A SURVEY OF RECENT APL SPACECRAFT POWER SYSTEMS

By Gregg A. Herbert
The Johns Hopkins University Applied Physics Laboratory
Laurel MD 20707

Submitted to Utah State University for their Conference on Small Satellites, October 7-9, 1987.

During the last 25 years APL has designed and built more than 50 small spacecraft, many being unique designs. The Power Systems of these spacecraft take many forms but almost all use a solar cell array and a Nickel-Cadmium battery. An overview of seven spacecraft power systems is presented. Four of the spacecraft are gravity gradient stabilized in a near-polar Low Earth Orbit (LEO). The remaining three spacecraft are spin stabilized, two with near-equatorial orbits. Both dissipative and nondissipative charge control systems are represented and the solar array configurations, all consisting of deployable panels, are described.

Power system performance parameters are given for each system including weight, power generation, solar cell area utilization and power system efficiency. The array generation capabilities range from 34 to 280 Watts. Power system efficiencies are on the order of 60 to 80% and experiments use from 20 to 63% of the available spacecraft power. Power generation per array weight is approximately 5 Watts/Kilogram and available power per system weight is 1-3 Watts/Kilogram for the spacecraft surveyed.

INTRODUCTION

The Johns Hopkins University Applied Physics Laboratory (APL) Space Department got its start with the proposal, design, devolopment and implementation of the Navy Navigation Satellite System (NNSS). Since this system was declared operational in 1966 the department has been responsible for several other spacecraft for the Air Force, NASA, the Navy and others which have performed a variety of scientific missions. All of the spacecraft designed and built at APL are considered small with the largest (in terms of power generation) being the GEOSAT-A spacecraft having an array capability of 282 Watts. Although many of the spacecraft designs have relied on "heritage" from the NNSS spacecraft, most have some unique features and it proved interesting to review several of the spacecraft from the viewpoint of power system performance.

In order to evaluate the performance of small spacecraft power systems, it is necessary to have something with which to compare. When conceptualizing about a new spacecraft design, with certain launch vehicle constraints, questions persistently arise such as: How much power can be generated by such a spacecraft? How much of this power can be made available for the experimenter's use? What will be the size and weight of the solar array and of the power system? In an attempt to answer these questions, this survey was done on seven previous APL spacecraft, five of which were launched on a Scout rocket and the remaining two on larger launch vehicles. The information contained within is intended to be used as a reference and as a comparison for other existing small spacecraft power systems and new designs. It is intended that the numbers in the tables will, for the most part, speak for themselves with little explanation.

Five tables are included presenting background information on the spacecraft and the power systems as well as general performance parameters for the solar cell arrays and power systems. The data is based on a combination of estimates, spacecraft test values and actual flight data. Even with the estimates and approximations, it is believed that the data is accurate to within a few percent (less than 5% in most cases). After presenting the data, a few conclusions are drawn and ideas are given for areas of improvement in a small spacecraft power system design.

SPACECRAFT/POWER SYSTEM DESCRIPTIONS

The following is a brief discussion of the seven spacecraft and their power systems. These seven were chosen because they are fairly recent (with the exception of the SAS series spacecraft) and because the required information seemed to be readily available. Table 1 lists the spacecraft along with launch dates, the sponsoring organization and orbits. All of the power systems utilize a deployable Silicon Solar Cell array and a rechargeable Nickel-Cadmium battery. The battery defines the primary bus voltage and battery charge control includes a temperature-compensated voltage limit in each system. Array isolation diodes, battery charge control electronics, converters and regulators and low voltage protection circuits are all considered as part of the power system. The main differences in the systems exist in the regulation and converter schemes and in battery charge control methods. These two areas are secondary to the purpose of this paper and, hence, are not addressed in any detail. Power dissipation within such electronics boxes was considered in the power system efficiency calculation and the weights were included for the power/weight numbers.

The Battery sizes, Bus voltages and Charge Control methods are listed in Table 2. SAS-A, HILAT and POLAR BEAR employ dissipative charge control systems, meaning that excess array power is discarded with the use of resistors and ends up as heat somewhere on the spacecraft. AMPTE-CCE and GEOSAT-A both use an FET switch to short out array circuits resulting in a nondissipative system. SAS-C charge control is accomplished with a combination of a low power shorting and a small linear shunt. NOVA-III uses a Pulse-Width-Modulated (PWM) regulator which reduces the operating voltage of the array when the power is not needed. This is also considered to be a nondissipative system although some power is converted to heat within the regulator in proportion to the excess array power.

