations

PCB	Dissipation, W
Tx	0.164
Rx	0.067
IHU	0.090
PSU	0.010 - 0.043 (in sun)
Battery	0.010 – 0.210
Experiment	0.000
Total	0.341 – 0.574

Table 2 Hot Case, Tx On, Full Sun PCB Power Dissipations

PCB	Dissipation, W
Tx	0.290
Rx	0.067
IHU	0.090
PSU	0.370
Battery	0.010 – 0.210
Experiment	0.100
Total	0.927 – 1.127

4 Heat Sinks

To get rid of the heating caused by the above noted sources requires a heat sink. The ultimate heat sink for a satellite is that of radiation to the "cold black space" envelop surrounding the structure. In reality the space heat sink is not at absolute zero, but actually at about 4 Kelvin as a residual from the originating big bang. From a thermal standpoint of a satellite operating from 250 to 310 Kelvin there is not much difference in the space heat sink from 0 Kelvin, thus the 4 Kelvin is commonly ignored.

Internally in the Fox-1 spacecraft the effective heat sink (by conduction and radiation) for the electronics is the spaceframe. I shall not qualify this situation further until I get into the discussion of the thermal design as there are some things that are different in Fox-1 than I have seen in any of our prior AMSAT spacecraft.

5 Thermal Control Methods

All spacecraft temperatures are controlled by the nature of their external coatings. The values of absorptance, α_S, and emittance, ε_{IR}, are very important and are determined by careful measurement. Most materials have been so characterized and books have been written to document these properties. Prior to the advent of our interest in spaceflight, such documentation was generally not very well understood, available, or documented. Sixty years ago in colleges and universities the Heat Transfer courses generally only gave passing interest to the issues of radiant heat transfer. These days such oversight cannot be tolerated for aspiring satellite engineers.

In most spacecraft programs a significant effort is expended to insure that the coatings, both inside and outside the spacecraft, are well understood. In fact substantial control of the temperatures of a spacecraft can be achieved by such methods as the use of Multi-Layer Insulation, MLI; or solar absorbing coatings, generally metallic films; or solar reflecting coatings, such as second-surface mirrors of quartz or particular types of Teflon. Internally material coatings also play a part in the control of electronic box temperatures.

In the case of the Fox-1 spacecraft we do not have any liberties in adjusting the external coatings as the spaceframe size and features precludes any specific area for the use of thermal control materials save, perhaps, a small amount of gold plating on the soar panels. The eight 8.5x108mm rail surfaces must have a specified finish of aluminum oxide coating while the remainder of the area is spoken for by the solar cells and their PCB mounting boards. These surface properties are what they are and there is no latitude for any adjustments. This is a situation that I have not heretofore seen or had to work with as it places some rather serious restraints on the thermal design (sadly taking matters out of the hands of the designer).

As will be seen, the temperature variations of the spaceframe are substantial. Normally electronic modules or, in the case of Fox, just PCBs are mounted directly to the spaceframe. If that is done, as has been done in the past, the PCB temperature variations would also be substantial along with the spaceframe. These temperatures are somewhat in excess, both too warm and too cold, of what would be desirable for the electronics and so steps need to be taken to alleviate this situation if possible, as will be seen.

Other factors noted in dealing with the electronic PCBs is that we have electronic functions on a single small PCB that in the past required a whole sizable module to accomplish. In addition the power dissipation of the whole spacecraft is at the most a very few watts. To combat the spacecraft temperature variations, as reflected in the PCB temperatures, the mounting of the PCBs will provide a conductive coupling of the boards together as a stack as shown in Figure 2 and then provide as much thermal isolation of this stack from the spaceframe as possible, through the use of Delrin plastic mounting blocks, so that the limited power dissipation of this PCB stack can be used for self-heating to provide a more benign temperature environment for these electronics. This has been seen analytically and needs to be capitalized for the flight hardware. This whole process needs to be approached carefully. This process affects a number of factors, such as: PCB size; PCB stack assembly; PCB stack mounting to the spaceframe; and the PCB stack radiant heat transfer to the interior of the spaceframe.

