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Synchronous Orbit Power Technology Needs

L. W. Slifer, Jr. and W. J. Billerbeck

APRIL 1979

National Aeronautics and
Space Administration

Goddard Space Flight Center
Greenbelt, Maryland 20771

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ABSTRACT

The objective of this paper is to define the needs for future geosynchronous orbit spacecraft power subsystem components, including power generation, energy storage, and power processing. A review of the rapid expansion of the satellite communications field provides a basis for projection into the future. Three projected models, a mission model, an orbit transfer vehicle model, and a mass model for power subsystem components are used to define power requirements and mass limitations for future spacecraft. Based upon these three models, the power subsystems for a 10 kw, 10 year life, dedicated spacecraft and for a 20 kw, 20 year life, multi-mission platform are analyzed in further detail to establish power density requirements for the generation, storage and processing components of power subsystems as related to orbit transfer vehicle capabilities. Comparison of these requirements to state of the art (INTELSAT-V) design values shows that major improvements, by a factor of 2 or more, are needed to accomplish the near term missions. However, with the advent of large transfer vehicles, these requirements are significantly reduced, leaving the long lifetime requirement, associated with reliability and/or refurbishment, as the primary development need. A few technology advances, currently under development, are noted with regard to their impacts on future capability.

†This work was supported by COMSAT Laboratories (INTELSAT) and NASA/Goddard Space Flight Center.

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INTRODUCTION

The definition of research and development needs for the future and the judicious apportionment of funding for that research and development are common and difficult problems. In order to solve those problems, it is necessary to estimate what the requirements for the future will be. To some extent, a crystal ball is needed, but frequently, past developments can provide good guidelines to the future. The future requirements, based on those guidelines may well be controversial in specific details, but can be quite valuable when used in general form.

Extensive use of spacecraft in Geosynchronous Equatorial Orbit (GEO) has been made for communications. Review of past development and expansion in this area and projection into the future provide a basis for estimating power requirements for future GEO missions. Comparing these to the state of the art for the various spacecraft power subsystem components for power generation, energy storage and power processing, provides a basis for estimating development needs. It is the purpose of this paper to illustrate this approach in determining power technology needs for GEO spacecraft and, in addition, to provide some indication as to how far current technology development programs go toward meeting those needs.

GROWTH IN COMMUNICATIONS TECHNOLOGY

Active repeater satellites have become a routinely accepted means of relaying electronic communications on a commercial basis. Some measure of the growth and international acceptance of this technology can be seen from the rapid increase in number of earth stations in the INTELSAT system, as shown in Figure 1. Although television is perhaps the form of communication most widely recognized by the general public, voice communications far exceed the traffic volume represented by television. This growth in service, characterized in terms of 4 KHz bandwidth one-direction links, is shown in Figure 2. The continuing exponential growth undoubtedly has been caused by a number of factors, including the reductions in rates shown in Figure 3.

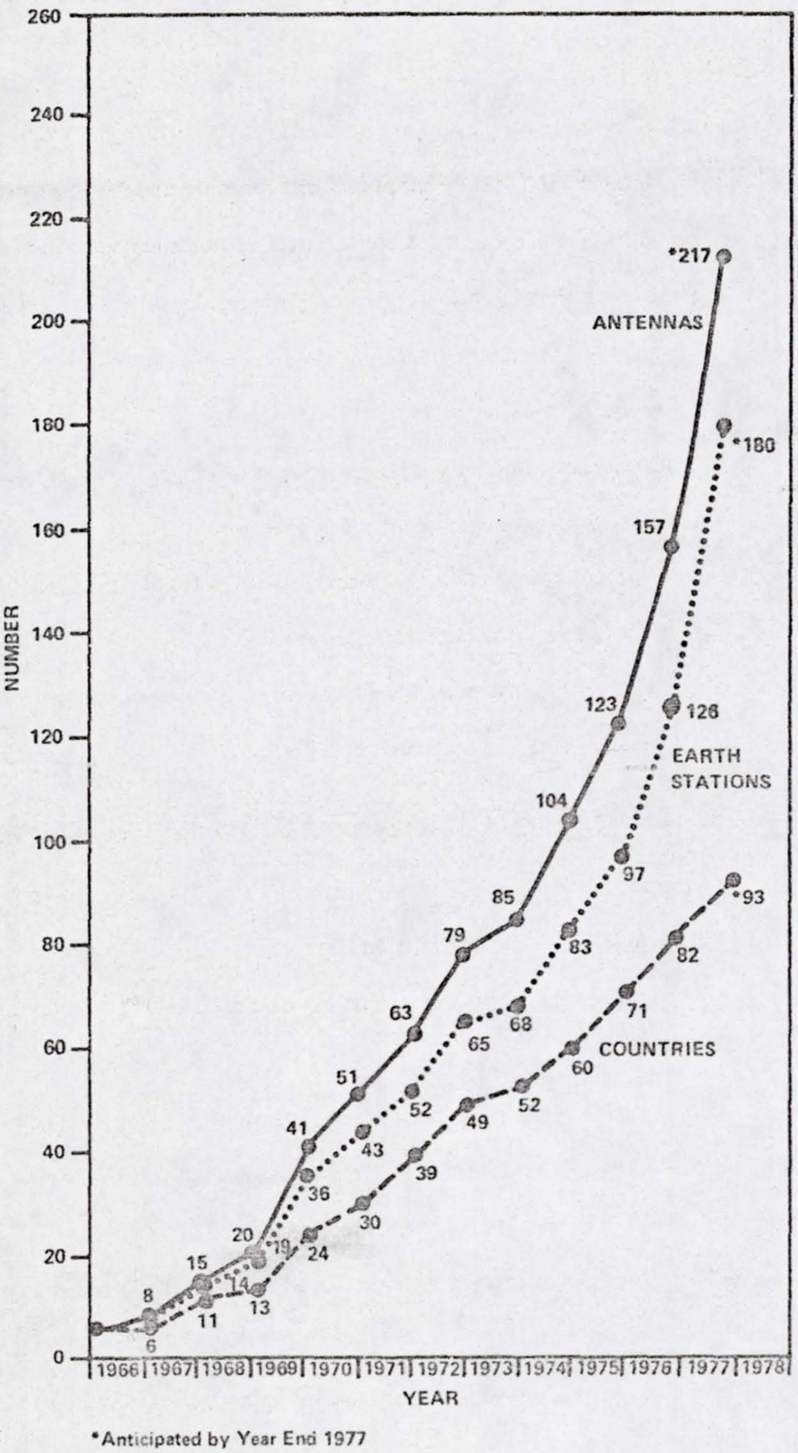

Figure 1. Earth Stations in the INTELSAT System (Year-end Totals)

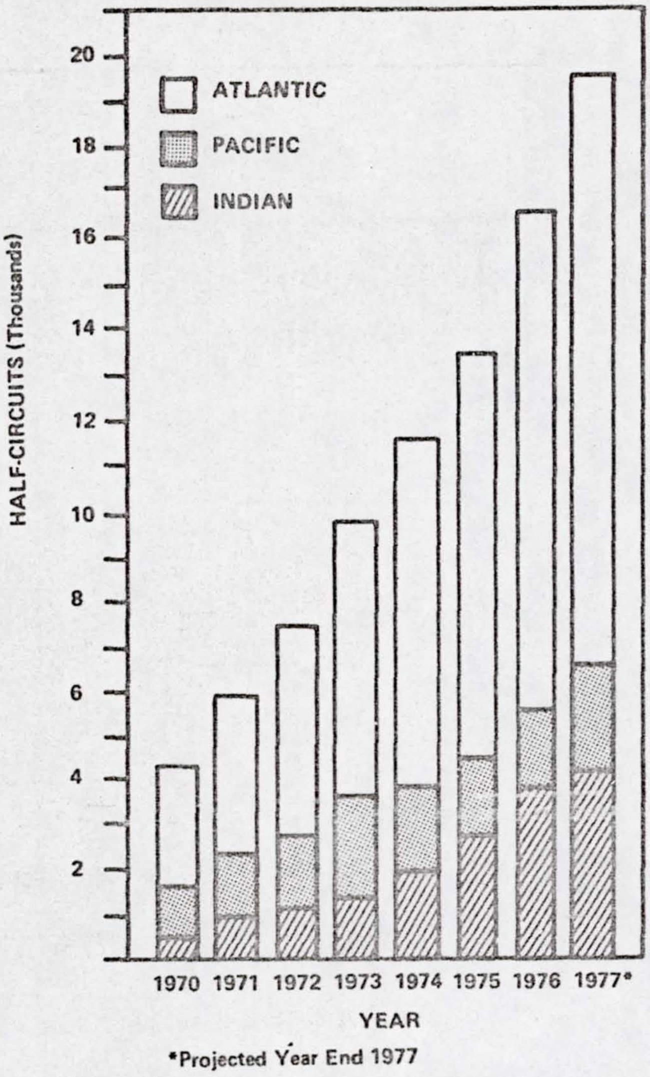

Figure 2. Full-time Satellite Use By Region

In recent years much attention has been given to increasing satellite communications capacities through such means as clever modulation techniques, narrower antenna beams, and polarization diversity to allow frequency reuse, and through more efficient electronic devices. In addition, effort has been focused on increasing the electric power available for communications equipment. Many spacecraft subsystems typical of those aboard current communications satellites have fallen into design patterns in which changes are evolutionary, rather than revolutionary. The electric power subsystems for communications satellites, to a large degree, fit this category.

