THE SMALL ASTRONOMY SATELLITE (SAS) POWER SYSTEMS

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NASA's Small Astronomy Satellites were designed and developed by The Johns Hopkins University's Applied Physics Laboratory for use in the early 1970's. They were Scout launched, spin stabilized satellites intended for a near equatorial, 550 Km, circular orbit.

Since the spacecraft were designed so that their spin axis could be pointed in any direction in inertial space, an omni-directional solar cell array was required. An additional significant constraint on the solar cell array design was that the effect of atmospheric drag on the spacecraft stabilization had to be minimized. These requirements led to the need for an electronic battery charge control system with a wide dynamic range. The final design incorporated a moveable solar cell array with periodic control and a charge control system capable of accommodating large array and load power variations with low internal power dissipation.

The final power system design was intended to accommodate any small, spin stabilized spacecraft. Analyses and resulting hardware are described for the solar cell array and power system.

INTRODUCTION

The SMALL ASTRONOMY SATELLITE (SAS) program was started in late 1966. At that time NASA/GSFC requested the Applied Physics Laboratory to design a spacecraft from which a series of astronomy experiments could be conducted. The final design concept was a cylindrical control section capable of having a separately developed experiment attached. This control section or package provided structural support, attitude detection, attitude control, power, command, telemetry and RF systems for the spacecraft. It was a basic structural and electrical system which could be easily adapted to various astronomy experiments without a major redesign effort and one that could be launched on a small vehicle such as the Scout rocket. The Applied Physics Laboratory was responsible for the design and fabrication of the control section as well as the integration of the subsystems, including the experiment, and for the qualification and launching of the spacecraft.

Three spacecraft were flown. SAS-A carried an X-ray experiment designed by American Science and Engineering of Cambridge, Massachusetts. SAS-B was equipped with a gamma ray experiment designed by NASA's Goddard Space Flight Center. SAS-C was a second generation spacecraft with improved attitude control, more power and enhanced data systems. It carried a more sophisticated X-ray experiment designed and developed by the Massachusetts Institute of Technology. Therefore, there were two control section designs: one for SAS-A and B and another for the SAS-C spacecraft.

The SAS mission was to map the celestial sphere for X-ray and gamma ray sources. Therefore, the spacecraft were all spin stabilized so that the sensors, which pointed normal to the spin axis, could scan a band of the sky with each revolution. So that the entire sky could be eventually investigated, the attitude control system was equipped with a magnetic torquing system capable of pointing the spin axis anywhere in inertial space. This required the solar cell array to be omnidirectional, capable of supplying the spacecraft electrical load for any position of the spin axis.

Each of the spacecraft were launched into circular, near equatorial, orbits of approximately 550 km altitude. They were launched from the San Marco Equatorial Range in Kenya, East Africa in the early 1970's. The launch site and the relatively low orbit were chosen to maximize the allowable spacecraft weight. However, the low altitude led to additional constraints on the solar cell array design. There was concern that an imbalance in the atmospheric drag (and to a lesser extent the magnetic forces) on the solar panels would cause non-uniformities in the rotation rate of the spacecraft, producing errors in the experimental measurements. Therefore, it was decided to position opposite solar panels in the same plane and to use a solar cell layout which would produce equal and opposite magnetic moments.

THE SOLAR CELL ARRAYS

Therefore, the general requirements of both solar cell array designs can be summarized:

1. Omnidirectional power generating capability.

2. Minimize the disturbing torques on the spacecraft.

   a. Minimize the aerodynamic torques. (Opposite or co-axial solar panels to be coplanar)

   b. Minimize magnetic torques. (Identical solar cell patterns on opposite solar panels)

3. Minimize weight.

4. Maximize end of life (EOL) power.

SAS A/B SOLAR CELL ARRAY

The SAS A/B solar cell array consisted of four solar panels, with cells on both sides, which were hinged to the spacecraft, stowed along the last stage of the Scout Rocket during launch (Figure 1) and deployed into fixed position for flight (Figure 2). Since co-axial panels were also coplanar, the area projected normal to the sun-line changed through a wide dynamic range with each spacecraft rotation for many sun-satellite orientations. This caused a variation in array current with a range that depended upon the angle between the spacecraft spin axis and a line to the sun (Psi). The current variation was an extreme of nearly ten amperes for angles of Psi=40 (or 140) degrees (Figure 3). Figure 4 shows the average, maximum and minimum solar array current of the main array as a function of Psi. The auxiliary array had a similar pattern and contributed an additional eight percent of the current. The angle between the solar panel normal vectors and the spin axis was selected to be 60 degrees in order to maximize the minimum average power from the array.

With such a wide possible variation in the generated array current, it became necessary to design a charge control system capable of shunting up to 10 amperes (approximately 110 watts). Linear (dissipative) shunts were used with the characteristics shown in Figure 5. Four of these three ampere shunts were used in parallel for a total capacity of 12 amperes to form the external shunt. A total of eight shunt drivers, containing the power transistor, were used, four for each of two redundant systems. To dissipate the heat external to the spacecraft, the shunt drivers were mounted on the ends of each of the four solar panels and the corresponding resistor was etched from stainless steel and distributed over one side of a 1.3 square ft section of each panel (Figure 6).

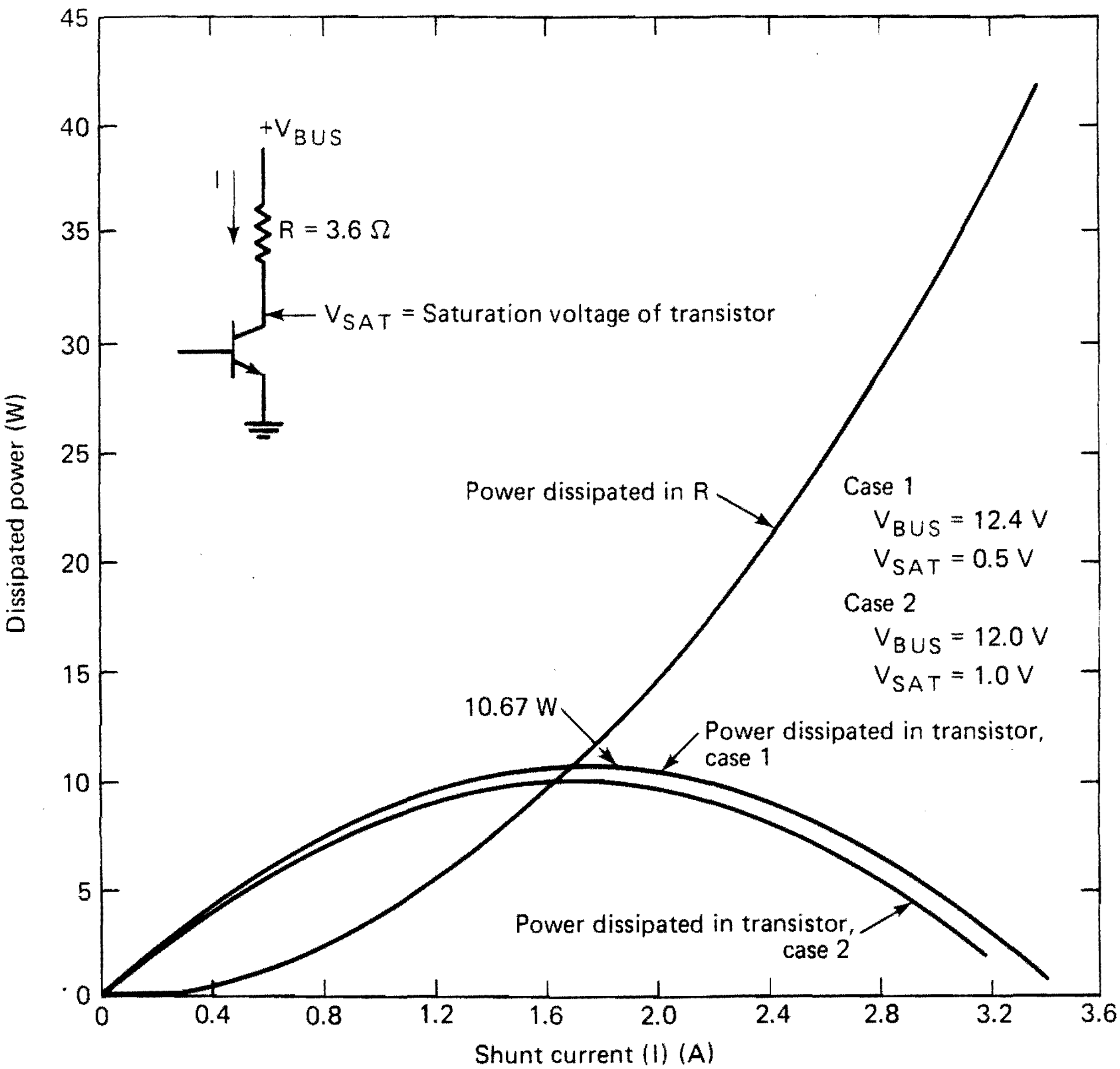
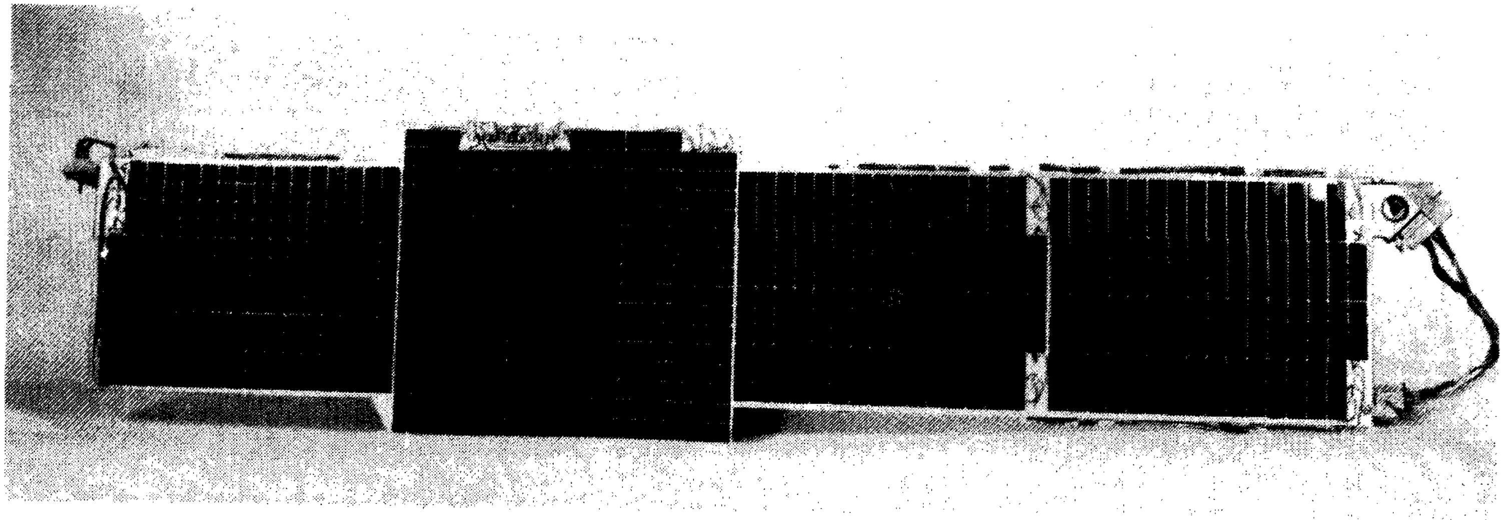


