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POWER REQUIREMENTS FOR COMMERCIAL COMMUNICATIONS SPACECRAFT*

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This paper presents historical data on commercial spacecraft power systems and relates their power requirements to the growth of satellite communications channel usage. On the basis of these data as well as input from other sources, some approaches for estimating future power requirements of this class of spacecraft through the year 2000 are proposed. The key technology drivers in satellite power systems are also addressed. The paper concludes with a description of several technological trends in such systems, focusing on the most useful areas for research and development of major subsystems, including solar arrays, energy storage, and power electronics equipment.

INTRODUCTION

Commercial communications satellites have become a significant portion of the total U.S. spacecraft market, with sales to both domestic and foreign entities. Their importance can be measured in several ways, including level of engineering design effort, number of projects, and construction costs, to name a few. An interesting illustration of their market potential can be seen in a recent NASA flight manifest (ref. 1), which shows that 36 percent of the assigned shuttle payloads for the 1984 to 1987 period consist of commercial communications spacecraft.

The prominence of this class of spacecraft requires that future NASA research and development programs be aimed at improving the technology to produce increasingly sophisticated U.S. engineering and commercial efforts in this area.

GROWTH IN TELECOMMUNICATIONS TRAFFIC

Active repeater satellites have become a routinely accepted means of relaying electronic communications for commercial purposes. Although television is perhaps the form of international communications most widely recognized by the general public, voice communications far exceeds the international satellite-borne traffic volume represented by television. This growth in service (ref. 2), characterized by 4-kHz bandwidth 2-way links currently in service within the INTELSAT system, is shown in figure 1. The continuing growth of this traffic undoubtedly is the result of several factors, including the rapid growth of international trade, as well as the reductions in rates, as shown in figure 2.

Basic data on the characteristics of the INTELSAT geosynchronous spacecraft, an enabling element in the INTELSAT communications systems, are shown in table 1. Each of the INTELSAT satellite series listed consists of a group of similar spacecraft, typically numbering between four and eight. The INTELSAT I through IV series are now

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deactivated; the INTELSAT IV-A, V, and the first V-A now provide all international operational service. Additional INTELSAT V and V-A series are being prepared for flight, and the INTELSAT VI series of spacecraft is currently in the construction and testing phases.

In recent years, much attention has been given to increasing satellite communications capacities through sophisticated techniques such as digital modulation, narrower antenna beams, and polarization diversity to allow for frequency reuse.

An analysis of the DC power required at the spacecraft's main bus terminals for each 2-way telephone channel (two half-circuits in communications parlance) is plotted versus time, as shown in figure 3. This decreasing power per channel is indicative of the results of these technological improvements. In addition, effort has been focused on increasing the available electric power for communications equipment. Many spacecraft subsystems typical of those on board current communications satellites have matured so much that changes to the overall system configuration are evolutionary rather than revolutionary. However, new devices and techniques are continually being introduced to improve their performance. To a large degree, electric power subsystems for communications satellites fit this category.

TRENDS IN POWER REQUIREMENTS

An interesting example of the growth of DC power requirements for commercial satellites is again provided by the INTELSAT spacecraft series. The proportions of primary power distributed to various loads on a typical spacecraft are shown in figure 4, taken from reference 3. All earlier designs used microwave-repeater-type devices in which most of the power goes to traveling wave tubes with a DC-to-RF conversion efficiency of 30 to 40 percent. More recently, solid-state RF power amplifiers, with longer life and higher reliability, are being introduced in the lower frequency bands. These amplifiers presently have a DC-to-RF conversion efficiency of about 30 percent.

A historical survey of overall DC power requirements for the INTELSAT spacecraft reveals continual and reasonably predictable growth over time. These data are presented in figure 5, with plotted points showing the total DC power load on the main bus vs the date of actual or anticipated launch of each spacecraft.

Several approaches can be taken in estimating future power requirements for this class of microwave relay spacecraft. A simple extrapolation from the historical trend of power vs time would lead one to expect a spacecraft power load of about 12 to 25 kW by the year 2000.

A more elegant approach rests on a number of assumptions. The first of these is the anticipation that international traffic will continue to grow at the same compound rate as that of the last 10 years. A spacecraft average channel occupancy rate of perhaps 60 percent can be selected; then, assuming a DC-power-per-channel that is based on an extrapolation of the historical trend shown in figure 3, and assuming the number of satellites in the constellation, the DC power required per spacecraft can be calculated.

For example, in the year 2000, if the present rate of international satellite communications traffic growth continues unabated, the number of half-circuits in use worldwide might be extrapolated to be 900,000 to 1,330,000. A more detailed formal study of expected satellite channel requirements developed from a communications point of view (ref. 4) anticipates international satellite traffic of between 752,000 and 1,140,000 channels by the year 2000. Thus, the simple traffic extrapolation seems reasonable. Assuming an extrapolated ratio of 19 telephone circuits per DC watt of spacecraft bus power, and a 60-percent channel usage rate, the total in-orbit power required for the whole constellation might be about 70 to 117 kW.

Various scenarios can be developed, to determine the number of satellites in the constellation. These scenarios must involve complex tradeoffs related to orbital slot availability, launch vehicle capabilities, and launching costs, as well as questions about the operational complexities of operating a large number of spacecraft. The latter is probably dependent to a large degree upon the amount of automation on the ground and in the spacecraft. Assuming a total of 10 similar operational spacecraft in orbit, each one would have to support a DC load of about 8 to 12 kW, whereas an assumption of 20 operational spacecraft would imply a DC load of about 4 to 6 kW on each one. Based on the present distribution of traffic at the three orbital locations (10-percent Pacific, 20-percent Indian Ocean, 70-percent Atlantic), the constellation of 10 similar satellites probably represents the minimum reasonable number required to fully utilize the Pacific satellite. At the other extreme, 20 satellites is probably a reasonable maximum for control center loading, since some in-orbit spares (probably older spacecraft) must also be operated and maintained.

This series of assumptions leads to an expectation of a requirement for an international communications spacecraft with a DC-load capability in the range of approximately 4 to 12 kW by the year 2000. Allowing some power margin, this translates to an end-of-life (EOL) solar array capability of approximately 5 to 14 kW.

IMPACT OF WEIGHT REDUCTION

It is possible to use already published data to estimate the value of weight and power savings in terms of potential revenue on commercial spacecraft. For instance, if improvements in power system efficiency are made that allow for additional telephone channel capacity, the expected lifetime revenue payback can be calculated from the product of the number of channels per DC watt (table 1), the leasing price per channel-year (fig. 2), the spacecraft design lifetime, and the estimated ratio of channels leased compared to full load. With the first launch of a spacecraft, the channel load is usually low; the size of the load grows with time in operational service. It is considered unwise to operate at 100-percent capacity, since this leaves no reserve for peak communications loads, and might also preclude switching to a backup in case of a transponder failure. Between 70 and 80 percent is considered a conservative maximum operating capacity under normal conditions. Using such assumptions, revenue increases can be calculated to be between $200,000 and $300,000 per watt saved by spacecraft power-system efficiency improvements.

Similar calculations can be performed on the potential revenue increases associated with spacecraft mass savings. In this case, the number of communications circuits can be calculated, divided by the total mass of the spacecraft communications system plus power system, to yield the number of telephone channels per pound. This number can be multiplied by the same revenue and channel occupancy factors as before to come up with the potential revenue increase per pound of spacecraft mass saved. Calculations have produced numbers in the range of $300,000 to $500,000 per pound. Based on his calculations, Raab (ref. 5) cites a value of $500,000 to $1 million per pound. In either case, power or weight reduction, the total amount saved must be large enough to allow for an additional transponder on board, and the additional communications capacity must be marketable to actually realize any additional revenue. Spacecraft life and reliability should not be compromised in any way, since this would have a detrimental effect on the desired result.