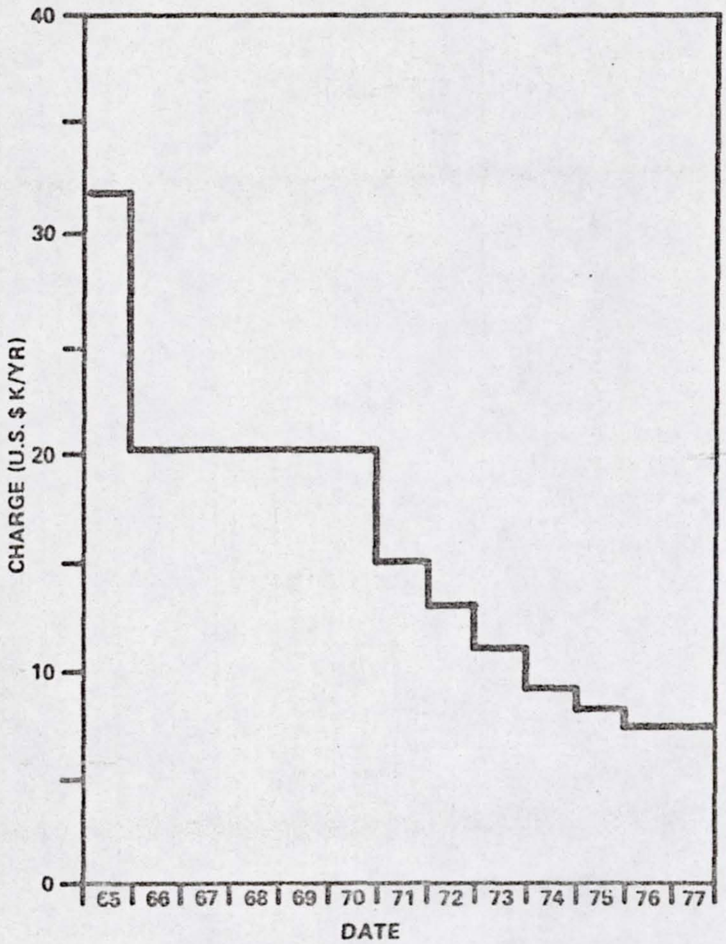

Figure 3. INTELSAT Satellite Utilization Charge Per Channel

Growth in Power Requirements

An interesting example of the growth of DC power requirements is provided by the INTELSAT series of spacecraft. All of these use microwave-repeater-type devices in which the majority of the power goes to traveling wave tubes having a DC-to-RF conversion efficiency in the neighborhood of 30 percent. As shown in Figure 4, the DC power requirement has increased with each new series of spacecraft.(1) The plot of *operational* spacecraft used in U.S. Domestic service, Figure 5, shows similar trends.

The relationship between DC load power and the number of telephonic half-circuits in the INTELSAT spacecraft is shown in Table 1. It should be noted that the design lifetime requirement

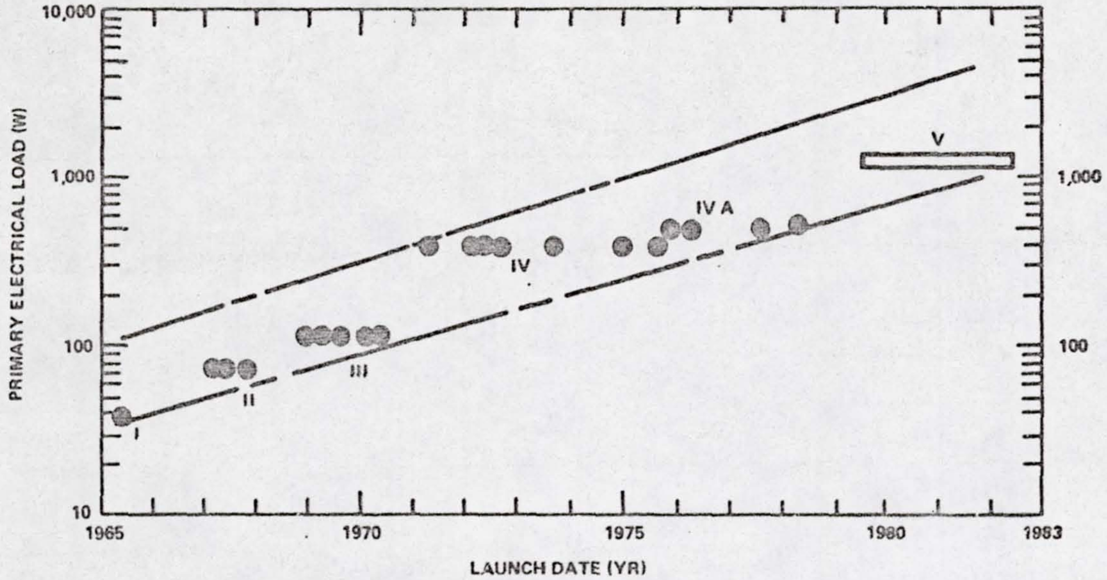

Figure 4. Electrical Load Power for INTELSAT Spacecraft

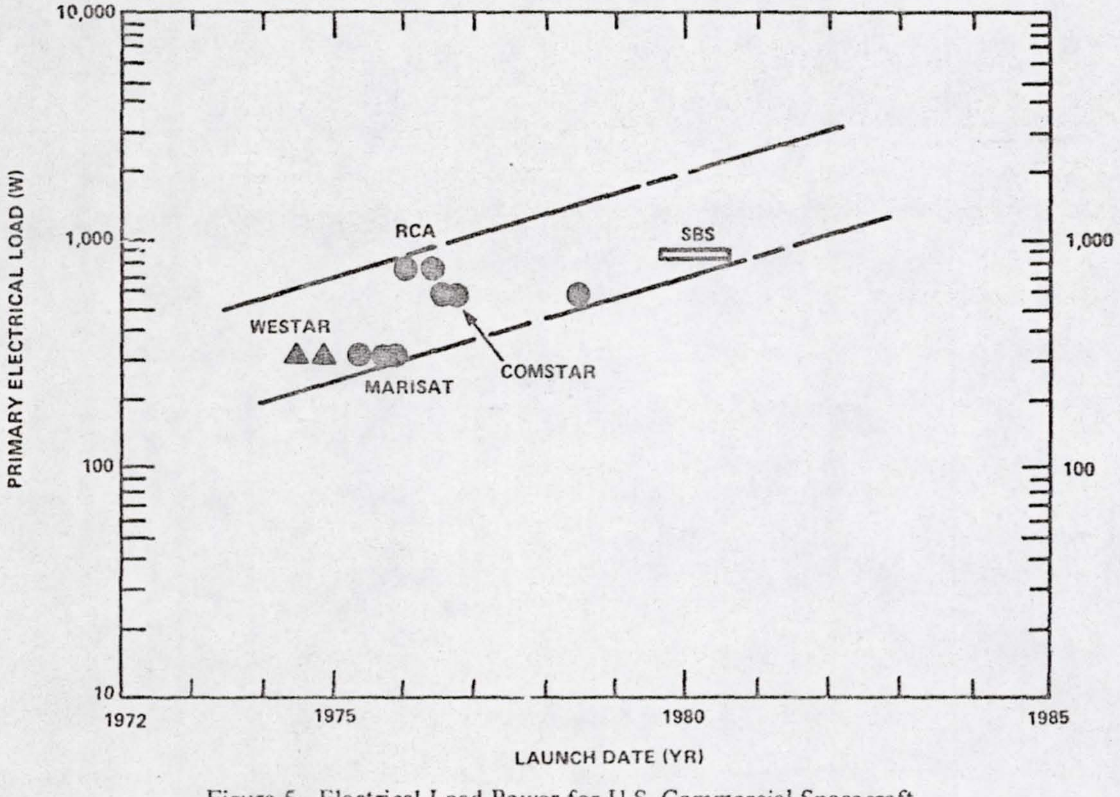

Figure 5. Electrical Load Power for U.S. Commercial Spacecraft

Table 1

Growth of INTELSAT Spacecraft

	INTELSAT Satellite					
	I	II	III	IV	IV-A	V
Year of 1st launch	1965	1967	1968	1971	1975	'79 to '80
Drum dimensions (cm) dia hght	72.1 59.6	142 67.3	142 104	238 282	238 282	– –
Overall deployed height (cm)				528	590	1585
Mass (kg) at launch in orbit	68 38	162 86	293 152	1385 700	1469 790	1870 1014
Primary load power (w)	40	75	120	400	500	975
Active Transponders	2	1	2	12	20	20–30
No. of tel. circuits	240*	240	1200	4000	6000	12000 +2TV
Design lifetime (yr)	1.5	3	5	7	7	7/10**

