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# NIMBUS POWER SYSTEMS
(1960 - 1969)

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## ABSTRACT

The Nimbus program is a major research effort in the development of satellite technology for meteorological purposes. Global cloudcover information is provided by sensors mounted on a 3-axis stabilized, earth-oriented spacecraft. Electrical power for the spacecraft and sensors is provided by a solar-conversion energy-storage subsystem.

This paper describes the evolution of the power supply subsystems designed for use on the Nimbus spacecraft. The original design, generated during the period 1961-1963, used dissipative regulation techniques and simple battery protection circuits. Battery overcharge protection was provided by ground control of special spacecraft loads. This type of system was successfully flown on Nimbus 1 in August 1964 and on Nimbus 2 in May 1966.

The second generation design, currently being fabricated for test and evaluation, is intended for operational flight use in late 1967. The system concept, based on research and development effort in the period 1962-65, incorporates nondissipative regulation techniques and automatic battery overcharge control.

A new concept, intended for use in a third generation Nimbus power system, is presently being explored. Recent advances in power conditioning technology will allow substantial gains in terms of extended spacecraft life or increased load capability. The new technology, called maximum power point tracking, assures more optimum source-to-load energy transfer. Thus, the increase in available solar cell energy caused by periodic, large, low-temperature excursions of the Nimbus solar array can be fully utilized. The implementation of the tracker technique into the Nimbus multiple storage module concept provides a flexible and unique system wherein the number of storage modules may be easily adjusted to satisfy mission requirements.

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## INTRODUCTION

The Nimbus program, a major research and development effort in the use of satellite technology for meteorological purposes, was conceived in the summer of 1959 and initiated in 1960 by the National Aeronautics and Space Administration (NASA). The program envisioned the development of a complete system (Figure 1) which would include:

- An amply powered, earth-oriented spacecraft capable of supporting a variety of experiments designed for observation of the earth's atmosphere and for rapid transmission of the collected data

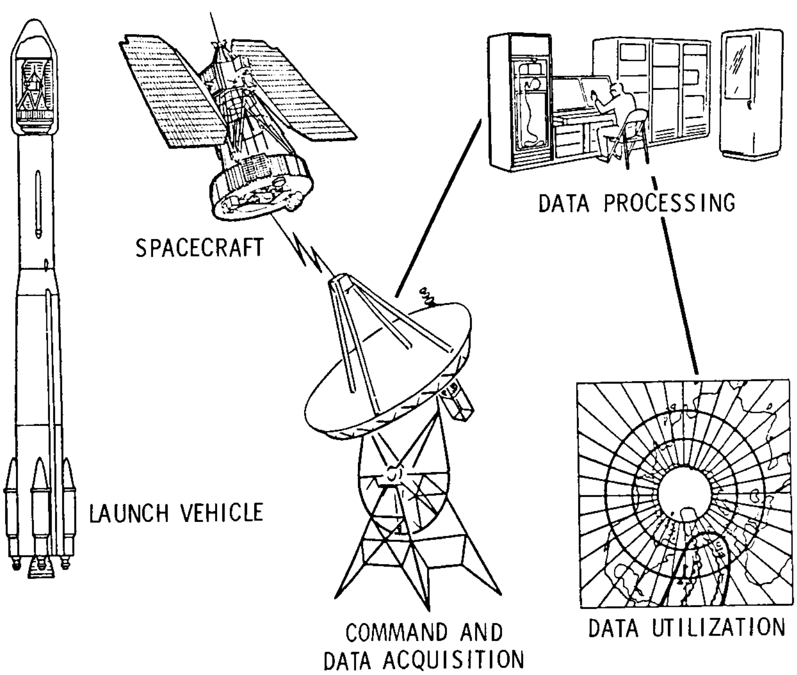


Figure 1. Nimbus System Complex

- An orbit providing worldwide coverage on a daily basis for prolonged periods of time

- A launch vehicle able to place the spacecraft into the required orbit with a high degree of reliability

- A ground command and data-acquisition system to control the spacecraft in orbit, and to receive data

- A sophisticated ground data-processing and transmission system to provide the users (meteorologists and atmospheric physicists) with information in a convenient and meaningful form within an appropriate time scale

A spacecraft which met the foregoing objectives was designed, built, and prepared for flight, a reliable launch vehicle was selected, and a ground complex to support the mission was constructed and placed into operation.

In August 1964, the first Nimbus, Nimbus A (Figure 2), was launched from the Western Test Range (WTR), Vandenberg Air Force Base, California, and was designated Nimbus 1 on achievement of orbit. Nimbus C, originally the backup to Nimbus A, was launched May 1966 and is carrying the full complement of experiments carried on Nimbus 1, plus additional experiments. Nimbus B and D will carry a number of major new meteorological experiments.

The Nimbus program is the subject of reports by Stampfl (1962), by Stampfl and Press (1962), and by Press (1965). The purpose of the present paper is to describe the evolution of the solar-conversion energy-storage power-supply subsystems designed to provide continuous regulated electrical power to the meteorological experiments on the Nimbus spacecraft.

## NIMBUS SYSTEM CHARACTERISTICS

To meet the requirements of the Nimbus program, the basic spacecraft called for a support structure to house the basic meteorological sensors recording atmospheric phenomena, such as global television picture coverage of daytime cloudcover, measurement of infrared and reflected radiation, and measurements of the earth's heat balance. An earth-oriented, three axis stabilized platform appeared best able to meet these requirements. As the sensors must continually point at the earth, a circular near-polar orbit was selected to provide coverage of the various latitudes of the earth. Launch azimuth was chosen to make use of the earth's rotation to provide longitudinal coverage. A low orbital altitude was dictated by launch-vehicle, camera-resolution, and communications-system considerations.

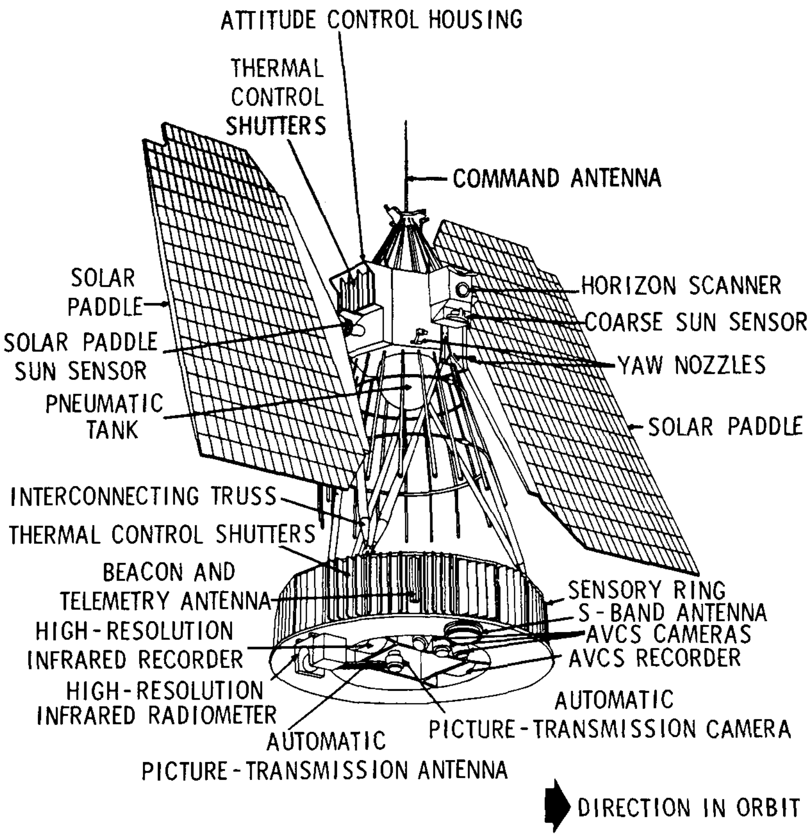


Figure 2. Nimbus 1 Spacecraft

The exact orbit for the Nimbus system was chosen after considerable study. Because the orbit plane of a satellite drifts around the earth, choice of an 81 degree angle from the equator toward the southwest results in an orbital drift corresponding exactly to that of the earth's movement around the sun. Thus, the earth-sun line is in the orbital plane (Figure 3). As a consequence, the solar-power collectors need rotate about only one axis to maintain a sun-oriented position. Launch time was chosen to provide a daytime equatorial crossing which would occur at local "high noon." The resulting high-noon sun-synchronous orbit provides a feasible picture-lighting condition for the televisi( sensors.

