ΔΙΙΔΡΥΜΑΤΙΚΟ ΠΡΟΓΡΑΜΜΑ ΜΕΤΑΠΤΥΧΙΑΚΩΝ ΣΠΟΥΔΩΝ

ΔΙΑΣΤΗΜΙΚΕΣ ΤΕΧΝΟΛΟΓΙΕΣ, ΕΦΑΡΜΟΓΕΣ και ΥΠΗΡΕΣΙΕΣ

Power Systems

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ΠΑΝΕΠΙΣΤΗΜΙΟ ΠΑΤΡΩΝ
UNIVERSITY OF PATRAS

ΕΛΛΗΝΙΚΗ ΔΗΜΟΚΡΑΤΙΑ
Εθνικόν και Καποδιστριακόν Πανεπιστήμιον Αθηνών
ΙΔΡΥΘΕΝ ΤΟ 1837

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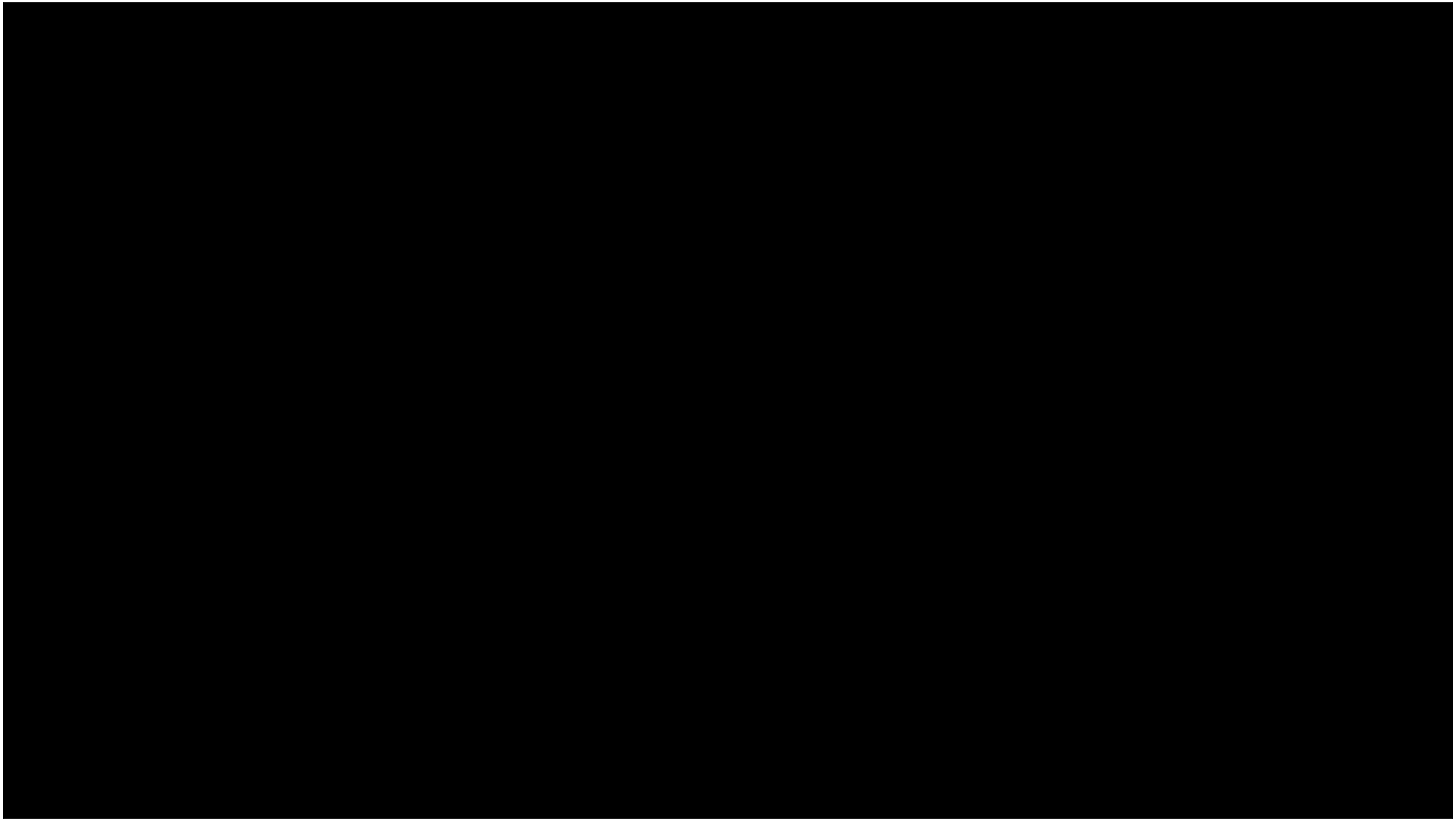
Starship – the store till now



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‘Test as you fly – fly as you test’

Course Content

- **Week 1:** Brief history & context: Background to the development of space, agencies, space history/policy, space economics, funding, future missions – case studies.
- **Week 2:** Introduction to space system design methodology: requirements, trade-off analysis, design specifications, system budgets. Introduction to space system architecture. Launch Vehicles.
- **Week 2:** Space and Spacecraft Environment: Radiation, vacuum, debris, spacecraft charging, material behaviour and outgassing.
- **Week 3:** Orbit Mechanics: celestial mechanics, orbits, trajectory design and spacecraft maneuvers
- **Weeks 4-10:** Spacecraft sub-systems design: Structure & configuration; **Power, the power budget and solar array and battery sizing;** Communications and the link budget; Attitude determination and control; Orbit determination and control; propulsion Thermal control, assembly, Integration and Test processes; Launch campaign; Space mission operations. Space Project Management, review, tutorial problems/exam mock up

Spacecraft Power Subsystems

(Chapter 13.2 Understanding Space)

Spacecraft Power Subsystems

Power systems are needed to operate satellite on-board electronic systems and payloads, they comprise:

- PRIMARY SOURCE OF ENERGY CONVERSION

  Chemical - batteries (rockets)
  Nuclear - radio isotrope thermoelectric
  Solar - photo-voltaic 'solar' cells