Fig. 5 Power dissipated in each external shunt element versus shunt current.



Solar panel with solar cells

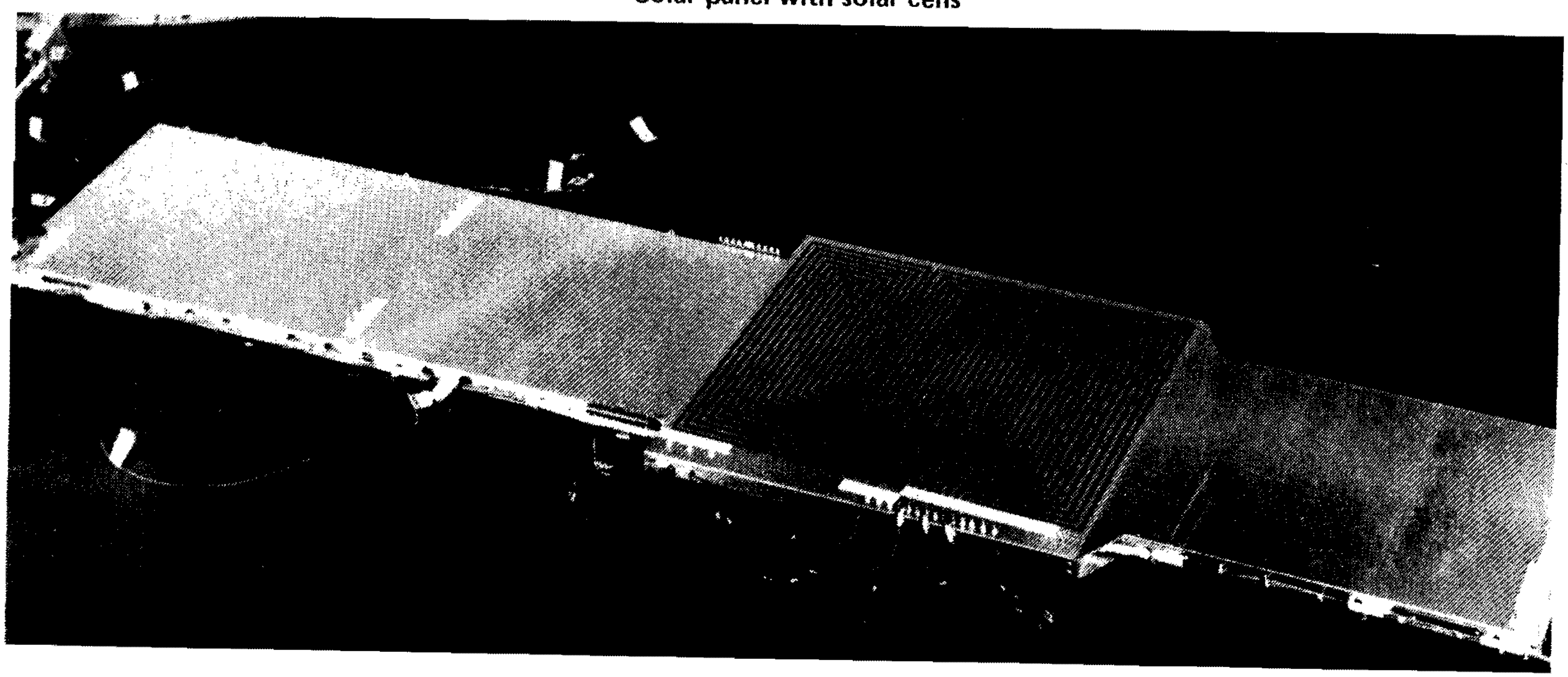


Fig. 6 Solar cell panel and substrate.

Also, a fifth, slightly smaller (~2.8 ampere) internal shunt was provided to maintain a minimum battery temperature of 55 deg F. Its shunt driver and distributed resistor were mounted on the main instrument deck.

SAS A/B POWER SYSTEM DESCRIPTION

A block diagram of the SAS A/B power system is shown in Figure 7. It consisted of the previously described solar cell array, a single nickel-cadmium battery of eight 6 ampere-hour cells connected in series, two redundant battery charge control systems, a low voltage sensing switch, power conditioning regulators and converters and a number of heaters. The system voltage varied from a maximum of 12.4 volts to a minimum of 8.8 volts with a nominal value of 10.7 volts. A summary of the weight and power required by the power system components is shown in Table 1.

The solar cell array consisted of three separate sections to provide the necessary power to the spacecraft, including the experiment. The main and auxiliary arrays supplied sufficient power in sunlight to sustain the 27-watt load and recharge the battery. The auxiliary array will recharge the battery if it is removed from the bus. The -Z solar cell array was mounted on the bottom of the spacecraft to supply heat to the experiment when the sun was near the -Z axis.

The spacecraft orbital period for this altitude is 96 minutes, thirty-six minutes in eclipse and 60 minutes in sunlight. For a near-equatorial orbit the eclipse time is almost constant. The 6A-H battery supported the 1.5 ampere-hour eclipse load and any periodic loads that exceeded the array capability in sunlight.

Two redundant battery charge control systems were used: the battery current and voltage limiter (BCVL) and the Charge Regulator And Monitor 2 (CRAM 2). The BCVL was the simplest of the two. Its function was to actuate the linear shunts to limit the battery charge rate to a fixed maximum (4 amperes) and to limit the battery charge voltage to a maximum that is a function of battery temperature (Figure 8). CRAM 2 provided the same functions as the BCVL, but it also incorporated an electronic coulometer to keep a running count of the battery state of charge. The coulometer could be commanded to operate open loop to monitor the state of charge of the battery so that it could be telemetered to the ground station. Or, it could be operated closed loop to reduce the battery charge current to a small 'trickle' charge rate when the battery reached full charge. The voltage limiting characteristics of either the BCVL or CRAM 2 would function even when the battery was removed from the bus (the 'Solar Only Mode'). Therefore, even if the battery were to fail, the spacecraft would function in sunlight as long as the array current exceeded the load current since the voltage limiters acted like large capacity zener diodes.

The purpose of the Low Voltage Sensing Circuit (LVSC) was to protect the battery from cell reversal if its charge should be depleted due to an overload or insufficient array current. It also defined the lowest bus voltage at which the loads were operated. When the bus voltage dropped below 8.8 volts (1.1 volts per battery cell), the low voltage sensing circuit caused the battery and all non-essential loads to be removed from the bus. With the command system and a few 'keep alive' heaters as the only loads, the now isolated battery would then be recharged from the auxiliary solar array.

SAS-C SPACECRAFT

The SAS-C spacecraft was a second generation design of the Small Astronomy Satellites. It was intended to be a universal satellite, capable of accommodating a wide range of experiments in any orbit for either earth pointing or spin stabilized missions. It was physically larger than the SAS A/B spacecraft. Its experiment had more sophisticated X-ray detectors and the supporting subsystems were also enhanced. The attitude control system had more precise attitude determination and more flexible control over the measurement procedures. A closed loop control system enabled the spin rate to be controlled more precisely by using a rate gyro to sense the spin rate. A delayed command system and variable format telemetry were also incorporated. The required power increased from 27 watts to 60 watts, leading to a re-design of the power system.

