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A Century of Aerospace Electrical Power Technology

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Introduction

THE evolution of the aerospace technology is marked by a series of technology achievements across the spectrum of the engineering disciplines, and one of the more interesting stories in the history of the aerospace industry relates to the development of electrical power systems for aircraft and spacecraft. The research and development effort leading to today's systems has spanned 10 decades and has continued through world wars and economic depressions. It was dependent on a marriage of ideas from virtually all fields of engineering: electrical, mechanical, materials, industrial, chemical, computer, and aerospace. It has achieved a remarkable state of technology even though it was never as prominent as the developments in propulsion systems, as elegant as breakthroughs in avionics, or as intensely studied as structures and aerodynamics. It has always been the "other" system, one that everyone assumed would be there at the appropriate time. And it was.

The evolution of aerospace electrical power systems from the early days of powered flight through today is marked by a combination of serendipity, insight, innovation, and hard work. In this paper we will recall some of the early difficulties faced by aircraft designers, the processes through which they addressed the challenges, and how they went about creating the technology breakthroughs leading to today's aircraft and spacecraft electrical systems. At various points in the review, we pause to probe more deeply into the key technologies of that day to give the reader a better appreciation of the ingenuity of the early engineers who created this technology. Such a review could not hope to cover all of the topics of relevance or of interest, and so a subjective selection has been imposed, hopefully one that does not do too great an injustice to aerospace power history or to those who created it.

The story is presented in two parts: first, the development of aircraft systems and, then, spacecraft systems. This is done, as the reader will see, because of the differences in the constraints, operational environments, and demands on the two systems. In one important aspect, however, they do share at least one common trait: not surprisingly, each benefited from an opportunity to harvest existing technologies until the demand for power exceeded the ability of these existing technologies to address the need for power. Only then, sparked by investments from the government, the airlines, a host of component manufacturers, and the aircraft builders, did serious research and development efforts begin.

Aircraft Electrical Power

Beginnings

Even the earliest powered aircraft depended on an electrical power system, albeit a simple, dedicated one. As we can imagine, in the first days of powered flight the focus was on designing a machine that could fly and be controlled and with the development of an engine that was light enough and powerful enough. The early Wright gliders, of course, needed no electricity, because they carried no engines or instrumentation (other than a piece of wool attached to the wing to indicate airflow). These gliders first appeared at the beginning of the 20th century, and with them the Wright Brothers introduced two key improvements to the new disciple: the elevator for steering and the ability to maintain balance and vary lift by flexing the rear edge on the aircraft. Once the gasoline engine was introduced, the aircraft became dependent on electrical power for ignition, and this dependency has continued to grow with time. With the first powered flights in 1903, they also introduced a new discipline—aircraft electrical power. No real thought was given to the need for electrical power because it was good enough then to take off, fly, and return safely. Any research that was done focused on aerodynamics, structures, and lightweight engines. This was the situation in early aviation that continued until the First World War.

The early electrical system for an aircraft was based on that developed for automobiles. This was done for two reasons: first, it was the most available technology, and second, the requirements for electricity on the early airplanes were minimal to none. Those aircraft that operated on the Otto-cycle principle universally depended on self-contained magnetos for the ignition system. In those days the entire electrical system consisted of the on-off switch for the magneto! It is easy to forget that for Lindbergh's famous flight in 1927 *The Spirit of St. Louis* had no generating system aboard at all. It had two magnetos for engine ignition and, because of weight considerations, carried no radio. The first radio transmission from

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an aircraft had taken place, 16 years earlier, in 1911, but even then a magneto rather than a battery or generator powered the device. It was soon thereafter, in the mid-1910s, that batteries became the preferred means of supplying electrical power. This situation continued in many aircraft until the beginning of World War II. The radio batteries were simply recharged after each flight. As early as 1917, a few radios were powered by airscrew-driven generators but not by generators linked directly to the aircraft engines. Within two decades even that situation was to change. As airspeeds increased, the increased drag from airscrew-driven generators became unacceptable, and so in the early 1930s generators driven directly by the aircraft engines began to be introduced.

Early Systems

World War I moved airplanes from sporting machines to fighting machines, and with that transition came the need for power for rudimentary instrumentation. At first, it was only lighting for the instrument panel and later the radio. These first requirements were met with the aid of dry batteries alone, but that situation was destined to change.

The start of our modern electrical power systems came about not as the result of a forward thinking, but rather because of contractors' inability to meet delivery constraints. The manufacturers of the Liberty engine used in World War I were not able to obtain appropriate magnetos for the ignition systems and so were forced to use batteries. The lead-acid batteries were kept charged by a generator driven by the engine, and the aircraft electrical system was born. Once magnetos became again available, they were employed to power the ignition system, and the generator-battery system was used for the lights and instruments that were becoming part of the airplane. The automobile 6-V dc power system became the standard for the day. As the power requirements increased, 12-V systems were adopted, the change driven by weight constraints placed on the distribution bus. An increase in power at constant voltage demands an increase in current and so more copper in the wire. Because each time the voltage doubled, the weight of the wiring halved, it was not long before the 28-V dc generators (Fig. 1) were adopted to power motors and a variety of equipment operating between 18 and 32 V. Normally, the voltage regulators were set to hold 28.5 V on the bus. This system was the standard through much of World War II, although a few aircraft began the migration to alternate power sources −120 V dc and ac systems operating at frequencies from 60–2400 Hz.

In 1917 the U.S. Army Signal Corps worked with Bell Laboratories to install radios in several military aircraft. The radios were powered with batteries that were kept charged by generators, powered by either airscrews mounted in the slipstream or by direct coupling to the aircraft engine. Although most of these systems were dc, one early ac system did appear; a 900-Hz generator, which saw only limited use, was developed for powering a spark transmitter.

In the early 1920s regularly scheduled mail service was the impetus for further electrically powered equipment being added to the increasingly sophisticated airplane. Controls, instrument lighting, landing lights, electric starters, and radios all required power, and so engine-driven generators were needed to keep the batteries, still the primary electrical power source, properly charged. One of the earliest (∼1920) generators to see service was a constant-current 250-W device that was inadequate almost the day it came into service. Here we see one of the first examples of a recurring theme—no matter the capacity of the power source, demands will be greater. Higher-power generators, lighter batteries, and better voltage regulators became the order of the day. There were not many alternatives to the standard lead-acid battery, but improvements were steady in the areas of generation and voltage control.

Much work was done to increase the reliability and reduce the weight of generators. The first self-excited dc aircraft generators were simple devices. The voltage is collected from the armature windings through brushes that made contact with a rotating commutator. The output was directed to the appropriate load as well as supplying the shunt-field excitation. In this configuration the magnitude of the voltage depended on the field strength (which was proportional to the current passing through the shunt field) as well as the rotational speed of the armature.