Adverse factors affect how well this PCB stack is thermally isolated from the spaceframe. In addition to the overall thermal conductance of the Delrin mounting blocks, we will have sizable conductance of the electrical connections between the solar panels and antenna to the PCB stack. While numerically these conductance values look small, their influence is profound. For example, the following **Table 3** values have been estimated and used in this modeling:

Table 3 PCB Stack to Spaceframe Thermal Conductance Values

Element	Conductance, W/℃
Delrin Mounting Blocks	0.009435
±X Solar Panel Connectors	0.00330
±Y Solar Panel Connectors	0.00551
±Z Solar Panel Connectors	0.00342
Tx & Rx Coaxial Cables	0.00330
Total Conductance	0.02496

6 Analytic Methods

The first steps of building, or composing, a thermal analytic model is to create a numeric geometric model for radiation analysis, both for internal spaceframe radiation and for the external orbital heating from the sun and earth. Creating the required numeric model calls for a close look at the mechanical design such as shown in Figures 1-3. For Fox-1 we see that it is composed of a collection of flat plates whose geometry can be numerically described and located in the relative spacecraft coordinate system. The spacecraft can then be geometrically located in the orbital coordinate system.

For this radiation analytic effort we employ a suite of Monte-Carlo ray tracing analysis software contained in a package called NEVADA. For the internal radiation evaluation RENO software (part of NEVADA) is used to report the radiant interchange factors while for the orbital heating of the solar radiation and earth emissions, VEGAS software is used. Both of these programs use the same, or similar, input files. Models can be assembled using somewhat expensive automated software, but we have felt that these steps are not needed. Most automated programs create rather large and highly detailed models that have much more complexity than is justified by the thermal needs of the structure. These programs provide output files that are used directly with the SINDA thermal analyzer, or with further text editing to create the needed files.

For the thermal analysis proper a different numeric model is created for use with the SINDA/G package. The spaceframe is composed of a description of the spacecraft surfaces as nodes with thermal conductors (either by thermal conduction or radiation) linking those nodes. Creating this type of model of a spacecraft greatly depends upon the experience of the analyst to insure that a meaningful model is assembled but one that is not so complex that it cannot be understood. Highly detailed models can be created but that added detail is of little value as there is, generally, little interest or value in knowing the temperature gradient across a surface. You can see that the KISS principle is at work here.

These software packages are not recommended for the uninitiated to use as I have found, even with many years (≈50) using them, which I have had to discover new features, capabilities and ways of doing things.

7 Data Handling

In the process of understanding the output data of this analytic effort, a series of Microsoft Excel spreadsheets (SS) were created. Such a tool has been shown to be needed to make sense of handling a large amount of output data. Despite the relative simplicity of the thermal analytic model there is a need to be able to clearly understand these results and to portray the results in a manner that allows the engineer to understand how model variations affect the end results.

This effort evolved further with an expansion of the SS that incorporated the input files for VEGAS and SINDA as well as their output files. This SS tool has helped us understand and document the analytic work flow and the overall analytic effort immensely. The expanded SS have also greatly helped by reducing the amount of manual labor in the extensive text editing needed in certain areas of the thermal analytic effort, as well as incorporating the very useful graphical presentations of output data. The SS effort has also made possible an understanding of how satellite motions can be modeled in the desired tumble motions and to properly code those into the VEGAS input file.

8 Analytic Results

The available spreadsheet results are usefully expressed in several charts to illustrate the Fox-1 temperatures and incident and solar cell generated power levels. Other papers will be used to discuss the power results as that subject will require some close attention to illustrate those issues. This paper will only discuss the satellite thermal performance and the resulting temperatures. This task is not as simple as it may seem as we have run analyses for many situations of orbital orientation, “warm sun and “cool sun” heating values and satellite orientation. This has resulted in quite a few models that were used to examine the boundaries of spacecraft operation. Providing just a couple of snapshot view of this data in this paper does not reflect all of the work done.

A further consideration is the modeling of electronic power dissipations of PCBs that have yet to be designed. These dissipations can then be considered as educated targets for the thermal model and the electronic designers both. The target for the average spacecraft power dissipation is to not exceed the minimum power generation of about 2.1W. For that this thermal model used the dissipation numbers of Tables 1 & 2 for the PCBs.

Early on in this effort it was realized that nominally connecting the PCBs to the spaceframe could bring some extreme, and undesirable, temperatures to the electronics. This resulted in assembling, in the thermal model, a conductive stack assembly of the PCBs and a minimum conductance to the spaceframe which has been a challenge for the spacecraft mechanical designer. This consequently resulted in a moderation of the PCB temperatures. These results are shown in Figures 4 Spacecraft Temperatures, 5 PCB Temperatures, and 6 Solar Powers, for the 33% maximum eclipse orbit, with minimum solar radiance (cool Sun), illustrating that PCB self heated temperatures ranged from about -11.3°C to -4.7°C with the battery thermostatically controlled between +7°C to +8°C. These temperatures are a bit cool but workable for the electronics.