- POWER STORAGE

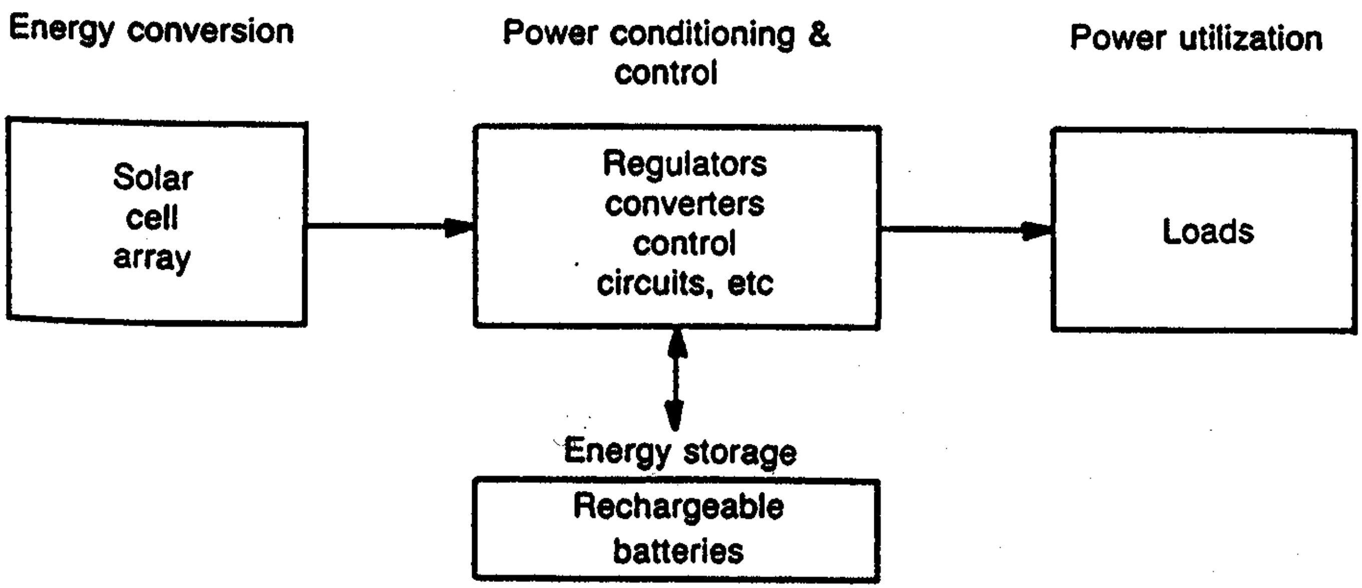
  Re-chargeable Batteries
  - *Nickel-Cadmium*
  - *Silver-Cadmium*
  - *Silver-Zinc*
  - *Nickel-Hydrogen*
  - *Lithium-Ion*
  - *NiMH*

- POWER CONDITIONING

  Battery charge/discharge control
  Voltage regulation
  Load switching & protection
  Voltage/Current monitoring

Spacecraft Power Subsystems

BASIC CONFIGURATION OF A SPACECRAFT POWER SYSTEM



Spacecraft Power Subsystems

SOLAR POWER:

- Solar Constant = 1353 Watts/sq.metre
- Solar Cell Photo-voltaic Conversion Efficiency

Silicon	*= 12% to 15.5%*
Gallium Arsenide	*= 16% to 20%*
Indium Phosphide	*= 16% to 20%*

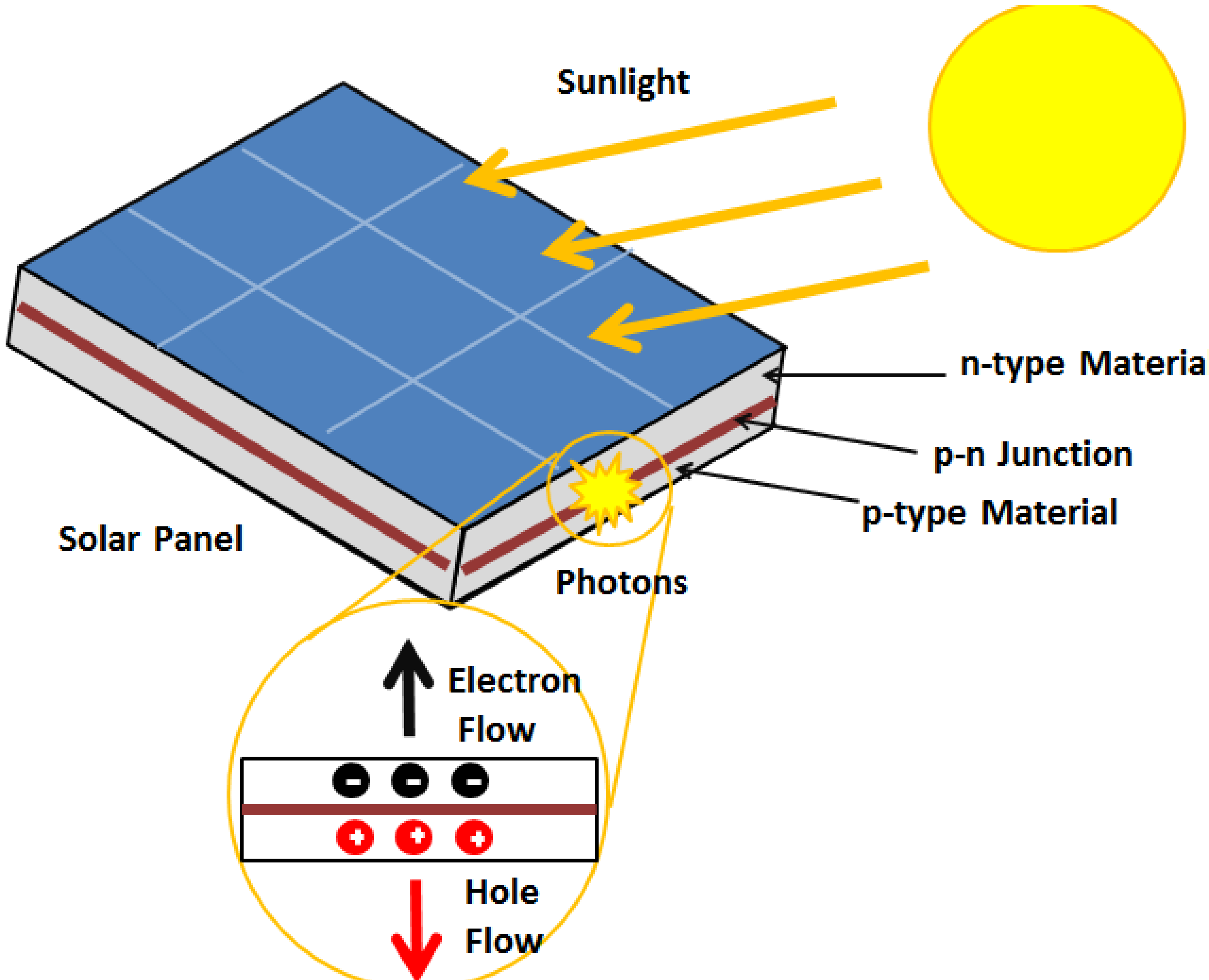
- Cost

Silicon	*= $50 -75k/sq.metre*
Gallium Arsenide	*= $200-400k/sq.metre*

- New Developments

Tandem cells	*- p-n junctions back-to-back*
	- GaAs on Si
GaAs/Ge	*- to reduce weight & cost*
Triple Junction cells	*- ~36% efficiency*