SAS-C SOLAR CELL ARRAY

The requirements for the solar cell array design were much the same as they were for SAS A/B. However, the need for increased power, coupled with the requirement for a universal satellite led to a unique solar cell array configuration (Figure 9). Co-axial panels were again co-planar. However, each panel was broken into three inter-hinged segments, precluding the necessity of interfacing with the rocket. It was also a step in the direction of modular design since from one to six segments could be used as needed. In the launch configuration the three panel segments were folded together, along the side of the control section and held in place by cables that were terminated by despin weights (Figure 10). After injection into orbit the despin weights and cables were released, allowing all four panels to deploy to the horizontal position. By sending a ground command to an onboard stepper motor system each co-

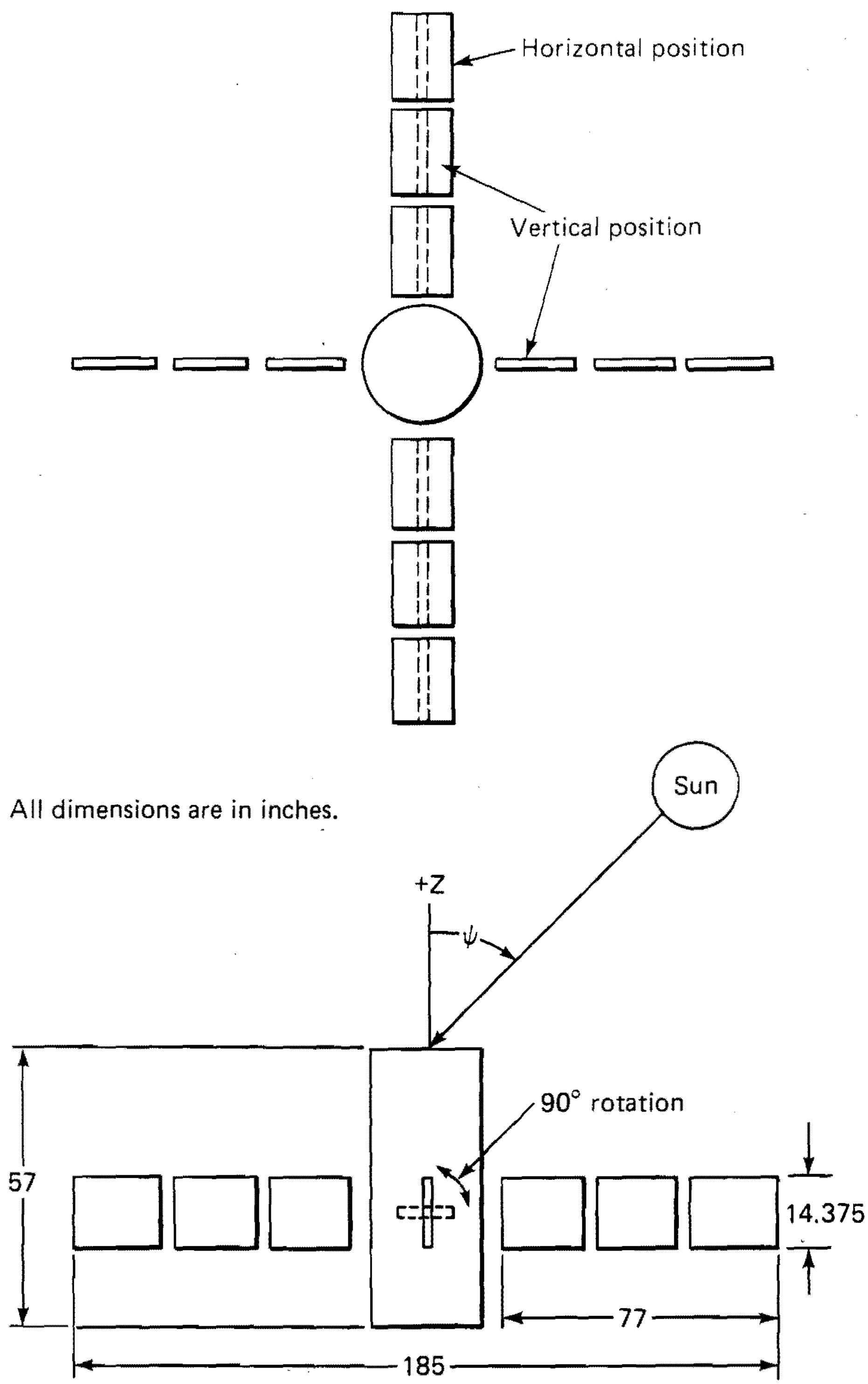


Fig. 9 SAS-C solar panel configuration.

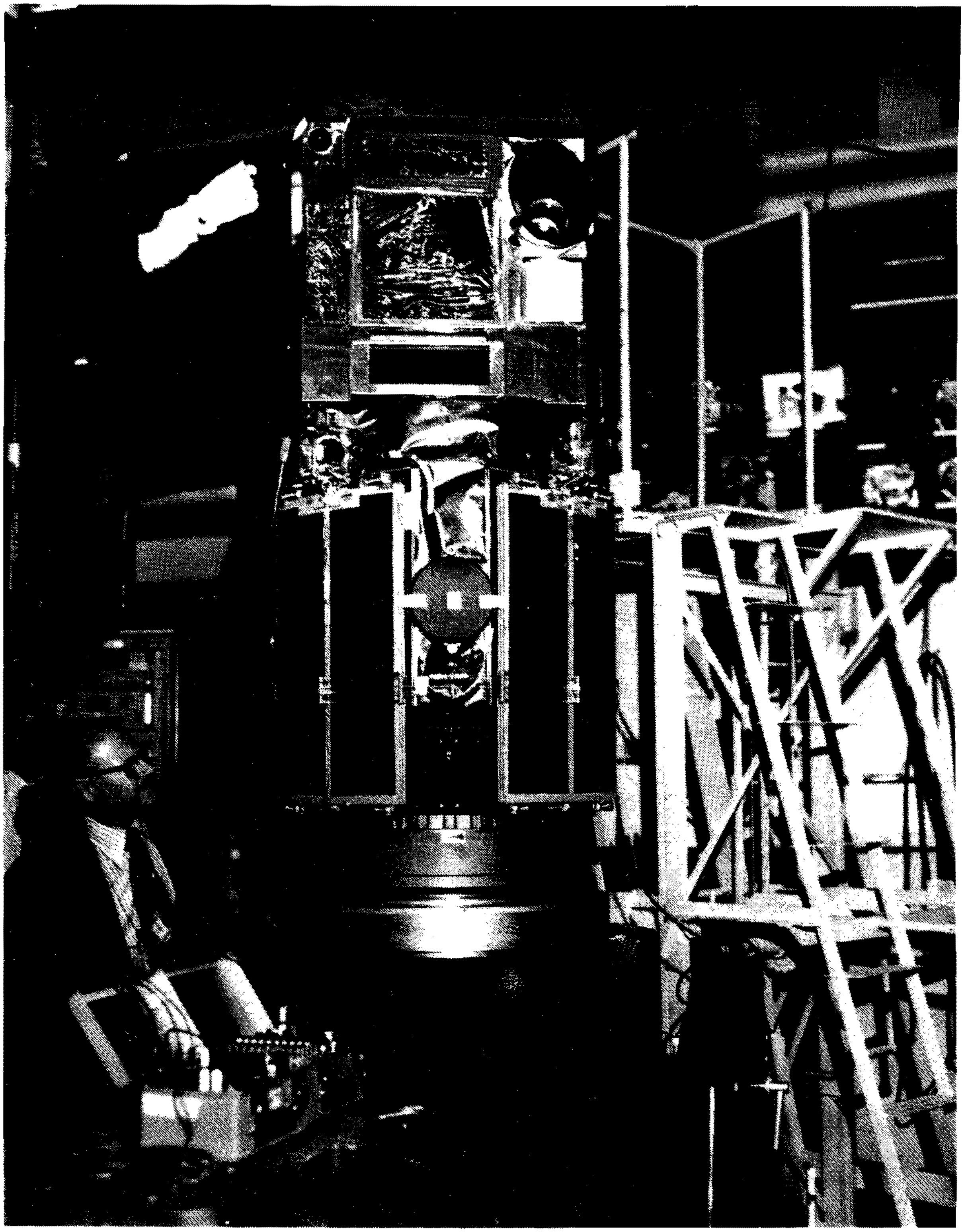


Fig. 10 SAS-C spacecraft, launch configuration.

axial pair of panels was then rotatable about its long axis, allowing for periodic optimization of the solar array configuration.

Although an infinite number of configurations are possible, the three basic orbital configurations are shown in Figure 9. The available power corresponding to these configurations are shown in Figure 11. After release of the despin weights, the four panels were all deployed to the horizontal position, providing up to 160 watts when the sun is near the Z-axis. This was plenty of power to supply the electrical load and to energize heaters on the end of the spacecraft facing away from the sun. Extra heater power may be required for some missions when the sun is near the spin axis since there is a high thermal resistance at the mounting interface between the experiment and the control section. However, if large amounts of heater power are not required, then two panels could be turned to the vertical position (or any position in between) to provide approximately 65 watts whenever the sun is within 55 degrees of the Z-axis. When the sun is within 35 degrees of the satellite's equator, all four panels should be turned to the vertical position to maximize the power. For all configurations, distinct minimums of 63 to 65 watts occur at angles of PSI equal to 55, 90 and 125 degrees.

Therefore, periodic rotational control of the solar panels provided substantial flexibility in the choice of power availability. It also allowed a crude control of the average array current (a fall-back mode of operation) if the charge control system had failed. And, since the stepper motors position the panels to the proper angle, the primary hinges were a relatively simple design.

Figure 12 shows the extremes of solar panel temperature for these panels in a 550 Km, circular orbit and Figure 13 shows the solar cell voltage - current curves for these extremes of temperature. The solar cell circuits and associated wiring were arranged so that their resulting magnetic moments tended to cancel.

SAS-C POWER SYSTEM DESCRIPTION

A block diagram of the SAS-C power system is shown in Figure 14. It consisted of the solar cell array, a single nickel-cadmium battery of 12 nine ampere-hour cells connected in series, redundant CRAM 3 charge control systems, a low voltage sensing switch, power conditioning regulators and several 'loads off' heaters. The system voltage varied from a maximum of 18.6 volts to a minimum of 13.2 volts with a nominal value of 16.1 volts. A summary of the weight and power required by the power system components is shown in Table 2.