These early generators started with those that could be easily adapted from the automobile use. The 28-V systems, common to the day, were not small, even by today's standards. Table 1 compares

Table 1 Characteristics of early 28-V dc generators

Continuous rated current, amperes	200	300	400
Operating speed			
Base, rpm	2200	4550	4100
Maximum, rpm	4500	8000	8000
Weight			
Total, lb	48	47	60
Power density, lb per kW	8.0	5.3	5.0

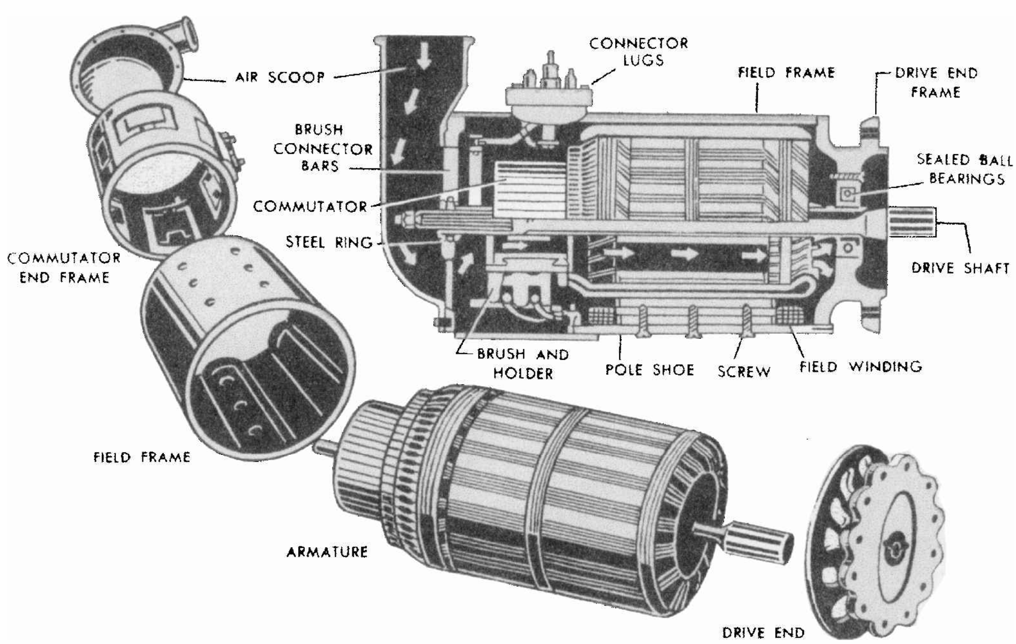

Fig. 1 Early high-output 28-V generator, which was to dominate aircraft power generation systems for almost a quarter of a decade.

several of these generators that started to appear in the period between World Wars I and II and stayed in use through the 1940s.

Generally these generators were self-excited designs whose operation at startup depended on the presence of a small amount of flux produced by residual magnetism in the pole pieces. The generators were designed with as large a number of armature windings as possible because voltage output increased and ripple decreased with an increasing number of windings.

Three general types of dc generators were available—series, shunt, and compound wound designs—the difference among them caused by the relation of the field windings to the external load circuit. Series generators, those in which the field winding was simply in series with the external circuit, were not favored at all because of the inability to regulate the voltage with even slowly changing loads. As the current to the external circuit increased, the current increased in the field windings, and so the output voltage increased. The opposite occurred, of course, when the load current decreased.

Shunt generators, those in which the field winding was in parallel with the external circuit, also had restrictions, namely, their inability to maintain a constant voltage in the face of rapidly fluctuating loads. Here, an increase in the load current was related to a corresponding increase in the current through the armature winding. Because there was some resistance in the armature, the Ohmic loss in the armature was reflected in a decrease in the terminal voltage. As a consequence of the current in the field coil following the terminal voltage (the field coils are in effect in parallel with the armature), a weaker field further reduced the terminal voltage.

Although compound generators, those designed with both a shunt and a series field, tend to behave much like a series generator, there is a sufficient enough loss in the generator caused by Ohmic losses that the terminal voltage varies less than with either of the other two, and hence came to be a favorite for aircraft use.

All dc generators (and indeed ac generators as well) faced two issues critical to proper operation: the need for voltage regulation and, in multiengine aircraft, a way to balance the loads across the generators.

Voltage regulation, the ability to maintain a constant voltage supply from the generator in the face of varying rotational speeds and loads, was addressed early in the 1920s. Although much experience in voltage regulation was certainly present from the operation of terrestrial systems, aircraft generator regulation and control presented some new problems because of the environment in which the system was to perform. Rapidly changing temperatures, vibration, shock, uneven cooling, changing pressures, and quick acceleration, as examples, were key environmental conditions that affected both the lifetime and the accuracy of the regulators. But, the problems were identified and, with time, successfully addressed, and a regulation accuracy of $\pm 2\%$ for the full range of the generator load and speed was typical for even the early systems.

Among the factors that determine the voltage output of a generator, only the strength of the field current, as shown in Fig. 2, could be easily controlled. Several methods appropriate to the aircraft environment were explored, and all shared a common goal (changing the value of the variable resistor in series with the shunt field), but each in its own ingenious way.

One of the first methods used was the Tirrill regulator (a vibrating contact device that was a predecessor of transistor/pulse-width modulation devices), common in large electrical power generation facilities, but not yet adapted by the automobile industry. Because of its construction (a rapidly opening and closing switch), it was (electrically) very noisy and so, interfered with radio operation. Because it was the only regulator readily available at the time, it was used for a while, but because the radio was perhaps the principal load for the electrical system, this incompatibility forced a look at other options. The regulator system that emerged was the carbon-pile regulator that saw service into the 1950s.

The carbon-pile voltage regulator used a stack of carbon disks that were packaged and shock mounted as shown in Fig. 3. The resistance of the stack varied inversely with the pressure applied to compress the disks—an increase in the pressure squeezed the stack and in doing so decreased the resistance. If the pressure were relaxed, the stack expanded slightly, and the resistance increased because the disks were no longer in firm contact. The stack was designed so that the pressure depended on two opposing forces: a spring that compressed the stack and an electromagnet, which allowed the stack to relax, as shown in Fig. 4. In effect, the carbon pile was used as a variable field resistor: if the voltage of the generator rose, the pull on the stack electromagnet increased, the resistance of the pile increased, and because this resistance was in series with the field windings, there was a drop in the field strength, and the generator voltage dropped. The carbon-pile regulator became the standard for many years and remains in limited use today even though more precise and reliable methods of voltage regulation have been developed.

Multiengine aircraft were beginning to appear with increasing regularity about this time and with them the need to devise methods of operating, in parallel, the generators driven by each engine. Parallel operation was not mandatory, but did offer a number of advantages. Parallel generators provided greater electric motor-starting

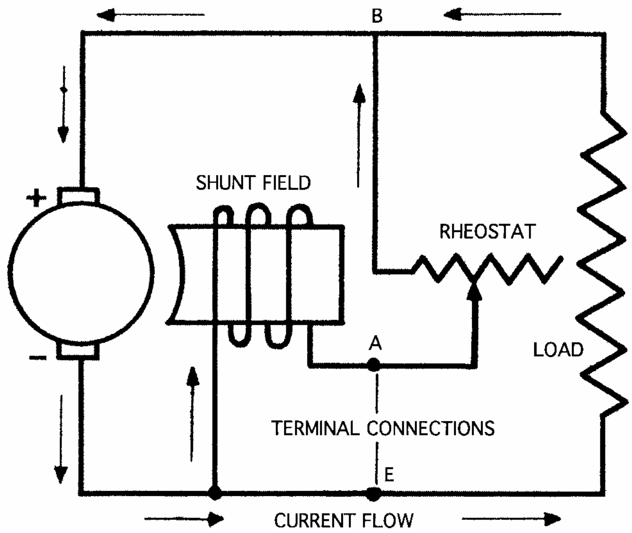

Fig. 2 Regulation of the generator voltage by a field rheostat, considered the most direct way of controlling the system voltage.

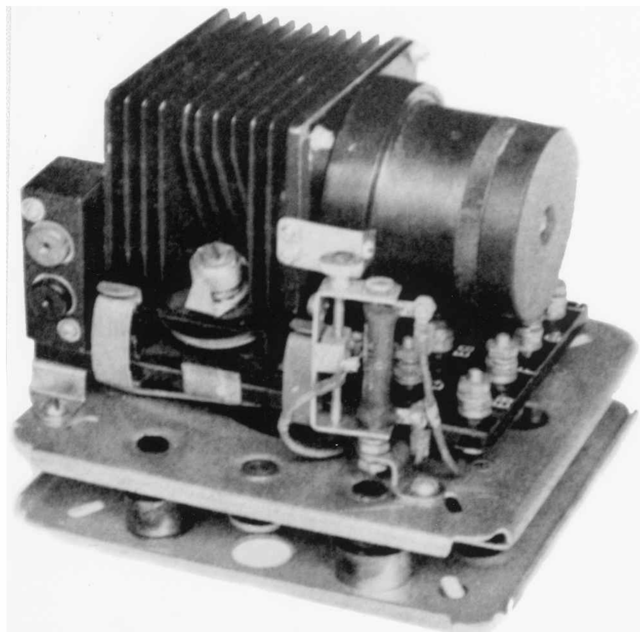

Fig. 3 Carbon-pile voltage regulator from the 1940s.