*No multiple access.
**Incentive 7 yrs; design 10 yrs.

for these spacecraft increased as each new design was laid down. This brings up the aspect of economics, since orbital lifetimes can obviously reduce costs and perhaps increase profits. As has been pointed out,(2) the revenue potential of these satellites is quite large, since each one can handle a very large volume of communications traffic. Assuming that mass saved in other subsystems can be effectively utilized by incorporating additional R.F. transponders, thus providing additional communications channels, one analysis(3) shows that in synchronous orbit commercial service the revenue potential of one kilogram of satellite in-orbit mass is of the order of 1 to 2 million dollars.

PROJECTED MODELS

In order to estimate requirements for power subsystems for future GEO spacecraft, three models were projected. These models are a mission model, an orbit transfer vehicle model, and a mass model for power subsystem components. It is emphasized that these models were projected for estimating purposes only and do not represent proposed, planned or approved projects.

Mission Model

The mission model, Table 2, was developed to establish a basis for estimating future power requirements. As such, it incorporates only missions with power requirements in excess of 2 kw. It should be noted, however, that there are many potential missions requiring lower power levels which do not impact the power requirements, but would certainly take advantage of any advances in technology.

Table 2

Mission Model for GEO Spacecraft

Year	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95
Spacecraft	Power (kw)															
Emergency I	2							2						2		
Emergency II			5							5						5
Hotline		2					2					2				
Intergov't I				2					2					2		
Intergov't II					3.5					3.5					3.5	
Intergov't III								5					5		5	
Electronic mail I					5				5				5			
Electronic mail II							15				15				15	
Voting/polling I							X				X				X	
Voting/polling II														50		
Pwr. module																2×10^3
Mobile comm. tech.				2												
TDRSS follow-on							5		5		5		5			
Public service								5			5			5		
Broadcast						2					2					2
Multi-beam comm.						3			3			3			3	
Geosynch. R&D												2				
Deep space relay										7						

NOTE: Life science, astrophysics, planetary, solar-terrestrial – no drivers, but will use what is available.

This model is a combination of elements from various sources.[4,5] The main features observed for this model are:

1. Communications continues as the primary application for GEO spacecraft.
2. Expansion of communications into new areas is projected.
3. The high power drivers (circled on the mission model) can be considered nominally as extensions of past power requirement trends (see Figure 6). A reference dedicated spacecraft and a reference multi-mission platform, in agreement with this model, were selected for further analysis.

Orbit Transfer Vehicle Model

With the future availability of the shuttle and the possibility of assembly of large spacecraft or platforms in low earth orbit (LEO), previous limitations on size become less significant. However, mass limitations remain critical and are a major factor in the transfer of spacecraft from LEO to GEO for operational use. This mass limitation is brought into consideration, using the orbit transfer vehicle model, Table 3.

The model projects an interim orbit transfer vehicle (IOTV) in 1985 with capability somewhat improved over that of the inertial upper stage (IUS) presently under development. Following the IOTV, development of large stage transfer vehicles with significantly increased capabilities: first, an interim large stage (ILS) vehicle in 1988, followed by an advanced large stage (ALS) vehicle in 1990; are projected. This model is used as a basis for approximating the spacecraft mass that can be put into GEO as various capabilities become available. That is, it is used to establish approximate mass limits for spacecraft as a function of time.

Table 3

Orbit Transfer Vehicle Model

Transfer Vehicle	Projected IOC* Year	S/C Mass to GEO (kg)
IUS	1980	2268
IOTV	1985	4000
ILS	1988	16,500
ALS	1990	24,000

*Initial operational capability.

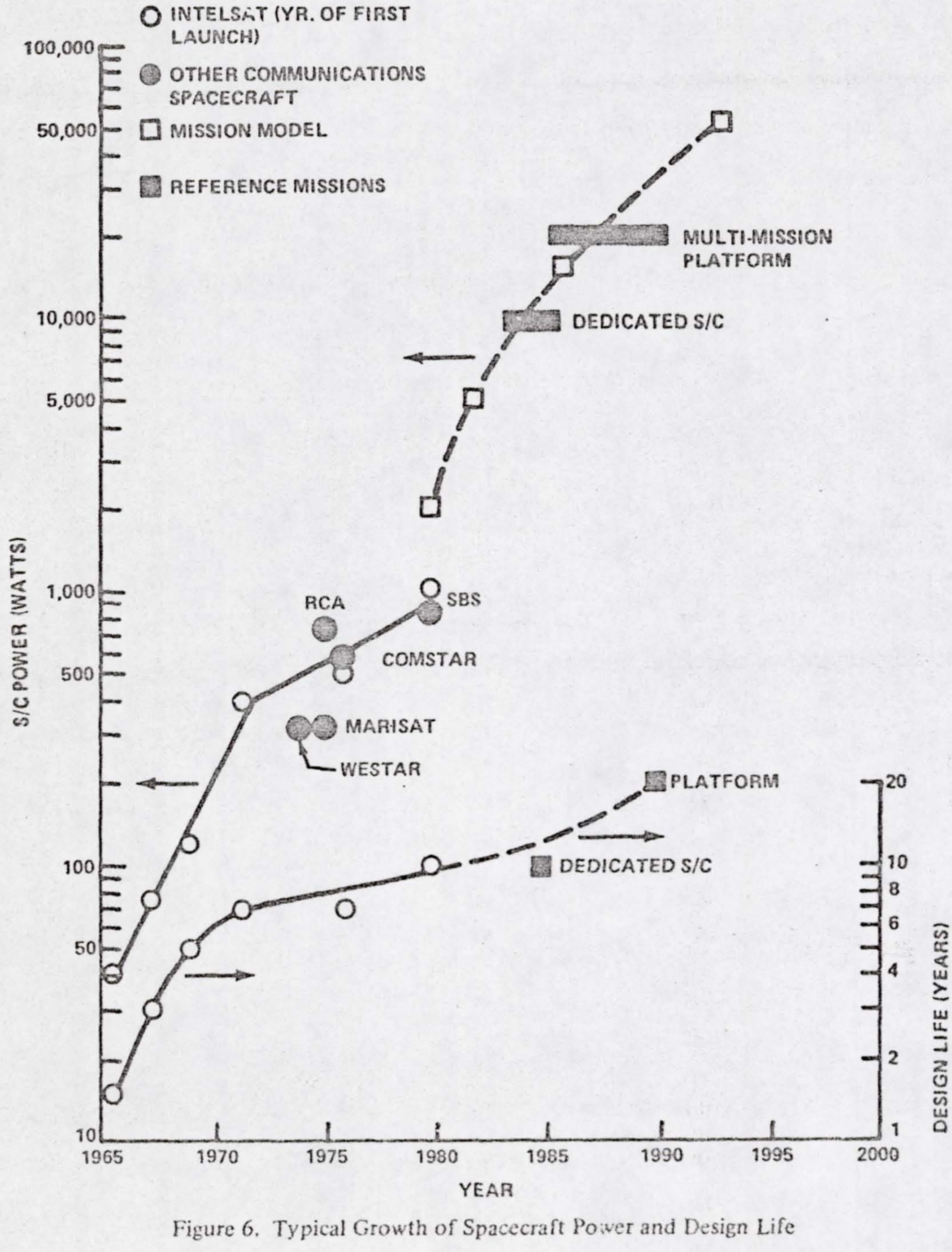

Figure 6. Typical Growth of Spacecraft Power and Design Life

Mass Model for Power Subsystem Components

Since spacecraft mass limitations will, in turn, result in limitations for the power subsystem and its components, a mass model for power subsystem components was developed, Table 4. This model is based on the INTELSAT spacecraft power subsystems, using mass ratios to define the model. Table 5 summarizes these values for the various INTELSAT power subsystems. It can be seen that, excluding INTELSAT-I (Earlybird) where there is uncertainty as to actual power subsystem mass, the percentage of spacecraft mass available to the power subsystem has continually decreased to a level of about 17.5%. Simultaneously spacecraft power requirements have increased an order of magnitude. These considerations led to the selection of maximum available ratio of power subsystem mass to spacecraft mass of 17.5% for the model, even considering another order of magnitude or more increase in spacecraft power. (For comparison, this ratio for spacecraft of the Tracking and Data Relay Satellite System, TDRSS, is 15.9%.)