Another distinctive feature of the Nimbus system is the complete modular design approach (Figure 4). A separate and independent attitude-control subsystem, together with a sensory ring design consisting of 18 separate modular bays, allows for separate development, evolution, and improvement of individual subsystems with a minimum of interference problems. This flexibilit) feature permits the smooth product improvement and evolution which is part of the Nimbus concept. Such a design has great flexibility, and the basic spacecraft offers a general-purpose platform for a multitude of experiments.

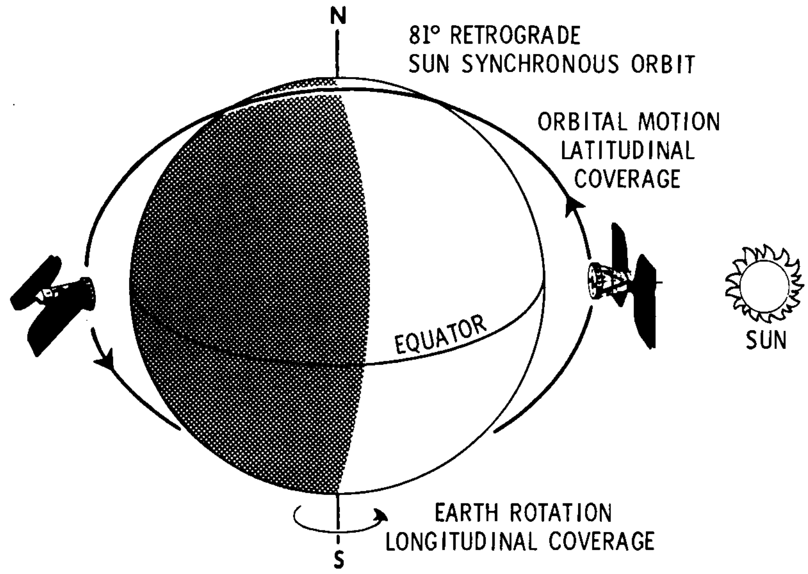


Figure 3. Nimbus High-Noon Sun-Synchronous Orbit

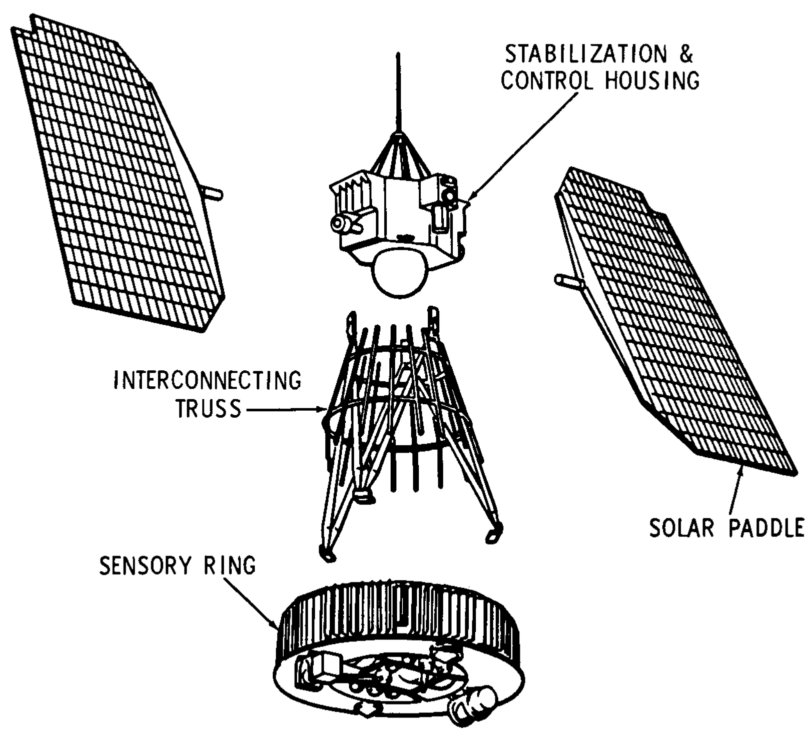


Figure 4. Nimbus Module Design

## POWER SYSTEM CONSIDERATIONS

With guidelines established for the basic spacecraft design, work began on the various subsystems. An immediate task was to define the interfaces between the subsystems and establish the general power-system configuration. Items of particular interest to the power subsystem were the interface with the attitude-control subsystem, the anticipated power-demand profile, the choice of an output voltage, and the interconnection of the various elements of the power system.

### SUBSYSTEM INTERFACES

Two important areas were investigated almost immediately: the interface between the attitude-control subsystem and the power subsystem, and the output voltage level of the power subsystem. By placing all the attitude-control subsystem work in one area, mutual problems between the three-axis stabilization requirements and the single-axis paddle-rotation requirement could be resolved quickly. The interface with the solar paddles is a shaft to which the paddles are attached by split clamps. In the center of the shaft is a connector tying the electrical output of the paddles to the rest of the spacecraft through a set of slip rings.

### OUTPUT VOLTAGE SELECTION

The output voltage (-24.5 volts dc, regulated to ±2 percent) was chosen on the basis of experience gained in the TIROS program. Voltage requirements of the television sensors and other subsystems conflicted with the then existing low $V_{CE}$ ratings of the available power transistors. A compromise was made at the largest possible value of voltage consistent with adequate derating of the power transistors. A second factor in choosing the voltage was the need to make maximum use of instrumentation and circuitry already under development for TIROS, in order to reduce the cost of developing the overall system and to ease the tight schedule requirements.

### POWER DEMAND PROFILE

The next major consideration in designing the power system was the power-demand or load profile. Figure 5 shows the power-demand profile for a typical Nimbus spacecraft. Several features of this profile are noteworthy: First, the spacecraft has a minimum load which the input power must meet, or the spacecraft will be lost. Second, the experiment or sensor load falls naturally into two sections, an earth-day load and an earth-night load. The earth-day load consists of those sensors which are used to view the illuminated

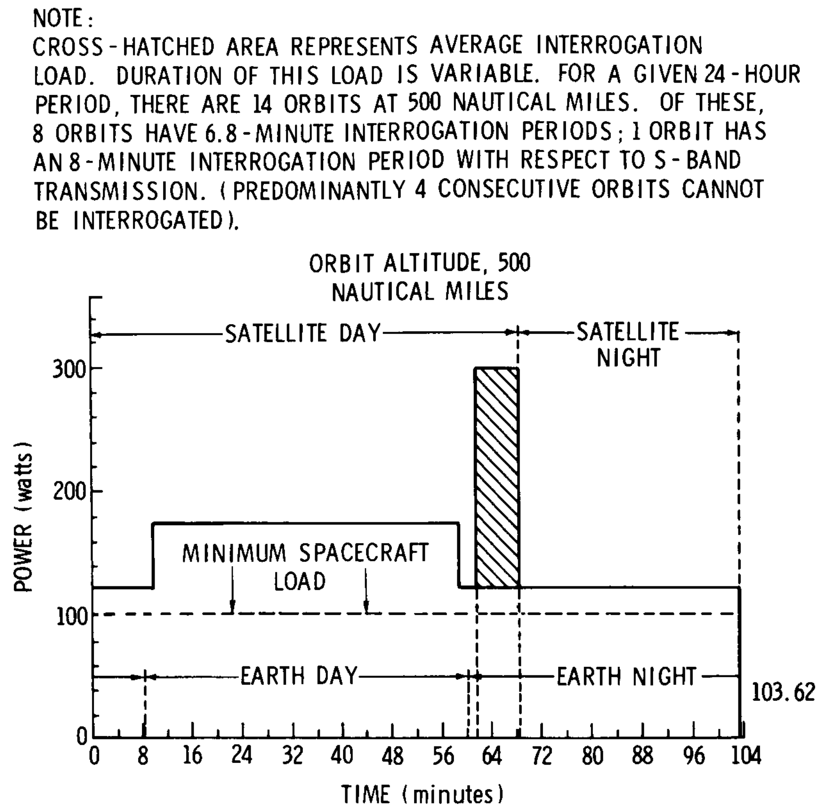


Figure 5. Nimbus 1 Power Demand Profile

portions of the earth; these are turned ON automatically just after the spacecraft begins to view the sunlit portion of the earth, and are turned OFF just before the spacecraft begins to view the dark side of the earth. Conversely, the nightload sensors are intended to view the dark portion of the earth; these are also automatically turned ON and OFF as the spacecraft moves into proper position.