The solar cell array was limited by a redundant, battery charge control system (CRAM 3). Like its predecessor on SAS A/B, CRAM 3 sensed the battery current, voltage and temperature redundantly. It operated either of two possible voltage limiters, current limiters, and optional coulometers. The voltage limiters each had two voltage-temperature limit transfer functions, shown in Figure 15, selectable by ground command. Either of the two redundant coulometers could be deselected (turned off), operated in a simple monitor mode to track the battery state of charge, or actively operated as part of the charge control system in parallel with the voltage limiter. In this latter mode, the percent recharge was selectable by ground command. Figure 16 shows a typical orbital cycle when the voltage limiter and coulometric charge control functions were operated in parallel. During battery recharge, the voltage limiter first caused a reduction in charge current. However, after the proper recharge was reached (101 to 125% of the previous discharge), the coulometer caused the battery charge rate to be reduced to a small 'trickle' charge rate of C/90.

The big improvement in the SAS-C charge control system over that of SAS-A/B was its ability to control much larger solar array currents with much lower power dissipation as well as reduced variations in the power dissipation. This so-called 'nondissipative' charge control system was achieved by segmenting the solar cell array into 24 parallel circuits, each of which was controlled by a transistor switch in the solar array controller (SAC). When the SAC's controlling transistor was off, it would allow the solar cell circuit to deliver its current to the bus as usual. However, when the current from the circuit was not needed, CRAM 3 switched the transistor to saturation, short circuiting the array, causing its removal from the bus with a low power dissipation equal to the maximum solar cell circuit current times the transistor saturation voltage.

Since the variations in the solar array current are relatively slow, this coarse 'digital' control of the solar array current was accomplished at a slow switching rate, depending primarily on the spacecraft spin rate. A small linear shunt, with a capacity of 1.5 solar cell circuits was employed as a vernier to provide fine, continuous control of the battery current. That is, a required change in battery current was provided by a corresponding, complementary change in the linear shunt current. When the linear shunt could not compensate for the required current change, either because it was full or empty, CRAM 3 would short or open digital shunt(s) until the dissipative shunt was back in the linear range. Note that the actuation of a digital shunt would not always result in a solar array current change since, at any given time, some of the solar cell circuits would be facing away from the sun. Therefore, the sequence of switching the circuits was structured to minimize the number of dark circuits that could be consecutively encountered when actuating the digital shunts.

Still, the switching time required for the digital shunts would have resulted in some low rate current and voltage transients on the bus when the battery was disconnected (solar only). Therefore, a small active ripple filter (ARF) was used to act as an enhanced capacitor (a current source or sink) to provide or accept any short duration bus current changes during the 'solar' only mode.

CONCLUSIONS

If this power system were to be redesigned for a spacecraft today, there are some things that would be done differently. The nominal bus voltage would be increased to 28 volts and electrical filters would be placed on the bus between the solar array and the spacecraft body to attenuate noise picked up by the solar panels. Also, the charge control system would be redesigned to take advantage of today's higher density chips and microprocessors, and FET transistors would be used for the shunts due to their low saturation voltage.

However, the following concepts would be retained. The solar panels would be rotatable, allowing periodic optimization, for any spin stabilized or earth pointing (one spin per orbit) spacecraft. The solar array and charge control system would be modular, allowing easy adaptability to different payload sizes. The solar cell circuits and wiring would be routed in such a way as to minimize the resulting magnetic field. The charge control system would be relatively non-dissipative, minimizing the thermal control problems associated with changes in spacecraft attitude and/or electrical load. Most spacecraft, especially scientific ones, do not need to operate 100% of the time. For these missions 'solar only' operation is genuinely useful and is a real alternative to battery operation. Therefore, I believe that a power system that is going to be useful for a large class of missions should include the ability to operate in sunlight without the battery.

Based on the SAS experience, there is a tendency to stay with a shunt type charge control system since it is simple, has low power dissipation and introduces little noise to the bus. Another advantage to the shunt type system over the series and pulse width modulated types is that in the minimum power condition at the end of life, the shunt system requires no power.

Table 1

SAS A/B POWER SYSTEM WEIGHTS AND POWER REQUIREMENTS

Item	Weight (Lb)	Power (W)
Solar Cell Array (17.04 Sq Ft)	26.9	N. A.
Battery (8 6 AH Cells)	7.1	3.75
Charge Regulator and Monitor (CRAM 2)	2.8	0.33
Battery Current and Voltage Limiter (BCVL)	0.1	0.05
Shunts (4)	1.8	N. A.
TOTAL	38.7	4.13

Table 2

SAS C POWER SYSTEM WEIGHTS AND POWER REQUIREMENTS

Item	Weight (Lb)	Power (W)
Solar Cell Array (30.0 Sq Ft)	35.1	N. A.
Solar Panel Rotation System	3.1	9.7 *
Battery (12 9 AH Cells)	13.1	5.0
Charge Regulator and Monitor (CRAM 3)	2.3	0.9
Solar Array Controller (SAC)	2.2	3.0 *
Linear Shunt and A. R. F.	1.3	0 to 30 * 8.0 *
TOTAL	57.1	5.9

* Only when energized. Normal load at end of life is 0 watts.

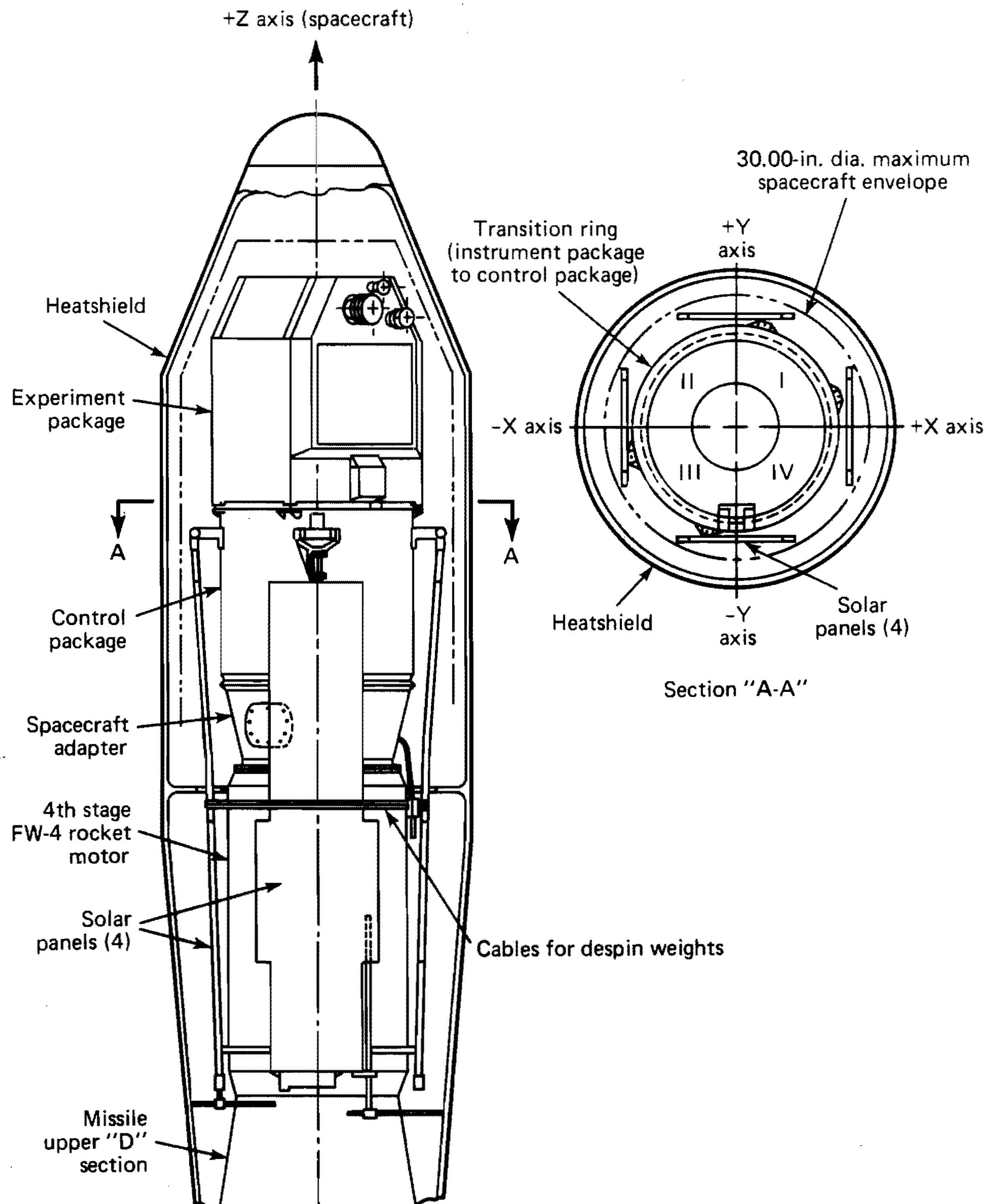


Fig. 1 Launch configuration SCOUT/SAS-A.