Table 4

Mass Model for Power Subsystem Components (mass in kg)

Transfer Vehicle (Time Frame)	IUS (1980-1984)	IOTV (1985-1987)	ILS (1988-1990)	ALS (1990-)
Spacecraft	2268	4000	16,500	24,000
Pwr. subsystem (17.5% of S/C)	397	700	2888	4200
Array (36% of pwr. subs.)	143	252	1040	1512
Storage (36% of pwr. subs.)	143	252	1040	1512
Pwr. mgt. (28% of pwr. subs.)	111	196	808	1176

Further breakdown of the power subsystem mass among the power subsystem components was determined using the breakdown for INTELSAT-V, Table 6. These mass ratios (percentages) within the power subsystem were used to define the mass model for power subsystem components for the various transfer vehicles (see Table 4).

Table 5

Summary of INTELSAT Spacecraft Characteristics

Spacecraft	I	II	III	IV	IV-A	V
S/C power (w)	40	75	120	400	500	1000
S/C Mass (kg)	38.6	86.2	151.5	700	790	1014
Power Subsystem mass (kg)	>6.22	>24.8	34.9	136	148	178
Power Subsystem mass ratio	>0.161	> 0.287	0.230	0.194	0.187	0.176

Table 6

Mass Breakdown for INTELSAT V Power Subsystem Components

Component	Mass (kg)	Comp. %
Power Subsystem	178.0	100
Array	64.4	36.2
Batteries	63.7	35.8
Power processing	49.9	28.0

REFERENCE SPACECRAFT AND PLATFORM

Reference Dedicated Spacecraft (1985 Time Frame)

A dedicated spacecraft is defined here as one with only a single mission, such as communications. The reference dedicated spacecraft is assumed to be launched into LEO by shuttle and transferred to the GEO using either the IUS or IOTV as available. It has a 10 kw power requirement and a design life of 10 years (see Figure 6).

Solar Array. The solar array for the dedicated spacecraft must deliver 10 kw of power for spacecraft use at the end of life (EOL) of 10 years. Typical (INTELSAT-V, for example) designs include the step by step addition of 10% for battery charge (nominal for GEO missions), 5% for load contingency and 10% design margin. These result in an EOL power requirement of 12.7 kw for array output. Array degradation of 25% for 10 years in GEO, Figure 7, leads to a beginning of life

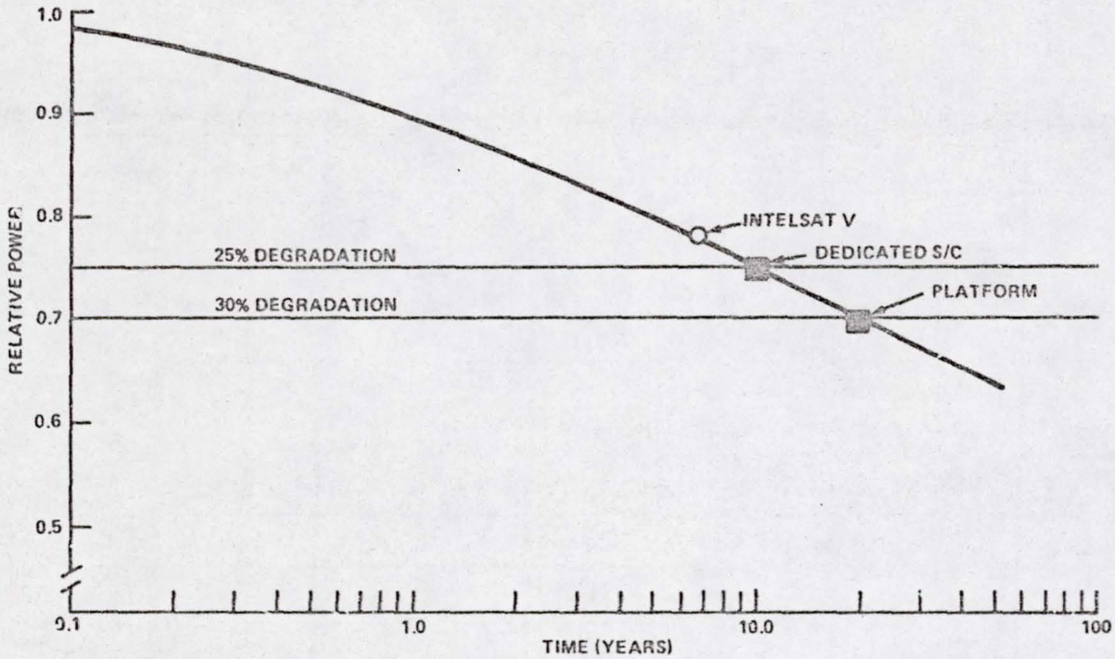

Figure 7. Solar Array Degradation in GEO

(BOL) requirement of 16.95 kw for the array. With the IUS vehicle, where 143 kg are allocated for the array, this represents a power density requirement of 119 W/kg. With the IOTV, where 252 kg are allocated, the power density requirement is 67 W/kg.

Battery. The spacecraft batteries must deliver 10 kw of power during eclipse. For GEO, the maximum eclipse period is 1.2 hours, resulting in a 12 kw-hr requirement for usable energy storage. Currently the design limit for Ni-Cd battery depth of discharge is 50% for a 10 year life in GEO.(6) This results in a minimum energy storage capacity (static) requirement of 24 kw-hr. For a state of the art system, such as the multi-mission modular spacecraft (MMS), batteries are made up of 50 AH cells with each battery having 22 cells in series to provide a bus voltage of 26.4 volts. For such a state of the art, low voltage system, 19 MMS-type batteries would be needed to accommodate the 24 kw-hr storage requirement.

In this design, the requirements for usable energy density are 84 W-hr/kg and 48 W-hr/kg for the mass allocated for IUS and IOTV respectively. Similarly, the static energy density requirements for the batteries are 175 W-hr/kg and 100 W-hr/kg for IUS and IOTV respectively.

Power Processing. The power to be processed within the spacecraft is 10 kw. Dissipation of excess power, prior to radiation degradation, or resulting from factors such as overdesign or failure of portions of the payload are not included in this figure.

Density requirements for power processing are 90 W/kg (11 kg/kw) and 51 W/kg (20 kg/kw) respectively for the IUS and IOTV mass allocations.

Reference Multi-Mission Platform (1985-1990 Time Frame)

A multi-mission platform is defined here as an assembled platform to which a variety of equipments designed to perform a variety of missions (services or functions) are mounted. Several subsystems such as those for station keeping, attitude control and power can be utilized in common. The reference platform, as with the dedicated spacecraft, is assumed to be launched by shuttle and assembled in LEO. It is then transferred to GEO by the IOTV, the ILS or the ALS as available. It is defined to have a 20 kw power requirement and a design life of 20 years (see Figure 6).

Solar Array. The spacecraft load of 20 kw leads to an EOL requirement of 25.4 kw for the array, using the same additional battery charge, load contingency and design margin as for the reference dedicated spacecraft. In this case, however, for a design life of 20 years, a degradation of 30% (see Figure 7) leads to a BOL requirement of 36.3 kw. Array power density requirements are found to be 144, 35, and 24 W/kg for the IOTV, ILS and ALS respectively.

Energy Storage.* For a 1.2 hour eclipse the 20 kw load results in a 24 kw-hr energy discharge. This, in turn, leads to total energy storage requirements of 48 kw-hr with a 50% discharge allowance or 30 kw-hr if an 80% discharge allowance is attained. In either case, the usable energy density requirements are 95, 23, and 16 W-hr/kg for IOTV, ILS and ALS respectively. However, static energy density requirements are 190, 46, and 32 W-hr/kg for a 50% depth of discharge and 119, 29, and 20 W-hr/kg for an 80% depth of discharge for the respective transfer vehicles.

Power Processing. For the 20 kw platform requirement, the allocated masses for power processing lead to density requirements of 102 W/kg (9.8 kg/kw), 25 W/kg (40.4 kg/kw), and 17 W/kg (58.8 kg/kw) for the IOTV, ILS, and ALS respectively.

*The change in terminology from "battery" to "energy storage" has been intentional in recognition of the possibility that an alternative storage system will be used for the platform.

Power Density Requirements

The requirements determined above for the reference dedicated spacecraft and for the reference multi-mission platform power subsystem components are summarized in Table 7 in comparison to the state of the art (INTELSAT-V) data. This table shows the need for improvements, many of them by a factor of 2 or more, over the current state of the art if larger power systems (10 or 20 kw) are to become a reality in the near term (1985). However in the far term (beyond 1988), the projected large stage transfer vehicles would make major reductions in these requirements.