Details of the solar array configuration are discussed for each system below as a supplement to Table 3 which contains some array/cell facts and figures.

SAS-A

Four panels with cells on both sides make up the SAS-A solar array. The panels are orientated along the spacecraft X and Y axes with a rotation angle (defined as the angle between the panel normal vector and the spacecraft Z axis) of 60 degrees. To shield the relatively low level of proton and electron radiation, 6 mil coverslides are used with an antireflective coating and a blue filter. The solar cells are N/P type Si with a base resistivity of 2 ohm-cm.

SAS-A was designed to fly in a near-Equatorial Low Earth Orbit (LEO) for a period of at least 1 year. The spacecraft is spin-stabilized about the Z axis.

Table 1
SPACECRAFT SUMMARY

Spacecraft	Launch date	Sponsor	Orbit : apogee (km) perigee (km) inclination (deg.)
SAS-A	12 DEC 1970	NASA	532 502 3
SAS-C	7 MAY 1975	NASA	516 509 2.9
HILAT	27 JUN 1983	DNA	807 800 82
AMPTE-CCE	16 AUG 1984	NASA	49684 1108 4.8
NOVA-III	12 OCT 1984	NAVY	1178 1178 90
GEOSAT-A	12 MAR 1985	NAVY	818 761 108
POLAR BEAR	13 NOV 1986	DNA/AFSD	1012 970 89.6

Table 2
BATTERY/CHARGE CONTROL DEFINITIONS

Spacecraft	Battery Definition	Bus Voltage (Nominal on discharge*) (Volts)	Battery Charge Control Method
SAS-A	8-series 6 A-h Ni-Cd cells	10.2	Temp. Comp. Voltage Limit, Current Limit and %Ret/I Limit Linear Shunt
SAS-C	12-series 6 A-h Ni-Cd cells	15.3	Same as SAS-A
HILAT	8-series 12 A-h Ni-Cd cells	10.2	Temp. Comp. Voltage Limit Linear Shunt
AMPTE-CCE	2 batteries 22-series 4 A-h Ni-Cd cells each	28.0	Temp. Comp. Voltage Limit Array circuits shorted with FET switches
NOVA-III	12-series 12A-h Ni-Cd cells	15.3	Temp. Comp. Voltage Limit PWM Regulator
GEOSAT-A	2 batteries 22-series 20A-h Ni-Cd cells each	28.0	Temp. Comp. Voltage Limit w Coulometer and %Ret/I Limit option, Array shorted with FET switches
POLAR BEAR	8-series 12 A-h Ni-Cd cells	10.2	Temp. Comp. Voltage Limit Linear Shunt

* Based on a per cell discharge voltage of 28.0/22 = 1.273 V/cell

that the battery charge regulator is micro-processor controlled and has increased flexibility and features. Solar cells used are the same as those on AMPTE-CCE but coverslides are only 6 mil here because the radiation environment is much less severe. Again, much redundancy is included in this two battery system.

This is a gravity gradient stabilized LEO spacecraft.

POWER SYSTEM PERFORMANCE PARAMETERS

Solar Array Performance

Tables 3 and 4 give the Solar Array details and the relevant performance numbers for each array. The total area is the total cell area of the array based on the total number of cells and nominal cell sizes of either 4 cm^2 or 8 cm^2. Since each array is connected to a battery bus the number of cells in series is set by the warmest end-of-life (EOL) panel operating into the highest expected battery voltage plus any diode, telemetry resistors and line voltage drops. The SAS-A, HILAT and POLAR BEAR arrays all feed an 8 cell battery, the SAS-C and NOVA-III arrays are connected to a 12 cell battery and AMPTE-CCE and GEOSAT-A both employ two parallel 22 cell batteries.

Hinges, spars and rotation mechanisms are included in the array weight totals because these components sometimes improve the array power generating capability. Considering all possible sun angles with respect to the orbit plane, the minimum orbit average power of each array is determined using an orbit average panel load voltage and beginning-of-life (BOL) currents. Eclipse time is taken into account implying that if a particular array were to be flown in an orbit other than the one it is in the array performance may differ.