It is clear that if the above conditions are realized, there are very strong motivations for minimizing power system mass and maximizing power system efficiency in commercial communications satellites. Therefore, R&D efforts that are successful in reducing power system mass will provide a large payoff for this class of spacecraft.

SOLAR ELECTRIC GENERATORS

Several important technology changes that significantly affect the mass and conversion efficiency of photovoltaic arrays were pioneered during the last decade. These modifications are gradually being applied on operational spacecraft as they progress through the steps of production process refinement, experimental flight, and finally, after they are fully qualified by testing and certified for operational use.

Solar Cells

In the mid-1960s, the silicon photovoltaic cell had reached a rather stable design status with a conversion efficiency plateau of about 10-1/2 percent. Detailed analyses of the sources of energy loss within the cell (ref. 6) highlighted promising areas for improvements in performance. At the same time, a laboratory development effort at COMSAT produced a new silicon cell called the "violet cell" (ref. 7). This triggered a new burst of silicon cell development (ref. 8) which has produced laboratory cells with conversion efficiencies as high as 15-1/2 percent (see fig. 6).

These laboratory developments on the silicon cell component are now being rapidly exploited in operational programs. The hybrid-type cell has been used in a variety of spacecraft programs, and cells closely approaching a 20-mW/cm^2 level were flown on the NASA International Sun-Earth Explorer spacecraft. The USAF supported an extensive qualification program for the 80-mW, textured cell (sometimes referred to as the K7 cell) which is now in operational use on drum-spinner type arrays in the SBS, INTELSAT VI, and other satellite programs. Of course, the mass and area of solar arrays that use these higher efficiency cells can be reduced almost in proportion to the efficiency ratios. In some cases the gains are slightly less because of higher electron degradation rates or increased operating temperatures.

Another possibility, which is still in the prototype production stage, is the 50-μm (2-mil) thick silicon cell. Development work on this component was originally sponsored by NASA through JPL (ref. 9). Conventional cells have a power-to-mass ratio of about 100 W/kg when they are covered with an equivalent thickness of quartz. By contrast, these new thin cells can produce about 1,000 W/kg bare, and using 50- to 100-μm covers, could possibly achieve something in the vicinity of 300 to 500 W/kg covered. In both cases, the mounting and interconnection provisions are not included in the mass. These thin cells have an additional advantage--they degrade less than the conventional 200- to 250-μm cells in the geosynchronous orbit electron radiation environment (ref. 10).

Efficiency improvements from current research and development on advanced cells (such as GaAlAs and multi-bandgap cells) could also have significant effects on the power-to-mass ratio.

Solar Array Structures

Solar array hardware is gradually incorporating these solar cell improvements into operational spacecraft. The structure of these arrays is also changing in an evolutionary way. Most of the early INTELSAT and U.S. commercial spacecraft have been drum spinners with a honeycomb sandwich panel construction. This structure consisted of epoxy-bonded glass fiber skins bonded to a vented aluminum honeycomb spacer. In structural design terminology, the face sheets are the load-bearing member, and the honeycomb spaces them apart to increase the structural moment of inertia to provide the desired panel stiffness. In most cases this required stiffness is dictated by the vibration environment encountered during launch.

The simple drum-spinner array has proved quite reliable, but as Barthelemy has pointed out (ref. 11), it has required roughly three times as many solar cells as a sun-oriented array. Their maximum power capability was limited by the volume available within the launch-vehicle heat shield. The weight of these drum-spinner arrays has been approximately 100 to 160 kg/kWe at the end of the mission (see table 2). The early RCA SATCOM sun-oriented arrays with aluminum-faced honeycomb panels were considerably lighter, weighing about half as much as the drum spinner arrays.

As payoffs brought about by a reduction in the mass of these commercial satellites are recognized, the new array of structural designs have begun to reflect aerospace industry advances in lightweight structures. For instance, the SBS and Anik-C drum-spinner designs with extending skirts, built by Hughes Aircraft Co., have an epoxy-bonded Kevlar high-strength-to-weight-ratio fiber in the face skins. The much larger (12-ft diameter) INTELSAT VI arrays (ref. 12), built by Messerschmidt-Buclow-Blohm (MBB) in Europe, are similar to them in design concept and materials (see fig. 7). In another example, the INTELSAT V flat, sun-oriented solar arrays (see fig. 8), which were also designed and built by MBB (ref. 13), use woven graphite fibers in epoxy composite skins. In this case, a Kapton layer is placed under the solar cells to insulate them from the conductive graphite.

The laboratory developments (listed in the lower portion of table 2), in combination with the solar cell advances mentioned above, promise continuing area efficiency and power-density improvements significantly better than present operational designs. Much of this improvement will probably result from improved structural concepts investigated in R&D work. It should be noted that much of the recent R&D work on arrays suitable for use on spacecraft in the 5- to 10-kW range (which is a range of interest for communications spacecraft) is being pursued in Europe (ref. 14). This advance in technology is beginning to result in production contracts.

Current R&D designs of flexible roll-out and flexible fold-out arrays are in the 35- to 60-W/kg (34- to 18-kg/kWe) power-density range, and the future possibility of arrays with power densities of 110 to 200 W/kg has been suggested in conceptual design studies (ref. 15). Lightweight solar concentrators with gallium arsenide cells for photovoltaic conversion also offer some promise for the future (ref. 16). However, no concentrator designs have been described in the literature that are presently weight-competitive with the advanced flat solar arrays.

ENERGY STORAGE

Ni-Cd Batteries

The rechargeable nickel-cadmium (Ni-Cd) alkaline cell has been used to supply primary power during eclipse in all operational U.S. domestic and INTELSAT communications spacecraft launched before 1983. In particular, the backlog of orbital experience, high-rate, deep-discharge capability, and long storage life appear to be key qualifications of this type of cell.

Detailed analyses of a number of these Ni-Cd batteries designed for geosynchronous orbit missions reveal that the total energy density available from new cells at 100 percent depth of discharge is relatively constant at about 26 Wh/kg. The cell weight constitutes about 82 percent of the weight of a typical flight battery, with the wiring, connectors, electronics, and structure making up the remaining 18 percent.

Typically, from one-third to one-half of the power subsystem mass in these spacecraft consists of batteries. The principal variables that determine the delivered energy density are actual depth of discharge used and redundancy strategy.

This is shown graphically in figure 9. Most of the U.S. commercial and INTELSAT craft using Ni-Cd batteries have utilized the series cell-redundancy approach and have been operated in the discharge range of 35 to 55 percent of total electrochemical capacity as a maximum (ref. 17). The energy density, in terms of total mass and actual watt-hours delivered to the load during the longest eclipse, is shown in table 3 for several of these spacecraft.

In-orbit reconditioning allows longer mission life and deeper discharging of the Ni-Cd batteries in the newer designs, resulting in somewhat higher energy densities for more recent spacecraft, as shown in table 3 (ref. 18, 19).

$Ni\text{-}H_2$ Batteries

Rechargeable nickel-hydrogen cells have been under development since the original work began at COMSAT Laboratories in 1970. Under USAF and INTELSAT sponsorship, there has been an extensive parallel development of two different cell designs. The INTELSAT cell design was first flown experimentally on the NTS-2 satellite (ref. 20), with successful results. It was then introduced into operational use as part of the INTELSAT V spacecraft series. Because the battery had to be a direct physical and electrical replacement for a Ni-Cd battery package (ref. 21), the delivered energy density was nearly unchanged at about 17.6 Wh/kg, as shown in table 3. The INTELSAT V-A series, which will have a larger DC load and therefore a greater battery depth of discharge, is expected to produce an improved energy density of about 19.6 Wh/kg. The new G-STAR and RCA SPACENET batteries were initially planned as $Ni\text{-}H_2$ designs, and with these designs engineers have been able to reach energy densities of almost 20 Wh/kg delivered to the load (ref. 22).