Warmer results are seen in Figures 7 Spacecraft Temperatures, 8 PCB Temperatures, and 9 Solar Powers, for a full-sun orbit, with maximum solar radiance (hot Sun), showing the PCB stack running from +5.0°C to +6.7°C with the battery operating at+7.4°C again thermostatically controlled.

9 Future Work

The analytic results shown to date have promise for being able to offer a modest temperature environment for the spacecraft electronics. Along the way, a very capable set of analytic tools have been created to speed up the process of creating these analyses. It is clear that more work is needed before creating flight hardware. The added work needed will be discussed below.

Another issue that needs attention is that of the PCB dissipations. We have used one nominal set of estimated power values that may not be like the final values. Clearly the spacecraft must be able to operate within the boundaries of the available power generation. Having the high efficiency UTJ solar cells has made CubeSats possible in the first place, but that power is still highly limited and must be employed wisely.

The Fox-1 program so far has been aimed toward the Fox-1 mission, with a relatively simple set of electronics and with no deployable solar panels. For the future, however, we will be aiming for Fox-2 with its increased complexity and capability. Fox-2 may need to have deployable solar panels so as to provide the added power for the electronic payload.

One issue that has not been addressed is the variable power consumption than may be needed for different parts of an orbit as in eclipse. Other ultra-low-power operations need to be defined and quantified.

Another modeling variable is that not all of the solar power may be needed for the whole orbit, yet our current models unrealistically uses all of the power all the time that it is available. This modeling issue was addressed more than twenty years ago in the Microsat program. This FORTRAN code, which is still available, removed the rated generated power from the solar panels, operated the necessary equipment and charged the battery until its rated energy is stored and then, with less than full power needed from the panels, lowered the power removed from the solar panels, effectively adding heat to those panels. Battery stored energy was removed as needed during eclipse. This code dynamically models the complete energy balance of the satellite.

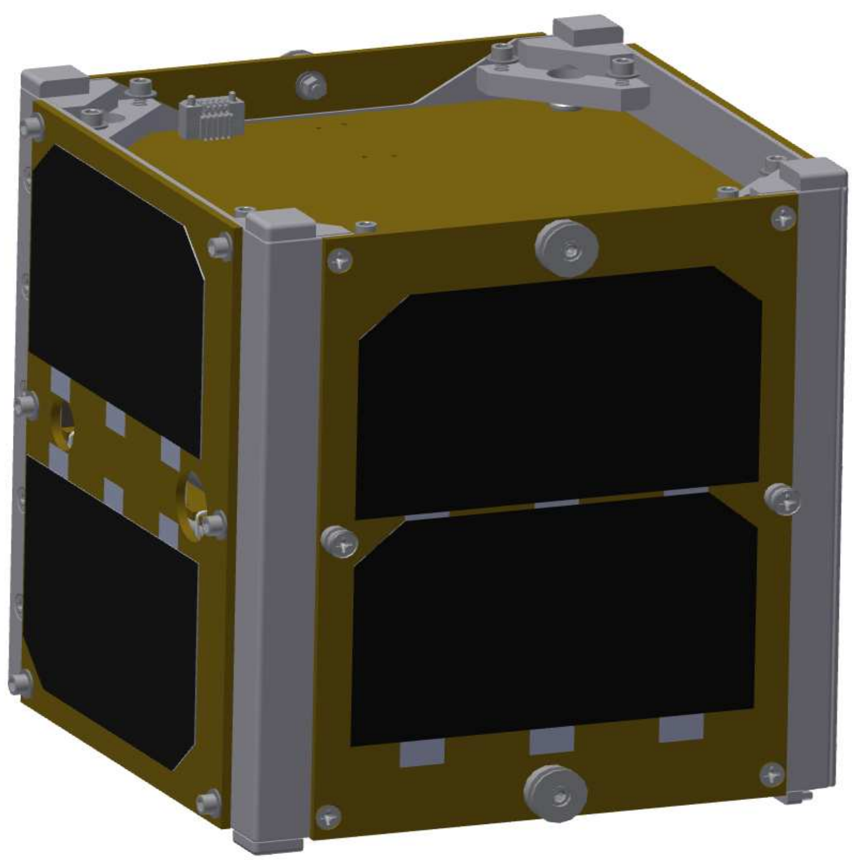

Figure 1 Fox-1 spaceframe assembly.