The first parameter which can be used to compare systems is the Power Generation per unit Array weight. AMPTE-CCE is the highest because it has a planar array which is always within 20 degrees of the sun and the orbit has a high sunlight to shadow ratio. All of the other arrays can be compared using this number along with final column which gives a measure of the cell area utilization. Most significantly, the NOVA-III array (cruciform configuration with 2 panels rotatable) proves to be better the GEOSAT-A array (conical configuration) in both catagories (these two spacecraft are in similar orbits with solar cells that were manufactured at approximately the same time). The remaining four spacecraft are all comparable in area utilization and all used solar panels that were made in the late 1960's or early 1970's. SAS-C is better in terms of power per unit weight because it used curved panels which could be made thinner while retaining similar rigidity.

Table 3
SOLAR ARRAY DETAILS

Spacecraft	Configuration: Number of Panels and Orientation	Cells: Number and Size	Layout: Total Series/ Parallel	Tot. Area (Sq. m)
SAS-A	4-Panel, 2 sided Fixed Cruciform	6712 2x2 cm	31/216	2.68
SAS-C	4-Sets of three 2 Sided Cruciform X & Y axis pairs Rotatable	10944 2x2 cm	57/192	4.38
HILAT	4-Panel, 2 sided Fixed Cruciform (OSCAR)	6688 2x2 cm	38/176	2.67
AMPTE-CCE	4-Panel Fixed Planar	1776 2x4 cm	111/16	1.42
NOVA-III	4-Panel (6-sides) Cruciform X axis pair Rotatable	7776 2x2 cm	54/144	3.11
GEOSAT-A	8-Pairs of two Fixed Conical	12032 2x4 cm	94/128	9.63
POLAR BEAR	4-Panel, 2 sided Fixed Cruciform (OSCAR)	6688 2x2 cm	38/176	2.67

Table 4
SOLAR ARRAY PERFORMANCE

Spacecraft	Weight (kg)	Minimum Orbit Avg. Pwr (Watts)	Power Gen. per Unit Wt. (W/kg)	Power Gen. per Unit Area (W/Sq.m)
SAS-A	12.2	34	2.78	12.69
SAS-C	17.7	67	3.80	15.30
HILAT	14.9	42	2.82	15.73
AMPTE-CCE	12.0	130	10.83	91.55
NOVA-III	19.0	100	5.26	32.15
GEOSAT-A	59.2	282	4.76	29.28
POLAR BEAR	14.9	42	2.82	15.73

System Performance

Not all of the power generated by the solar arrays is available for use by the spacecraft. There is a significant portion that is "lost" within the remaining power system components. Batteries and converters are not 100% efficient; diodes, charge control electronics and telemetry resistors all dissipate power. Heaters were not considered as part of the power system even though they are often used to help maintain battery temperatures.

Table 5 lists the total system weight for each spacecraft power system as well as the system efficiency which is defined as the ratio of the power available to the spacecraft (other than the power system) and the power generated by the solar array. An important system "figure of merit" can then be calculated from these numbers: the power available per unit system weight. It was of interest to see what percentage of the available spacecraft power was used for experiments. In most cases this number could have been higher but some power is discarded in the process of battery charge control. This is a necessary condition resulting from conservatism in the original power system design and analysis. Finally, to help understand the previous numbers the energy storage was examined since the Ni-Cd batteries usually constitute a large portion of the total system weight. The battery sizing is highly dependent on the orbit (eclipse time), the load profile and mission lifetime so it must be considered when evaluating the overall system performance. Here again, the AMPTE-CCE spacecraft should not be compared to the rest because of its unusual orbit. In comparing the remaining systems one notices that the NOVA-III system appears to be the best in terms of power available per unit weight but it should be pointed out that there was not the same weight concern on GEOSAT-A because it used a different launch vehicle. The newer spacecraft (AMPTE-CCE and GEOSAT-A) would look even better than they do when compared with the older spacecraft (SAS-A, SAS-C and the OSCAR based HILAT and POLAR BEAR) in the power/weight catagory if the redundancy levels were comparable. GEOSAT-A has the largest battery (in relation to generated power) which gives it a higher reliability (longer battery lifetime) but worsens the power/weight number for the system.

Power system efficiencies followed a basic trend of rising as the generated power rises. There is a noticeable difference between the larger GEOSAT-A system and the other smaller systems. With the larger system a higher percentage of power can be devoted to experiments. There must be an overhead for the RF, Command, Telemetry and Attitude subsystems that does not increase linearly with spacecraft power. NOVA-III is misleading in this catagory because it is an operational spacecraft and the "experiment" was chosen to be the Transmitter. The OSCAR spacecraft (HILAT and POLAR BEAR) have relatively large batteries which can be attributed to early design conservatism and the fact that the OSCAR was originally designed to be an operational spacecraft (long lived) like NOVA. AMPTE-CCE has the smallest relative battery which is reasonable since it has the lowest percent shadow.