To attain complete earth coverage, television and infrared data are stored on magnetic tape and then transmitted to earth once each orbit. To take 100 minutes of data, and transmit these data to earth during a 5-minute pass over a ground station, requires a high-rate data-transmission system. Such a system uses a large amount of power, hence, the large power spike shown in the demand profile. Note that the time appearance of this spike is a function of many variables such as season of launch, interrogating ground-station location, and operational plan. The pulse can occur entirely in the sunlight, partially in sunlight, partially in dark, or entirely in the dark. The power system must be designed to handle all these cases.

## PACKAGING AND STORAGE RELIABILITY CONCEPTS

The reliability of the power system rests on redundancy, pre-usage conditioning of semiconductor devices, and a rigorous qualification-test program. The Nimbus modular concept has led quite naturally to use of the building-block technique. Basic elements of the system were designed to be complete working units. The system was then built by connecting the working units in parallel, a minimum number to meet the design requirements, plus an additional number to provide redundancy. Use of pre-conditioned transistors takes advantage of the process in which weak elements are screened out of flight hardware, resulting in greater reliability for the design and for the end product. As the program progressed, several components and designs were revised when a particular component type was found unable to pass the preconditioning test with a satisfactory yield.

The Nimbus hardware qualification test program is designed to produce reliable flight hardware through rigorous environmental testing. Each subsystem design is qualified to prototype levels of humidity, acceleration, vibration, and thermal-vacuum cycling. Environmental limits of these tests are far more severe than the anticipated flight environment, thus ensuring adequate design margins within the subsystem. Hardware must be operational during these tests, and must survive without any permanent degradation of performance.

Flight hardware is qualified as a subsystem in less severe vibration and thermal-vacuum cycling environments. Hardware must be operational throughout the duration of the tests with no measurable degradation of performance. The subsystems are then assembled and integrated into a flight spacecraft which is tested as a unit in vibration and in a thermal-vacuum chamber. During the thermal-vacuum test, the entire spacecraft operation is controlled through the RF links. Hardwiring connections are limited to solar-array power simulation and those required for thermal and electrical safety precautions. At the completion of this series of tests, a flight power subsystem has demonstrated satisfactory performance through a simulated launch, simulated solar-array acquisition of the sun, and 300 to 400 orbit cycles of operation at both high and low thermal-vacuum conditions.

## INTERCONNECTION OF POWER SYSTEM ELEMENTS

Finally, the interconnection of the power system elements was considered. The primary source of power was to be the sun. Solar cells would convert energy received from the sun to electrical energy.

The converted energy which supplies spacecraft loads is stored in electrochemical batteries for use during spacecraft eclipses. The batteries also supply short-duration peak loads which exceed the capability of the solar array. The load capability of the power system depends directly upon the size of the solar array. Trade-offs are possible between day-load and night-load power demands, by proper programming of the spacecraft sensors, as long as sufficient battery charge current is available to maintain a charge-discharge energy balance of the storage elements.

## NIMBUS 1 AND 2 POWER SYSTEMS

The first-generation Nimbus power-system development began in 1960. An ambitious program was undertaken to develop and qualify prototype hardware in a period of about 8 months, with flight hardware to follow about 4 months later. Conventional and proven techniques were emphasized, such as dissipative series regulation, current limited charge, and the use of TIROS type F nickel-cadmium storage cells. Circuits were to be as simple as possible, and charge control was to be maintained by ground-command-controlled load programming. The philosophy adopted in power-supply fusing was that circuits whose loss would not result in a spacecraft failure would be protected by fuses.

## POWER SYSTEMS DESCRIPTION

The Nimbus 1 and 2 power-supply subsystems, designed to meet the general requirements just stated, will provide no less than 160 watts of continuous spacecraft power for a minimum mission lifetime of 6 months. Functional subsystem requirements were:

- Acquisition of incident solar radiation, and photo-voltaic conversion to electrical power

- Energy storage and release by electrochemical means

- Regulation, control, and distribution of the power at a voltage level suitable to the spacecraft subsystems

The stated requirements are fulfilled by two major groupings of hardware: two solar-cell platforms attached to the control subsystem housing by platform driveshafts, and sensory ring hardware consisting of seven parallel-connected battery modules and a control or electronics module. Figure 6 is a functional block diagram of the Nimbus 1 and 2 power systems. The sensory ring, through a compartmented design and simple interface connections, provides a housing and independent thermal-control capability for each module. Table 1 shows physical dimensions and weights for each power-supply assembly and for the total system.

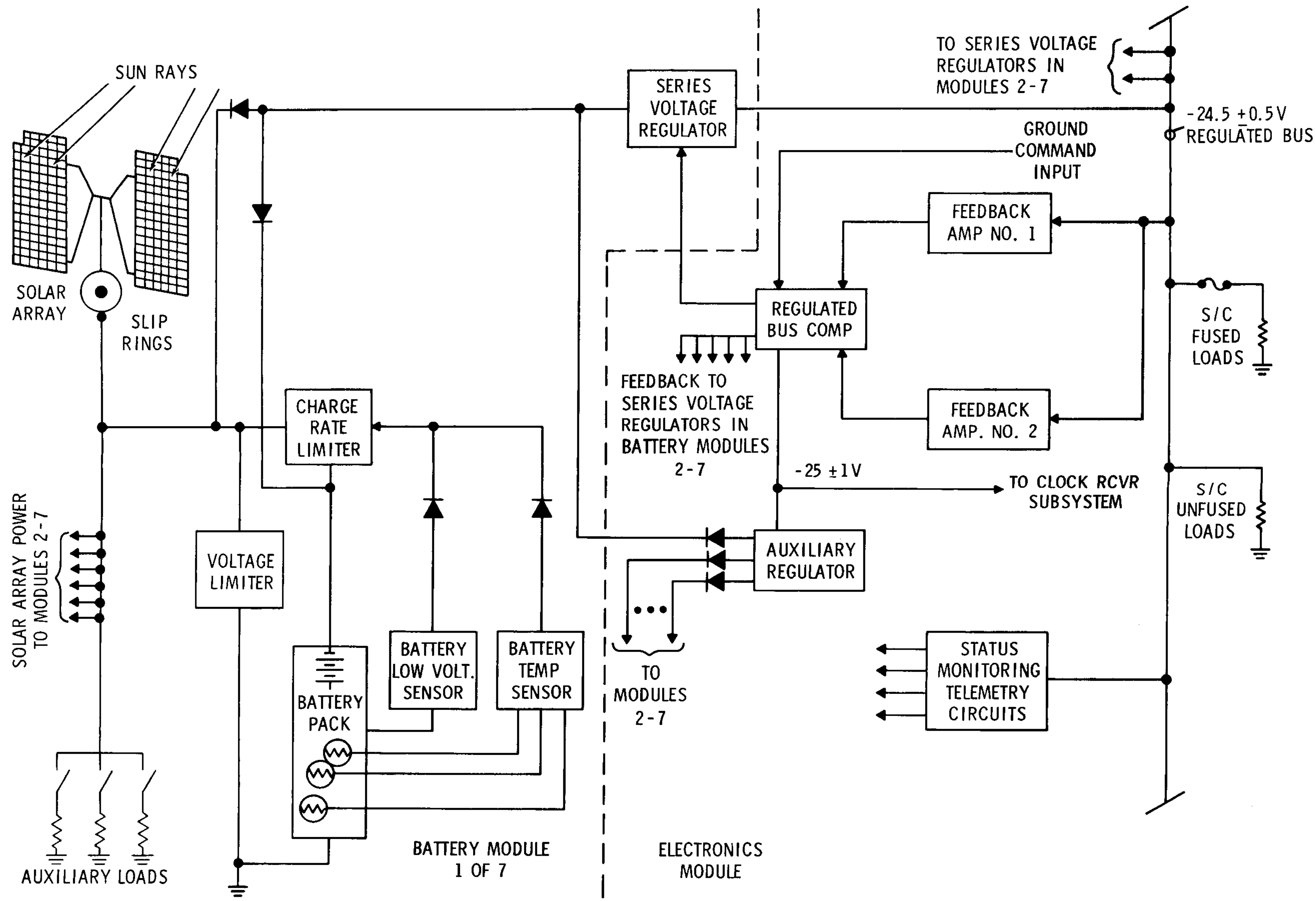


Figure 6. Nimbus 1 and 2 Power Systems, Simplified Block Diagram

Table 1

Size and Weight of Nimbus 1
Subsystem Assemblies

| Assembly | Dimensions | | | | Weight (lb) |
|---|---|---|---|---|---|
| | Width (in.) | Height (in.) | Depth (in.) | Volume (cu. in.) | |
| Single solar cell platform, including transistion section | 46.75 | 96 | - | - | 38.5 |
| Battery module | 6 | 8 | 6-1/2 | 312 | 15.2 |
| Electronics module | 6 | 4 | 13 | 312 | 6.8 |

Total power system weight is 190 lbs.