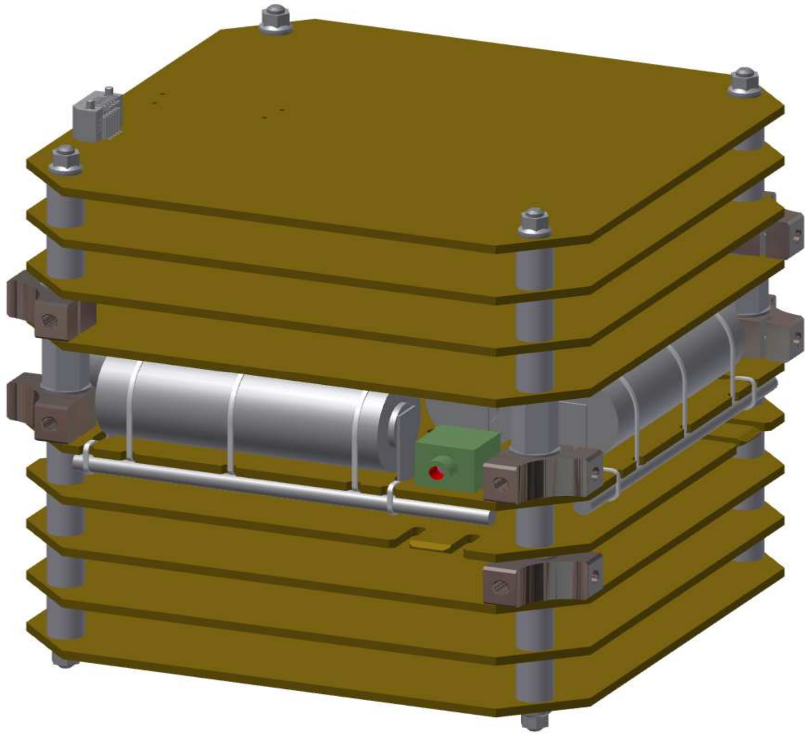

Figure 2 Fox-1 PCB Stack showing the battery cells in the center.

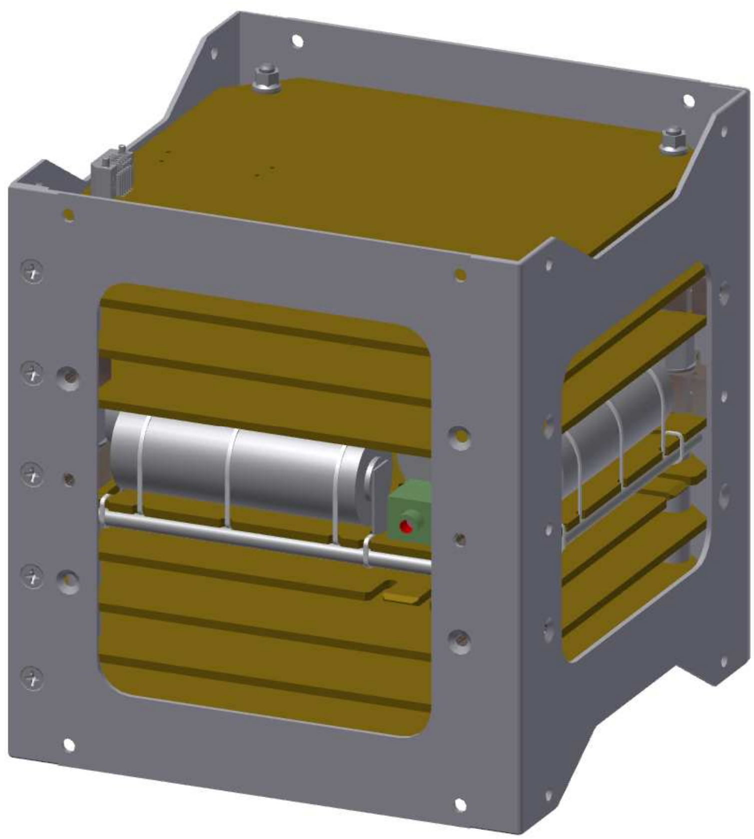

Figure 3 Fox-1 bare spaceframe with centrally mounted PCB stack.