Table 5
POWER SYSTEM PERFORMANCE PARAMETERS

Spacecraft	Weight (kg)	Power Avail. per Sys.Weight (W/kg)	System Eff.	Experiment usage fraction	Energy Storage Ratio* (Hours)
SAS-A	19.8	1.02	.59	.50	1.76
SAS-C	33.2	1.20	.60	.43	2.01
HILAT	26.2	1.08	.67	.37	2.86
AMPTE-CCE	29.6	3.18	.72	.38	1.69
NOVA-III	33.5	2.18	.73	.20	1.80
GEOSAT-A	125.0	1.87	.83	.63	3.90
POLAR BEAR	26.9	1.05	.67	.51	2.86

* Using the cell nameplate capacity X 1.25 volts/cell on discharge X the total number of cells divided by the power generation.

CONCLUSIONS

After reviewing seven small spacecraft power systems, several parameters were calculated and reported. These parameters include solar array performance and system performance. Omnidirectional array configurations that are used for gravity gradient stabilized spacecraft appear to be capable of delivering more than 5 W/kg of array power/array weight (NOVA-III for example). Twice this number is easily attainable in the case of an AMPTE-CCE type array with panels always pointed toward the sun. This supports the obvious notion that significant gains in power generation could be realized if panels employ a rotation mechanism enabling optimum orientation at all times. The SAS-C array compares reasonably well due to its thin fold-out panels suggesting that prudent use of available volume using folding panels could also improve the power generation potential of small spacecraft arrays.

At the system level the power available for use by other spacecraft subsystems and experiments is roughly 2 W/kg of power system weight. System efficiencies range from .59 to .83 and rise with overall power. Battery sizing plays an important role in the system performance since it is a large fraction of the system weight. There are not many large gains to be made in the power system power/weight ratio by improving any of the components other than the solar array. A new battery technology suitable for spacecraft of this size (stored energy typically less than 200 W-Hours) would prove helpful but the availability of such a device in the near future is uncertain. The experiments can expect to use roughly 35% of the power generated by the array (assuming a 70% efficient power system and 50% of that used by other spacecraft components). One final point is that the array power often varies greatly as the orbit precesses so with a sophisticated command system the operational mode of the spacecraft could easily be varied to take advantage of seasonal power excesses.

Small spacecraft will always have their place in the space program, so it is hoped that collected data such as that presented above will be useful in the future. Comparisons can be made, goals set and improvements inspired by examing such numbers. Ultimately, this will result in higher performance power systems and small spacecraft of greater value.

REFERENCES

1. ARTIFICIAL EARTH SATELLITES Designed and fabricated by the Johns Hopkins University Appied Physics Laboratory. SDO 1600, MAY 1987

2. Design and Test of the SAS-A Power System. R. M. Sullivan. TG 1106, MAY 1970

3. Design Specification for the SAS-C Power Subsystem. APL Drawing # 7233-9907, Released MAY 1974

4. Payload Description of the HILAT Spacecraft (P83-1) SDO/PAO-0348, FEB 1983

5. POLAR BEAR Power System Description, Analysis and Performance. G. A. Herbert, SOP-2-86-291, NOV. 1986.

6. Comparison of Recent APL Satellite Subsystem Weights for Future Weight Estimation Purposes. K. J. Heffernan, S1I-2-350, 4 MAY 1987.

7. NOVA-III Power System Performance. R. M. Sullivan and T. Floryanzia, S3P-1-064, 8 SEPT. 1986.

8. Survey of NOVA Electrical Loads. R. M. Sullivan, S4P-1-084, 27 SEPT. 1984.

9. Update of AMPTE/CCE Electrical Power Loads Determined from Recent Telemetered Data. R. M. Sullivan, S3P-1-041, 10 JUNE 1986.

10. CCE, Solar Array Power Availability as a Function of Time in Orbit. W. E. Allen, S4S-5-562, 23 APRIL 1982.

11. Post Launch AMPTE Solar Array Performance Continued. R. P. Pignataro, S3P-1-140, 13 JULY 1987.

12. Post-Launch GEOSAT Solar Array Performance. S. H. Lim and J. D. Boldt, S3P-1-120A, 8 APRIL 1987.