## PHOTOVOLTAIC ENERGY CONVERSION

Energy radiated from the sun and used in direct photovoltaic conversion provides electrical power for the spacecraft. A platform assembly of 2 x 2-cm phosphorus-doped silicon n-on-p cells constitutes the photovoltaic mechanism which converts the energy. The solar array intercepts solar energy at a rate of 5822 watts (140 milliwatts per square centimeter nominal intensity) during the sunlight period. Assuming a solar-cell module efficiency of 10.1 percent at air mass zero (25°C), the environment in space, and the power-supply operating characteristics, the array can supply approximately 460 watts at the design operating point at a temperature of 55°C at the beginning of life.

Each solar-cell platform consists of 5472 cells, a mounting structure, a transition section, a latching assembly, a drive motor with an associated gear-reduction unit, and a control-shaft clamp. The solar cells, mounted on one side of the aluminum honeycomb platforms and maintained continuously incident to the earth-sun line, can intercept a maximum of solar energy during the sunlight period.

## ENERGY STORAGE

The Nimbus system concept includes storage of energy in multiple parallel-connected battery packs to provide fail-safe redundance. A battery module contains a single battery pack composed of 23 series-connected nickel-cadmium cells. Nominal capacity of each cell is 3.5 ampere-hours for Nimbus 1, 4.5 ampere-hours for Nimbus 2. Depth-of-discharge is limited to a maximum of 20 percent during normal system operation. If a battery pack fails in a semi-to-full short mode, the failed unit automatically adjusts to accept only a trickle charge during satellite day, and automatically isolates itself from the remaining units during satellite night, so that normal system operation may continue. Six battery packs were sufficient to sustain full spacecraft system capability for the Nimbus 1 and 2 missions.

Inherent in the multiple battery-module concept is individual charge control for each battery pack. A charge regulator limits the maximum battery-charge current to 1.5 amperes. Two measures employed for battery protection can individually reduce the charging current to a nominal trickle rate of 300 milliamperes. Three sensors strategically placed in the pack monitor the pack temperature. If any one of the sensors reaches 65°C, the pack is switched to the trickle rate. Battery voltage is continuously monitored and, at a pack voltage of 21 volts or below, a trickle rate will be maintained. This guarantees maximum use of available system-recharge current should a battery pack fail catastrophically. The Nimbus battery modules also contain solar-array power limiters and status-monitoring telemetry circuits. A total system limiter consists of seven parallel circuits to provide redundancy and maximum protection against catastrophic failures. The limiter prevents voltage excursions of the array beyond 40 volts to protect the various regulation circuits from over-dissipation caused by excessive array voltage. Circuits whose status is monitored to ascertain system performance include battery temperature, battery voltage, and battery-charge and battery-discharge current. The output of each status-monitoring point is suitably conditioned for input to the telemetry subsystem.

## POWER CONTROL AND DISTRIBUTION

The electronics module performs a centralized power-control and distribution function for the subsystem. Each spacecraft subsystem receives power from a common bus at a -24.5 volt ±2 percent level. Exothermic pyrofuses are provided for each noncritical load. Power control is accomplished by a dissipative regulation system. Each of the battery modules contains a separate load-current-regulating element preadjusted to share the total load current and driven by a common load-bus-derived error signal. Individual regulator elements, fused for short-circuit protection, are monitored by telemetry to indicate overall status. An isolated connection to the input of each element from its associated battery pack and/or the solar array is made through a disconnect diode. The regulated bus voltage is sensed in the normal manner, and one of two redundant feedback amplifiers feeds back an error signal to the regulating elements. A regulated bus comparator senses the main bus voltage for low or high limit conditions; if the bus voltage exceeds the design limits, an error-switching signal initiates switchover to the redundant amplifier. A ground-commanded switchover facility is provided as an emergency backup. An auxiliary regulator, receiving a proportionate amount of input power from each of the battery modules, provides power for the comparator and associated ground-command channels in the spacecraft clock receiver. The independent power source ensures proper comparator and clock-receiver operation should a sudden loss of the main bus occur. The electronics module also provides status-monitoring circuits, including unregulated and regulated bus voltages and currents, and the operational status of the feedback amplifiers.

## PERFORMANCE AND RESULTS

Nimbus 1. The Nimbus 1 spacecraft was launched from the Western Test Range on August 28, 1964. The power supply functioned properly during both the nighttime and daytime portions of the orbit to provide all power required by the spacecraft. The solar array provided an average current of 13 amperes, which was within design specification, and a maximum power output of 470 watts, which exceeded the original maximum power requirement by 20 watts. The supply delivered -24.5 volts regulated within ±2 percent. Battery voltages were maintained within specification with the use of proper auxiliary and compensating loads. No degradation of the solar cells' power output was observed during the short life of the spacecraft. On September 23, a failure in a mechanism of the attitude and control system providing solar paddle rotation caused a lockup leaving the solar paddles in an immovable position. The reduction in power meant that minimum spacecraft power requirements could no longer be met, and eventually the spacecraft lost stabilization and meteorological usefulness.

Nimbus 2. Nimbus 2 was successfully launched on May 15, 1966. The spacecraft achieved a near-perfect orbit. The power supply met specifications and performed properly.

## NIMBUS 1 AND 2 SYSTEM IMPROVEMENTS

Of primary importance to the design of optimized power systems is the efficiency of dc voltage regulation. As noted earlier, severe schedule constraints and adherence to state-of-the-art technology required the use of dissipative regulation in Nimbus 1 and 2. During the fourth quarter of 1962, an ambitious program was undertaken to develop a 500-watt pulse-width-modulated switching regulator capable of 90-percent average orbital efficiency, compared to 75-percent efficiency for the dissipative model. The regulator was to be completely adaptable to the Nimbus modular power-system concept. Once developed, the regulator would increase the power available for spacecraft use from 160 to 195 watts (average) during an orbit. Development work was successfully completed in July 1964. Figure 7 shows salient results for the load power and input-voltage ranges noted. After a series of spacecraft compatibility studies, the design was adopted for use in the Nimbus B power system scheduled for flight in the last quarter of 1967.

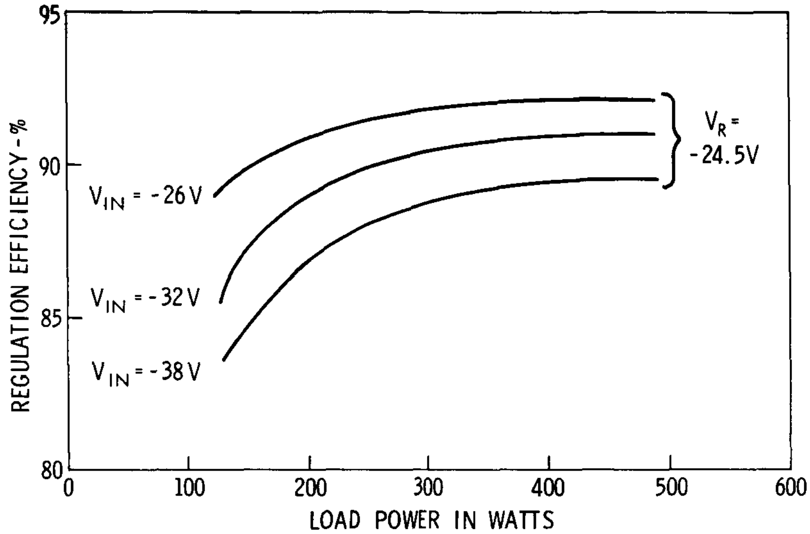


Figure 7. Nimbus B Switching Regulator Efficiency Characteristics

## NIMBUS B POWER SYSTEM

Figure 8 is a functional block diagram of the Nimbus B power supply. As a result of mission similarities, the solar array is identical to that used on the Nimbus 1 and 2 spacecraft. Nimbus B uses 8 rather than 7 battery modules in conjunction with an electronics module to support increased mission requirements. The remainder of the Nimbus B power system, although employing Nimbus 1 and 2 design concepts and nomenclature, differs markedly in the areas of battery-overcharge control, battery protection, handling of excess array power, and (as previously noted) dc voltage regulation.