Several R&D efforts are also under way on bipolar and common pressure vessel $Ni\text{-}H_2$ designs that may reduce battery size and mass by simplifying the structural design of the pressure vessel (ref. 23). These efforts may eventually prove successful in further improving the energy density of nickel-hydrogen batteries.

One area that might help improve the life and reliability of both Ni-Cd and $Ni\text{-}H_2$ battery systems is research in parameters that control the growth of microcracks in sintered nickel electrodes. This crack growth with cycling, illustrated in reference 24, is not well understood; however, it is one of the key degradation mechanisms found in both types of cells.

ADVANCED ENERGY STORAGE SYSTEMS

Advanced electrochemical energy-storage devices and various other mechanical and thermal energy systems have been studied by many investigators. The key requirements of simplicity, long life expectancy (verifiable by testing), modularized design that allows implementation of redundancy with minimum weight penalty, tend to eliminate many of these candidate energy storage systems.

For instance, an in-depth technical comparison, including hardware development on a high-speed composite-wheel energy storage system, indicated that although the wheels themselves have a high energy density, when all auxiliary equipment such as motor/generators, power conditioners, and redundant wheels and control equipment are included, the overall energy storage system density for a geosynchronous mission with a 2.5-kW load is about 13.5 W/kg (ref. 25). This amount is considerably less than that of existing flight hardware Ni-H_2 batteries, and taken together with system complexity and other concerns regarding large, extraneous torques on the vehicle, make it an unlikely candidate for commercial geosynchronous communications applications.

Regenerative H_2-O_2 fuel cells have been studied as a possible spacecraft energy-storage system. These multicell designs with separate fuel cell and electrolysis cell units appear feasible. Both acid and alkaline systems have been studied in the past. Under NASA sponsorship, systems studies and laboratory demonstration testing of alkaline fuel cells with electrolyzers is continuing (ref. 26). Estimates of total energy storage subsystem energy density for this type of system are between 80 and 120 Wh/kg for a 35- to 350-kW load. This may be a very satisfactory system in this high-power range. However, a successful single-cell regenerative H_2-O_2 system has not been developed for spacecraft loads in the 4- to 25-kW power range, and a scale-down of the separate electrolyzer system to that level would probably result in much lower energy densities.

Surveys of the advanced secondary electrochemical systems, which are candidates for the next major leap forward in energy density beyond nickel-hydrogen, indicate that the high-temperature sodium-sulfur cell is probably the most attractive one to consider for development (ref. 27). This cell, which has a theoretical energy density of 790 Wh/kg and uses a beta-alumina solid material to separate the electrodes, was invented in the 1960s. Research efforts have been pursued by several U.S. organizations, including Ford and General Electric, and work has also been conducted in the U.K. and in Japan. The work has moved beyond the basic research stage, and prototype rechargeable cells have been developed for several transportation and terrestrial power-topping cycle applications.

These sodium-sulfur cells, as shown in figure 10, have demonstrated good performance in high rate discharge, efficient recharge, and reasonable discharge regulation. Safety issues have also been addressed. In the sizes and charge/discharge rates of interest for these satellites, cell energy densities of 100 to 120 Wh/kg have already been successfully demonstrated (ref. 28), and some cells with twice that amount of energy density have also been reported (ref. 29).

For development and testing aimed at specific NaS battery applications in spacecraft, several battery system studies have been reported (ref. 30), and others are under way. Laboratory testing that simulates the cyclic conditions anticipated in orbital applications (ref. 31) is also being conducted.

One fundamental technical question that constitutes a barrier to future work with NaS cells in geosynchronous applications is related to cell life. The existing Ford cell design has been demonstrated to have a mean lifetime of about 3 years at full operating temperature, with one deep cycle per day (ref. 32). This 1,000-cycle lifetime is probably satisfactory for commercial satellite applications. What is not clear, however, is whether longer calendar lifetimes that approach the desired 7 to 15 years could be expected with these cells in a geosynchronous orbit mode. For instance, it might be possible to place them on standby most of the time, perhaps at a lower temperature in the 225 to 250°C range, and bring them up to full operating temperature for only the 90 days per year when deep discharges are required. Tests of this kind would demonstrate whether the cell is ready for engineering development into a flight design, or whether further R&D on corrosion or other failure mechanisms is needed. Since all other attributes of the NaS cell appear quite attractive, this seems to be a question worthy of investigation.

Most of the power used on board communications spacecraft flows through electronic power conditioners (EPCs) into the RF transmitting devices, as shown in figure 4. When traveling wave tubes (TWTs) are used, each high-voltage EPC unit typically weighs about the same as the TWT. An EPC unit is dedicated to each TWT, including both operating transponders and back-up units. Thus, the total weight of these units constitute a significant portion of spacecraft in-orbit mass.

Empirical studies of these high-voltage EPC units have shown that over a reasonable range, their mass is a linear function of output power. As shown in figure 11, the mass is strongly affected by the converter switching frequency. Similar trends have also been found in empirical studies of low-voltage DC/DC converters.

This can be explained simply by considering the converter as an "energy ladling" device that stores a certain quantity of energy during each cycle, and then releases it to the load. The physical size and weight of this main energy storage device (which can be an inductor, a transformer, or a capacitor) is related to the amount of energy stored. At a given power level, as the switching frequency is increased, the amount of energy stored per cycle is reduced in inverse proportion, and the mass of the energy storage components can be significantly reduced. The mass of the input and output filters can also be reduced at the higher frequency of operation because they too can be considered to be energy storage components.

The availability of space-qualified, high-voltage field-effect transistor (FET) power devices now makes it possible to operate practical DC/DC converters at much higher frequencies than ever before. For example, a 600-W electronic power conditioner, using a fairly conventional push-pull, current fed topology, was designed for 150-kHz operation and was built to run a high-power, direct broadcast TWT (ref. 33). This reduced the weight of the EPC unit by approximately half, compared to the 50-kHz production design, as shown in figure 11. If implemented throughout the particular spacecraft of interest, it could save a total of 42 pounds.

Much higher operating frequencies are possible by using the FET switches in electronic power converters, and work is under way in several organizations to investigate high-efficiency designs operating in the megahertz region. If these efforts are successful, some very lightweight power converters should result. There are two key background areas that require study to promote a great advance in this technology. First, a firstrate, widely available computer program for the design of high-frequency power transformers and inductors is needed. Such a program would allow rapid iteration and optimization of magnetic component designs on the computer, instead of the repetitive "cut and try" approach that is used almost universally in the industry today.

A second requirement is for more comprehensive research studies and publication of high-efficiency DC/DC power converter and regulator circuit topologies suitable for operation in the megahertz region, with capabilities for power output in the range of 5 to 500 W. This is the region of interest for communications satellites. Many typical tube or solid-state communications transponders require a DC input power of about 10 to 30 W. Some solid-state units are expected to have power requirements as low as 5 W, and the 500-W requirement appears to be at the high end of the current needs for direct broadcast transponders.

Another fundamental requirement is for more efficient power rectifiers with faster recovery times. As DC/DC converters move into higher frequency regimes, the rectification problem becomes much more significant, particularly in supplies for the solid-state RF amplifiers that require voltages in the 4- to 8-V region. If the rectifier forward-conduction drop is a sizeable fraction of 1 V, a significant loss in the overall conversion efficiency results. It appears that more R&D on components like power FETs optimized for synchronous rectification, or possibly the GaAs Schottky power rectifier (ref. 34), might provide solutions to this problem. If these device developments were brought to fruition, they would have sizable spin-off effects throughout the power processing industry.