Photovoltaic Effect - Φωτοβολταικά



Solar Panels

- The fuel for photovoltaic conversion comes from the photons captured in the solar panels of the spacecraft/satellite.
- Solar panels that are properly oriented toward the Sun can provide about 130 W/m2 and 50W/kg of power. Because solar cells mounted on the satellite’s body will not, in general, be optimally oriented, they can typically provide 30 to 35 W/m2 and 8 to 12 W/kg of power

Solar Eclipses – Ηλιακή Έκλειψη

- Depending upon the position and the orbit of the satellite, it will be subjected to eclipses in which the solar panels will not be able to receive sufficent photons for electric production.
- Especially for satellites in LEO, there will be regular eclipses throughout the day and around 40% battery power may need to be used for each orbit. Each eclipse can last as long as 35 minutes.
- In GEO orbits, an eclipse can occur in certain seasons and it can last for about 72 minutes.

Performance Parameters for Solar Arrays/Panels

- There are two figures which can be used to measure the performance of a space solar array.
- Power per unit mass (W/kg)
- Power per unit area (W/m2)
- Typical values are - BOL 30-40 W/kg and 90-110 W/m2 - EOL (End of Life) can vary in space conditions

Satellite Power System Elements

- Satellite power systems consists of primary and secondary energy sources and a power control network which regulates the distribution of power.
- Primary Energy Soruce converts fuel into electrical power. (or photons to electricity)
- Secondary enegy source is required to store enery and deliver electrical power to the satellite system and the payloads when the primary system is not available.
- Power control network is required to deliver appropriate voltage-current levels to all spacecraft loads as and when required.

Primary Power Systems

- For majority of the satellites, the primary power system consists of using solar power systems (photovoltaic) through the means of a solar array in order to achieve that objective.
- A solar array is an assembly of thousands of solar cells connected in way to provide appropriate power levels as needed for the particular operation of the satellite.
- Solar systems will power the satellite’s payload and subsystems. Also, extra power needs to be produced for charging.

Secondary Power Systems

- Secondary power systems are those systems which are used for providing power when the primary power is not available.
- Especially for satellites with solar arrays this means that the batteries must provide power during eclipses and that the array must recharge the batteries in sunlight. (more frequently in LEO)

Batteries in Secondary Power Systems

- The main function of a battery operation in satellites is the way in which the reliability and charge efficiency are related to charge control.
- Parameters of importance for batteries include:

- Charge / Discharge rate (cycles)

- Depth of Discharge (percentage of the battery capacity discharged during the eclipse) [βάθος αποφόρτισης μπαταρίας]

- Extent of overcharging
- Thermal Sensitivity
- Temperature Gradients

Battery Types in Satellites

- In the 1960s, nickel-cadmium (NiCd) was the primary technology used for satellites, and it is still used to some extent today for LEO satellites that require lower levels of power. NiCd batteries are reliable, simple to manage, have a low self discharge, and a strong heritage in space.
- In the 1990s, nickel-hydrogen (Ni-H2) batteries began to replace NiCd, especially in GEO satellite because of their high energy over mass ratio.
- Now Lithium ion batteries are becoming the new standard, Combined with higher energy and better charge efficiency, this lets the satellite maker reduce the size of the satellites' solar panels.

Satellite Orbits and Batteries

- The orbit of the satellite will have a great impact on the batteries which will need to be present on the satellite.
- LEO satellites take around 90 minutes to circle the earth. In this position, the satellite is eclipsed for 30 to 40 minutes per day so the batteries endure about 5,000 cycles per year.
- GEO satellites take 24 hours to rotate the earth; and, for eclipse seasons (two 45-day periods a year), they use batteries for 0 to 72 minutes per day. Batteries for such GEO satellites — used primarily for telecommunications, military, and meteorological systems — must last 15 to 18 years.

Importance of Depth of Discharge (DOD)
ʹΒάθος Εκφόρτισηςʹ

- Depth of discharge can determine the lifeterm of a battery in a satellite. 100% DOD means a complete discharge.
- In order to preserve battery function, lower depth of discharge is needed.
- 20% DOD is commonly used for space NiCd batteries to guarantee 5 years of life in LEO.
- Since GEO satellites will undergo less eclipses, higher DOD can be used.
- DOD also determines mass due to number of batteries required. If a satellite requires 2 kW of power, then battery mass will range from 50 kg to 100 kg at 100% DOD and from 250 kg to 500 kg at 20% DOD (because you need more batteries for the same function).

Solar Array Design

- The solar array mass must take into account the battery charging requirements along with load requirements.
- If the satellite requires 2kW to function and if the energy consumed by the batteries by the loads during the eclipse is 2kW-hr;then the array must be able to generate will be around 3 kW of power.
- Thus, the array mass in this example will be around 85 to 120 kg
- Current rigid panel solar arrays on US satellites have specific powers in the range of 15W/kg to 30 W/kg.
- Array's exposure to temperature must also be taken into account.

Power Management and Distribution (PMAD)

- Power Management and Distribution subsytem (PMAD) is an integral part of power system design for Earth orbiting satellites.
- Electronic loads within the satellite need different voltages to operate and and special power demands need to be met for variety of components during the mission. Early satellites used 28 V DC.
- PMAD will handle all of this such as controlling and monitoring batteries, monitoring degradation of solar arrays, and switching of the load distribution system.

How to Design the Perfect Power System for Earth Satellites?