## BATTERY MODULES

Each Nimbus B battery module contains a series string of 23 4.5-ampere-hour nickel-cadmium storage cells connected to the solar-array bus by a closed-loop charge controller, instead of the open-loop bi-level charge-rate limiter used on Nimbus 1 and 2. Figure 9 shows the operation of the controller, In Region I, charge current is maintained as close as possible to the maximum allowable rate dictated by battery requirements and energy-balance considerations.

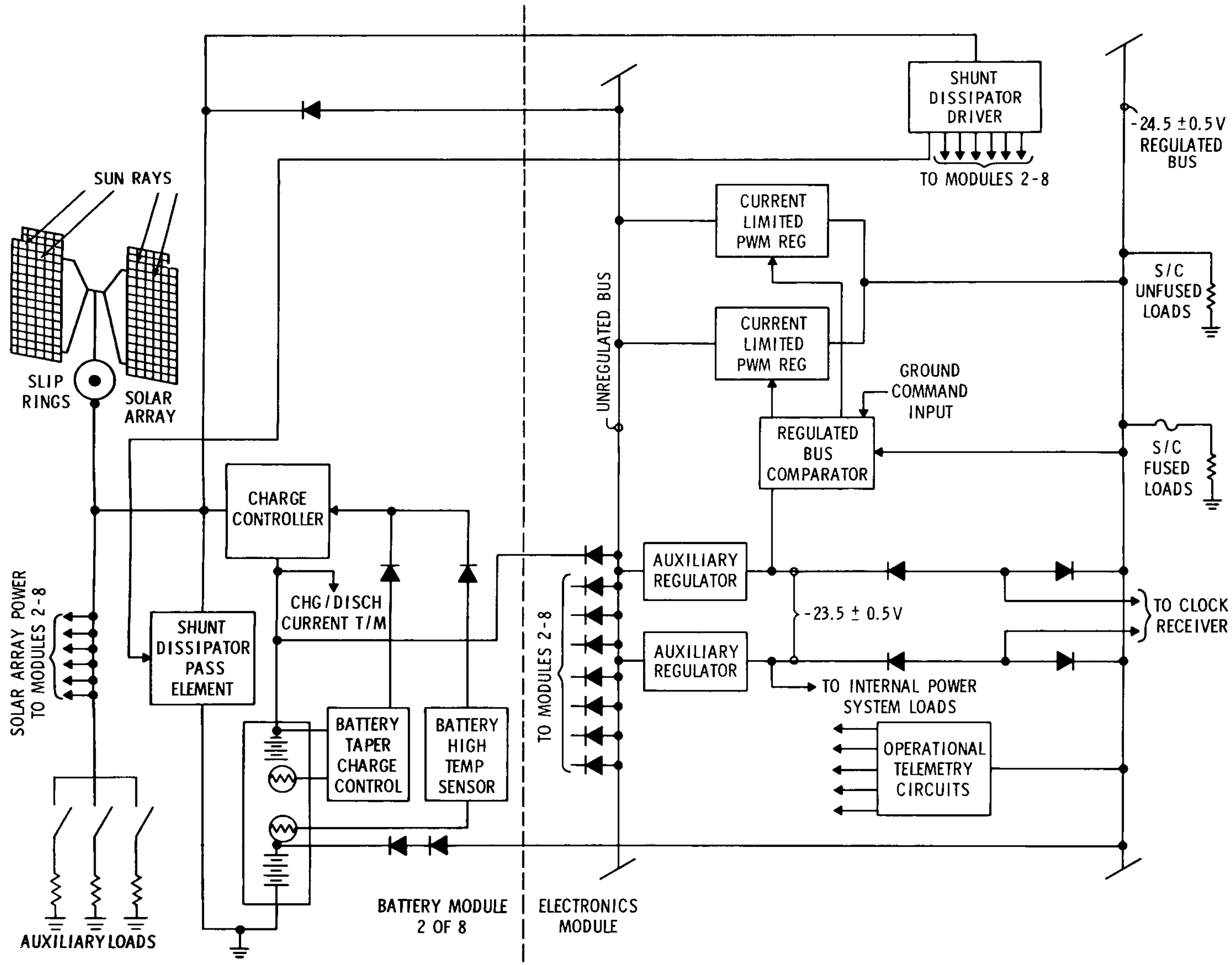


Figure 8. Nimbus B Power System, Functional Block Diagram

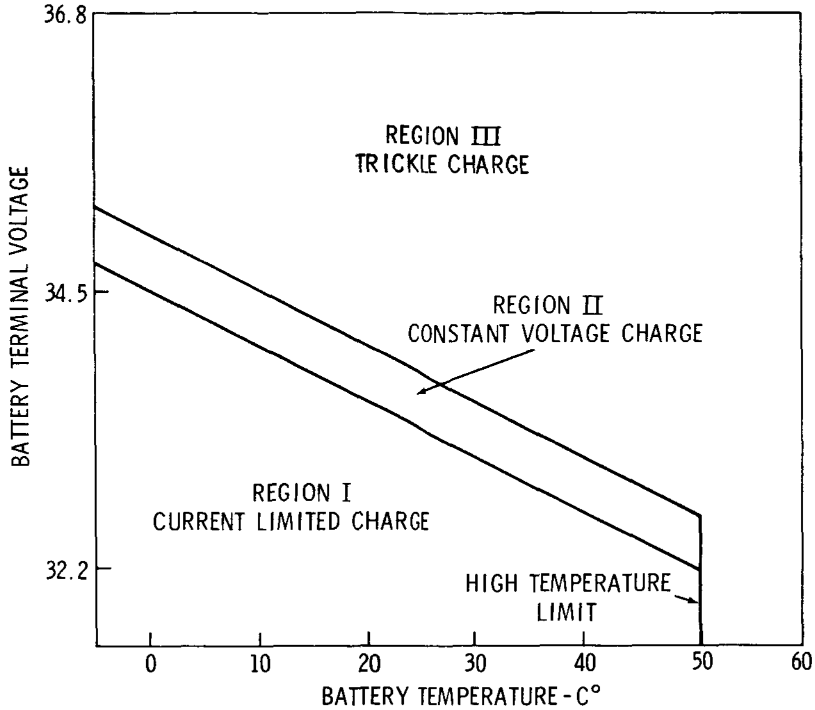


Figure 9. Nimbus B Charge Controller, Regions of Normal Operation

As battery recharge occurs, either or both its terminal voltage and temperature will increase. In Region II, it is desirable to limit the voltage to which the battery may charge and adjust the limiting voltage as a function of battery temperature. From a battery standpoint, temperature-compensated, voltage-limited-charge is desirable to prevent non-reversible hydrogen evolution within the cell under overcharge conditions. In Region III, the charge controller operates as a constant current regulator in essentially the same manner as in Region I, except that the regulated current becomes the trickle charge current. Conditions leading to Region III operation are excessive battery-pack temperature or a combination of battery voltage and temperature which can force the voltage/ temperature (Region II) circuits to reduce charging current to the trickle charge limit. A ground-command override circuit permits restoration of normal charging should an emergency arise.

Telemetry voltages proportional to battery temperature, voltage, and current are provided as module outputs.

A shunt dissipator in each battery module provides control over approximately 75 watts of solar-array bus power. A single feedback amplifier in the electronics module controls all shunt regulators in the system.

ELECTRONICS MODULE

As shown in Figure 8, the electronics module contains the shunt-regulator amplifier, a regulated bus comparator, redundant auxiliary regulators, subsystem operational telemetry, and redundant pulse-width-modulated switching regulators.

The switching regulator can deliver up to 20 amperes load current at -24.5 volts with an input voltage range from -26 to -55 volts. In addition, an output current limiting feature designed into the regulator will afford short-circuit protection. The resulting regulator can sustain a continuous zero-impedance short-circuit load without internal damage.