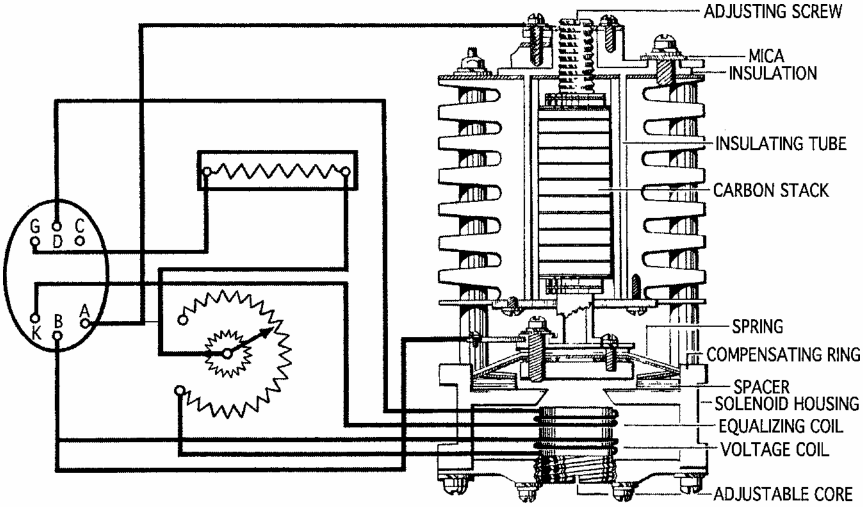

Fig. 4 28-V voltage regulator showing the internal circuits.

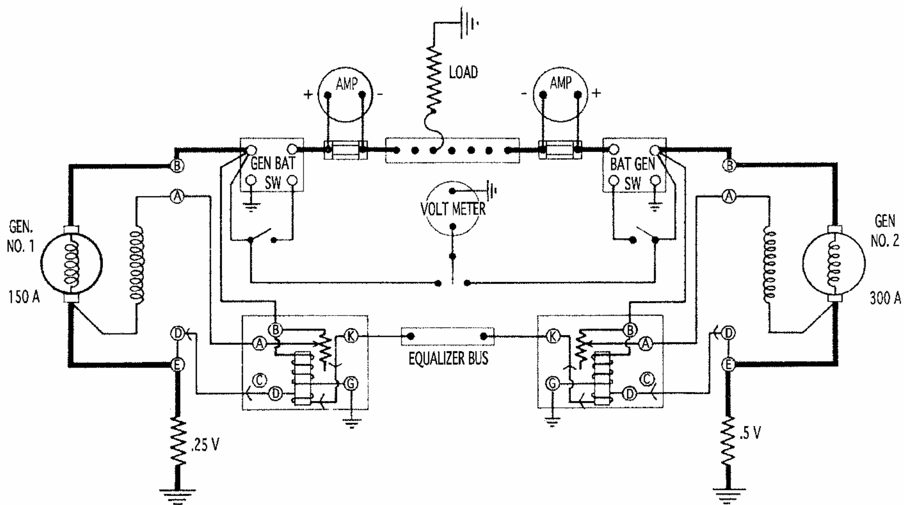

Fig. 5 Twin-engine current equalizer circuit from the 1940s.

and fault-clearing capacity and, perhaps most important, a critical degree of redundancy in the face of engine or generator failure or the appearance of faults along the distribution path. These advantages came with a price: an increase in the weight and complexity of the control system, the need for heavier switches to accommodate the larger currents that would accompany parallel operation, and the need for load balancing. Load balancing required that the current from each of the generators be the same. It was reasonable to look to the voltage regulators as a natural means of maintaining equal currents from individual generators. This ability to distribute the load equally among the generators operating in parallel was accomplished using one of several methods, each invariably linked to the operation of the voltage regulator.

One such representative method is shown in Fig. 5, which depicts a twin-engine equalizer circuit schematically. Note that there are two windings on the voltage-regulator solenoid—one for voltage control and the other related to current balance. For clarity, the carbon-pile regulator circuit for each generator is also shown. In this example, generator 1, on the left, is providing 150 A while generator 2, on the right, is providing 300 A. This current imbalance must be corrected and was done so through the use of the calibrated resistor shown between each generator and ground. In the case of generator 1, the 150 A passing through that resistor produced a 0.25-V voltage drop, while for the same value of resistance a 0.50-V drop was seen for generator 2. This meant that point E for generator 1 was at a lower voltage than the corresponding point E for generator 2, with the result that current would flow from point E of generator 2 to point E of generator 1. In doing so the current would pass through both carbon-pile regulator solenoids, but in doing so would aid the voltage coil in regulator 2 and oppose that in regulator 1. The result is that the voltage from generator 2 decreased and that from generator 1 increased to the point that the currents from each were balanced.

Although there were some advances in electrical power generation through the period up to the early 1930s, primarily through the initiative of the airline companies, most of the technology investments in aviation remained focused on the aerodynamics of aircraft structures and on the engines. The Boeing 247, first produced for passenger service in 1931, added electrically operated, retractable landing gear. The 247 carried 10 passengers at 160 mph (257 km/h), and with its appearance all other transports were made obsolete. Because aircraft of the era seldom flew at altitudes higher than 8000 ft (2450 m), there was no large power requirement for passenger comfort. As a reference point, two 50-A, 15-V generators provided ample power for the 247. Change was on the horizon.

Starting in the 1930s, with the growth of passenger service, the demands for electrical power continued to rise, driven at first by the need for increased passenger comfort. This rapid growth is shown in Table 2. Electrical loads such as reading lights, ventilation systems, and improved radios were the primary drivers.

Table 2 Growth of early commercial passenger service and its impact on power

Year	Passengers flown, M	Total miles flown, M	Average available electric power, kW
1925	Nil	——	——
1930	2	10	2
1935	10	60	7
1940	30	150	16

The period leading up to World War II was an age of discovery. The Ford trimotor airplane, a rugged commercial aircraft of the mid-1920s, carried no generating system at all. The DC-3 aircraft made its commercial appearance inn the mid-1930s and boasted 21 seats, cantilever wings, all-metal construction, two cowled Wright SGR-1820 1000-hp radial engines, retractable landing gear, trailing-edge flaps, automatic pilot, and two sets of instruments. These first DC-3s had two 50-A, 12-V generators and two 88-A-h, 12-V batteries to provide the power for 64 lights, 30 instruments, and a dozen motors and starters. This was also the beginning of the transition of functions from hydraulic to electrical operation.

Impact of World War II

The demands of World War II provided an even stronger impetus to the evolution of electrical power technology. The four-engine, long-range bombers that strategists were envisioning led the airplane builders back to the basics with a reevaluation of the purpose and requirements of an electric power system.

The electric power system should be designed to generate and distribute a variety of conditioned power to loads throughout the aircraft with maximum reliability, fault tolerance, minimum weight and space impact, and ease of maintainability. The system must not interfere with the operation of any other functions onboard and should be designed with component lifetimes compatible with the overall design philosophy of the particular aircraft. Finally, the electrical system needed to be integrated into the overall design with as much transparency as possible.

In the early 1940s there was a pressing need for improvements in efficiency throughout the aircraft, and electrical power was no exception. For an aircraft of the day designed to cruise for 20 h, about a half-pound of fuel was required for each pound of fixed weight, and, with existing engines, this half-pound of fuel corresponded to 1 hp-hour. With an assumed overall electrical system efficiency of 50% (reflected in generator, motor, and distribution losses), that corresponded to a need for 20 lb of fuel to drive a 1-hp motor for the flight. A 10% increase in electrical efficiency would yield a 3.3-lb savings in fuel weight. If that improvement in efficiency could be made with less than a 3.3-lb increase in weight, an overall improvement in aircraft efficiency could be realized.