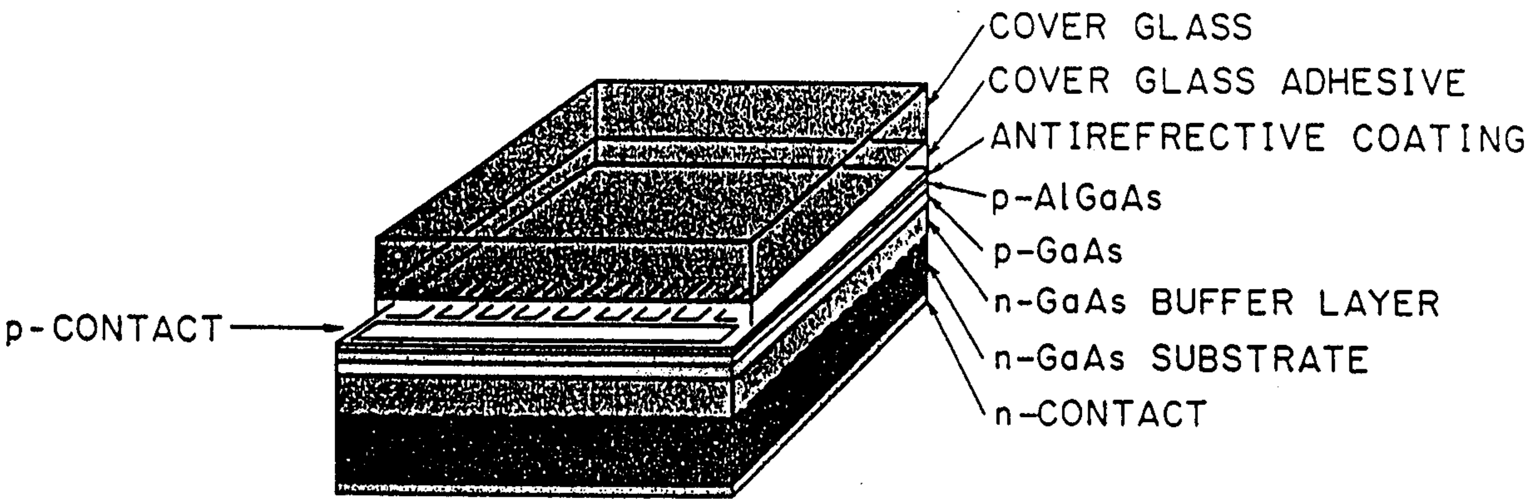
- Calculation of the exact orbit of the satellite
- The prior determination of the orientation of the satellite
- The requirements of the purpose of the satellite (communications, weather, GPS etc)
- Any extra payloads present on the satellite
- Calculation of the Required Solar Panel Array
- Calculation of the Requirements for Batteries
- Designing of the Overall System for Supplying Power to each Component

Power Subsystem Design Calculations

- It will be essential to calculate baseline power requirements for the satellite by
  - taking mission and payload inputs,
  - the method of stabilization (as it effects the solar array), - the type of solar cells used,
  - the rigidity or the flexibility of the solar array
  - the orbit in order to determine eclipse times, battery mass and battery requirements.
  - Probable losses in the system