The requirement for short-circuit protection arises primarily from the existence of fused "non-essential" spacecraft loads. If a fused load presents a short circuit to the power system, the regulator output voltage will fall because of the current-limiting feature. Although the limit value of current has been set low enough to protect the regulator, it may not be adequate to clear a large fuse in a reasonable time (i.e., less than 5 seconds). For this reason, the spacecraft storage batteries have been tapped at a voltage level of approximately 20 volts and have been "OR" gated into the regulator output bus. Figure 10 shows the resulting $V_o$ - $I_o$ characteristic for the regulator/battery-tap connection. In this case, any load which draws the regulated bus voltage down to the battery-tap voltage will see a low impedance bus capable of delivering in excess of 200

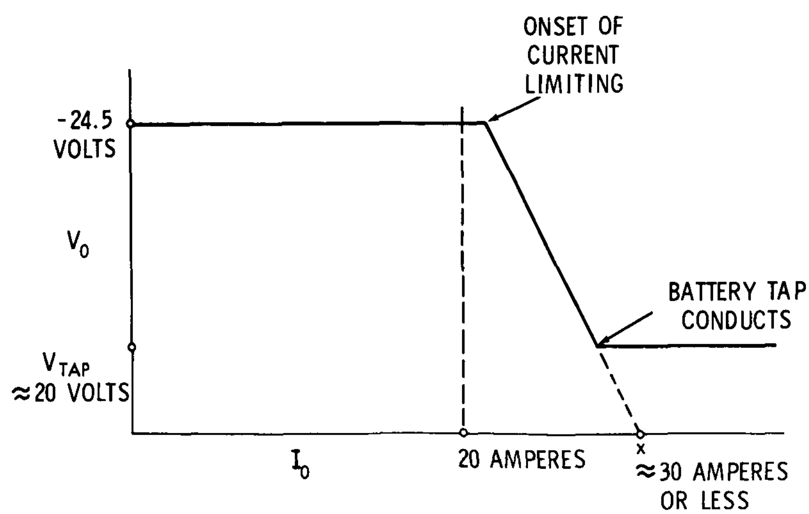


Figure 10. $V_o - I_o$ Characteristic for Regulator/Battery-Tap Connections

amperes. This current level is adequate to clear any of the fuses anticipated for use in Nimbus in a few milliseconds.

Figure 11 is a functional diagram of the regulator system, which provides redundancy for all essential functions except input and output filters. The action of the voltage control circuits is obvious. The current control, which senses output current through the indicated sensor, acts to inhibit the voltage feedback and assume control of the regulator whenever an excessive level of output current is reached.

A regulated bus comparator functions in a manner identical to that of the Nimbus 1 and 2 systems. When a change of more than ±2.0 volts from the regulated -24.5 volts occurs, the comparator provides the logic and control necessary to turn ON the standby regulator and shut OFF the defective or failed unit. Independent ground-commanded switch-over facility is again provided.

Whereas the Nimbus 1 and 2 systems included a single auxiliary power source to provide critical operational backup in case of temporary loss of the

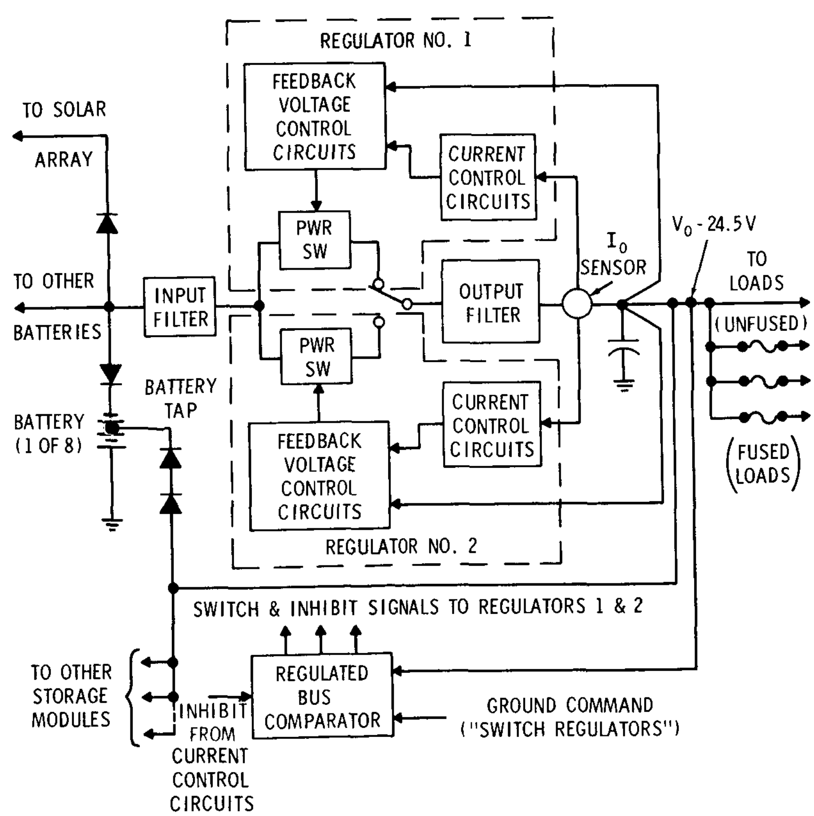


Figure 11. Nimbus B Regulator System, Functional Block Diagram

main bus, the Nimbus B system contains two auxiliary regulators: one unit provides power for battery-module protection circuitry, and the other provides power for the electronics-module circuitry. Each auxiliary regulator is diode-gated with the regulated bus to provide power to a redundant pair of spacecraft clock receivers if the main bus is temporarily lost.

As noted previously, the electronic module includes a common driver for the battery-module shunt dissipators. A shunt dissipator limits the maximum voltage that can appear on the solar-array bus, thus preventing high-voltage damage to the load regulators or excessive dissipation in the charge controller. During conditions of low spacecraft power demand, the power-subsystem operating point will attempt to move along the solar-array I-V curve in a direction towards the array open-circuit voltage. For a current demand of 5 amperes, the operating point could be at array voltages exceeding -60 volts at cold-array temperatures. Normally, the shunt dissipator will be inoperative. The array-operating voltage will typically range between -30 and -35 volts during an orbit, depending upon battery state-of-charge and temperature.

The electronics module also provides operational-status telemetry for regulated and unregulated bus voltages, solar-array and regulated bus current, regulator operational status, auxiliary regulator voltage, and shunt dissipator current.

Engineering and prototype system hardware is under construction, with prototype qualification testing scheduled for completion in late 1966. Flight hardware will be constructed and qualified in mid-1967 for a scheduled launch date of late 1967.

## MAXIMUM POWER TRACKING SYSTEM

Upon completion of the Nimbus B 500-watt switching-regulator development, it became apparent that the techniques evolved could be extended to increase the efficiency of the power system. It was decided to investigate this possibility on a parallel-effort basis and for later incorporation into flight hardware.

In the present system, when the array emerges from the earth's shadow, the low solar-cell temperature drives the array maximum-power point to a voltage far in excess of the range (determined by the maximum battery-charge voltage) in which the conventional system is designed to operate. The array-output voltage is clamped to a relatively low initial-recharge voltage level, plus a small charge-regulator drop. The resulting quiescent-power point is at a level far below the potential power capability of a cold-cell array. The advantage of having nearly the same current flow at the array output, but at a much larger voltage, is not realized.

Figure 12, based on actual data for an earth-orbiting spacecraft, typifies the situation. The dotted line indicates maximum available power during satellite day. The associated energy could be stored in the battery or delivered to loads. The solid line indicates the maximum power extracted from the array by the conventional system. In this case, use of maximum available power would allow roughly a 100-watt (or 20-percent) reduction in solar-array power requirements.

As the spacecraft goes further into the sun-illuminated portion of flight, the solar-cell temperature rises, reducing the maximum power-point voltage of the array to more nearly match the battery-charge voltage. After recharge of the battery, the charge mechanism reduces the charge current, creating a condition of excess energy (and, therefore, excess voltage) at the array output. Generally, the excess energy is absorbed either in the charge controller or in a specially designed excess-power shunt dissipator; in either case, intolerably large amounts of power may have to be dissipated. The resulting spacecraft thermal-balance problem becomes particularly severe where there is a large design difference between the beginning-of-life and end-of-life array output.