CONCLUSIONS AND RECOMMENDATIONS

An analysis of the historical trends in power requirements for commercial communications satellites has been conducted to predict those needs for the year 2000. Several approaches indicate a possible range of 12 to 25 kW, with a more likely DC load in the range of 4 to 12 kW.

Power-system technology drivers for this class of spacecraft have been identified, along with an overview of the development status of major system elements. Specific recommendations for future R&D work in each of these areas are mentioned. One of these recommendations is for the development of a solar array for the 5- to 15-kW range, with a power density greater than 50-W/kg EOL, which provides transfer orbit power and can be conveniently stowed within a launch vehicle. A second recommendation is for R&D to bring the high-energy density, rechargeable, sodium-sulfur battery cell to a technology readiness status for geosynchronous missions as rapidly as possible.

In the area of power electronics components, R&D on special FET switches for synchronous rectification, and possibly the GaAs Schottky power diode, appear to have potentially large payoffs. A good computer program for the design of high-frequency magnetic components that is made available on an industry-wide basis would greatly expedite the design of lightweight, efficient, high-frequency power processors. New studies on DC/DC converter and switching regulator topologies, aimed at utilizing the fast switching capabilities of power FET devices to produce lightweight power conditioners operating in the megahertz region, may also be useful.

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Table 1. Growth of INTELSAT Spacecraft

Parameter	INTELSAT Satellite Series							
	I	II	III	IV	IV-A	V	V-A	VI
Year of First Launch	1965	1967	1968	1971	1975	1981	1984	1986-87
Drum Dimensions (cm)								
Diameter	72.1	142	142	238	238	--	--	364
Height	59.6	67.3	104	282	282	--	--	613
Overall Deployed Height (cm)				528	590	1,585	1,585	1,180
Mass (kg)								
At Launch	68	162	293	1,385	1,469	1,870	1,870	--
In Orbit	38	86	152	700	790	1,014	1,014	2,250
Primary Load Power (W)	40	75	120	400	500	975	1,100	1,800
Active Transponders	2	1	2	12	20	23	32	48
Number of Telephones Circuits	240*	240	1,200	4,000	6,000	12,000	~15,000	~30,000
Design Lifetime (yr)	1.5	3	5	7	7	7	7	7

*No multiple access.

Table 2. Solar Array Characteristics

Spacecraft	End-of-Life Power (W)	Array Type	W/m^2	End-of-Life (kg/kWe)	Design Status
NATO III	375	Body-mounted	24.5	127.7	Flight
INTELSAT IV-A	522	Body-mounted	24.5	137.7	Flight
ANIK I-III	219	Body-mounted	23.8	151.6	Flight
ATS-6	600	Rigid, oriented	30	101.3	Flight
Orbital Workshop	12,240	Rigid, oriented	97	188.7	Flight
FilSatCom	1,470	Rigid, oriented	69	62.1	Flight
CTS-Hermes	1,045	Flexible	55.3	45.5	Flight
Hughes FRUSA	1,100	Flexible roll-out	70	28	Flight
SBS (Hughes)	710	Body-extended skirt	29	103	Flight
INTELSAT V (Ford)	1,290	Rigid, oriented	55	50	Flight
ECS (Fokker)	998	Rigid, oriented	75.3	48.5	Flight
MARECS (Fokker)	835	Rigid, oriented	63	58.5	Flight
STC/DBS (RCA)	2,200	Rigid, oriented	104	32.5	Under construction
INTELSAT VI	2,170	Body-extended skirt	35.3	77.2	Under construction
SKYNET 4 (Fokker)	1,400	Rigid, oriented	89.7	38.4	Under construction
TRW Lightweight	1,470	Rigid	75	41	R&D
MBB ULP	1,500	Semirigid	75	30	R&D
Aerospatiale/COMSAT	1,050	Flexible foldout	72	34	R&D
AEG Dora	6,600	Flexible roll-out	77	25	R&D
Lockheed SEPS	12,500	Flexible foldout	83	18	R&D
Fokker ARA MK3	2,000-4,000	Rigid	--	22-25	R&D

Table 3. Battery Energy Density Experience

Spacecraft Battery*	Delivered Wh/kg
INTELSAT III	13.9
INTELSAT IV	13.2
INTELSAT IV-A	13.2
INTELSAT V (F1-5) NiCd	17.6
COMSTAR	15.4
SBS	15.4
RCA SATCOM NiCd	16.5
NTS-2 Experiment $Ni-H_2$	17.6
INTELSAT V (F6-9) $Ni-H_2$	17.6
INTELSAT V-A (F10-14) $Ni-H_2$	19.6
GSTAR $Ni-H_2$	19.9
RCA SPACENET $Ni-H_2$	19.7

*Including instrumentation and hardware.

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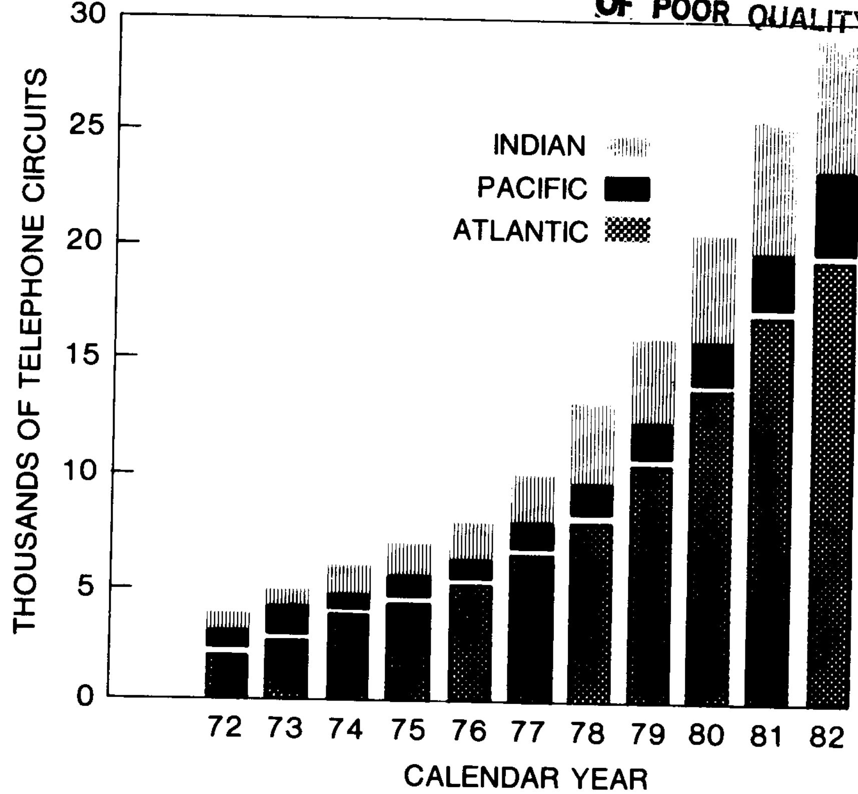