The possibilities for powering generators were also studied. In modern aircraft it is common to drive the generator through an accessory drive pad from the main engines, but other possibilities needed to be considered, from coupling the generator to the main engine through a flexible drive shaft to using exhaust gas turbines or auxiliary gasoline engines. Although today we might think that all electrical power comes from the main aircraft engines, in commercial airliners there are multiple generators that are driven by redundant means to guarantee electrical power under the most difficult circumstances.

In the late 1930s most military airplane design experience was for fighter aircraft that operated at lower altitudes and were small by comparison to the large bomber designs being considered. Early on, the aircraft engineers recognized that the lengths of heavy wiring needed to enable a 12-V dc system would be much too heavy for bomber use. Many different dc voltages were considered, and in the end the simplicity of going from 12–24 V carried the day. At the start of World War II, most of the larger aircraft relied on a 24-V electrical system. This was not universal however, and dc systems at 120 V were designed for some large military aircraft. One such class of aircraft was the cargo plane that needed the higher voltages to operate powerful hoists used during loading and unloading. The first, in 1939, was the Martin-built, U.S. Navy flying boat, a long-range cargo craft. The first aircraft for which the loads were specifically redesigned for 120-V dc operation was the Hughes H-4, again a flying boat. Three other aircraft, the Lockheed R60-1, the Northrop F89, and the Boeing B54 also used 120-V dc systems. However, the successful use of ac systems presented a strong alternative to higher-voltage dc use, and the extensive operational experience with ac was to lead to a decreasing interest in these dc systems.

The first of over 18,000 Consolidated B-24 aircraft first flew in 1939 and is generally acknowledged as the first aircraft to have an integrated electrical system. There were two 28-V, 200-A generators driven from the main engines, a motor-generator set each for 26-V and 115-V ac power, a gasoline-powered 28-V, 2000-W auxiliary power generator, plus batteries.

The most formidable aircraft to come out of World War II was the B-29, which generated over 50 kW of 24-V electrical power. It was, by the standards of the day, a huge machine, 99-ft long and with a wingspan of 141 ft. It had 1736 ft^2 of wing area that increased by more than 20% with retractable flaps.

Although too late to see action in the World War II, the Convair B-36, which first appeared in 1946, dwarfed the B-29 in almost every category. It generated over 120 kW of electrical power within its 162-ft-long fuselage. By comparison, the B-36 had almost double the wing span and triple the wing area of the B-29, not to mention more than twice the electrical power.

By this point the limits of the 24-V system had been reached, if not exceeded. At 24 V, a 120-kW system would need to transport and switch 5000 A. The weight of the conductors needed was prohibitive. A higher dc voltage, 120 V, was considered but rejected because of concerns related to arc interruption, machine commutation, and the difficulty of creating a variety of voltages. This led the aircraft electrical engineers of the day to consider a drastic alternative.

The U.S. Navy had been using another system, an 800-Hz, 120-V, single-phase, ac electrical system for its radios. The system was a solid performer for radio applications, but there was worry about its application to other aircraft systems. For example, would the necessity of going to capacitor-start motors or to a larger number of poles to obtain high motor speeds present difficulties? Alternating-current systems had captured the interest of the community, but what was the proper frequency?

At issue also was whether a constant-frequency or variable-frequency system was better. The constant-frequency option offered a number of advantages but required some type of variable-ratio transmission if the generator were to be operated directly from the aircraft engine. This was because engine speeds are not constant. In the case of the B-36, the engine speed range was almost 4:1 in takeoff vs cruise. If a constant-speed drive were needed, it would add weight and complexity to the electrical system design, but there was also the possibility of a variable-frequency system in which the frequency was proportional to engine speed. Such a system would be adequate for certain loads, but would not be acceptable for most other applications designed to operate at a fixed voltage. Although equipment such as radios and incandescent lights are insensitive to frequency because they either operate at any frequency or rectify the input ac to meet the specific need, other devices, such as induction motors, are quite sensitive to frequency and would require significant redesign to allow them to operate with a variable-frequency source.

There were other issues to be investigated also. Perhaps 800 Hz was the right choice for an ac system after all, but a competing design based on a 400-Hz, three-phase, 115/200-V system was suggested, and contracts were awarded for both prototypes. The 800-Hz system was put into a Boeing aircraft, the XB-15, and the 400-Hz system was installed in a Douglas aircraft, the XB-19. Two far-reaching decisions came from those experiments: first, the wisdom of an ac power system, and the need to have the generators driven directly from the aircraft main engines rather than from an auxiliary power source. Both experimental aircraft had experienced considerable difficulties with auxiliary engines used to power the generators.

Modern auxillary engines (called auxiliary power units, or APU) operate at constant speed, which facilitates the use of a constant-frequency converter.

But, to return to the 400-Hz decision, a standard had to be selected, and it was important to achieve some consensus regarding the frequency. A committee was formed with representatives from the U.S. Army Air Corps, the Bureau of Aeronautics, the British Air Ministry, and the electrical industry. The committee considered a number of options, including 360 Hz, because it was a multiple of the 60-Hz power in common use, in addition to 50, 60, and 800 Hz, and even higher frequencies. The higher frequencies were not pursued, presumably because skin effects above 400 Hz were not well understood. (The power ratings at this time were sufficiently low so that feeder sizing was minimally affected by skin effects. Such is not the case in modern systems where generating sources can exceed 180 kV-A. At this level skin effects are very important, and often parallel feeders are used.) The decision was made to adopt 400-Hz, three-phase, 115/200-V ac because it was that voltage that was seen as high enough to transmit high power over appropriate distances but low enough that it would present no unusual difficulties with

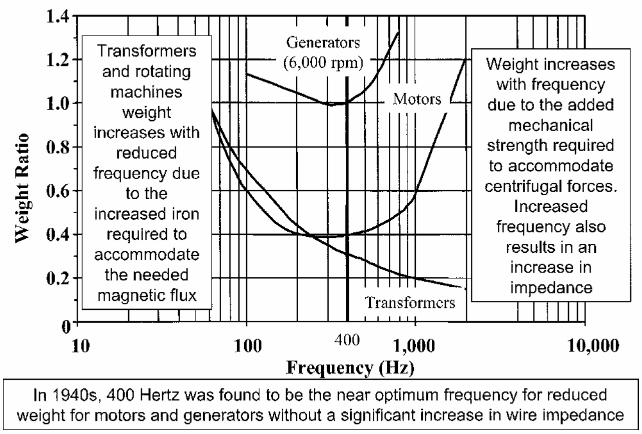

Fig. 6 Weight ratios for motors and generators plotted against frequency was part of the argument which to 400-Hz ac for aircraft. (Reproduced with permission of Boeing.)

load switching, corona at altitude, personnel hazard, or fault clearing. There was also the legacy of 12,000 rpm and 24,000 rpm motors that needed to be retained. The frequency should be high enough to drive motors at this speed without the need for motor poles. Recall that the frequency of the alternator (or motor) is the product of the number of pole pairs times the rotational speed divided by 60. The move to ac systems was further prompted by the nature of the high-power loads that were coming into use, most notably, the radar.

Some insight into the arguments leading to 400 Hz can be gained from Fig. 6. In plots of weight ratios vs frequency for generators operating at 6000 rpm, for motors, and for transformers, one can see a general, though broad, weight minimum around 400 Hz. The minimum is created because at lower frequencies transformers and rotating machines need more iron to accommodate the required magnetic flux, whereas at higher frequencies, weight increased because of the need for additional mechanical strength because of the higher rotational speeds. Increased frequency also creates larger impedances.

