Nanosatellite Power System Considerations

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INTRODUCTION

The capability to build complex electronic functions into compact packages is opening the path to building miniature satellites in the order of 1 kg mass, 10 cm across, packed with the computing machines, motion controllers, measurement sensors, and communications hardware necessary for operation.[1] Power generation will be from short strings of silicon or gallium arsenide-based solar photovoltaic cells with the array power maximized by a peak power tracker (PPT).[2] Energy storage will utilize a low voltage battery, with nickel cadmium or lithium ion cells as the most likely selections for rechargeables and lithium (MnO_2-Li) primary batteries for one shot short missions.

Based on a spacecraft requirement of 2-W orbit average power for a low Earth orbit (LEO) application with a 60/30 minute sun/dark ratio, the battery would need to deliver 2 W for 30 minutes, and to achieve satellite energy balance would need to be teamed with a solar array able to produce 3 to 4 W.

The spacecraft primary power bus (PBUS) which is the combined output of SA and battery, could be sized to directly power a transmitter for a communications intensive satellite and still be high enough in voltage for efficient conversion to the levels needed by the logic subsystems. These satellites would also use the standard techniques to save power: (1) dynamic adjustment of processing speeds and (2) powering down sections not in use.

GENERAL POWER SYSTEM CONSIDERATIONS

A typical power system is composed of the solar array for producing power from the sun, the battery for storing energy for use during the dark portions of the orbit, the circuitry for controlling and limiting the charge and discharge of the batteries, and power filters, converters, and switches that distribute the raw or conditioned power to the spacecraft subsystem loads.

Typically, one-third of the total weight of the spacecraft is taken by the power subsystem, with the batteries weighing one-third of the power system. The batteries not only supply energy during the dark portions of the orbit but provide the reserve energy to maintain the bus voltage during peak loads.

Selection of the cell type is dictated by the number of cycles and depth of discharge required over the mission life, operating temperature range, and average and peak discharge levels.

Sophisticated charge control can be done efficiently to implement protocols such as constant potential current-limited charging or by trickle charging after reaching a predetermined battery voltage.

ENERGY STORAGE

Nanosatellites would consume only 1 or 2 W of power, compared to medium to large contemporary satellites which draw 100 to several kilowatts of power. For smaller spacecraft requiring less than 200W, nickel cadmium remains a popular choice for reasons which include volumetric energy density, cost, and availability.

Now undergoing extensive evaluation for space applications, Lithium Ion or Li+ cells look to be promising alternates to NiCds with twice the energy density and 3 times the nominal terminal voltage per cell.

Li+ cell chemistry is based on intercalated lithium ions. Although the use of intercalated lithium reduces the energy density relative to the use of pure lithium foils, the cycle life is much greater and the voltage levels are almost as high. Many possible cathode materials are still being investigated in the search for a flatter discharge curve. Li+ cells must be protected from over voltage during charging and this requires each cell to have a voltage clamp which acts as a constant voltage current shunt around the cell. Given the low voltage bus requiring a few cells, the over voltage circuitry is modest overhead for the gains realized. There still remains important work to determine cycle life and optimum charge techniques before using Li+ cells for space use.

The battery type selected for an application can be based on a number of factors. Although cycle life is the most usual factor, other factors include volumetric energy density, and gravimetric energy density.

The Nanosat designer also has a choice between rechargeable and primary batteries. Lithium primary batteries have been flown on Shuttle Payloads and have an energy density tenfold greater than NiCds. They also can deliver significant number of 1C current pulses, generate no gas during discharge, are usable from -20C to +60C and have less than 0.5% self discharge per yr. for missions where long storage time is needed such as a NanoSatellite waiting for release from the mother ship to perform an Observer mission.

Another emerging cell type is based on thin-film technology.[3] Thin, low capacity devices based on successively plating layers of conductive base, cathode, electrolyte, and anode have already demonstrated encouraging cycle life performance.. Unfortunately, the capacities available from these cells are too low for consideration in NanoSatellite applications

For NanoSat applications, it is felt that a more traditional power bus architecture with centralized power source and charge control functions for the entire spacecraft will be more weight effective.