Recent advances in power-conditioning technology will substantially increase spacecraft life or load capability by assuring an optimum source-to-load energy transfer. Referred to as "maximum power tracking," this concept

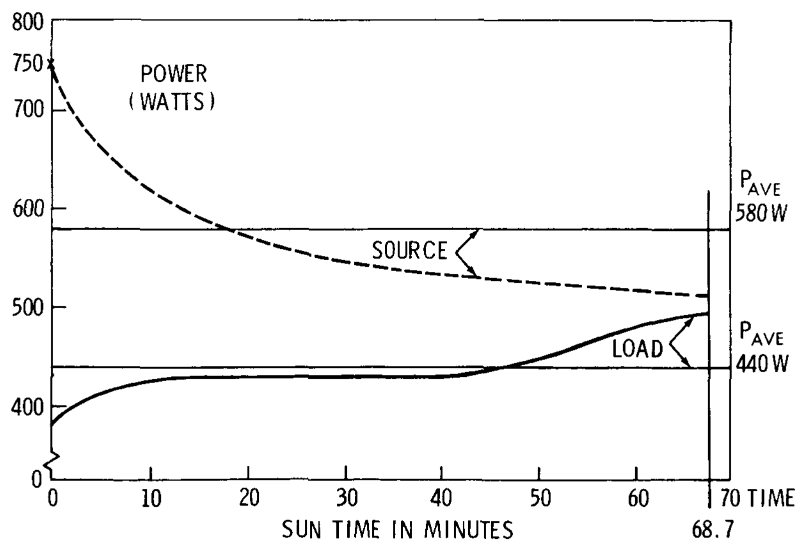


Figure 12. Maximum Available Array Power Used by a Conventional Power System vs Sun Time (For a Typical Earth-Orbiting Satellite)

involves delivery of maximum available source power (in this case, solar-array power) to the spacecraft, regardless of source variations caused by environment or time. This "optimum" transfer of energy results from adaptive control of pulsewidth-modulation (PWM) power-conditioning circuits. This system effectively matches the input impedance of the spacecraft to the output impedance of the solar array at its maximum power point. The spacecraft storage system plays an important role, in that any instantaneously available array energy not immediately required for spacecraft loads is delivered to storage. Storage overcharge control, which is fully automatic, is achieved by overriding the maximum power tracking control and intentionally mismatching the system and array to produce a less-than-optimum transfer of power. Load demand is, of course, always met on a preferential basis. Use of this "mismatching" technique completely obviates the need for a power sink such as a shunt limiter, and eliminates a severe thermal problem.

## MAXIMUM POWER TRACKING (MPT) COMPONENT DESCRIPTIONS

Figure 13 shows the impedance transformation properties of a PWM device. Here, the switch opens and closes at a constant repetition rate or frequency, f. The ratio of closure time t ON, to the period, T, is the duty-cycle, $\alpha$, which is controllable.

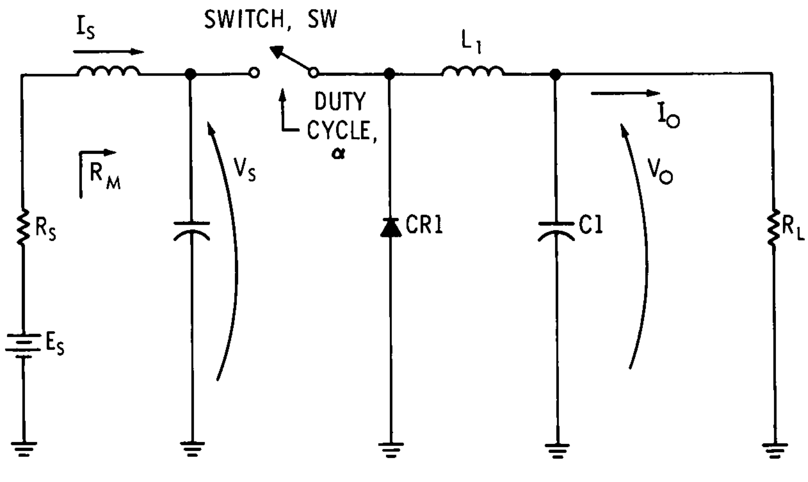


Figure 13. DC Impedance Transformation with PWM Techniques

If the average power delivered to $R_L$ is the dc power, and losses in sw and the filters are negligible, then

$$R_m = \frac{(V_s)^2}{P_s} \quad \text{where } P_s = \text{power delivered by source;}$$

but, if the switching network is essentially non-dissipative,

$$P_s = P_o = \frac{(V_o)^2}{R_L}$$

and,

$$R_m = \frac{(V_s)^2}{(V_o)^2} R_L$$

However, if the filter, $L_1 - C_1 - CR_1$, is an averaging filter,

$$V_o = \alpha V_s$$

and

$$V_s = \frac{V_o}{\alpha}$$

therefore,

$$R_m = \frac{V_o^{\ 2}}{V_o^{\ 2}} \frac{R_L}{\alpha^2} = \frac{R_L}{\alpha^2}$$

Therefore, controlling the duty-cycle of a PWM device will control the input impedance and thereby the source-to-load power transfer.

Placing a load directly at the source terminals obviously reduces the available source power, which is the power transferred to $R_L$ of Figure 13. However, this in no way impairs the ability of the PWM circuit to control the source operating point. In the system configuration of Figure 14, the load bus regulator draws load power from the source, thus reducing available source power.

In this system, the MPT functions primarily as a battery charger. By replacing $R_L$ of Figure 13 with a battery, the output voltage, $V_o$, becomes relatively independent of $I_o$. In this case, if the switch duty-cycle is varied over its range from zero to 100 percent, the output current, $I_o$, will also vary, and will be maximum when the power delivered to the battery ($P_o = V_o I_o$) is maximum. This occurs when the circuit is drawing maximum available power from the source.

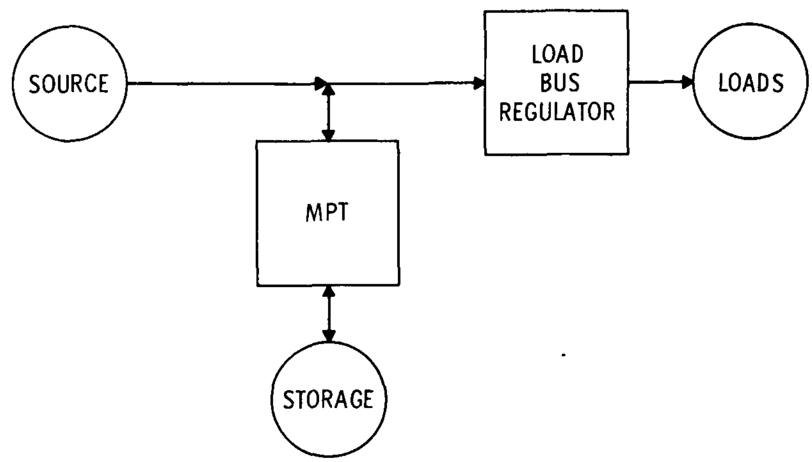


Figure 14. Basic Power-System Configurations

This, then, is the operating principle of the MPT. The duty cycle of a PWM power-switching circuit is slowly varied and a maximum condition in output current is recognized and maintained. Figure 15 is a functional diagram of the MPT, including battery-overcharge protection circuits.

As previously mentioned, the battery is protected against overcharge by overriding the power-tracking function and the controlling duty-cycle to achieve a non-optimum transfer of source energy.