Table 7

Technology Needs for Power Subsystem Components

	State of the art	Requirement				
		Dedicated S/C		Platform		
Transfer vehicle	–	IUS	IOTV	IOTV	ILS	ALS
Year (IOC)	1978	1980	1985	1985	1988	1990
Array (W/kg)	26.5	119**	67**	144**	35*	24
Storage (WHR/kg)†	18.3	84**	48**	95**	23*	16
Pwr. processing (W/kg)	20.0	90**	51**	102**	25*	17
Lifetime (yrs)	7	10*	10*	20**	20**	20**

*Advancement required
**Major advancement required (x 2 or more)
†Usable energy density

Lifetime Requirements

Since lifetime is related to improved reliability and/or refurbishment, and not related to vehicle capability, it remains as a development need, especially for the 20 year case.

Additional Requirements

In addition to the power density requirements derived above, many additional requirements must be simultaneously met. Some of these are implicit in the above derivation, and others result from operational constraints, environmental conditions, or interactions of the power subsystem within itself, with other subsystems, with the spacecraft and with the environment. Since these are already familiar requirements, they are listed below without elaboration.

Solar Array

1. The solar array must provide standby and housekeeping power for the spacecraft during transfer to GEO.
2. Array deployment (and probably retraction for dedicated spacecraft) is required.
3. The array must be orientable toward the sun, minimizing dynamic interaction with the spacecraft.
4. The array must survive thermal cycling during eclipse. Cycling will be approximately from +50°C to -200°C, with 880 cycles experienced by the 10 year spacecraft array and 1760 cycles experienced by the 20 year platform array.
5. The array must produce 17 kw of power (BOL) and 12.7 kw (EOL) for the 10 kw dedicated spacecraft and 36.3 kw (BOL), 25.4 kw (EOL) for the 20 kw platform.
6. Maximum degradation for the spacecraft is 25% in 10 years life and for the platform is 30% in 20 years life.

Energy Storage

1. The energy storage component of the power subsystem must provide 12 kw-hr of usable energy for the 10 kw spacecraft and 24 kw-hr of usable energy for the 20 kw platform.
2. It must provide energy for 880 and 1760 cycles, for the spacecraft and platforms respectively, at depth of discharge and operational temperature (for example, 50%, 15°C max. for Ni-Cd batteries).
3. Between discharge cycling, it must store energy for periods of 5 months each and at operating temperature, 20 periods for the spacecraft and 40 periods for the platform.

Power Processing. The power processing equipment must accommodate the following:

1. It must transfer 17 kw of power across the array/spacecraft interface or 36 kw of power across the array/platform interface and distribute it within the spacecraft or alternatively to various mission modules on the platform and then within the modules as required.

2. It must dissipate excess array power (7 kw BOL on the spacecraft, 16.3 kw BOL on the platform).

3. It must control bus voltages, which may be either regulated or unregulated, and regulate equipment voltages.

4. It must provide switching (high or low voltage; AC or DC), sensing, and fuzing as required.

5. It must perform at high efficiency.

Thermal Control. Thermal control of the subsystem components is required to provide for the following functions.

1. Minimize array temperature for best efficiency.

2. Control battery temperature within required limits (0°C - 15°C for Ni-Cd batteries).

3. Dissipate waste heat from:

   (a) Array excess power (40%)

   (b) Battery inefficiency (20%)

   (c) Processing Inefficiency (15%)

Subsystem Requirements

1. The power subsystem must be stable for the launch, transfer, and orbital environment, including orbital plasma effects related to spacecraft charging or to high voltage/plasma interactions.

2. All components of the power subsystem must be safe during fabrication and test (handling), safe for shuttle (manned) launch, and safe during docking and assembly operations.

FUTURE ADVANCES IN TECHNOLOGY

Solar Energy Conversion

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

Solar Cells. The silicon photovoltaic cell had reached a rather stable design status with a conversion efficiency plateau of about 10½ percent in the mid 1960's. Some detailed analysis of the sources of energy loss within the cell[7] high-lighted 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."[8] This triggered a new burst of silicon cell development[9] which has produced laboratory cells having conversion efficiencies as high as 15½ percent, Figure 8.

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

Another exciting possibility, which is still in the laboratory stage at this point, is the 50 micron (2 mil) thick silicon cell. The development work on this component is being sponsored by NASA through JPL.[10] 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 have the potential of producing about 1000 W/kg bare, and if 50 to 100 micron covers can be produced and handled, could possibly achieve something in the vicinity of 300 to 500 W/kg covered. In both cases above, the mounting and interconnection provisions are not included in the mass. However, a great deal of laboratory work remains to be done to learn how to routinely manufacture, cover, mount, and interconnect these cells. Of course the actual power to mass ratio for practical arrays is considerably below these figures due to the weight of materials and structures needed for interconnecting, supporting and deploying the solar cells as discussed below.

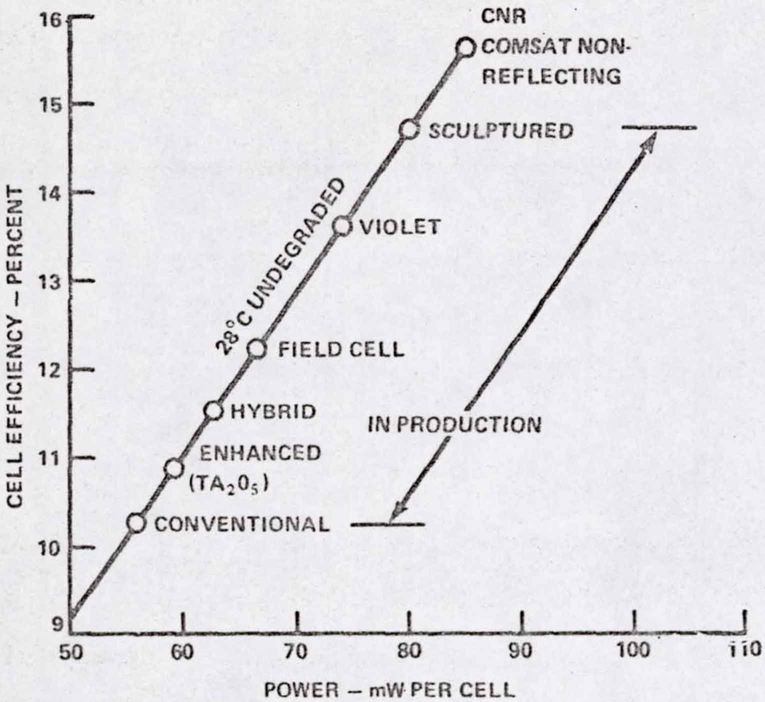

Figure 8. High Efficiency Solar Cell (2 x 2 cm) Performance

Efficiency improvements from ongoing research and development on advanced cells such as GaAlAs and multigap cells could also have significant effects on power to mass ratio.

Solar Array Structures. Solar array hardware is gradually incorporating the solar cell improvements into operational spacecraft as mentioned above. The structure of these arrays is also changing in an evolutionary way. With the exception of the RCA SATCOM which used a sun-oriented metallic panel with hat-shaped stiffening beams, all of the INTELSAT and U.S. commercial spacecraft presently in orbit have been drum spinners with a honeycomb sandwich panel construction. This structure consists of epoxy bonded glass fibre skins bonded to a vented aluminum honeycomb spacer. In structural design terminology, the fibreglass skins 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 drum-spinner array has proved quite simple and reliable, but as has been pointed out,(11) requires roughly three times as many solar cells as a sun-oriented array. The weight of these drum spinner arrays has been in the region of 110 to 160 Kg/kwe at end of mission using values taken from the above reference. (See also Table 8.) The RCA design is, of course, considerably lighter, weighing about half as much as the drum spinners.