Figure 1. Growth in Full-Time INTELSAT Satellite Use by Region

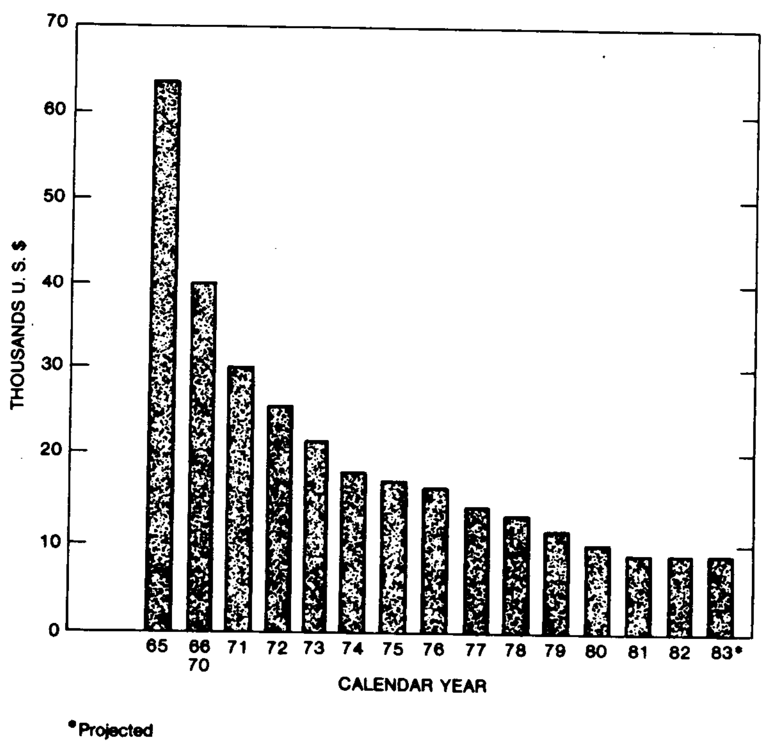

Figure 2. INTELSAT Satellite Utilization Charge

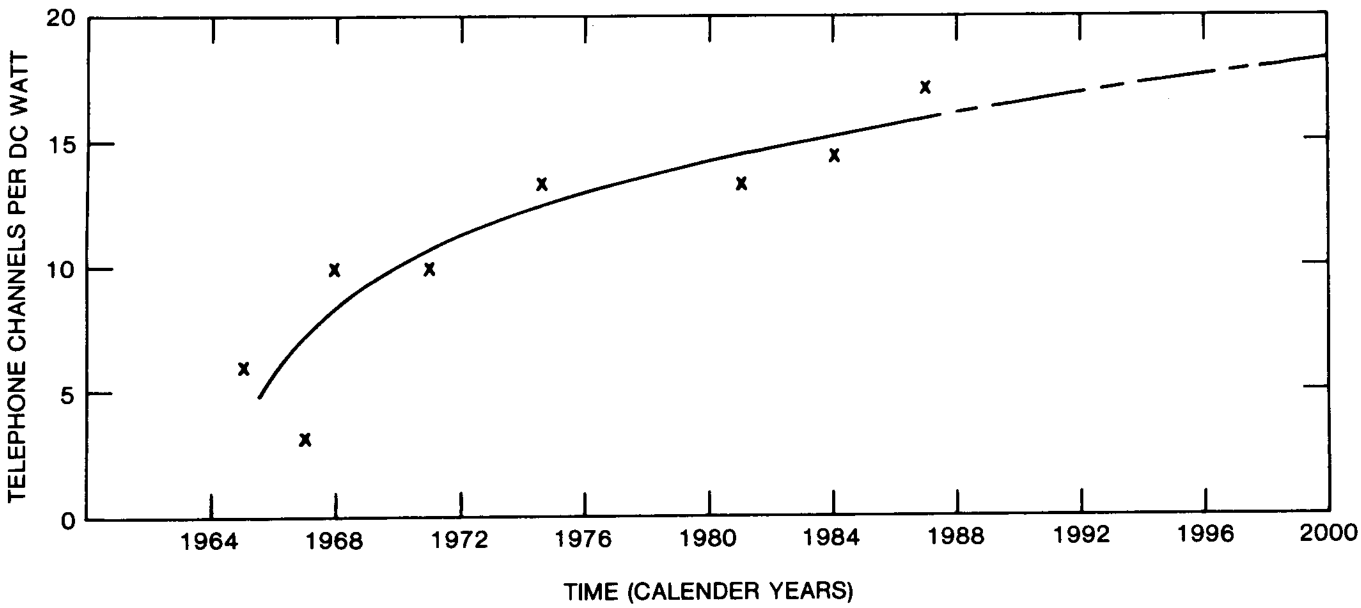

Figure 3. DC Bus Power Required per Two-Way Telephone Circuit

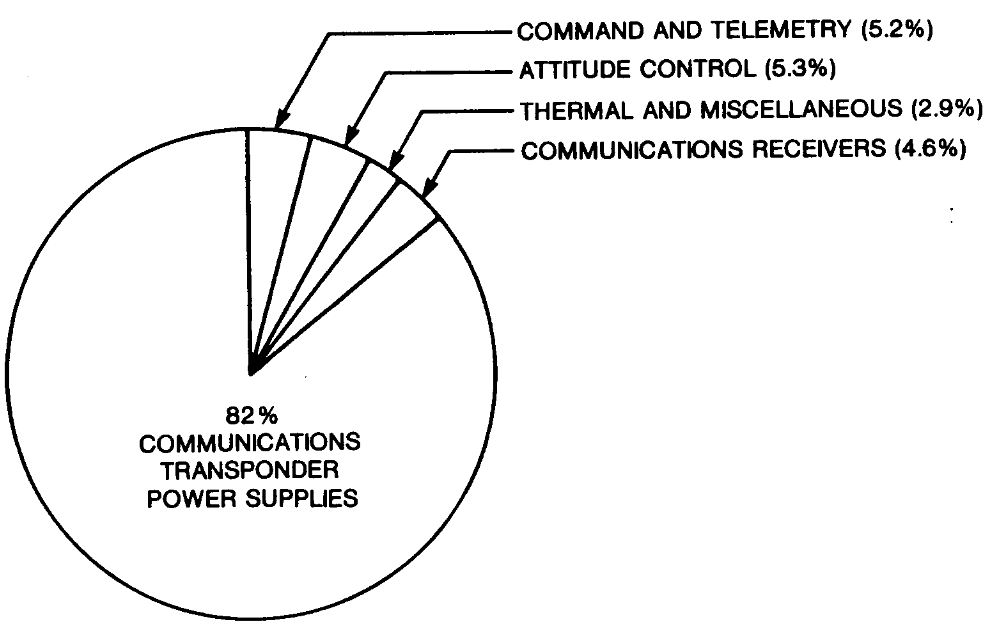