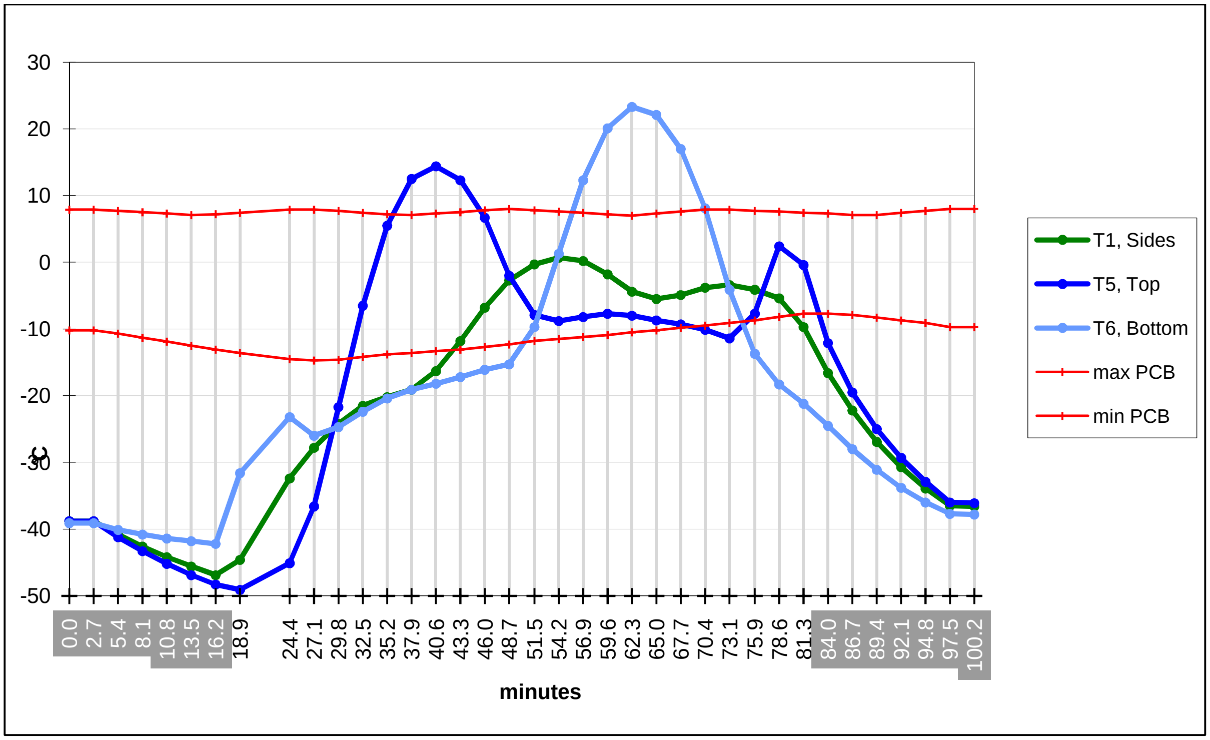

Figure 4 Fox-1 temperatures for33% eclipse orbit with thermally isolated electronics.