Table 8

Solar Array Characteristics (End of Life)

Spacecraft	Power	Array Type	W/m^2	Kg/kwe	W/Kg	Design Status
NATO III	375w	Body-mounted	24.5	127.7	7.83	Flight
INTELSAT IVA	522w	Body-mounted	24.5	137.7	7.26	
ANIK I-III	219w	Body-mounted	23.8	151.6	6.60	
ATS-6	600w	Rigid, oriented	30	101.3	9.87	
Orbital Workshop	12.24kw	Rigid, oriented	97	188.7	5.30	
Flt Sat Com	1.47kw	Rigid, oriented	69	62.1	16.1	
Communications Technology Satellite	1045w	Flexible	55.3	45.5	22.0	
Hughes FRUSA	1100w	Flex roll-out	70	28	35.7	
SBS	710w	Body-extended Skirt		103	9.71	Operational Designs
INTELSAT V	1290w	Rigid, oriented		50	20.0	
TRW Lightweight	1470w	Rigid	75	41	24.4	R&D
MBB ULP	1500w	Semi-rigid	75	30	33.3	
Aerospatiale/Comsat	1050w	Flex fold-out	72	34	29.4	
AEG Dora	6600w	Flex roll-out	77	25	40.0	
RAE Flat-Pack	250w	Flex fold-out	65.8	25	40.0	
Lockheed SEPS	12,500w	Flex fold-out	83	18	55.6	

As the payoffs in mass reduction of these commercial satellites have come to be recognized, the new array structural designs have begun to reflect some of the aerospace industry advancements in lightweight structures. For instance, the SBS and Anik-C drum spinner designs with extending skirts, Figure 9, now underway at Hughes Aircraft Co. are planned to have an epoxy-bonded Kevlar high strength to weight ratio fibre in the face skins. In another example, the INTELSAT V solar arrays, Figure 10, which are presently in the fabrication and testing stage use woven graphite fibres in the 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 8, in combination with the solar cell advancements mentioned earlier offer high promise for continuing area efficiency and power density improvements up to an order of magnitude or so better than the present operational designs. Much of this improvement will probably result from elimination of much of the substrate weight by moving to semi-rigid or flexible substrate concepts of the types investigated in the ongoing R&D work. Current R&D designs of flexible roll-out and flexible fold-out arrays are in the 30–60 W/kg (34–18 kg/kwe) power density range and the future possibility of arrays with power densities of 110 to 200 W/kg has been indicated in conceptual design studies.[12]

Energy Storage

Ni-Cd Batteries. The rechargeable nickel-cadmium (Ni-Cd) alkaline cell has been used to supply primary power during eclipse in all of the U.S. domestic and INTELSAT communications spacecraft flown to date. 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 analysis of a number of batteries designed for these geosynchronous orbit missions reveals that the total energy density available from new cells at 100 percent depth of discharge is relatively constant at about 26 watt-hours per kilogram. The cell weight comprises about 82 percent of the weight of a typical flight battery, with the wiring, connectors, electronics and structure making up the remaining 18 percent.

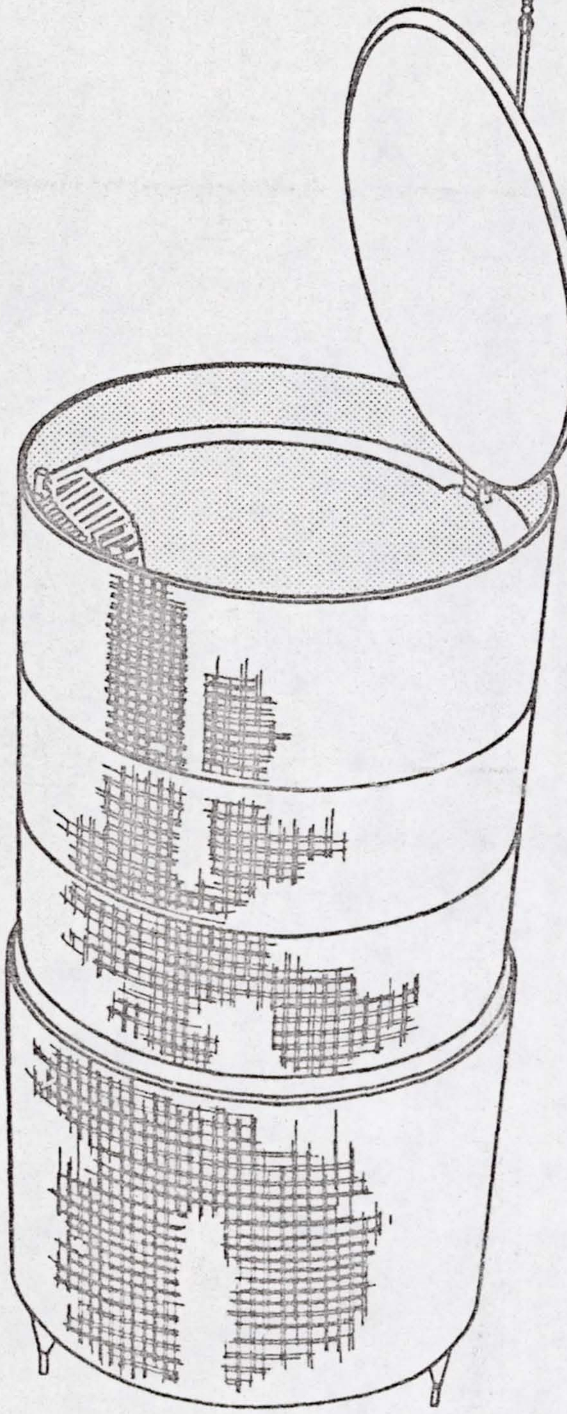

Figure 9. SBS/ANIK-C Extendable Skirt Spinner Array

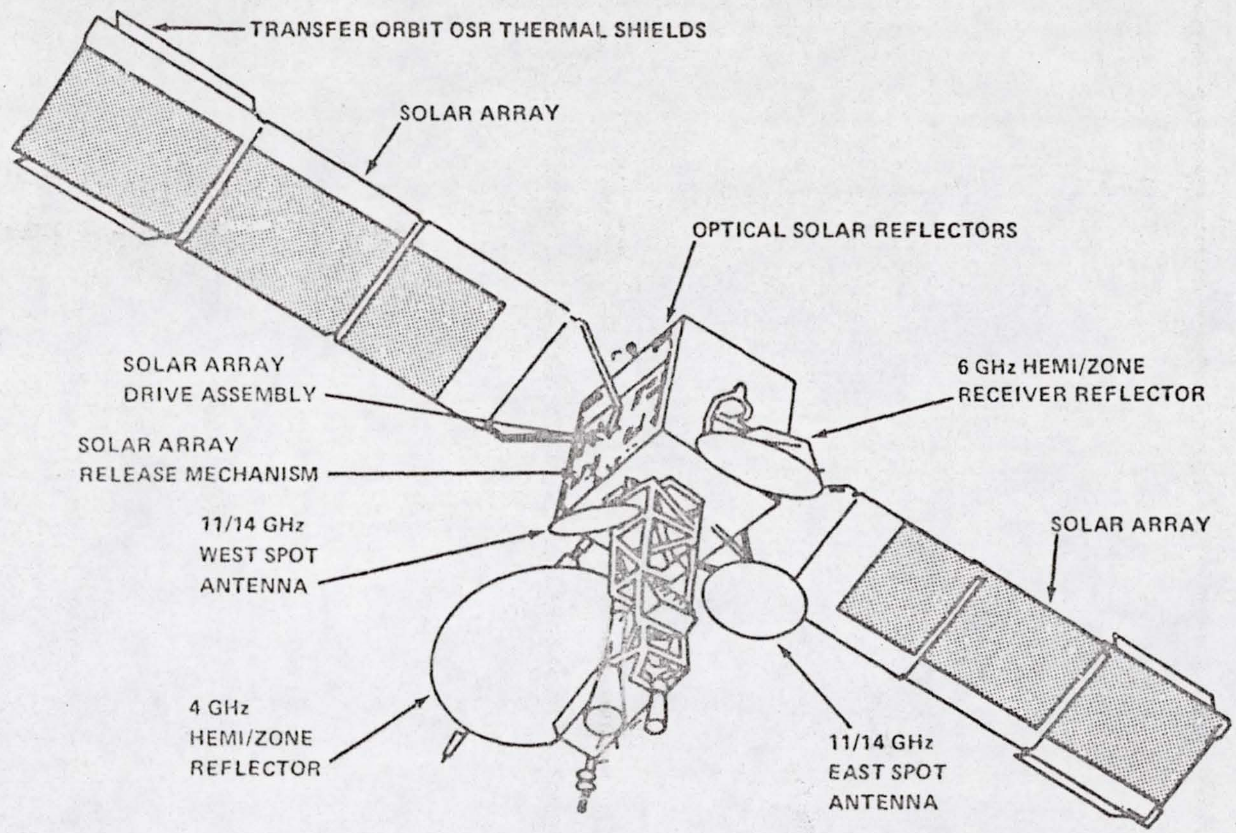

Figure 10. INTELSAT-V Spacecraft Configuration

Typically, from one-third to one-half the power subsystem mass in these spacecraft consists of batteries. The principal variables which determine the delivered energy density are the actual depth of discharge used and the redundancy strategy. This is shown graphically[1] in Figure 11. Most of the U.S. commercial and INTELSAT craft have used 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. The energy density in terms of actual watt-hours/kg delivered to the load during the longest eclipse is shown in Table 9 for a number of these spacecraft.