Spacecraft Power Subsystems

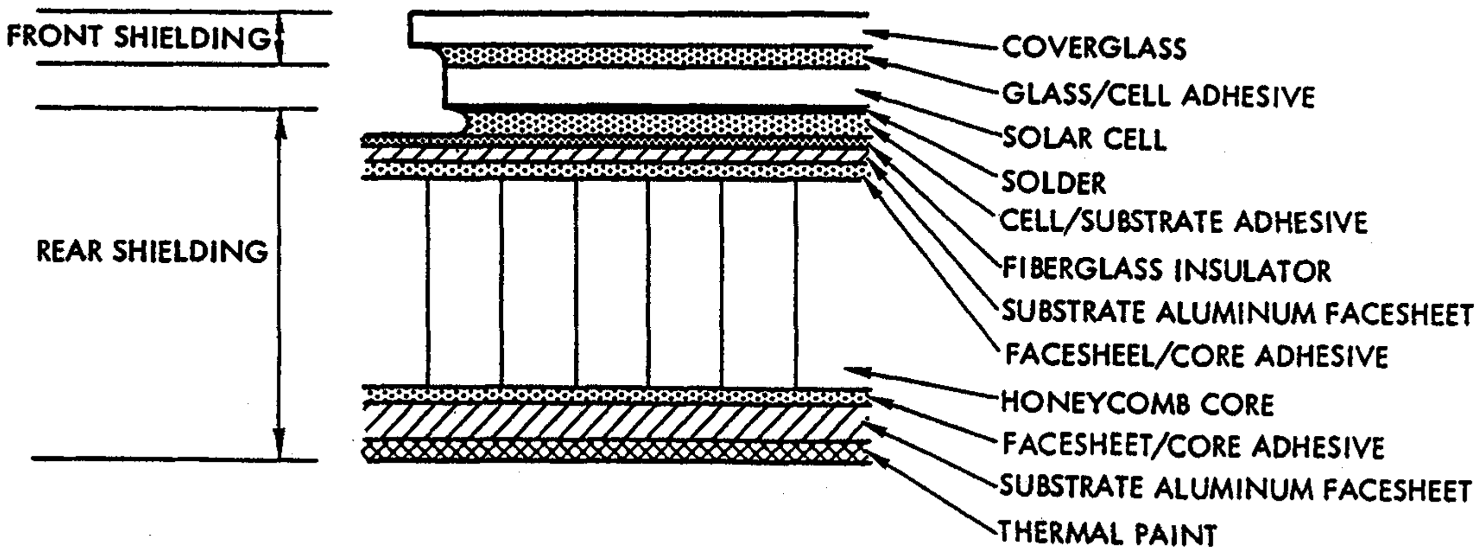
SOLAR POWER -SOLAR CELL CONSTRUCTION



- AlGaAs/GaAs Heteroface Structure

Spacecraft Power Subsystems

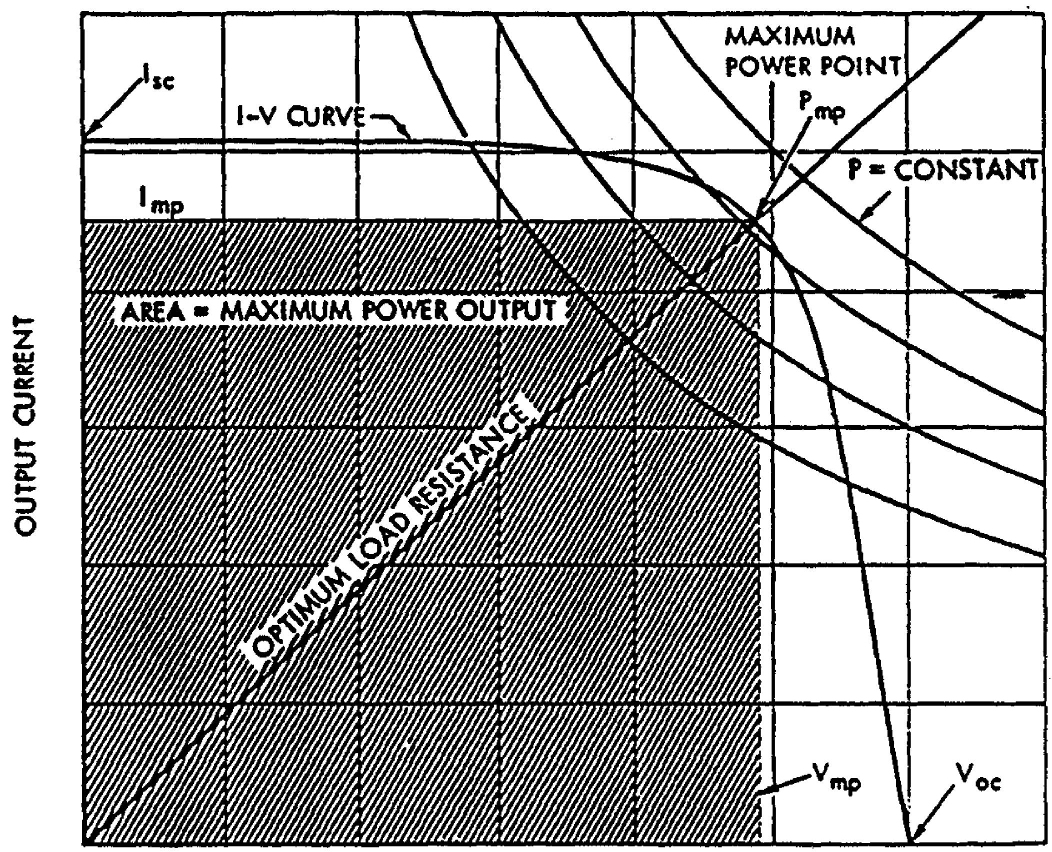
SOLAR POWER -SOLAR CELL CONSTRUCTION



Spacecraft Power Subsystems

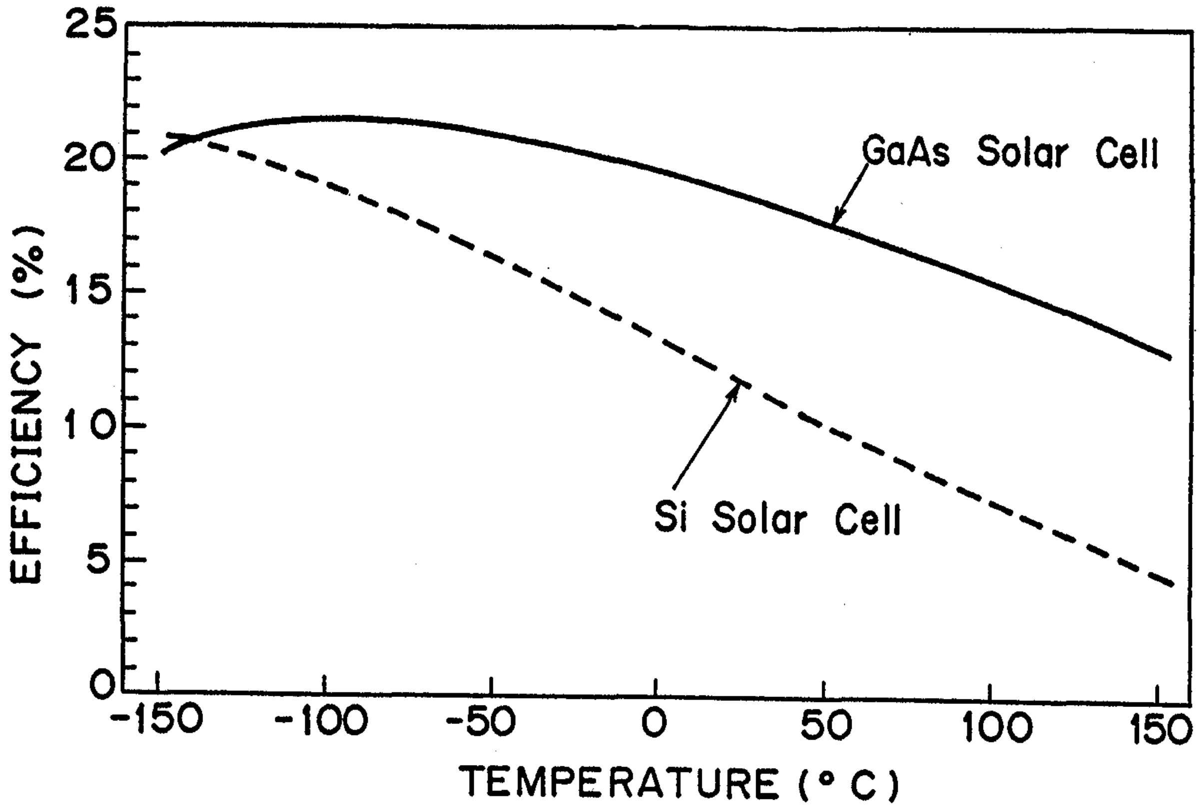
SOLAR POWER -SOLAR CELL ELECTRICAL CHARACTERISTICS

(I/V Curve)