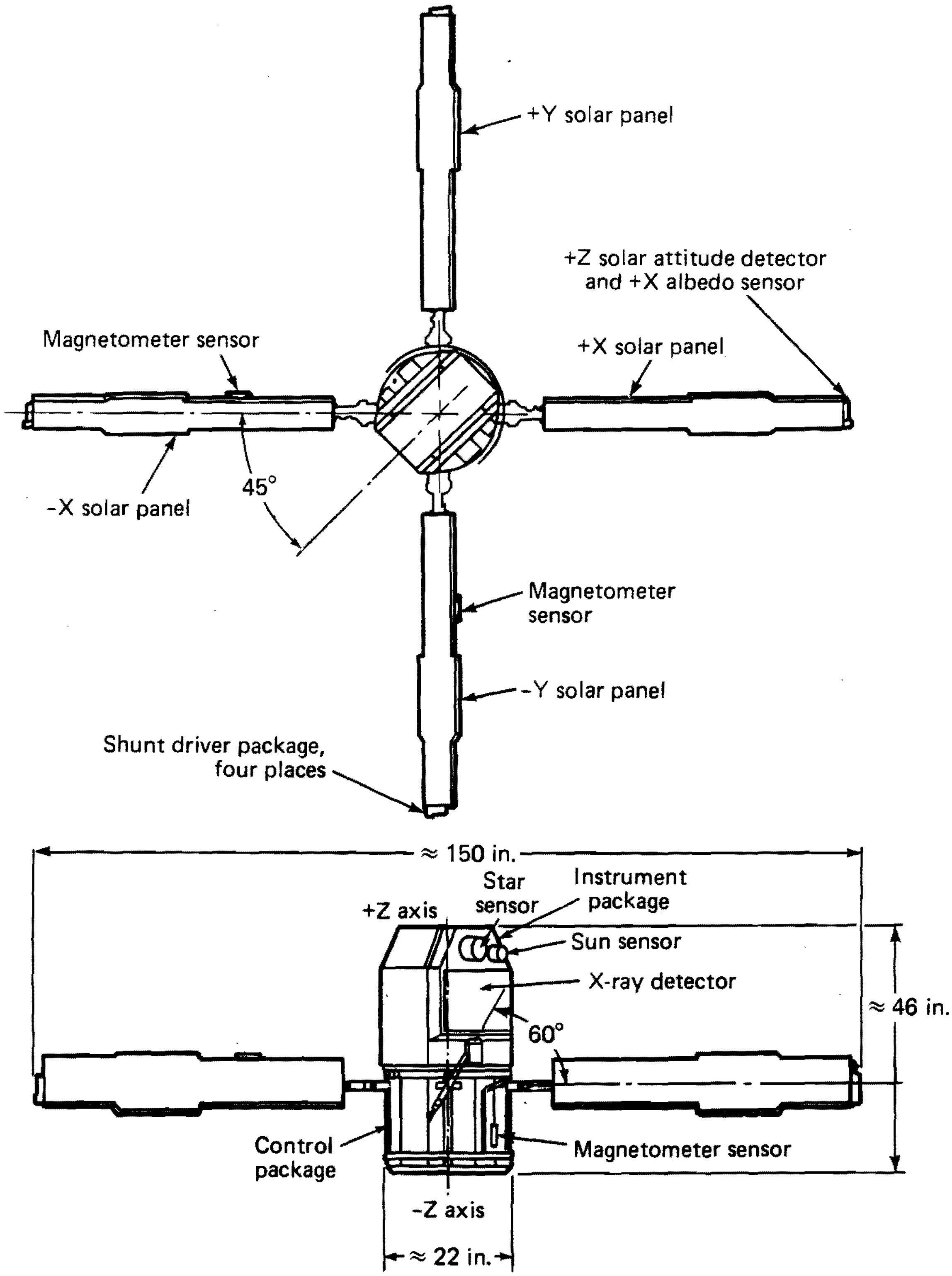


Fig. 2 SAS-A orbital configuration.

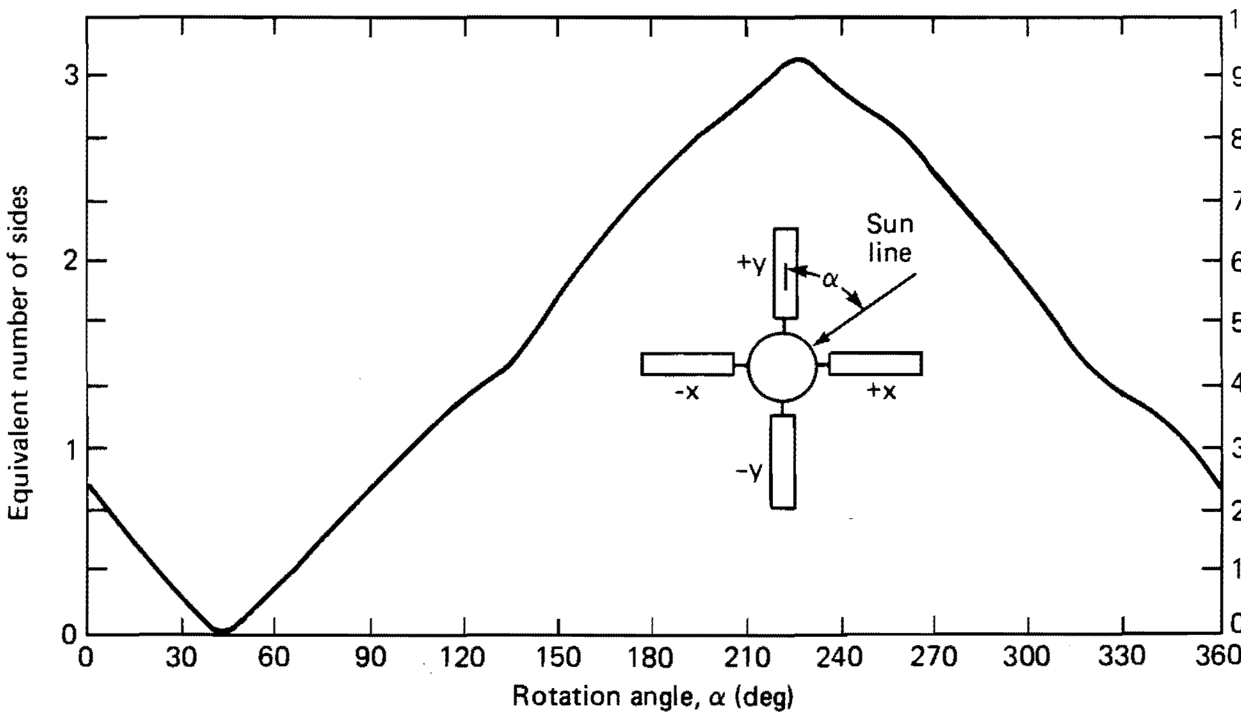


Fig. 3 $\psi = 40°$, solar array current (and equivalent number of solar panel sides at normal incidence to the sun line) versus rotation angle α.

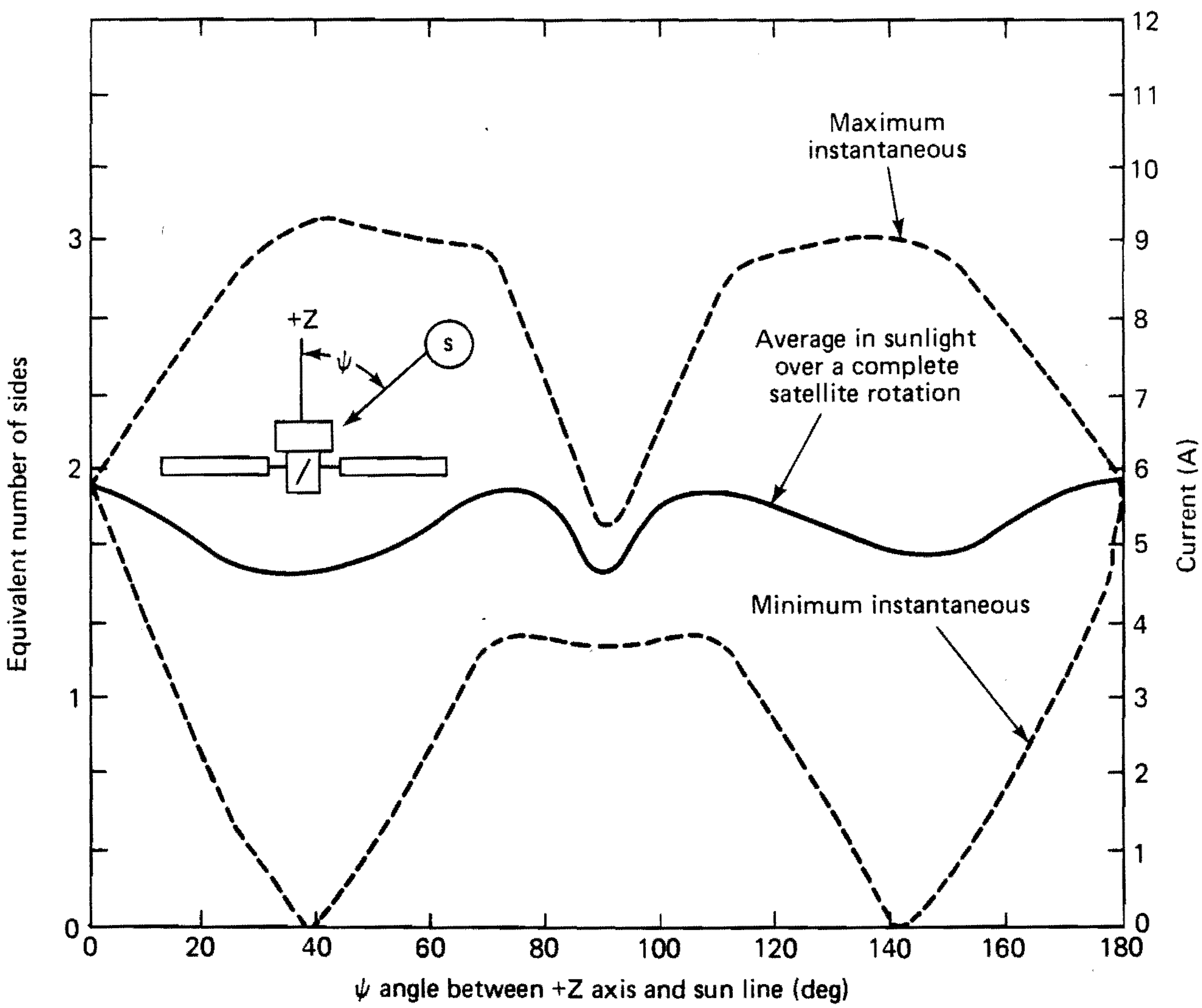


Fig. 4 Solar array current (and equivalent number of solar panel sides at normal incidence to the sun line) versus ψ.