Energy Density

The energy density of a single cell is usually determined by fully discharging it to an appropriate cutoff voltage over a 5-hour period. The energy density is a function of the cell capacity. Large nickel cadmium cells are usually assigned the number of 40 Whr/kg in cell sizes above 10 Ah. As cell sizes become smaller, the energy density is reduced as well as an increasing percentage of the cell consists of case, feed thrus, and other non-energy-producing components.

The energy density of a battery is reduced by 20 to 40% compared to the energy density of the cells because of the structures, wiring, and controls associated with the completed battery. These numbers are further reduced as the batteries are typically cycled to only 20 to 40% of their full capacity. The depth of discharge to which a battery is cycled is set by a combination of factors including temperature limitations, current strain rate limitations, life cycle requirements, and minimum end of discharge limitations. As an example, when 40 Wh/kg nickel cadmium cells are used in a 22-cell battery that is cycled to 20% DOD, the battery will have a usable energy density of 6.4 Wh/kg.

Although the cycle life of a complete battery system is viewed to be a strong function of the particular chemistry under consideration, it is often affected by other factors. Cell design codes, recharge protocols, DOD, battery temperature, etc., can result in factors-of-10 changes in the cycle life of an aerospace battery. These factors are usually under the control of the power system designer and/or satellite operator. Their impact on cycle life is investigated during the course of ground-based life cycle testing programs. As a result of extensive data bases, cycle life vs. depth-of-discharge relationships have been developed for each cell chemistry. This information can help guide the satellite design engineers.

Given the power requirements of a suggested nanosatellite and the cycle life requirements in low Earth orbit, only power systems based on commercially available nickel cadmium cells have a proven predictable flight history

Table 1. Summary of Battery Characteristics in Small Cell Sizes
(More information on these cell types can be found in Ref. 8.)

Cell Chemistry	Cycle Life Rating	Size Availability	Energy Density
Lead Acid	Insufficient	Should Be	Medium*
Lithium Ceramic	Encouraging	Not Likely	Low
Lithium Ion	In Development	Available	High
Lithium Organic	Insufficient	Available	High
Lithium Polymer	Insufficient	Should Be	High
Nickel Cadmium	Encouraging	Available	Medium
Nickel Hydrogen	Encouraging	Not Likely	Medium
Nickel Me Hydride	In Development	Should Be	Medium
Silver Cadmium	Insufficient	Not Likely	Medium
Silver Zinc	Insufficient	Should Be	High
Li MnO2	Primary Cell	Should Be	High+

*Low, < 15 Wh/kg; Med., 15-40 Wh/kg; High, > 40 Wh/kg; High+ >200 Wh/kg

If we project 5 to 10 years into the future, systems based on nickel metal hydride as well as intercalated lithium ion chemistries may be considered. The cell sizes and projected energy densities will be discussed in a later section. The relative energy density compared to nickel cadmium will be 1.5 for nickel metal hydride and 2.5 for lithium ion devices. For these small cell sizes, the energy densities at 100% DOD are approximately 20 Wh/kg, 30 Wh/kg, and 50 Wh/kg, respectively.

POWER SYSTEM OPTIMIZATION FOR NANOSATELLITES

Based on the current state of development of power system related technologies, the following subsystems form the basic blocks for a nanosatellite power system:

- Solar Arrays
- Solar Array Bus (SAB)
- Satellite Primary Bus (PBUS)
- Batteries
- Power Converters
- Battery Charge Circuits
- Satellite Logic Voltages
- Power Switching

Solar Arrays

Present plans are to use body-mounted solar cells. Small deployed arrays are a possibility if more power is required. The solar array, if confined to an area 10 cm in diameter on the top surface, will be inadequate to produce the required energy to operate the 2-W satellite when the recharge of the battery and the solar cell degradation factors are taken into account. Even with the very highest efficiency solar cells, the array needs to be kept pointing to the sun in order to obtain 2.6 W, which is about 1 W short of required. Dual junction cells operating at 25% are suggested for this application as being available in reasonable quantities. In addition, the packing factor of the cells needs to be as close to unity as possible. The circular outline also presents problems, as most solar cells are rectangular. The solar array arrangement is still under consideration.