Spacecraft Power Subsystems

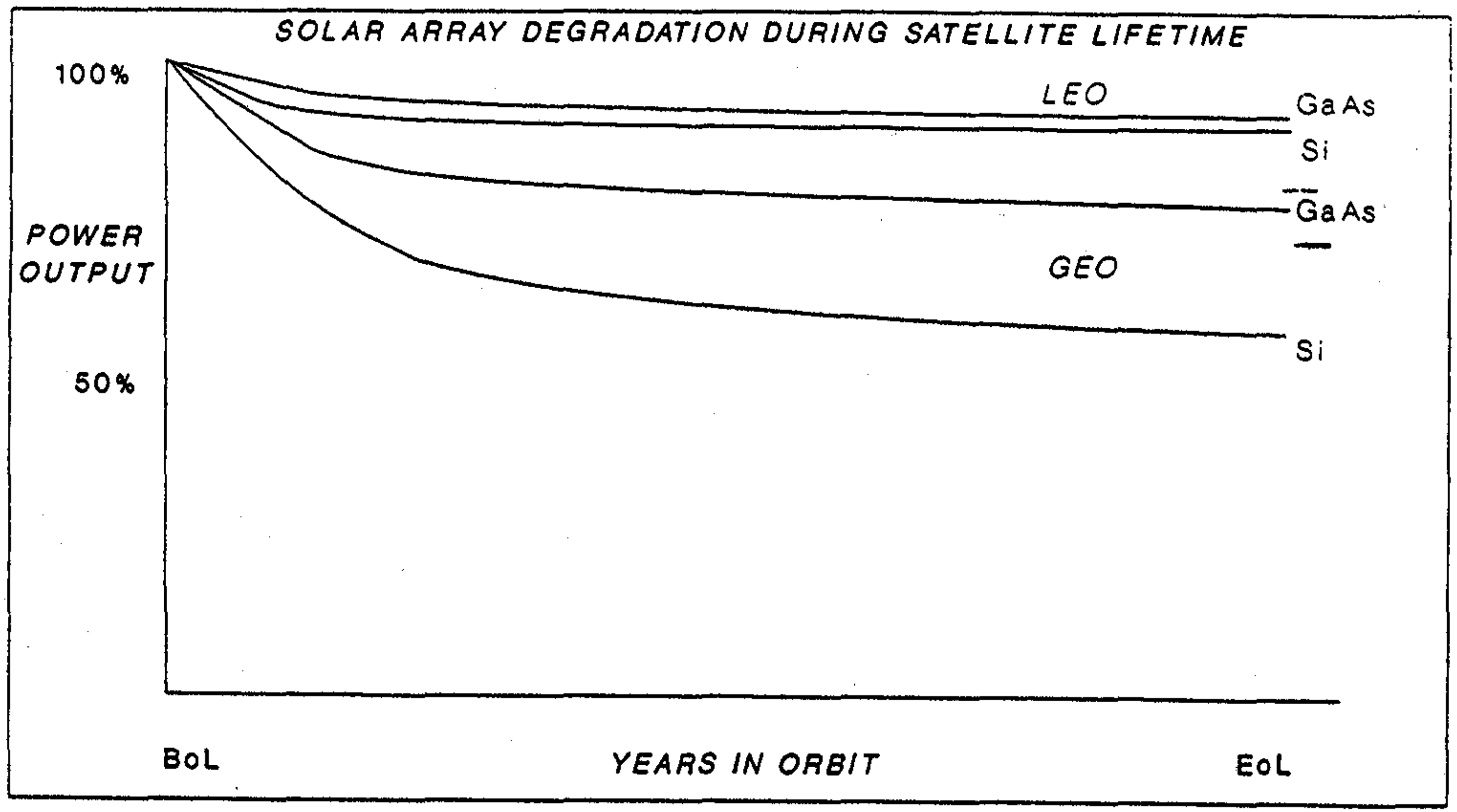
SOLAR POWER -SOLAR CELL ELECTRICAL CHARACTERISTICS



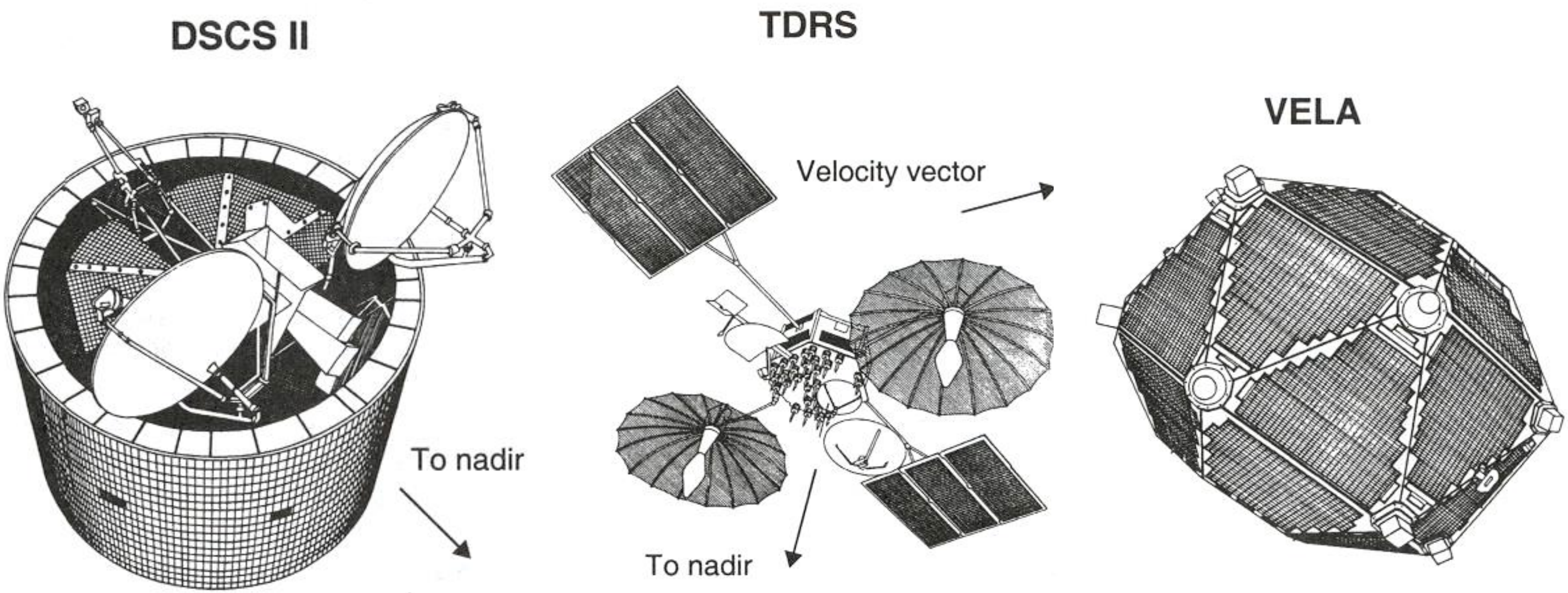
Spacecraft Power Subsystems

SOLAR POWER -SOLAR CELL ELECTRICAL CHARACTERISTICS

(Radiation Dose)



Solar Array Configurations



Spacecraft Power Subsystems

Power Storage

On-board secondary 're-chargable' batteries provide power to spacecraft sub-systems during eclipse periods and in response to peak demand in excess of the solar array capability.