The issue of single phase or multiphase ac was also addressed. Although it was true that single phase offers greater simplicity in wiring and switching configurations, single-phase motors required either capacitors or phase splitting to generate large starting torque. Three-phase ac systems offered the advantage of providing single-phase ac with no difficulty, using the winding space within the generator more effectively, and providing higher torque when powering motors. It can also be rectified to provide a variety of dc voltages with less ripple than single phase.

Figure 7 summarizes many of the technological milestones in the development of aircraft electrical power through the end of World War II.

Modern Systems

And so the stage was set for the new era of aircraft electrical power. The demand for electricity had continued to grow, the basic constraints limiting the generation and distribution of electricity at altitude were understood, the philosophical guidelines for the generation of power had been fixed, and the technologies to allow for greater use of electrical loads had arrived. The aircraft electric generation capacity curves shown in Fig. 8 also depict clearly the divergence of the lower voltage dc systems from the higher voltage ac systems that were mandated by the power levels that

ERA	Primitive/ Developmental	WWI	Transcontinental Air Mail Service	Commercial Air Lines	WWII	Late WWII Post WWII Jet Age
Engines	• Piston	• Piston	• Piston	•Multi-Engine Piston –DC3 –4 Engine	• Multi-Engine Piston	• Piston to Jet
Electrical Loads	• Engine Ignition	• Radios • Lights • Ignition	• Radios • Lights • Starters	• Lights • Instruments • Motors • Starters • Radio • Navigation	Same plus • Armaments	Same plus • Radar • Navigation *B36 Era begins
Electrical Supplies	• Magnetos	• Batteries • Wind-driven generators • Engine-driven generators • Magnetos	• 12 volt • Air screw engine-driven generators. • Lead-acid batteries	• 12 volt engine-driven generators	• 24 volt engine-driven DC generators, regulated	• 400 Hz, 115/200, 3ϕ engine-driven constant speed
Electrical Systems	None	← Generally automotive types Also Tirrel VR →		• Carbon-pile VR • Assorted controls	• Carbon pile Improved controls • Start of A/C electrical systems engineering	• Carbon pile to static VR • Coordinated controls E-M to static • CSD
	1903	1917	1924 (1927 Lindbergh)	1934	1939	1946 → now

Fig. 7 Early evolution of aircraft electrical systems. (Reproduced with permission of Hamilton Sundstrand.)

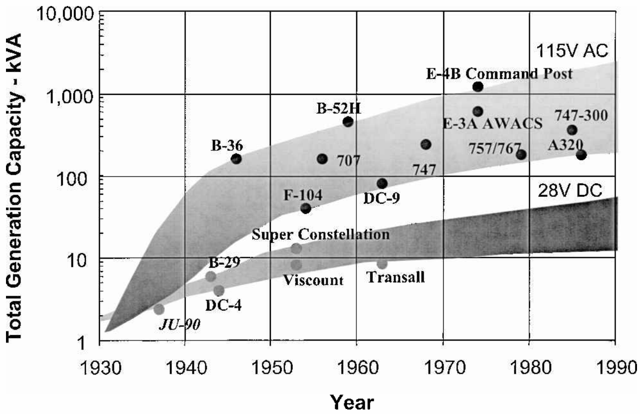

Fig. 8 Generating capacity for aircraft electrical power has continued to grow for the past 75 years.

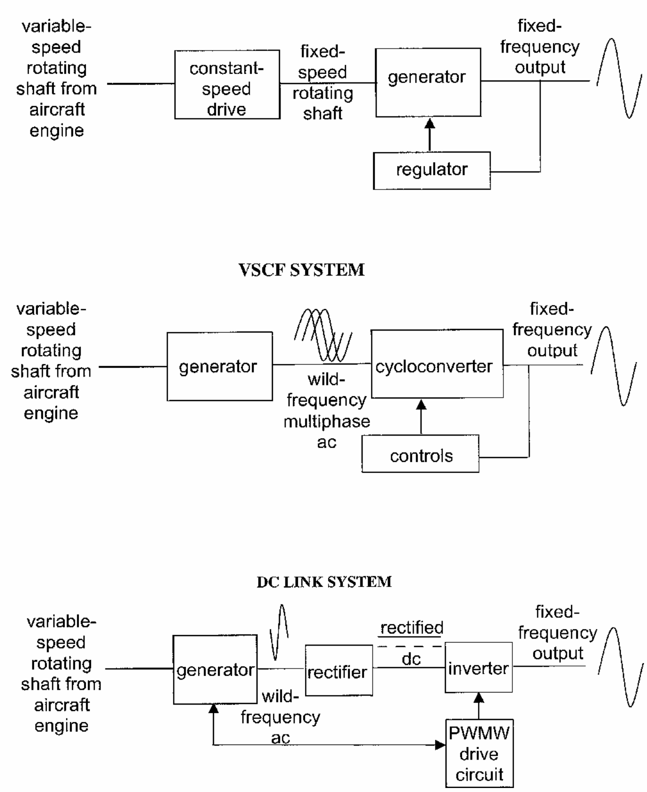

Fig. 9 Comparison of three primary methods of generating ac power on aircraft.

were required. The longstanding 28-V dc systems of the past were not adequate to meet the needs of the larger commercial aircraft or of the military customer. The historical impact of the B-36 is clearly evident in the figure. Not only was it the first aircraft to employ ac in a major way, but also moved the power-generation levels in aircraft more than an order of magnitude. The power levels available on the B-36 in the mid-1940s are equivalent to that seen today on many of the Boeing 700 series aircraft as well as the Airbus 320.

Most modern aircraft, both commercial and military, have come to rely on 400-Hz, three-phase, 115/200-V ac as the preferred electrical power source. One notable exception is the U.S. Air Force F-22, which came into service in 1997, which uses 270-V dc system (as does the U.S. Army Comanche helicopter), the decision having been based on weight considerations. The F-22 generator is also a six-phase machine rather than the more common three phase.

A synchronous generator, whose operation is based on rotation of a magnetic field, is at the heart of most modern aircraft electrical power systems. These generators can be built with any number of pole pairs by which the magnetic field will be made to rotate. The frequency of the voltage that is generated given by the number of pole pairs times the rotational speed (rpm) divided by 60. Thus, a four-pole generator (two pairs) rotating at 12,000 rpm will generate 400 Hz ac. The generator is synchronous in that the rotor will rotate in step with the movement of the magnetic field around the poles. Because the frequency of the ac voltage is proportional to the rotational speed of the generator, some methods of accommodating variable engine speed is needed when dealing with fixed-frequency systems.

Three primary methods are used today to generate the fixed-frequency, 400-Hz power for aircraft. The three are not at all equally used, however. Two of the three are dependent on power semiconductor electronics for both electronic switching and electronic power regulation. Each method also depends on a generator that is driven by the aircraft engine. But that is where the similarities stop. Two generate "wild frequencies" later converted through modern electronics to a fixed frequency, and the third produces a fixed frequency directly from the generator. They are compared in simple block form in Fig. 9. The upper diagram in the figure shows the principal components of the constant-speed drive system that provides a fixed rotational speed for the generator regardless of the actual main aircraft engine speed. The center schematic is that of a cycloconverter whose operation is based on the ability of fast electronics to synthesize a fixed-frequency ac voltage from a multiphase, wild-frequency output of a completely speed-unregulated generator. The bottom part of the figure is the dc link system that avoids the frequency issue by converting the generator output immediately to dc and then recreating a fixed-frequency ac voltage through an inverter. In all of these cases, it is important to remember that it is the generator which actually converts engine mechanical power to electrical power. The associated parts [e.g., constant speed drive (CSD), cycloconverter, dc link] that are discussed are present in the system simply to guarantee a fixed frequency of the ac. As we mentioned earlier, in some applications, there might be no need for a fixed frequency. Galley heaters, deicing equipment, incandescent lighting, avionics, and in-flight entertainment centers are examples of frequency-insensitive loads. In these cases a wild-frequency generator would suffice. For some aircraft electric motors it would be necessary to use motor controllers to convert the variable ac to a suitable frequency. Despite the advantages of tailoring motor operation to aircraft needs and even allowing soft-start control to minimize the impact of motor start on the system, these controllers would add weight and cost, and so their use would be avoided if possible. For some small fraction of aircraft equipment that requires 400 Hz, the need for conversion systems could not be avoided, and so some frequency conversion subsystem would be mandated. For commercial transport aircraft the required 400-Hz power might be as little as 5% of the total aircraft generating capacity.