Solar Array Bus

The solar array (SA) output supplies the battery charge regulators and the Primary Bus (PBUS) voltage regulator, the outputs of which are combined to form the PBUS voltage, which in turn is the primary power for the spacecraft loads. The array voltage is set so the battery charger and PBUS regulator operate near their highest efficiency when the SA's approach their end of life; that value is typically in the range of 6-8 V above the battery voltage. For an 8-V battery bus, the SA would be configured for an EOL output of 14-16 V, which would require a series string with about 10 high-efficiency, multi junction solar cells.

Maximizing Solar Array Power

Conventional solar array interfaces do not continuously set the array operating point at the "knee" of the voltage-current (V-I) curve where maximum power is produced. As a result, recharging deeply discharged batteries tends to depress the voltage reflected or "seen" by the array, cutting SA efficiency just when it is needed most. By using a peak power tracker (PPT) in series with the array, the maximum SA power can be generated regardless of the array's illumination, environmental conditions, or effects of aging. 2

The PPT technique uses a switchmode converter in series with the array to dynamically adjust the array output impedance to match the load. The PPT works by slowly increasing the series converter pulse width, so as to "load" the array, causing the array voltage to decrease. At the same time, the PPT measures the rate of SA voltage change and when the knee of the curve is passed, the rate of change starts to drop significantly. The PPT control loop detects this change and reverses course slightly to keep the system stable. The cycle is then repeated. Small PPTs have been demonstrated that realize a 10 to 15% power improvement.

Primary Bus (PBUS)

The PBUS is the spacecraft primary voltage bus formed by combining two sources: (1) the battery output and (2) the solar array output converted by the PBUS regulator to slightly above the maximum battery voltage. The PBUS voltage would be set high enough to directly power the primary loads on a communications or an imaging nanosat, e.g., a sensor array or a transmitter in the range of 0.3 W output. PBUS would also offer a voltage which could be efficiently converted up for higher power transmitters and converted down for the low voltage subsystems such as telemetry sensors and TT&C and ACS computers.

One possible PBS configuration would use either lithium ion or nickel cadmium cells in a nominal 7.2-7.5 V battery, resulting in a PBUS output of 9 V during sunlight, dropping to nominal 7 V as SA power drops out during umbra. A back up feature of PBUS is if the battery or charge regulator should fail, the satellite could still operate from the solar array but only of course during daylight.

Power Converters

Switch mode power converters are pulse-width-modulated down-converters, which are used in the PPT, in the battery charge regulator and in the PBUS regulator. The converters can be built using monolithic ICs and a few discrete parts with a 1-2 W output at 80-85% efficiency. More complex converters running at 600 kHz and using synchronous rectification to reduce switching device losses have 85% efficiency. However, for these low power applications, the small size and low overhead of the IC converters make them the preferred devices.

Batteries

The battery voltage is selected to minimize the number of cells, since the cell energy efficiency drops as cells get smaller, but still have high enough voltage to efficiently power the satellite subassemblies. The two battery types suggested as a result of the review presented in the Energy Storage section of this chapter are as follows: (1) six series-connected nickel cadmium cells and (2) two cells based on the emerging lithium ion chemistries. These batteries have nominal voltages of 7.5 V and 7.2V, respectively.

To supply the required power would require nickel cadmium cell sizes referred to as AA size (500 mAh). Test data for C and D sizes of nickel cadmium cells have shown reliable operation over the 5 to 15,000 charge and discharge cycles needed for 1 to 3 years of operation. Additional analysis and testing would be needed for the AA applications since performance comparable to the bigger cells has yet to be demonstrated for the AA size. Although there is insufficient cycle life data on the lithium manganese dioxide type of lithium ion cells, testing is in progress which holds promise for extended operation at 40-50% DOD. If lithium ion cells prove reliable, a battery consisting of 2 lithium ion cells at 40% DOD would have a weight savings of 30-40 gm, compared to a nanosat built with 6 nickel cadmium cells cycling to 50% DOD.