Geostationary satellites experience two eclipse 'seasons' in Spring and Autumn - total of 90 eclipses per year lasting up to 72 minutes each.

Low Earth Orbit satellites experience many more eclipses, dependent on orbit configuration, e.g.:

- 550 km polar sun-synchronous 12 am-pm
    - 15 eclipses each day
    - 30 minutes duration
    - 5500 eclipses each year

Spacecraft Power Subsystems

Secondary Batteries (re-chargeable)

A secondary cell converts electrical energy into chemical potential energy which can be stored until needed and then converted back into electrical energy.

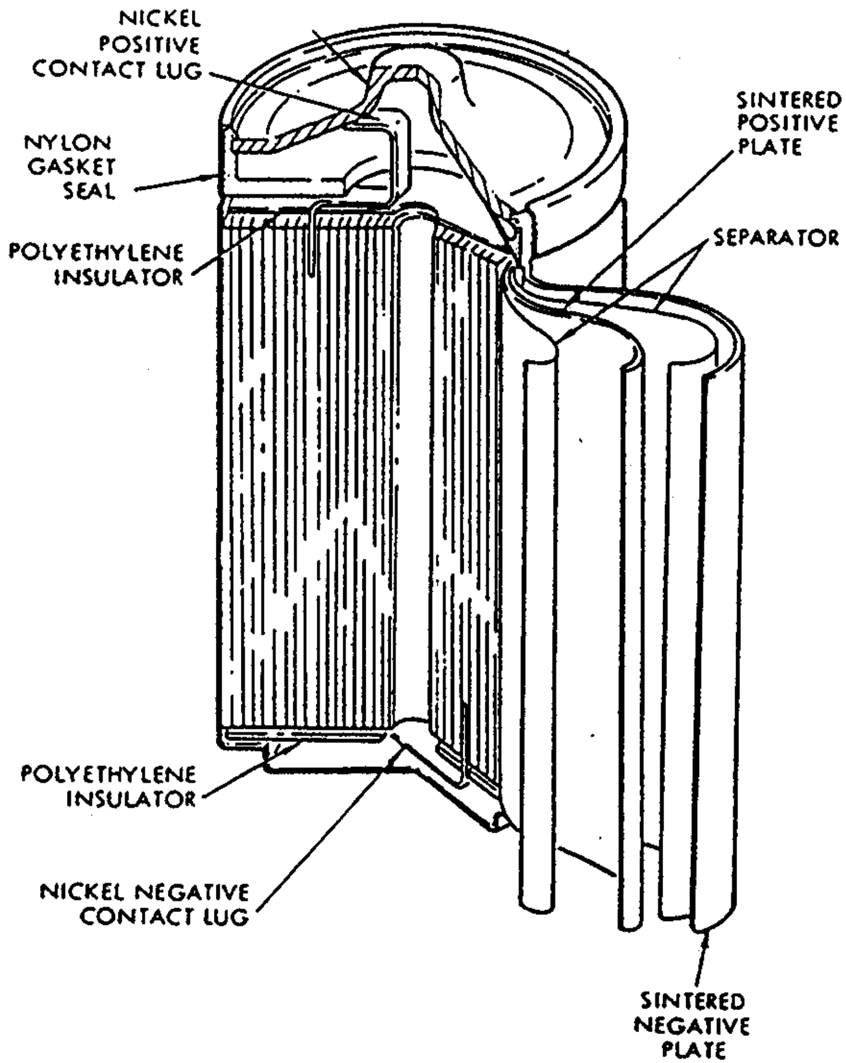
There is always some loss of energy (efficiency) associated with the charge/discharge process.

Different battery types exhibit different electrical properties and should be selected according to mission characteristics, typical battery cells for space use are:

- Nickel-Cadmium (NiCd) – now being phased out due to Cd
- Nickel-Hydrogen (NiH_2)
- Lithium-Ion (Li-Ion) – current technology

Spacecraft Power Subsystems

Nickel-Cadmium Cell Construction



Spacecraft Power Subsystems

Battery Selection Criteria

- Cycle Lifetime
- Depth Of Discharge (DOD)
- Mass, Energy Density (WHr/Kg)
- Terminal Voltage
- Temperature Range
- Storage Capacity (Ampere-Hours, A.Hr)
- Charge Method Used
- Battery Protection

Spacecraft Power Subsystems

Power Control, Regulation & Distribution

Why Do We Need It?

- The power from the solar array fluctuates dependent upon illumination angle and eclipse
- The battery voltage fluctuates dependent upon state of charge
- The spacecraft sub-systems and payloads generally require stable voltages and to be isolated from the effects of other systems
- It is necessary to monitor and switch ON/OFF the spacecraft loads

Spacecraft Power Subsystems

Unregulated Battery Voltage (UoSAT-3)