The CSD system has been the workhorse of commercial aviation since it was introduced by Sundstrand in the B-36 long-range bomber over 50 years ago. The CSD is the most popular method of generating the power, and the list of aircraft in which it operates is extensive: the Boeing 707/717/727/737/747/757/767/777, Douglas DC8/9/10, McDonnell-Douglas MD11/80/88, Airbus 300/310/320/319/321/330/340, and the Lockheed 1011, among others. The CSD does not itself generate any power, but rather serves as a compliant layer between the engine and the generator to guarantee a constant rotational speed of the generator's rotor (Fig. 10). The CSD, also known as a variable displacement hydraulic transmission, is a hydraulic-mechanical device that is typically mounted on an aircraft-mounted accessory drive or gearbox and so is driven directly from the aircraft engine. The input shaft to the CSD then rotates at a speed determined by the engine rpm, and as variations appear in engine speed so do they present themselves to the CSD. These variations in the engine speed can be as large as factors of 2:1 between takeoff and idle. The CSD accommodates these variations and, independent of the input rotational speed, delivers a constant output rotational speed (typically 6000, 12,000, or even 24,000 rpm) to a synchronous generator. Thus, the generator itself is shielded from the variations in engine speed by the CSD. Because the CSD had to be mated with a generator to produce power, in recent years the CSD has been packaged with the generator into a single unit, the integrated drive generator (IDG) (Fig. 11). The CSD offers an excellent example of technologies that changed to keep pace with an increasing demand for power. The original CSD on the B-36 did not contain a differential for speed summing, so that the entire load was transferred through the hydraulic unit. In later designs the hydraulic unit was used only for trimming the speed while the bulk of the power was transmitted directly through the CSD. Even though the reliability as well as power density of the CSD units have continued to improve remarkably over the last half-century (Fig. 12), interest has remained high in variable-frequency power system architectures, two of which we examine now.

The cycloconverter is a second method of generating the 115-V ac, 400-Hz, three-phase electric power from the aircraft engine's variable-speed accessory gearbox. The system consists of a lightweight generator that produces high-frequency (2 to 4 kH) ac and a conversion unit that transforms the variable-frequency output to constant 400-Hz power. At the heart is the cycloconverter that provides efficient ac-to-ac conversion. The generator is a high-speed (14,000–16,000 rpm) brushless machine with output voltage and frequency designed for optimal compatibility with the cycloconverter, a four-quadrant converter that uses two banks of silicon-controlled rectifiers (SRC) to combine the generator output into a constant 400-Hz sinusoidal output. In the absence of a gate signal, the SCR acts like a very high-resistance rectifier in both forward and reverse directions. However, when a gate signal is applied, the SCR behaves like a conventional rectifier and conducts only in the forward direction. Figure 13 shows a schematic for a three-phase, variable-speed, constant-frequency (VSCF) cycloconverter in simplified form. The circuit provides a single-phase output form a three-phase high-frequency input. The two phase-controlled rectifier banks (labeled positive and negative in the figure) alternately supply the positive and negative halves of the output current waveform, and the L/C filter attenuates the rectifier ripple. Figure 14 shows the 400-Hz waveform that results from the operation of the circuit. In a typical system a six- or nine-phase generator is used and two or three cycloconverter circuits are connected in parallel to form a single output phase, and three identical single-phase cycloconverters are used with the reference oscillators shifted by 120 deg to generate the three-phase output. These systems have been used primarily in military aircraft, including the U.S. Air Force F-117A, the U-2S, and the U.S. Navy F/A-18. This method and the dc link described next are both called VSCF systems, for reasons that are apparent.

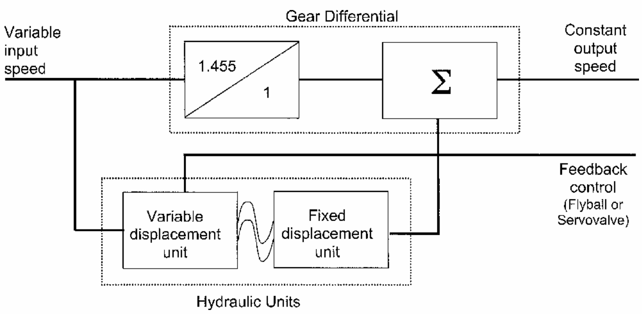

Fig. 10 Constant Speed Drive (CSD) block diagram. (Reproduced with permission of Hamilton Sundstrand.)

A second form of VSCF, the dc link, also owes it existence to the rectifier and bipolar junction transistor, although other types of power transistors such as MOSFETs and IGBTs can be used. This system rectifies the variable-frequency output of the generator, often operating at kilohertz frequencies, through a full-wave rectifier bridge to provide an intermediate dc power link. That dc power is then inverted to 400-Hz ac by means of a conventional three-phase inverter bridge, which chops the dc voltage into a pulse-width modulated output of three sine waves with harmonics. This output is then filtered to minimize the harmonics. A simplified schematic of this power inverter is seen in Fig. 15.

Variable-frequency technology is being used on a variety of aircraft, including the Bombardier Global Express and Dash 8-400, and commuter aircraft such as the ATR 42/72, the Bae ATP, the Saab 340 and 2000, and the Dornier 328 prop and jet.

A difficulty with variable-frequency systems might lie in the efficiency across the frequency range it must operate. If a generator is designed to provide a fixed power level over a range of frequencies, the design point must be at the lowest frequency. Generators operate by cutting flux lines, and if the rotation rate is lowered the number of flux lines must be increased—a tradeoff that translates into more weight (iron) in the generator than would be needed at the higher frequency. At the higher frequency a penalty is paid to achieve higher mechanical strength than is needed for lower speed operation. These constraints offer an advantage to fixed-frequency generators in many applications, an advantage that must be balanced by the higher overall reliability of the variable-frequency systems.

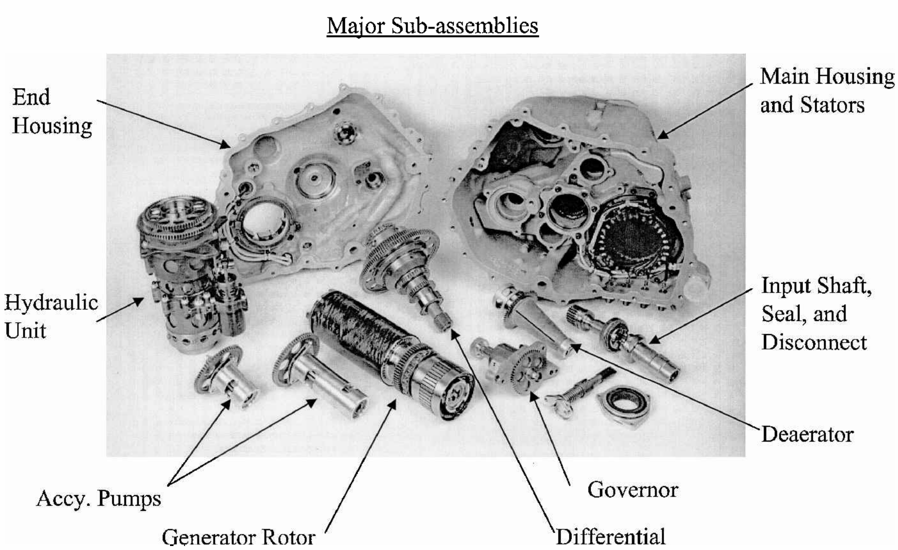

Fig. 11 Side-by-side integrated drive generator (IDG). IDG also is made in an in-line configuration. (Reproduced with permission of Hamilton Sundstrand.)