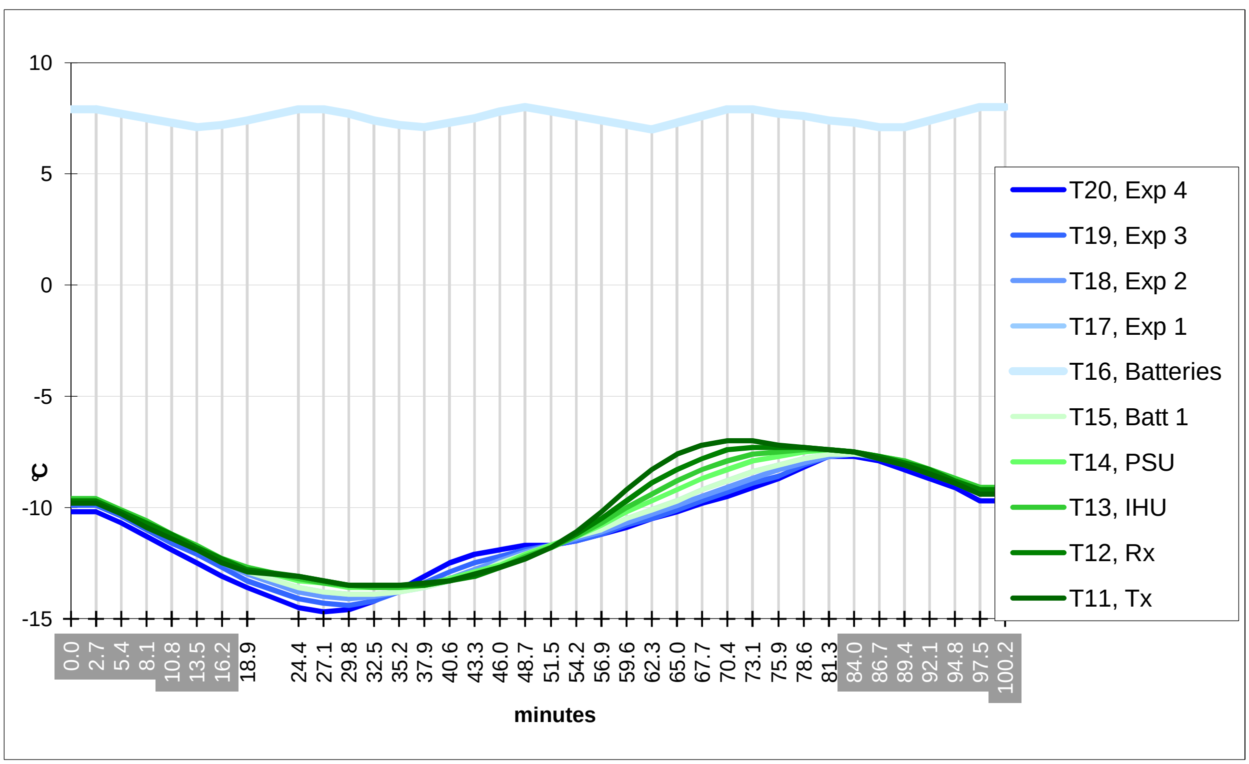

Figure 5 Fox-1 PCB temperatures for 33% eclipse orbit.

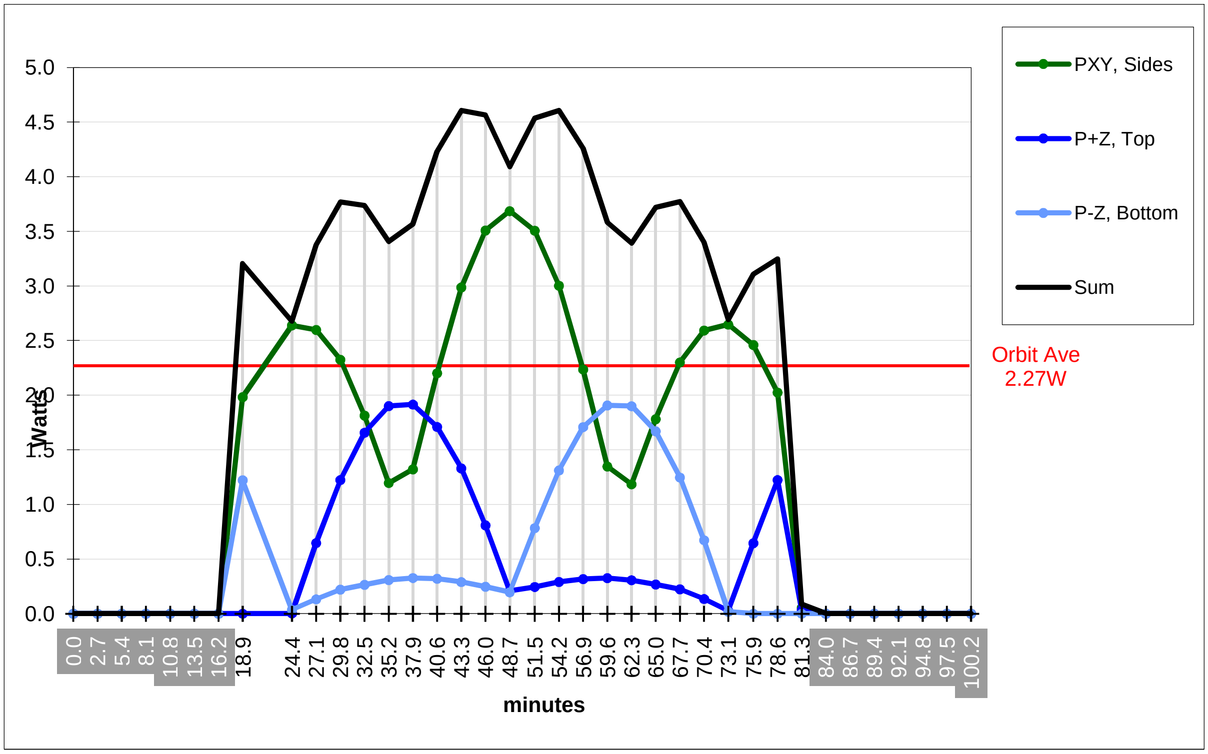

Figure 6 Fox-1 solar power generation by different panels for 33% eclipse orbit.