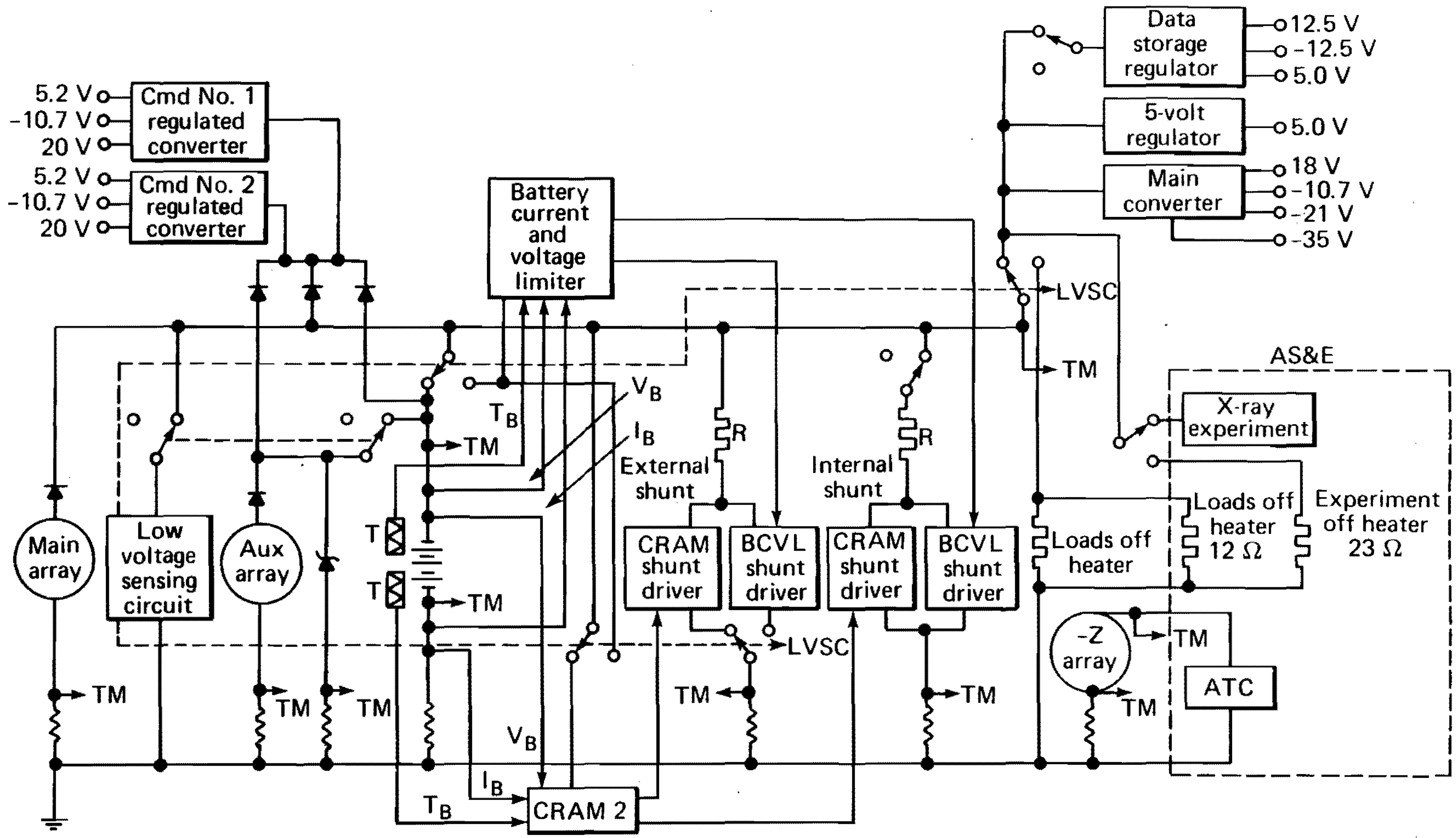


Fig. 7 SAS-A power system block diagram.

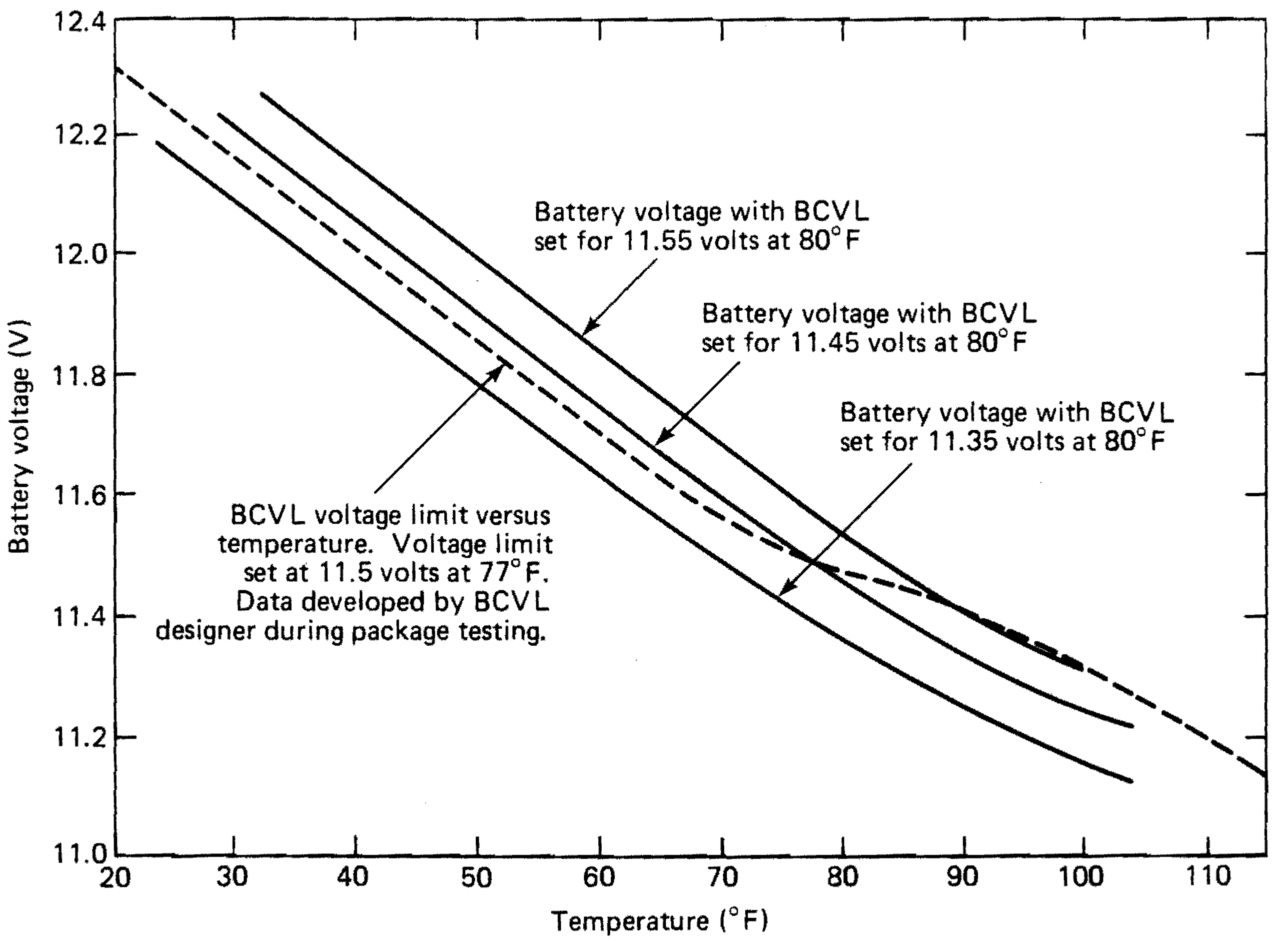


Fig. 8 High battery charge voltage versus temperature, BCVL control.