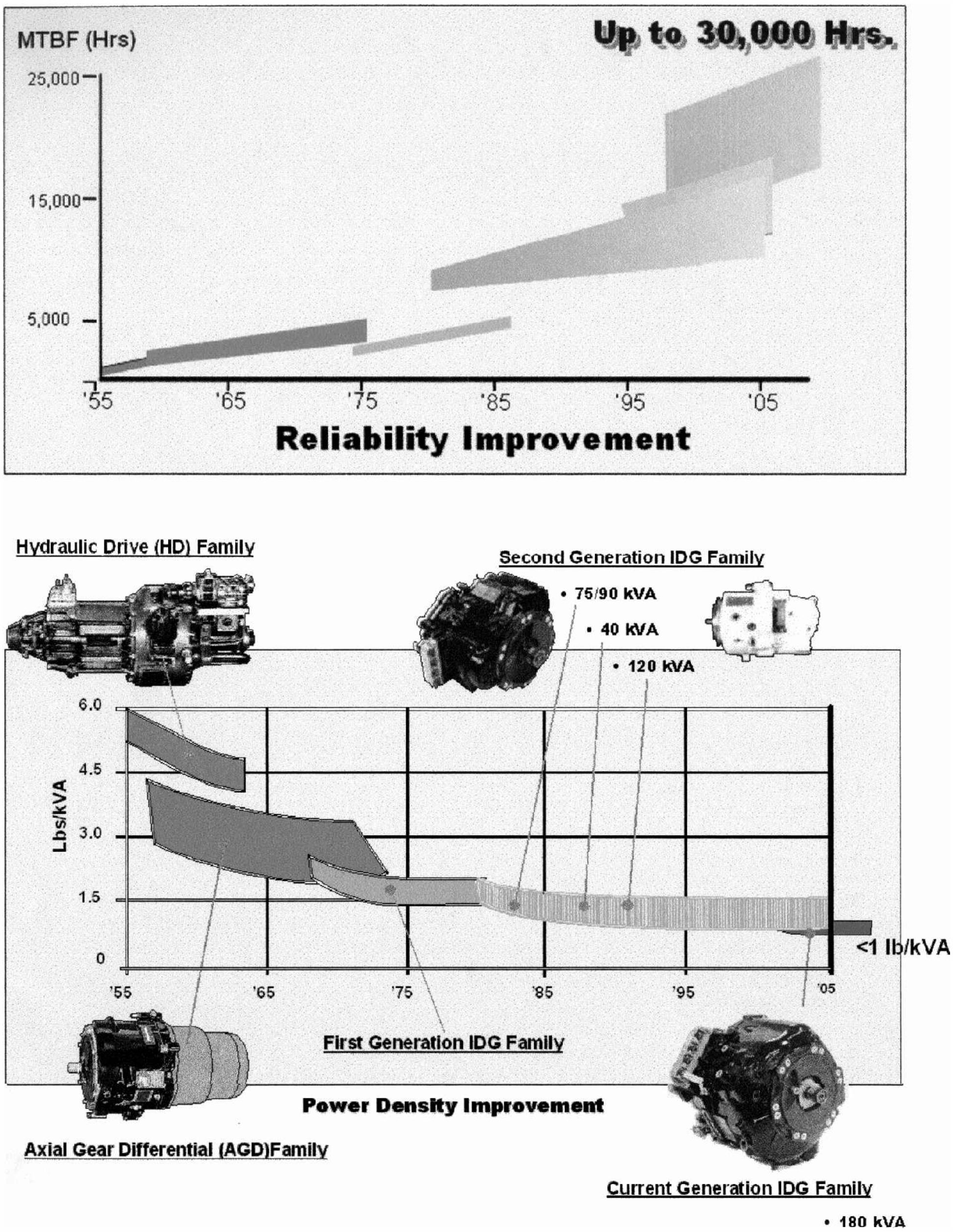

Fig. 12 Reliability and power-density improvements of CSD/IDG units over the past 50 years. (Reproduced with permission of Hamilton Sundstrand.)

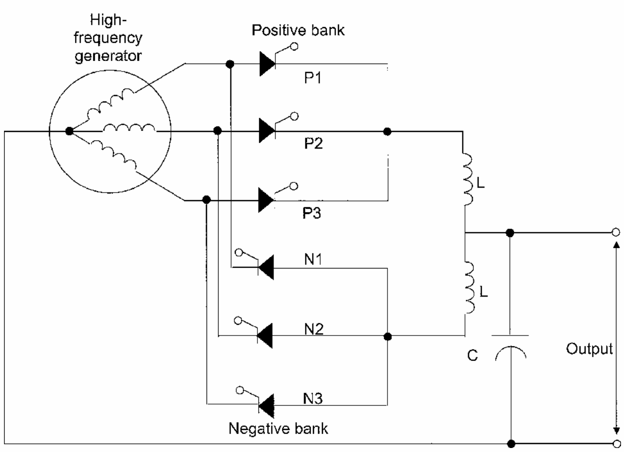

Fig. 13 Simplified schematic of the three-phase, VSCF cycloconverter. (Reproduced with permission of Smiths Aerospace.)

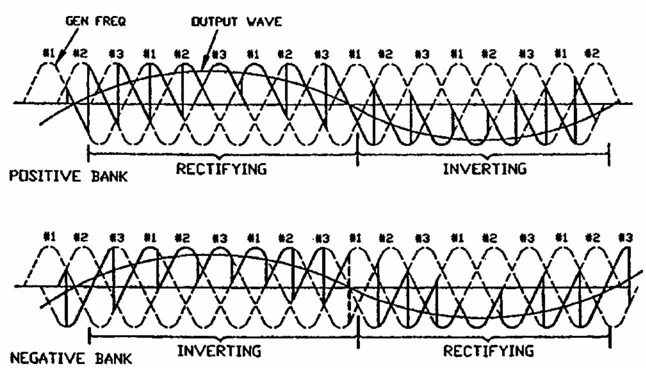

Fig. 14 Synthesized 400-Hz waveform obtained from the three-phase, variable-frequency voltage. (Reproduced with permission of Smiths Aerospace.)

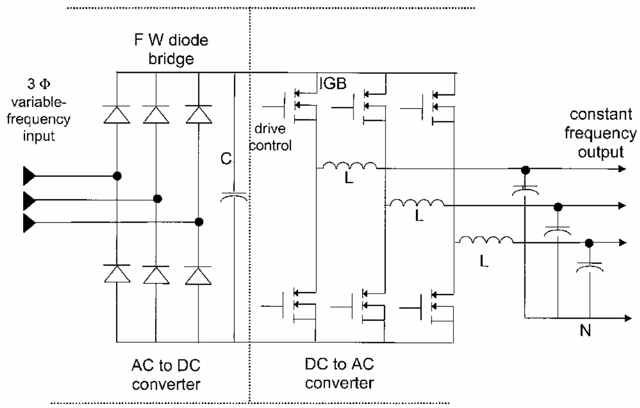

Fig. 15 Simplified schematic of the dc link inverter.