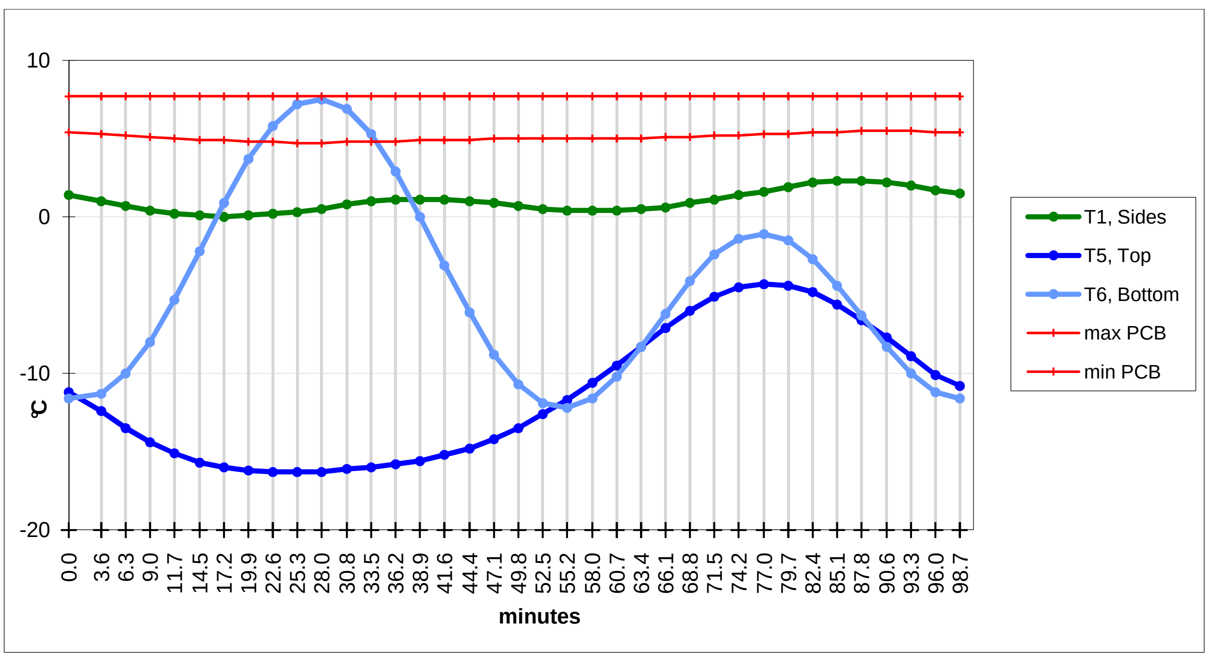

Fig 7 Fox-1 temperatures for full-sun orbit with thermally isolated electronics.