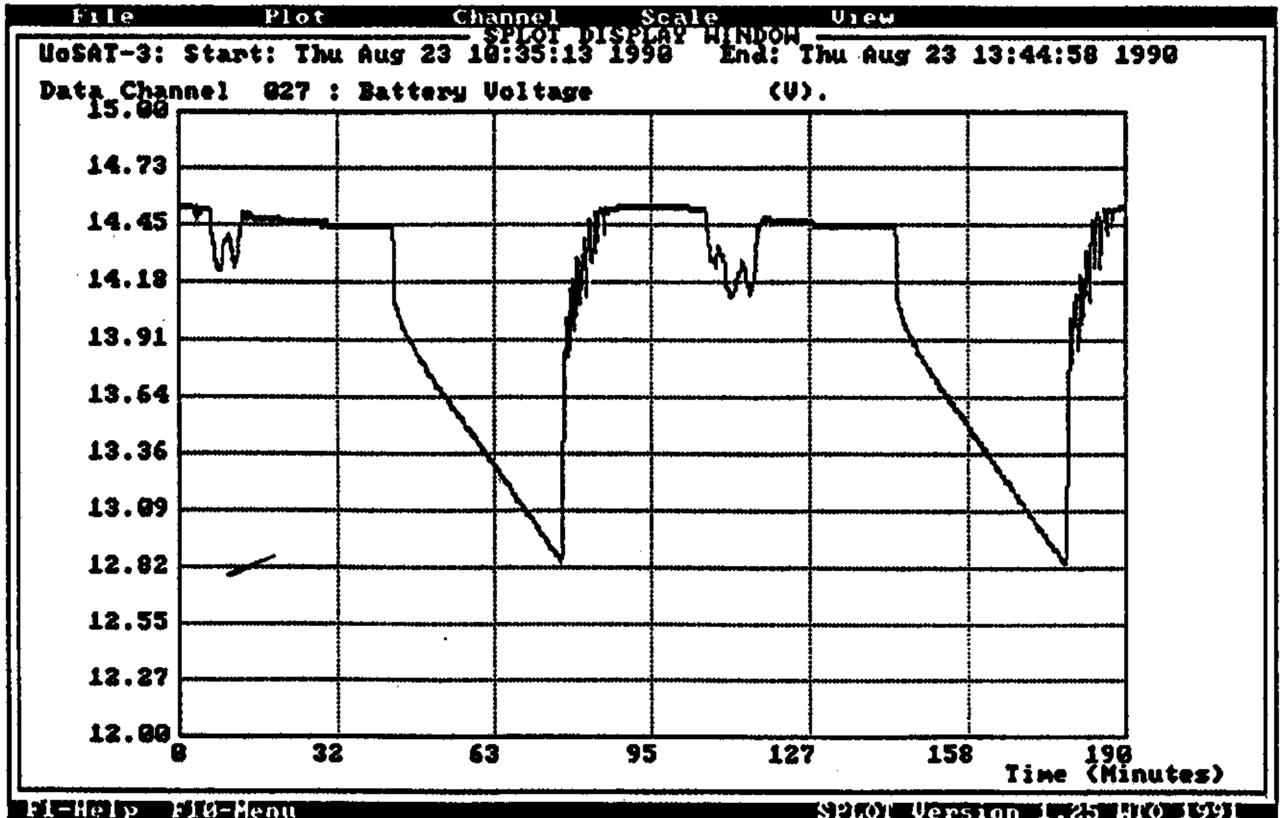


Figure 1 Battery Voltage

Solar Array Design Process

(1&2) Calculate power output of Solar Arrays

$$\underbrace{\frac{P_d T_d}{X_d} + \frac{P_e T_e}{X_e}}_{\substack{Total\ S/C\ energy \\ rqmt\ over\ one\ orbit}} = \underbrace{P_{sa} T_d}_{\substack{S/A\ must\ supply\ the \\ energy\ during\ daylight \\ portion\ of\ the\ orbit}}$$

- P_{sa} = power generated by solar array
- P_e and P_d = S/C power loads during eclipse and daylight
- T_e and T_d = times each orbit spent in eclipse and daylight
- X_d = efficiency getting power from S/A directly to loads (typically is 0.85)
- X_e = efficiency getting power from S/A to charge battery and then from battery to the load (typical value is 0.65)

Solar Array Design Process

(3&4) Determine size of arrays needed to generate power

$$P_{BOL} = \underbrace{(Flux)(\varsigma)}_{P_o \ \ from \ \ SMAD}(I_d)\cos\theta \left[\frac{\mathrm{W}}{\mathrm{m}^2}\right]$$

- P_o = power density output for cells (watts/m^2)
  - *Flux* (or P_i) = input solar power density (watts/m^2)
  - ζ (or η) = efficiency of solar cell material
- P_{BOL} = power density S/A's generate at beginning of life (watts/m^2)
- P_{EOL} = power density at end of life (watts/m^2)
- I_d = inherent degradation
- θ = sunlight incidence angle

Solar Array Design Process

(5) Account for degradation due to exposure to the space environment

$$P_{EOL} = P_{BOL} L_d = P_{BOL} \left(1 - \frac{\text{degradation}}{\text{year}}\right)^{(\text{lifetime in years})} \left[\frac{\text{W}}{\text{m}^2}\right]$$

- P_{EOL} = power density generated at end of life (watts/m^2)
- L_D = lifetime degradation
- Typical degradation/year:
  - 0.0375 for silicon in LEO
  - 0.0275 for GaAs in LEO

$$P_{EOL} = P_{BOL} L_d = (Flux)(\varsigma)(I_d)(L_d)\cos\theta \left[\frac{\text{W}}{\text{m}^2}\right]$$

Solar Array Design Process

(6) Find size of solar array needed at end of life

$$A_{sa} = \frac{P_{sa}}{P_{EOL}} \left[\mathrm{m}^2 \right]$$

Substituting in previous equations:

$$A_{sa} = \frac{\left(\dfrac{\dfrac{P_d T_d}{X_d} + \dfrac{P_e T_e}{X_e}}{T_d} \right)}{(Flux)(\varsigma)(I_d)(L_d)\cos\theta} \left[\mathrm{m}^2 \right]$$

Battery Design Process

Equation for battery capacity:

$$C_r = \frac{P_e T_e}{(DOD)Nn}\left[\mathrm{W \cdot hr}\right]$$

- C_r = total S/C battery capacity
- P_e = average eclipse load (watts)
- T_e = eclipse duration (hr)
- DoD = depth of discharge ($0 \le DoD \le 1$)
- N = number of batteries (need at least two if want some partial redundancy)
- n = transmission efficiency between battery and load (typical value is 0.9)

Battery Design Process

Finding *DoD*:

$$\#\,\text{Cycles} = (\#\,\text{years lifetime})\left(\frac{365.25\ \text{days}}{\text{year}}\right)\left(\frac{24\ \text{hrs}}{\text{day}}\right)\left(\frac{\#\,\text{eclipses}}{\text{hr}}\right)$$

Use chart / tabular data to determine allowable *DoD*

Spacecraft Power Subsystems

Fuel Cells

- Chemical reaction between H_2 and O_2 gives energy and water.
- Advantages:

  High Power / Mass ratio.

  By Product ‘Drinking’ water.
- Disadvantages:

  Limited supply H_2 and O_2.
- Used on short missions (Space Shuttle).

Spacecraft Power Subsystems

Nuclear Energy

- Radio-Isotope thermoelectric Generators (RTGs).
- Thermal Cycle Fission Systems.
- Advantages: Very Long Life
  Very High Power
  “Deep Space Missions”
- Disadvantages: Dangerous
  Very Expensive

Spacecraft Power Subsystems

Radio-Isotope Thermoelectric Generators

- Most Commonly used.
- Used on various interplanetary missions.
- Uses Energy from natural decay of Uranium or Plutonium.
- Thermo-couples convert the heat into Electrical Power.

Satellite Power Systems Examples

- **Check Example 13-2 Page 477 in Understanding Space**