Figure 4. Primary Power Distribution in a Typical Communications Spacecraft

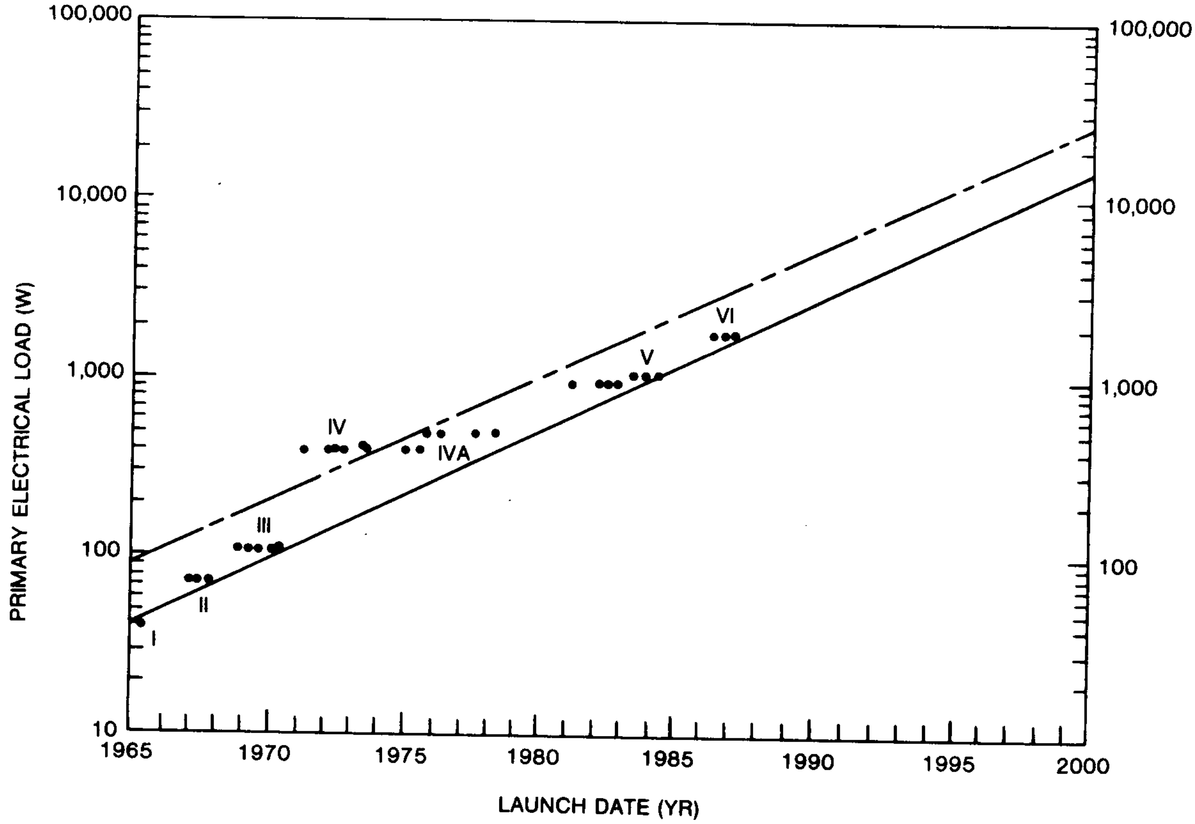

Figure 5. Electrical Load Power for INTELSAT Spacecraft

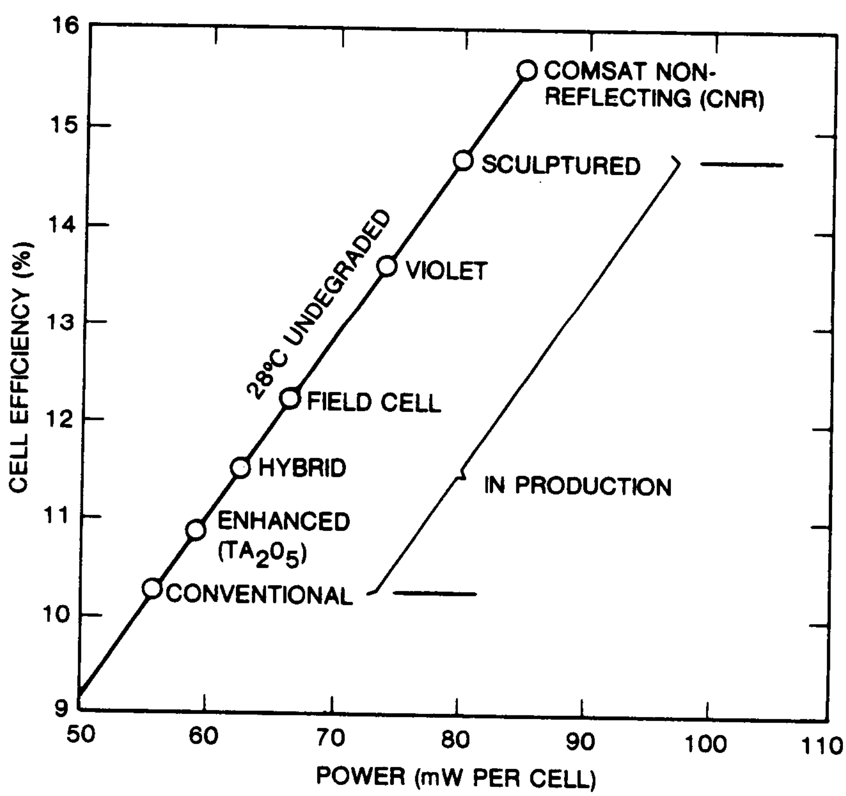

Figure 6. High-Efficiency Solar Cell (2 x 2 cm) Performance

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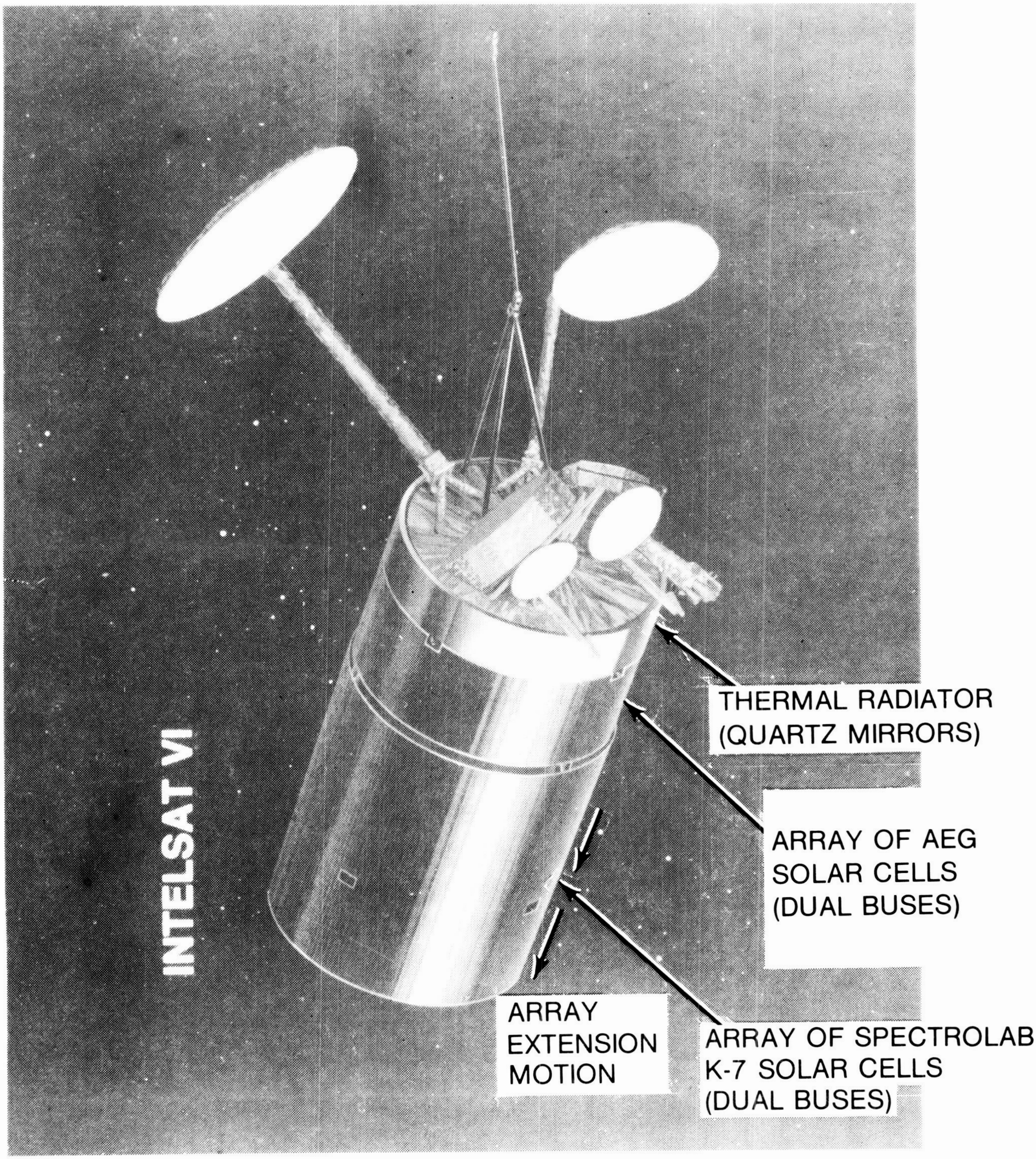