In-orbit reconditioning[13] is allowing longer mission life and deeper discharging of the Ni-Cd batteries in newer designs, resulting in the somewhat higher energy densities (see Table 9) for current spacecraft. It is hoped that additional R&D in this area will allow some further incremental improvements in life and energy density of Ni-Cd batteries in future designs.

Nickel Hydrogen and Other Advanced Couples. Advanced electrochemical energy storage devices which offer improvements in energy density have been investigated for a number of years. Of the large number of possible systems, the low-temperature Ni-H_2 alkaline system presently appears to possess the best near-term potential for synchronous orbit use.[14] These hermetically sealed secondary cells can be regenerated electrically, and require no pumps, or other moving parts. In addition, they have the high-rate discharge capability needed for operation in synchronous orbit spacecraft. The electrolyte is an aqueous solution of 35-percent potassium hydroxide.

A typical physical arrangement of these new Ni-H_2 cells is shown in Figure 12. A single-cell design is being pursued in this work for a number of reasons. It offers a high energy density and is advantageous in terms of reliability, since leaks or electrical faults can be isolated to a single cell by using diode bypassing techniques. Some extensive analytical work has been conducted to evaluate the energy density that can be achieved in practical designs of these single-cells. Specifically, computer models of the Ni-H_2 cells have been developed for parametric study. Further practical designs and fabrication studies, including structural and thermal aspects, have been done, including the appropriate testing.[15]

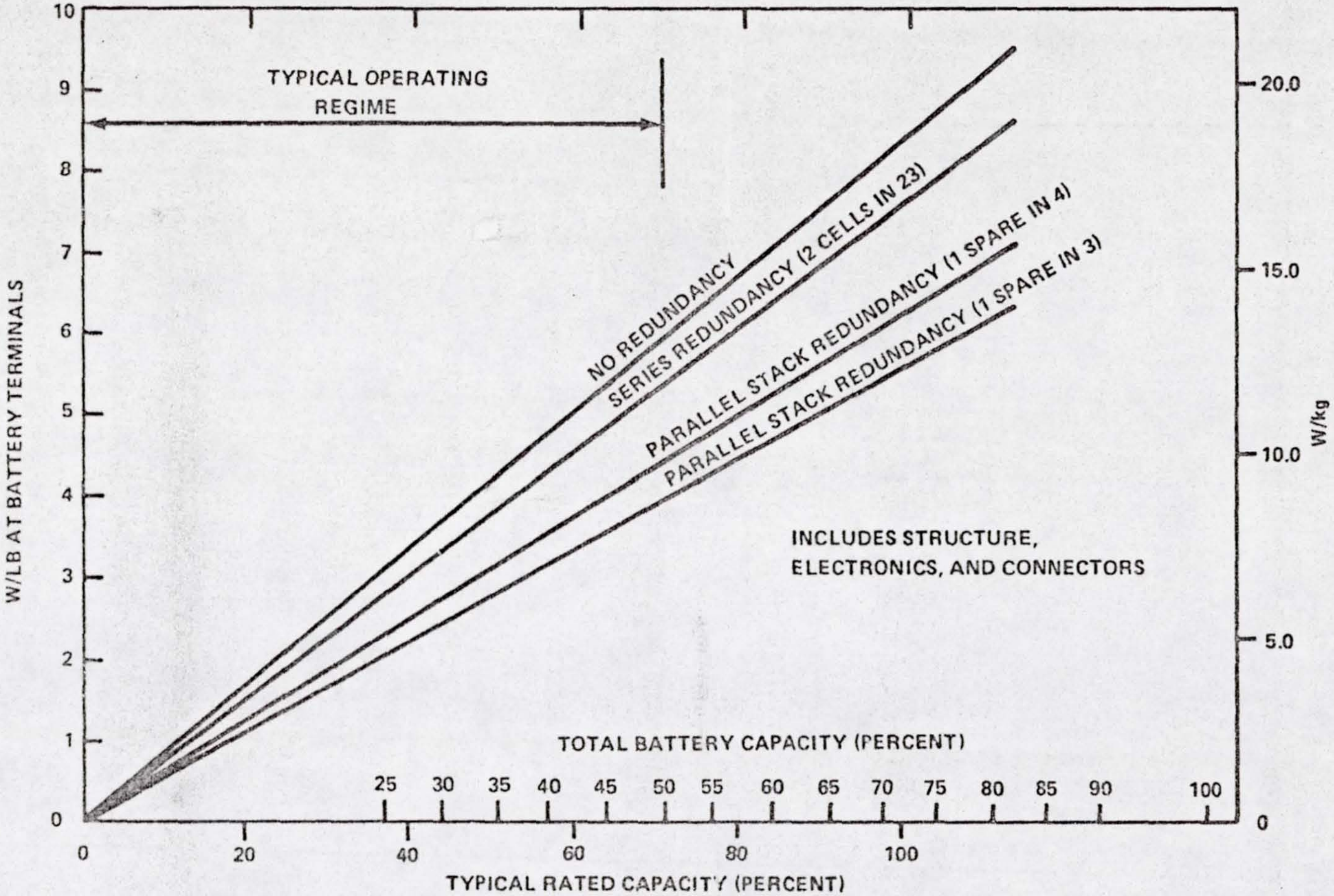

Figure 11. Ni-Cd Battery Energy Density vs. Depth of Discharge

Table 9

Battery Energy Density Experience

Battery Including Hardware	Delivered Watt-hrs/kg
INTELSAT III	13.9
INTELSAT IV	13.2
INTELSAT IVA	13.2
INTELSAT V	17.6
COMSTAR	15.4
SBS	15.4
RCA	16.5
NTS-2 Experiment ($Ni\text{-}H_2$)	17.6
Potential $Ni\text{-}H_2$ (Conventional Designs)	39 – 52
Potential $H_2\text{-}O_2$	44 to 110

The energy density which might be expected if $Ni\text{-}H_2$ cells were assembled into series strings for use as spacecraft batteries has been estimated. This estimate assumes the use of 35- to 50-AH cells. Since they are large enough to provide good cell energy density and have a capacity consistent with spacecraft designs in the kilowatt range. Data(15) indicate that a cell energy density of about 44 W-hr/kg was achieved for the 35-AH design; estimates based on uprating it to 50-AH indicate about 57 W-hr/kg at 100-percent depth of discharge, Figure 13. When these cells were assembled into a 10-cell battery stack with the necessary structural and thermal arrangements and all the usual flight accessories such as bypass diodes, wiring, and connectors, the energy density of the 35-AH unit was 39.2 W-hr/kg. The 50-AH unit was similarly estimated at 51.4 W-hr/kg (see Table 9).

The depth of discharge which can ultimately be used with $Ni\text{-}H_2$ cells in operational missions is not yet completely clear. Extensive laboratory cycling testing has been done at 50- to 80-percent depth of discharge and the flight experiment on the NTS-2 spacecraft was run quite successfully at a maximum depth of discharge of 60 percent.(16) Also it is not yet clear when the first application