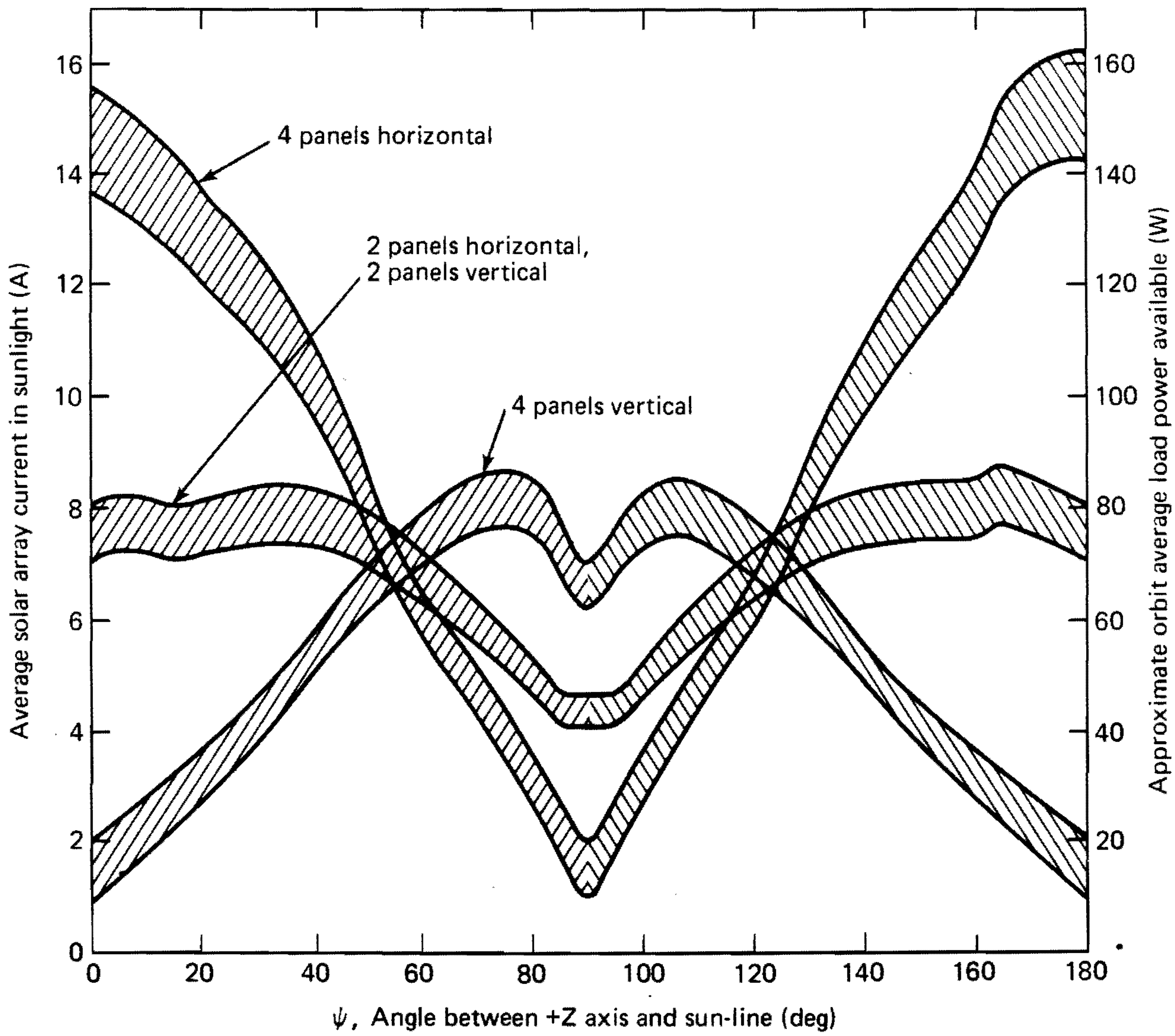


Fig. 11 SAS-C average current and power available.

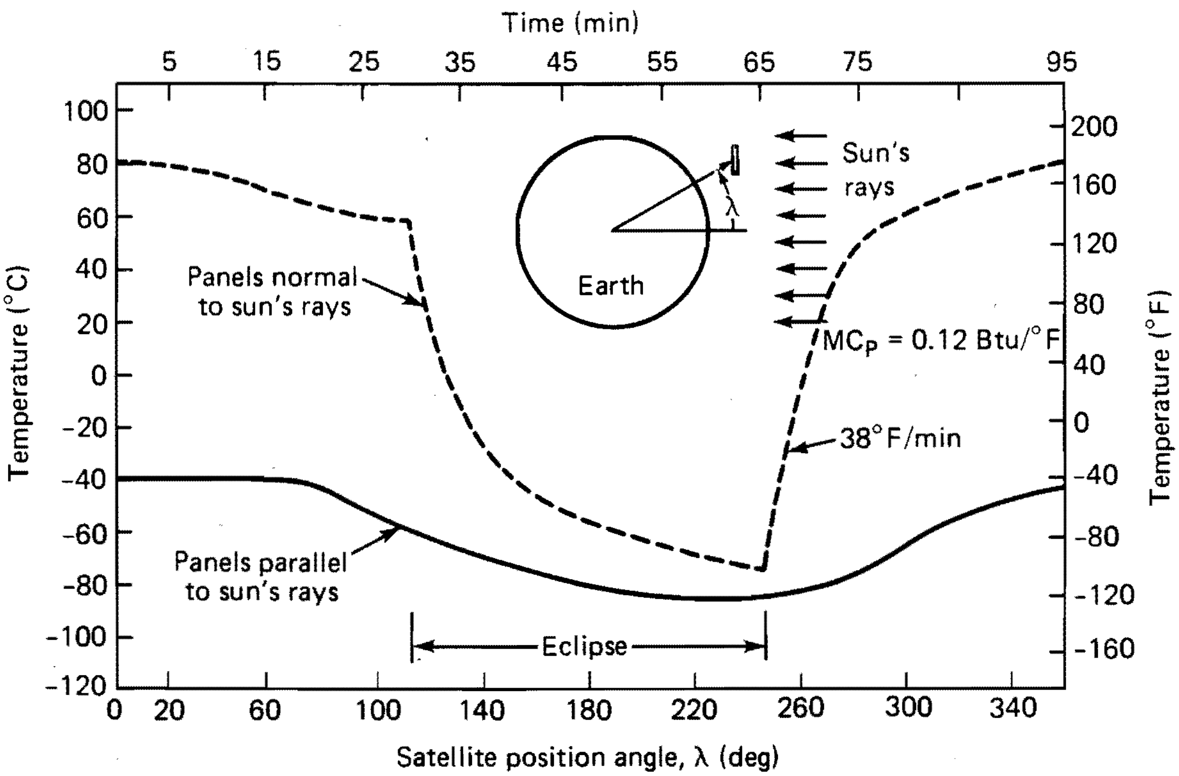


Fig. 12 Temperature extremes of SAS-C solar panels.

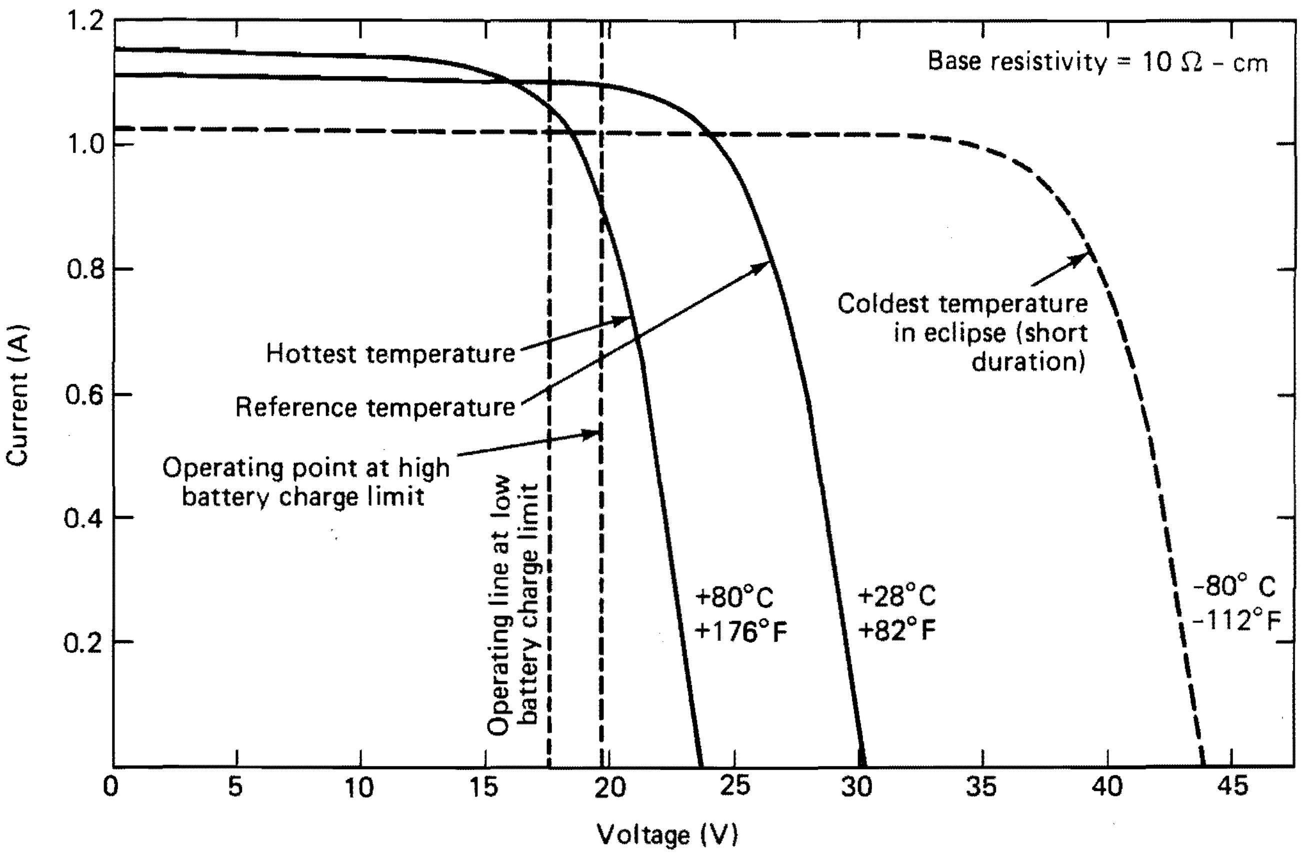


Fig. 13 I-V curve for a single solar cell circuit at different temperatures.