Battery Charge Regulator

Long-term lithium ion battery testing is needed to determine how to charge lithium ion cells for long cycle life. Since it is well established that present lithium-based cells have no inherent overcharge tolerance, the charger will need to protect each cell from over-voltage by performing precision end of charge (EOC) control. The charge regulator will also be able to adjust charge rates and EOC voltages based on battery temperatures or other environmental conditions. By comparison, the battery charger for nickel cadmium cells, which have inherent overcharge tolerance, can be a single unit for all the cells employing simple battery temperature compensation.

Regardless of battery selection, the charge regulator will be a switch-mode hybrid package, 80+% efficient, and capable of temperature-compensated constant current/constant voltage charging with lithium ion cells or with nickel cadmium cells.

Charger telemetry would include standard voltages, current, and temperature data and might include onboard fuel gauge computation as part of the charge regulator to aid in planning on-orbit operations. A typical value for the round trip energy efficiency for discharging and charging nickel cadmium batteries at the DODs under consideration here is 65-70%.

Logic Bus

The logic system will need regulated power stepped down from the PBUS and will use linear and digital integrated circuits (ICS) that run from 2.7 V to 3.6 V. The development of low voltage ICs substantially reduces the power requirements over 5 V systems and has the added benefit of lower effective switching noise. Since the complementary metal oxide semiconductor (CMOS) logic used for all the computing elements draws power directly in proportion to the clock rates, power savings can be realized from control firmware which adjusts the clock rates of the computing and control sections to just meet the processing task.

Lower voltage logic is expected to be more susceptible to single event upsets, but less susceptible to single event latchups compared to 5 V logic. Critical RAM data or program code memory upsets could be corrected by error detection and correction circuits (EDACs). EDACs assign check and correction codes to each data word allowing the EDAC to do a background check to correct single bit errors and to detect double bit errors.

Radiation tests of random access memories (RAMs) from select vendors[4] have shown that a single SEU event rarely hits two bits in a single byte. Thus EDAC is an effective approach to repairing memory SEUs in critical requirements and for protecting the satellite should a radiation blitz occur.

Other promising techniques to improve radiation tolerance include making relatively small changes to the basic gate layouts during the chip masking phase. Post wafer fabrication radiation hardening is also a possibility to improve SEU tolerance and total dose performance. However, for LEO orbits, the radiation levels after shielding by the silicon and aluminum satellite structures are well within present silicon-based IC tolerances.

Power Switching

Power switching is done by the power controller (PC), which routes and switches PBUS power or logic power to the various subsystems. Discrete switches within the PC need not be centralized but could be distributed onto the subassemblies, receiving on/off commands over a control bus. The PC would also generate the needed voltage, current, switch status and temperature telemetry. Power switching would employ smart IC switches, devices which integrate the FET switch and driver along with current limit and thermal protection to form a nearly indestructible switch.

SUMMARY

The power system technology requirements and architecture for a NanoSatellite design have been reviewed and a conceptual power system defined. For short mission life, Li primary batteries have high energy density and eliminate the solar array and charger overhead. For long missions, batteries based on the well established nickel cadmium chemistry or the much newer lithium ion chemistry would be used to store energy generated by high efficiency multi junction solar cells. To minimize the number of cells used in the battery as well as maximize the energy density of the cells used in the battery, a very low bus voltage should be used. The increasing availability of analog and digital devices and sensors able to efficiently operate below 5 V has greatly facilitated the practicality of low voltage bus architecture. There appears to be no insurmountable problems precluding building NanoSatellites on a production line which handles miniature parts, such as Laptop PCs, resulting in high quality low cost NanoSatellites.

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