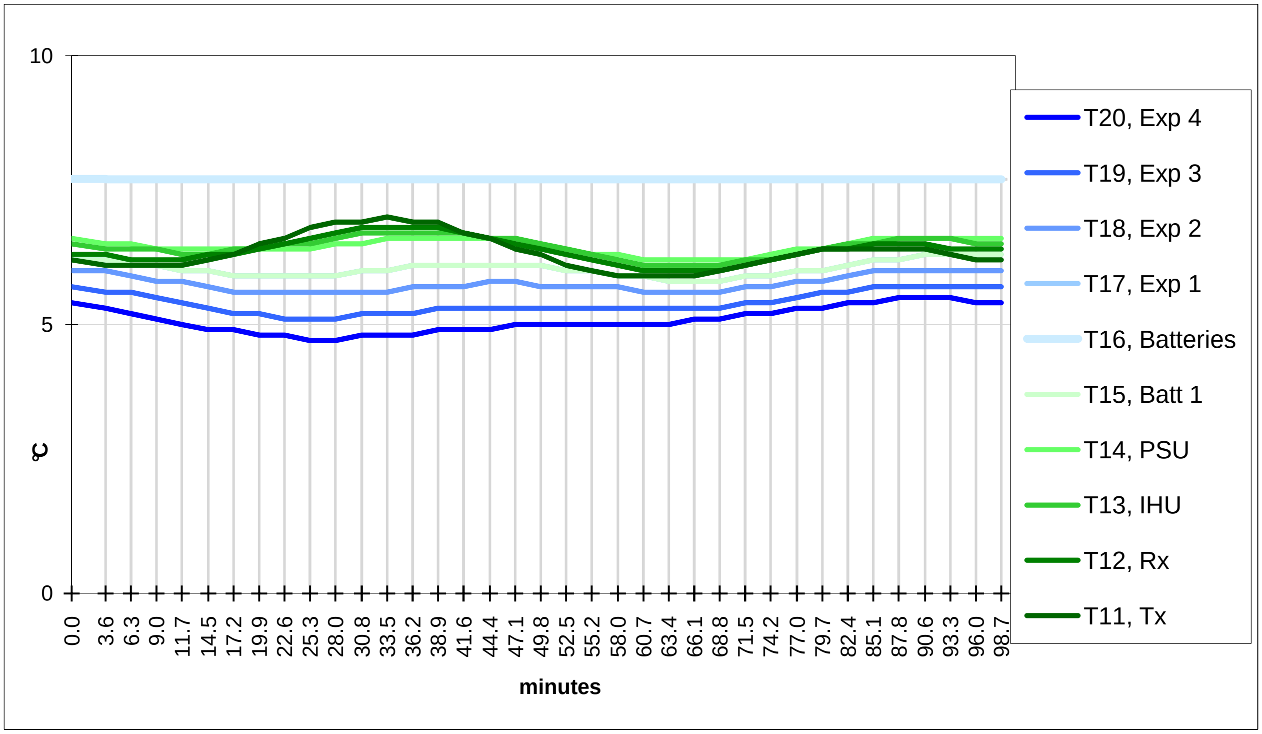

Figure 8 Fox-1 PCB Temperatures for full-sun orbit.

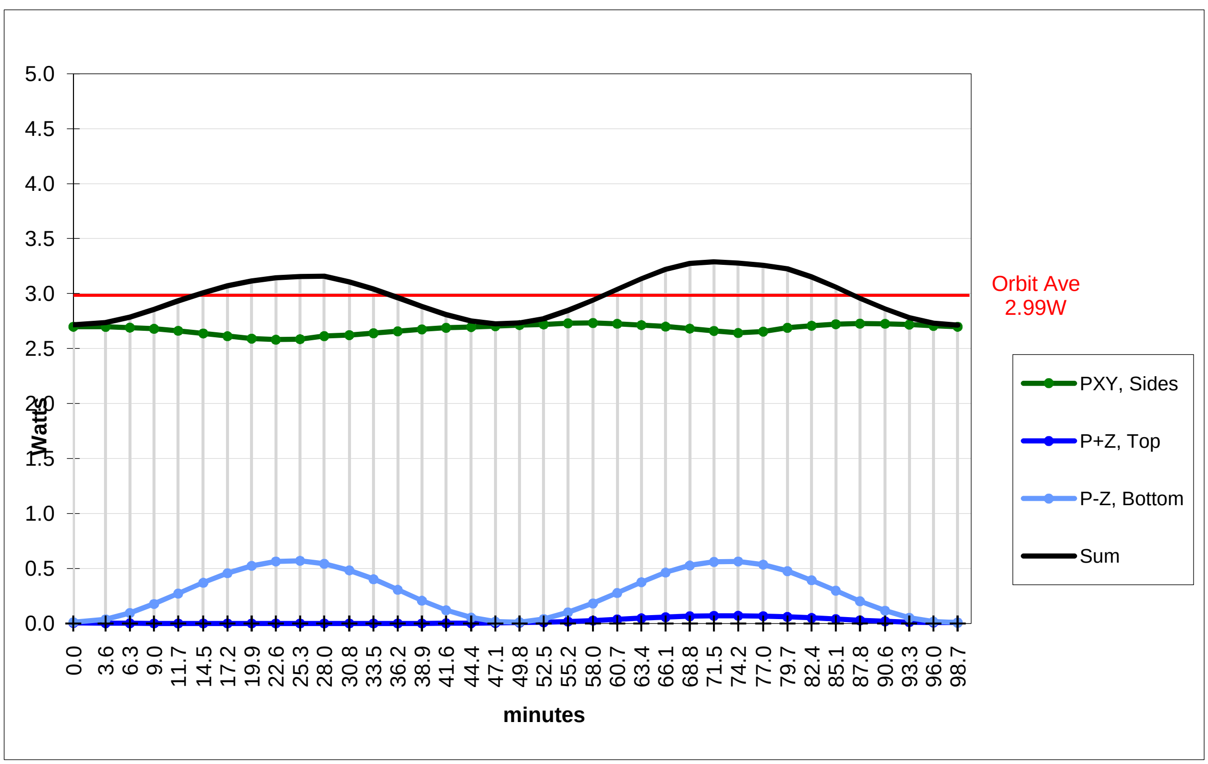

Figure 9 Fox-1 Solar power generation for full-sun orbit.

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[i] **AMSAT *Fox* - System Requirements**, Version 1.11, dated 02 August 2012.