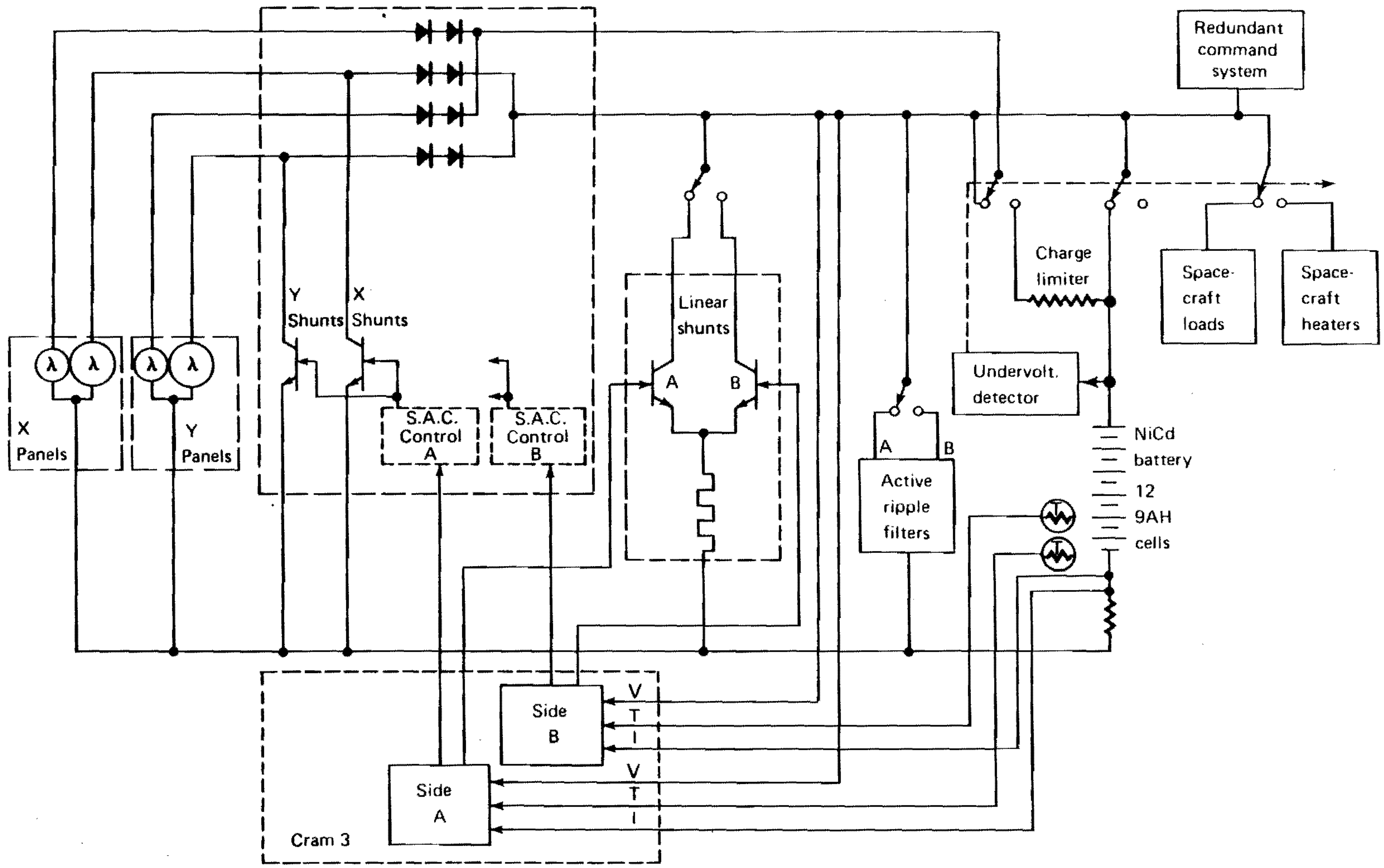


Fig. 14 SAS-C power system block diagram.

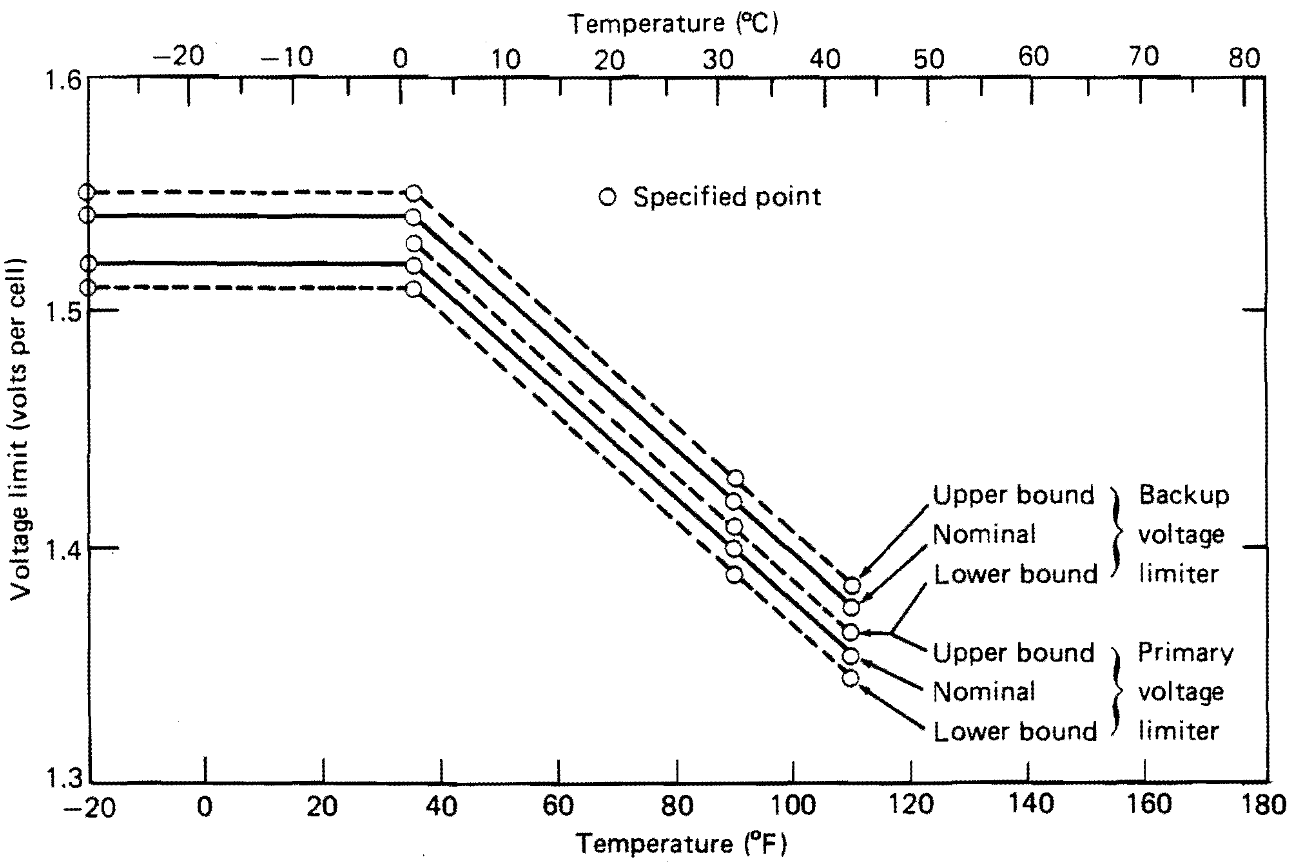


Fig. 15 CRAM III voltage limits.

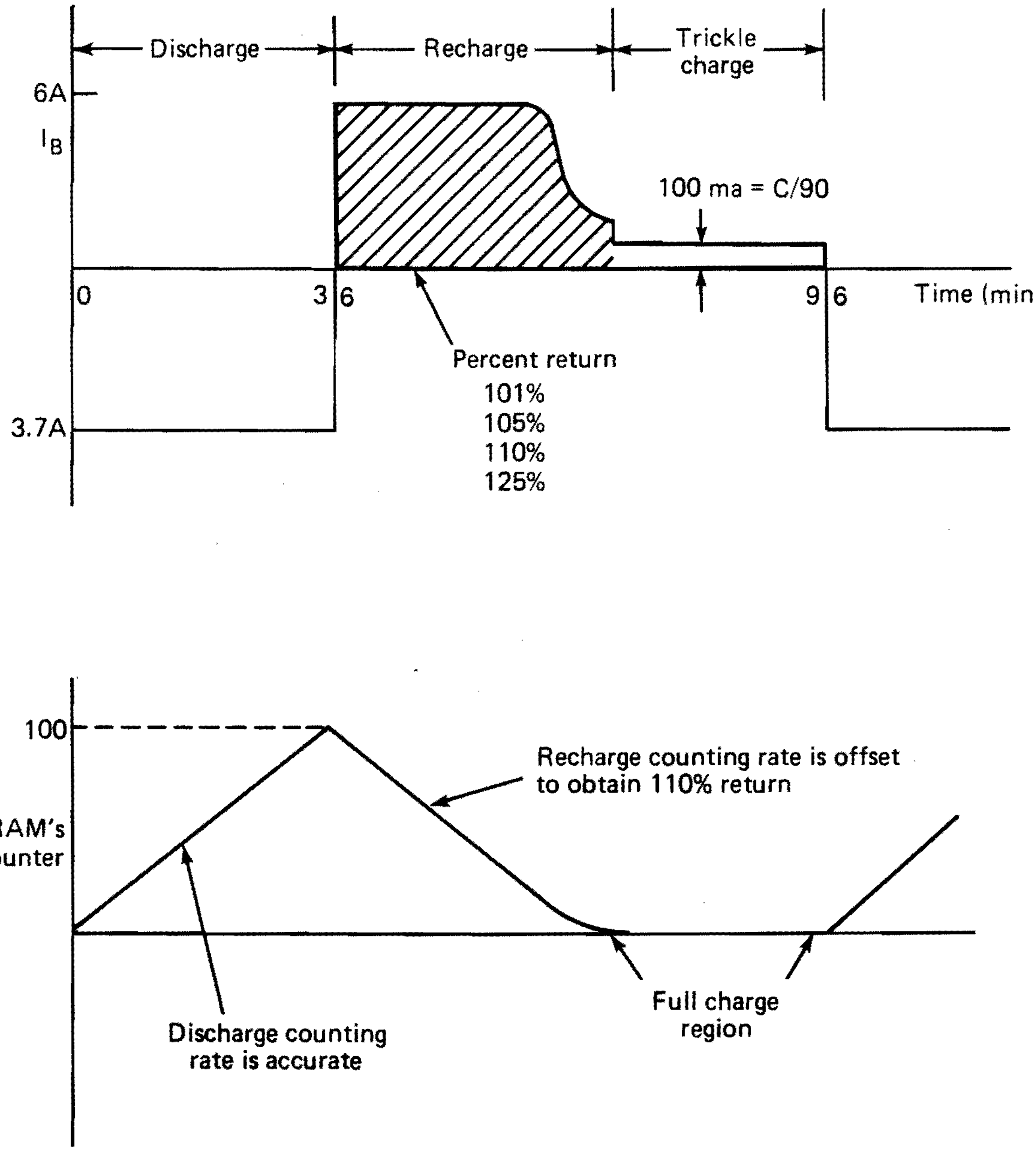


Fig. 16 Definition of battery charge regions.