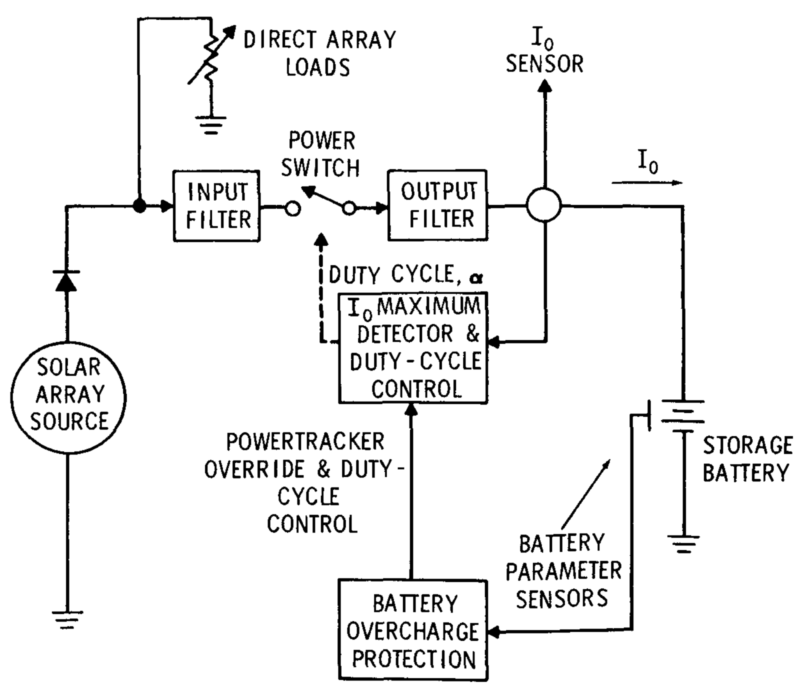


Figure 15. Maximum Power Tracker, Functional Diagram

## MULTIPLE BATTERY OPERATION

Figure 16 shows extension of the power tracker to systems with more than one battery. The system shown has one current-maximum detector for the total system and separate duty-cycle and overcharge control circuits for individual batteries. An active current-sharing scheme can overcome the unbalancing effects of differing battery voltages and other variable circuit parameters.

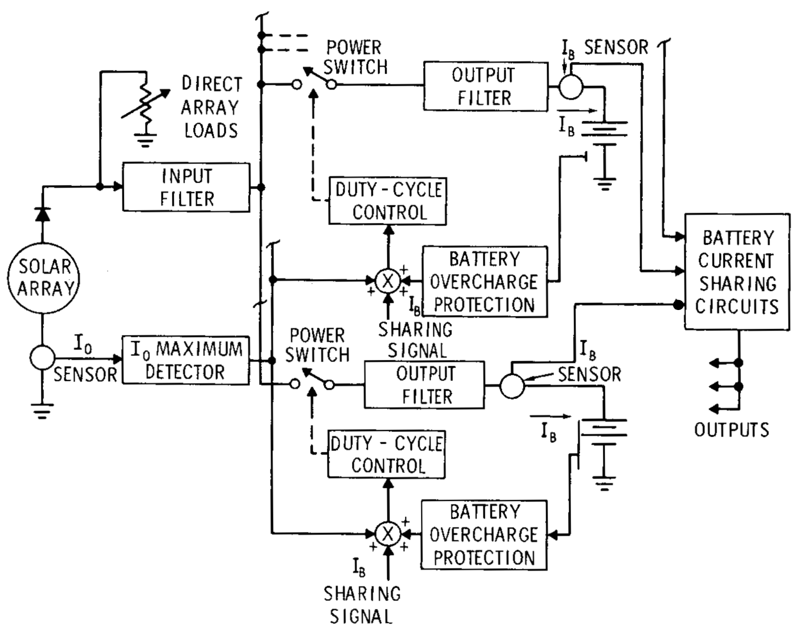


Figure 16. Power-Tracker Operation with Multiple Batteries

## BATTERY DISCHARGE

The battery discharges through isolation diodes which connect the battery terminals to the solar-array bus as in Figure 17. This function is identical with that in the two earlier Nimbus power systems.

The disadvantage of this discharge method appears when total spacecraft loads exceed available array power during the sunlit period of the spacecraft orbit. The heavy loading tends to lower the solar-array bus voltage to the point where the battery-isolation diodes conduct and cause the array to be clamped at essentially the battery voltage. For the duration of this heavy load, the maximum array power obviously cannot be delivered. Figure 18 shows available

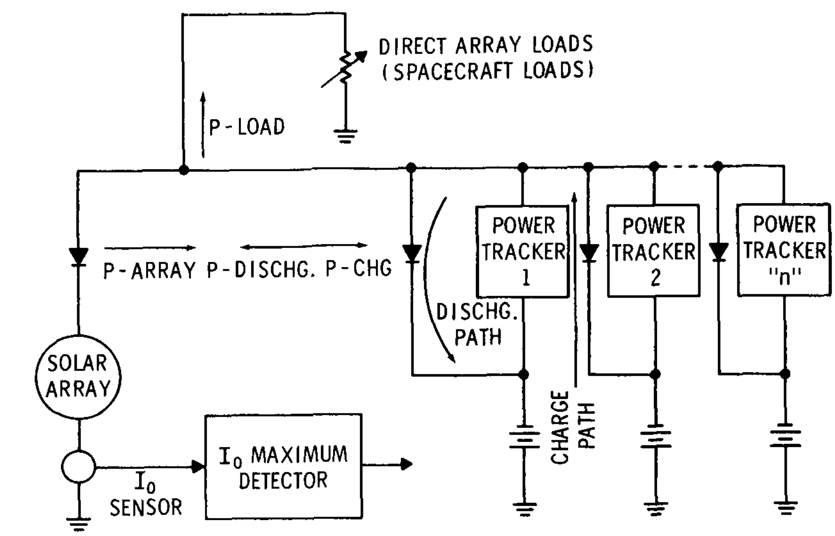


Figure 17. System Diagram Showing Battery-Discharge Paths

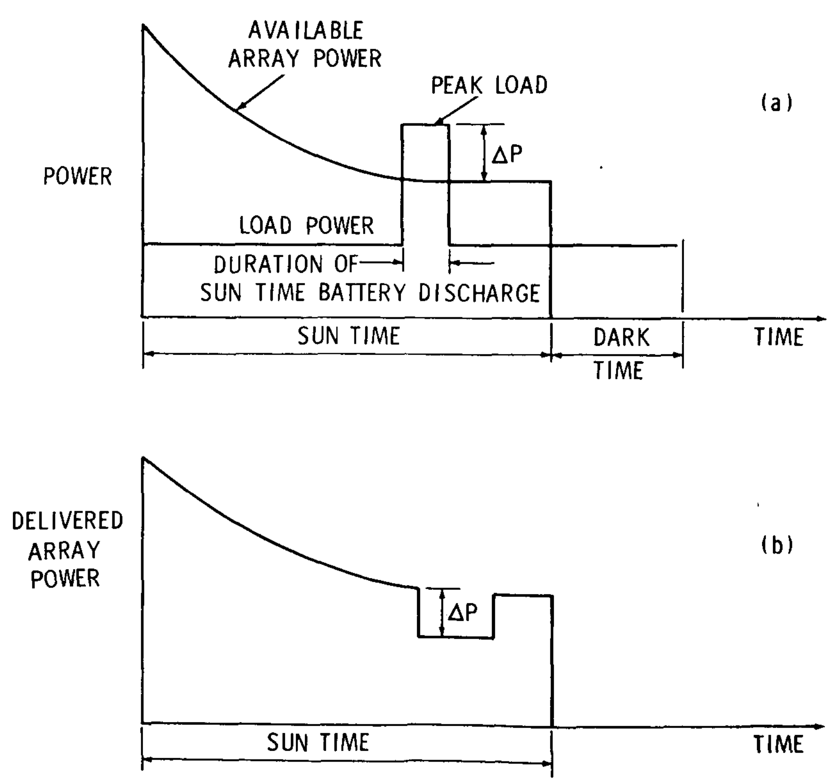


Figure 18. Array and Load Power vs Orbit Time

array power and a typical Nimbus-type load profile versus orbit time. When the load profile exceeds array power, the battery discharges into the array bus, thereby clamping that bus and reducing the power instantaneously delivered by the array to something less than full array capability.

Elimination of this disadvantage is not practical in view of the complexity of the additional hardware that would be required, and the fact that improvement would be slight for the load conditions stated for the Nimbus case.

## BATTERY CONSIDERATIONS

In the MPT system, the battery must be able to accept recharge at high rates. Without this capability, instantaneously available array power not required by the load may not be completely usable. An effective decrease in the average available array power would mean a loss in programming capability for a given array, or a larger array for a given load than would otherwise be required.

High-rate charging of conventional nickel-cadmium batteries seems to present no significant problems until the state-of-charge of the battery approaches 80 percent. At this point, generation of high, internal cell pressures and the possibility of hydrogen evolution render the high charge rate quite unattractive.

One answer to this problem is to use the "auxiliary" or "third" electrode nickel-cadmium battery cell, which can accept recharge in a cyclical routine at rates through the "C" rate.

## VOLTAGE REGULATOR

The voltage regulator in this system will be essentially the same one used in the second-generation Nimbus power system, except for greater input-voltage range and possibly increased load requirements.

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