Figure 7. INTELSAT VI Extendable Solar Array

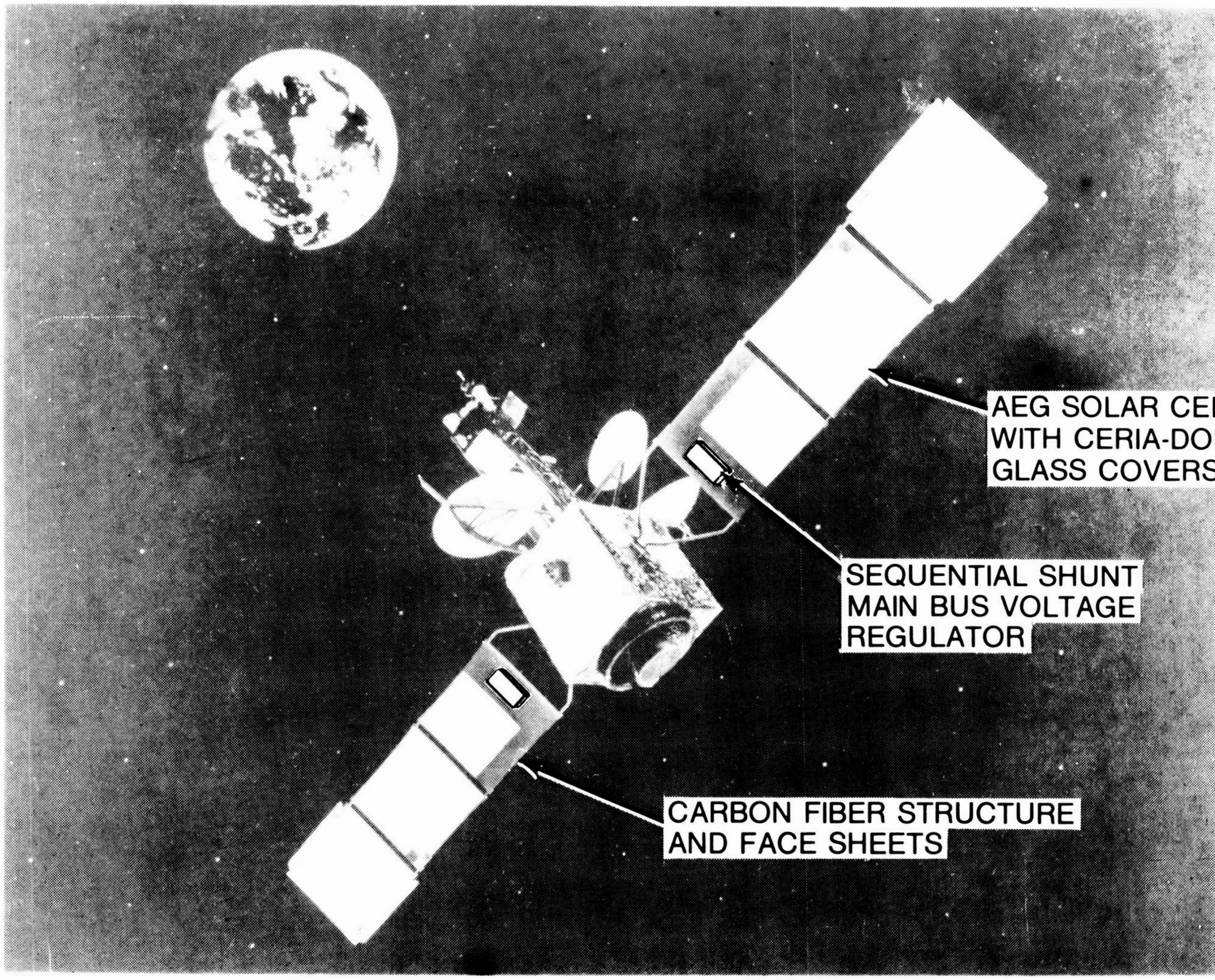

Figure 8. INTELSAT V Solar Arrays

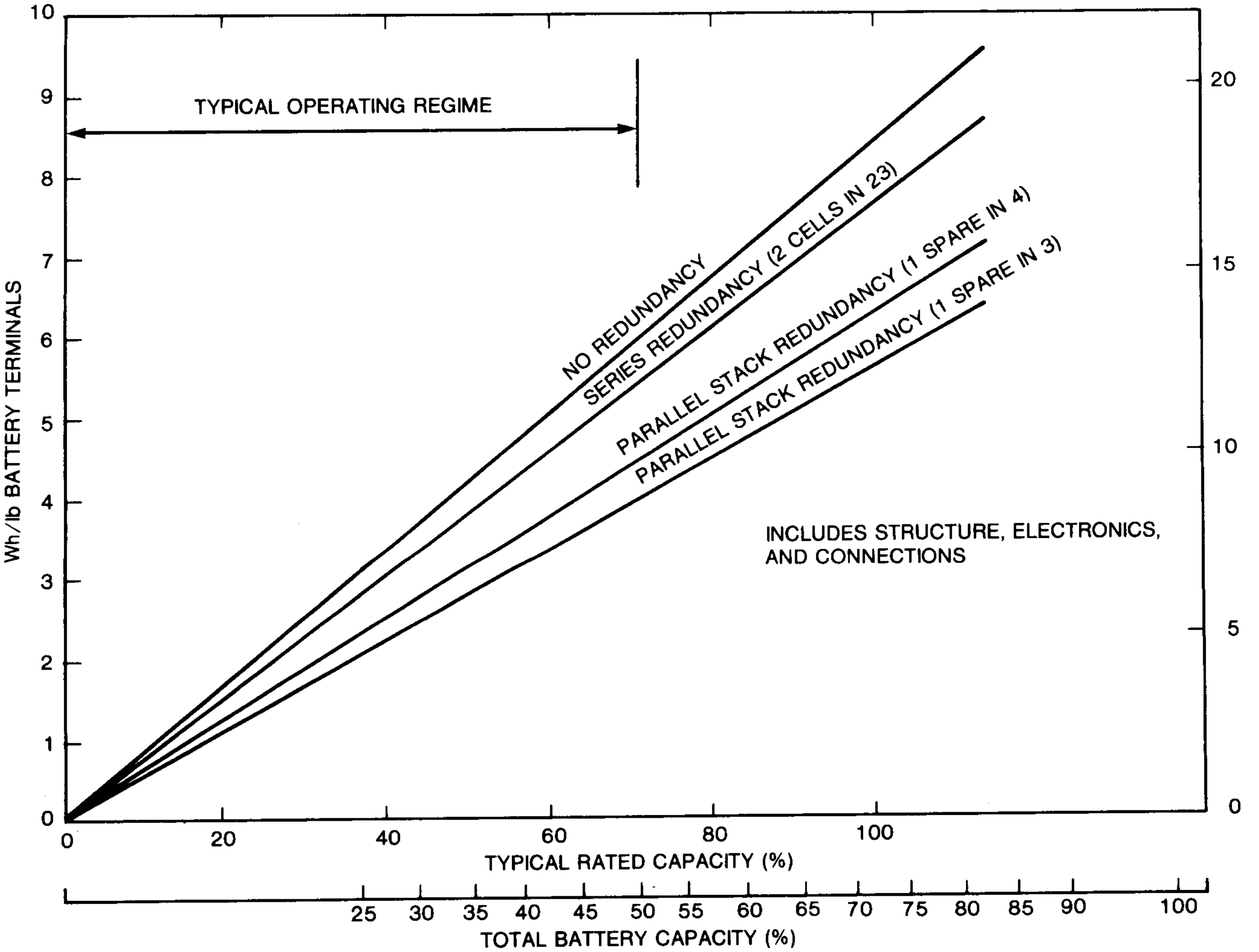

Figure 9. Ni-Cd Battery Energy Density vs Depth of Discharge