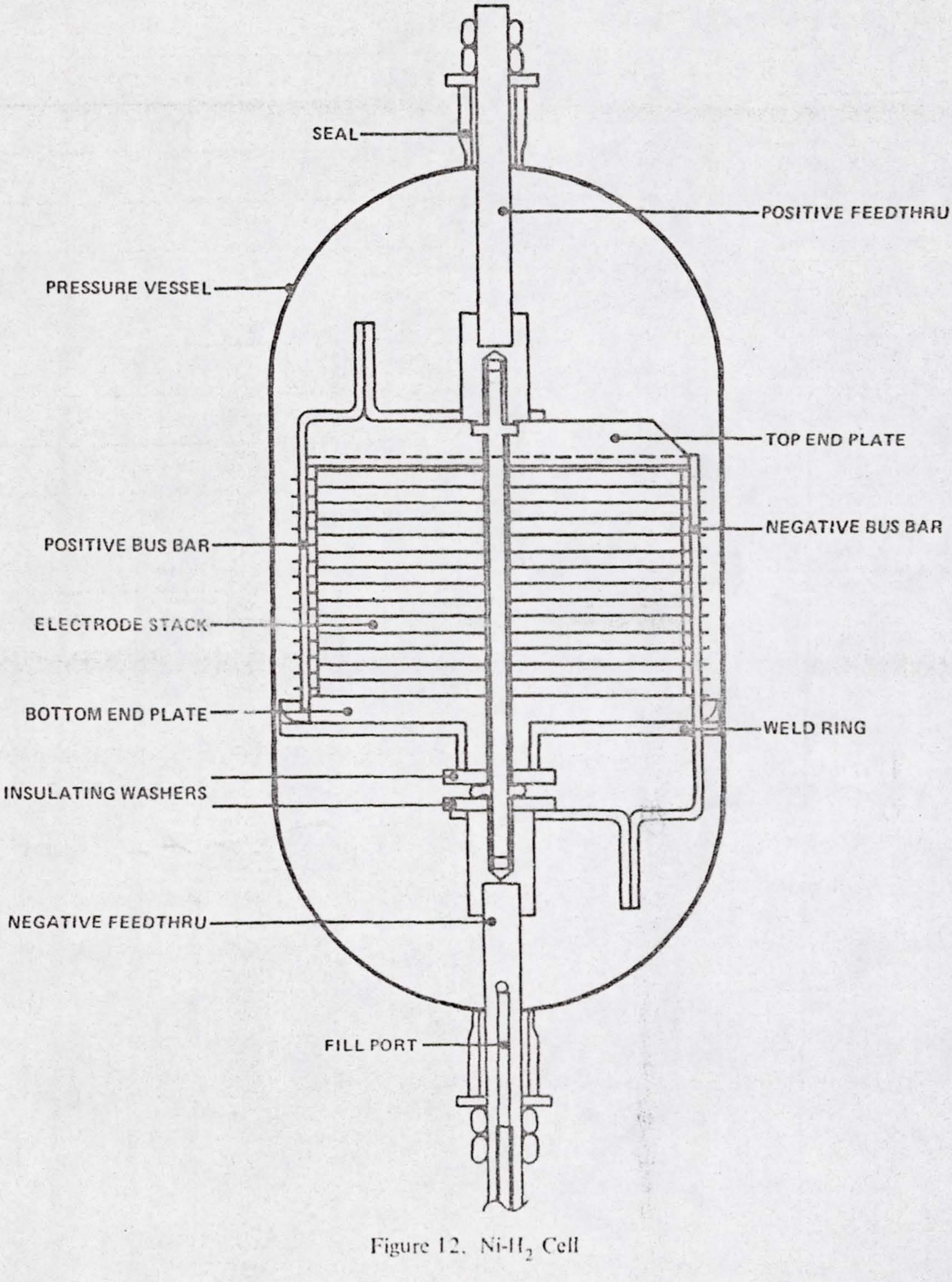

Figure 12. Ni-H_2 Cell

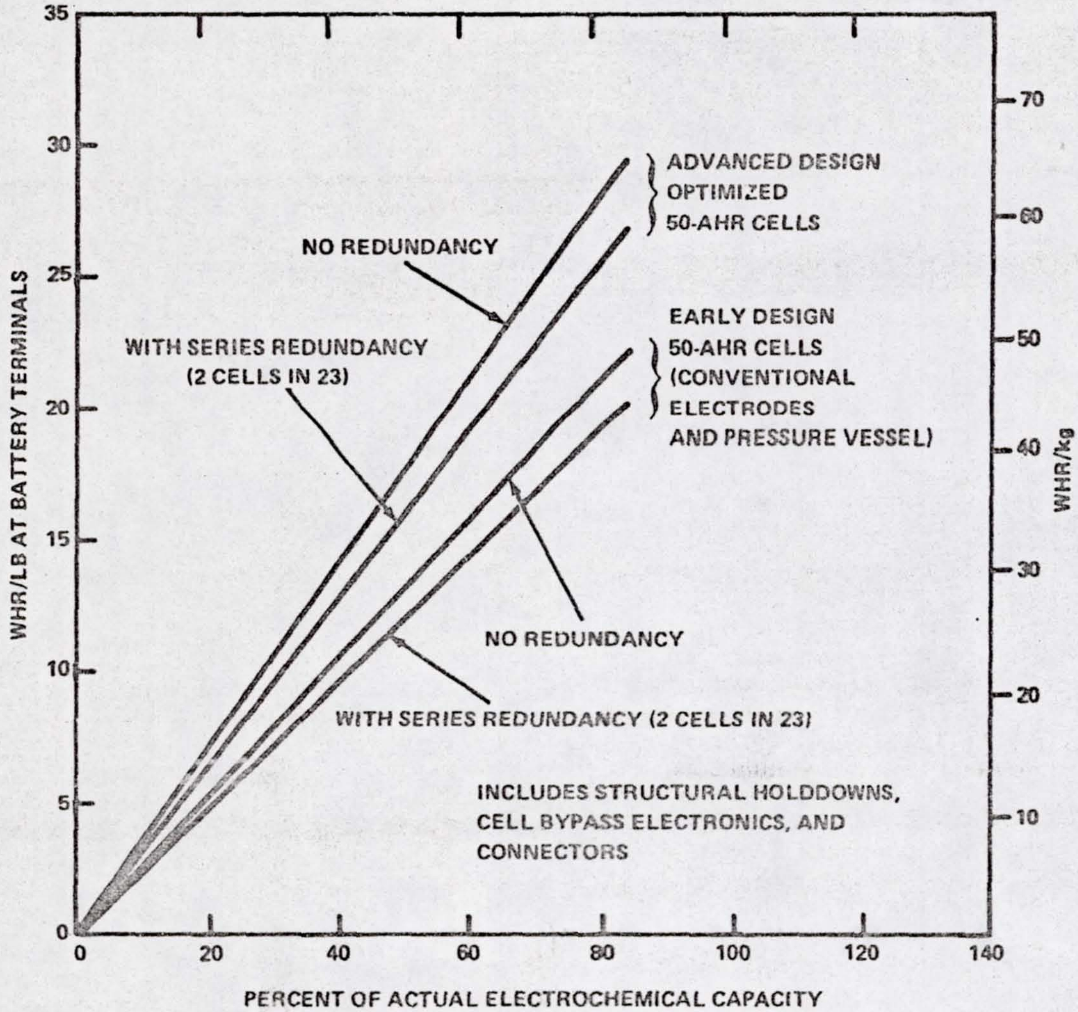

Figure 13. Analysis of Ni-H_2 Battery Energy Density

of Ni-H_2 batteries in operational communications satellites will take place, but the engineering and flight qualification work on batteries aimed at possible use in the later INTELSAT V craft is well underway.

For the more distant future, possible third generation aerospace energy storage systems are of interest. Some of those investigated in recent years include high-speed flywheels, the Ag-H_2 electrochemical system, several non-aqueous lithium couples, and the regenerative H_2-O_2 fuel cell. Inertia wheels, which can presently provide energy densities in the vicinity of 35 to 45 W-hr/kg offer some hope of higher performance with advanced materials. However, their inertial and reaction torques are a significant problem in many of the spacecraft applications. The advanced electrochemical systems appear to be a fruitful area for invention and innovative design ideas. For instance, various forms of the regenerative H_2-O_2 fuel cell demonstrated in the laboratory under several NASA-, USAF-, and INTELSAT-supported R&D programs have an attractive energy density in the vicinity of 100 W-hr/kg or more. An innovative approach which solves the pressure control and other reliability problems of this cell could represent a significant breakthrough in energy storage for aerospace applications.

CONCLUSIONS

1. There has been a rapid expansion in the use of geosynchronous equatorial orbit, especially for communications spacecraft.
2. Communications spacecraft have placed continually more stringent requirements on the power subsystems, both for total power load and for power density.
3. Projection into the future shows the need for major improvement in power density in all three components (power conversion, energy storage, power processing) of the power subsystem. It is anticipated that dedicated mission commercial spacecraft in the multi-kilowatt range will be designed in the next 3 - 5 years.
4. Projected needs in terms of power density or energy density are strongly dependent on orbit transfer vehicle capabilities.
5. Assuming transfer vehicle developments are realized with the high growth rates of the projected model used:
   (a) Launch of a 10 kw spacecraft prior to 1988 would still require major improvements in each of the power components. Availability of a technology ready flexible roll-out or flexible fold-out array in 1985 is a key element in reducing the need in array power density to one of minor improvement.
   (b) Launch of a 20 kw platform as early as 1988 (design in 1985) may be feasible by allocating "excess" array mass to energy storage and power processing.

(c) Advent of an advanced large stage orbit transfer vehicle (ALS) at a very early date with capabilities projected in the model would all but eliminate mass restrictions for the power subsystem components in the power range considered. However, in the more likely event that only the IOTV or possibly ILS are available in the needed time frame, significant power system advances are required.

6. Lifetime attainment (reliability and/or refurbishment) is problematical, especially for required lifetimes in excess of 10 years. Much development or refinement, particularly in the energy storage area, is needed to satisfy the projected 20 year lifetime requirement.
7. In the light of the model, current R&D efforts, and existing concepts for future development, the energy storage component of the power system appears to be the one most in need of attention.
8. In order to achieve technology readiness in the time frame of the model, R&D efforts aimed at accomplishing the required advances in power density and energy density must start at an early date.

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