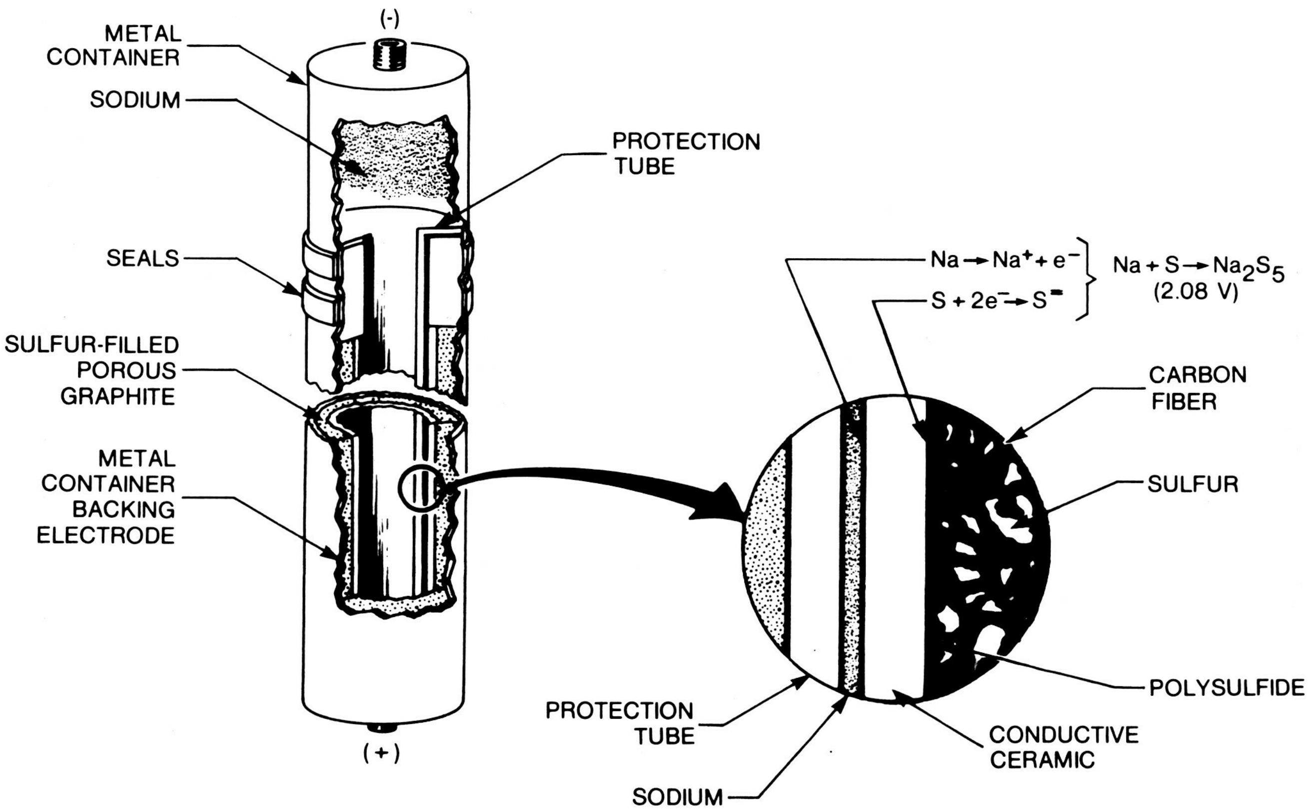

Figure 10. Schematic of Sodium-Sulfur Cell

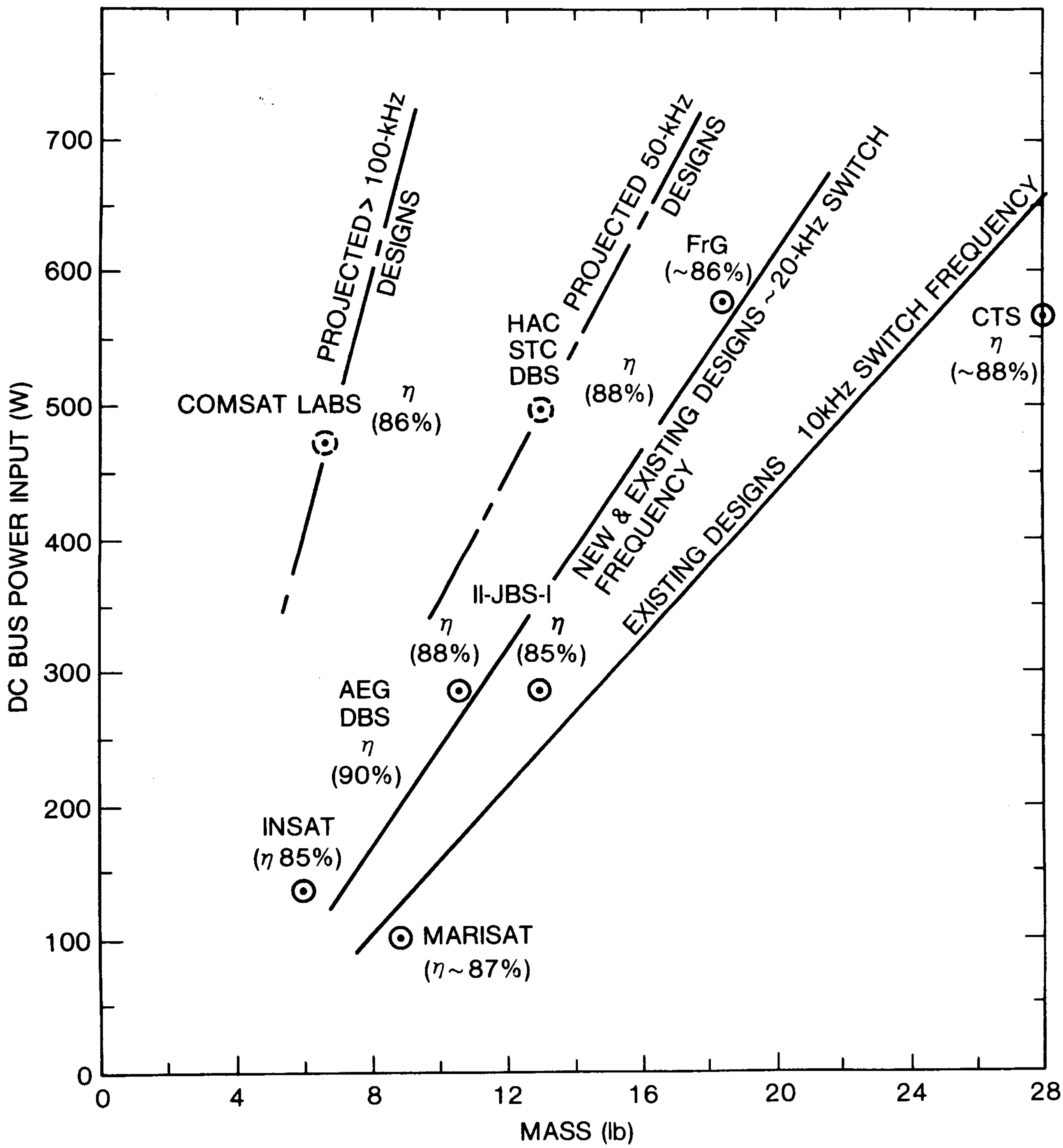

Figure 11. Mass of Electronic Power Conditioners for TWTs at High Power Levels