The Boeing 777 (Fig. 16) offers an insight into the complexity of a modern aircraft electrical power system. The system is a traditional hybrid of 115 V/400 Hz and 28 V dc. The power sources include two 120-kVA, 400-Hz, engine-driven generators with a CSD packaged as an IDG. There is an additional 120-kVA, 400-Hz, APU driven generator, two VSCF 20-kVA backup generators, and one 400-Hz converter, four 950-W permanent magnet generators integrated into the two backup generators, and flight-control batteries. As with many aircraft today, there is also a 7.5-kVA ram-air turbine, which is activated by dropping it into the airstream during an emergency. There are four 120-A dc transformer rectifier units to convert 115 V ac to 28 V dc. The central distribution panel (called the electrical load management system) controls distribution throughout the 777 although each of the separate generating channels (the 120 kVA IDGs and the 120 KVA APU) each have their own control and protection. Figure 17 shows the IDG installed on a main engine. Modern aircraft, with almost a megawatt

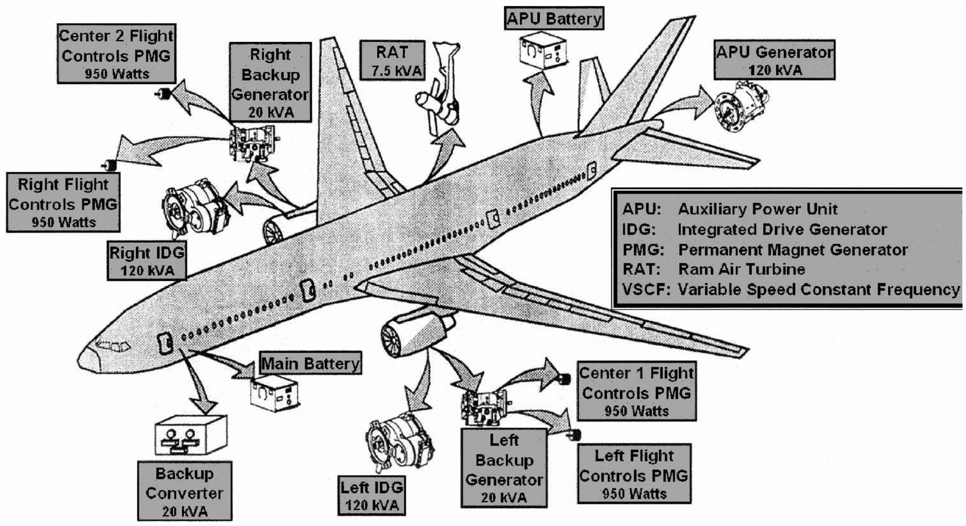

Fig. 16 Electrical system layout for the Boeing 777. (Reproduced with permission of Boeing.)

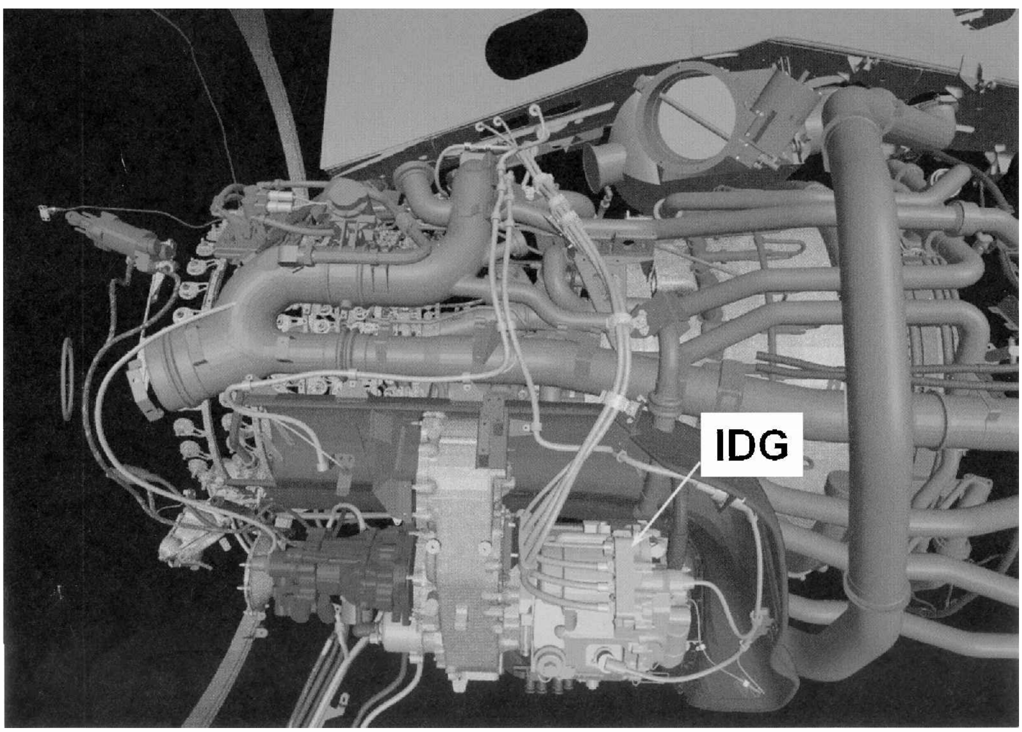

Fig. 17 Artist's view of the IDG mounted to a main engine. (Reproduced with permission of Boeing.)

Table 3 Current aircraft electrical power systems, their applications, and attributes (courtesy of Hamilton Sundstrand)

Power systems	Users	Attributes
28 V dc	General aviation Small business jets Turboprops	Low cost for starter, generator Heavy cable, high wire weight Limited to 12 kW (400 A)
270 V dc	U.S. military	Optimum input for radar and special-mission equipment Potential shock hazards Lower wire weight
Variable-frequency ac	Midsize business jets Large turboprops	Addresses the requirement for pneumatic start with narrow speed ranges Aircraft still has high dc loads, and ac is used for heating, deice
Constant-frequency ac (400 Hz)	Large business and regional jets Commercial transports Worldwide military	Optimized for ac-powered motor loads Cleanest power quality

of generating capacity and tens of miles of feeder, distribution, and control wiring, are a far cry from the simple magneto of the Wright Brothers' day.

Military aircraft typically generally use 115-V, 400-Hz, three-phase ac, and 28-V dc for power. The U.S. Air Force F-22 is unique in its choice of 270-V dc (based on a variable-frequency system), although 28-V batteries are used as backups for critical functions and for auxiliary power unit starts.

The U.S. Navy F/A-18 was the first aircraft to use a variable-speed, constant-frequency generator and in doing so eliminated the need for the constant-speed mechanical drive. The F/A-18 has two power busses driven by the two generators with automatic bus tie capability in case a generator should fail, although one generator can power the entire aircraft. This design feature might become more difficult in the future as electrical demands on the aircraft continue to increase.

The operational nature of the load has also added complexity to the modern aircraft power system. As one example, the active electronically scanned array radars are incorporated in the F-22, F/A-18E/F, and planned for the joint strike fighter, and place a stringent demand on the power interface because these very high electrical loads cycle on and off very quickly.

A summary of the various power options and their applications is presented in Table 3.

Tomorrow

There are many things about the future of aircraft electrical power that are hidden today, but some trends appear certain to continue. The demand for additional electric power will to grow as designers look to the future of more electric aircraft or (MEA). Increasingly, the functions that traditionally have been performed by hydraulics (and even earlier by cables) are migrating to electrical operation. Eliminating hydraulic lines is a way of increasing reliability and saving weight, but will require, perhaps, additional work on the design of small, high-torque motors. More sophisticated avionics will place an increasing demand on high-quality electrical power that is absolutely reliable. The expectations of passengers for more amenities driven by electricity will doubtless see the power requirement per passenger increase also.

The current blend of hydraulic, pneumatic, and electrical power for addressing both the flight loads (fuel feed and transfer, landing gear, flight control, avionics, deice, cabin environment, etc.) and the passenger loads (galleys, lights, entertainment, etc.) has been adequate. These and other loads will migrate to all-electric service in the MEA concept.

On the commercial side the projections suggested by Airbus are impressive: the current A340 aircraft generates 360 kVA of 400-Hz electrical power. The A380 will grow that number to 600 kVA using variable-frequency generators. Within perhaps a decade this number might grow to greater than one million VA as hydraulics are completely eliminated and electric actuators drive flight controls, landing gear, and brakes. Present technologies for power generation, control, and distribution might be strained as the demand for onboard power in a modern commercial aircraft continues to grow.

For the military, with no passenger considerations to accommodate, the path to an all-electric aircraft might be even shorter. Even as early as the 1940s, Douglas engineers considered replacing hydraulic with electric power in flight controls. The state of the art in power generation and conditioning did not allow for such a bold move then, but the concept remained in the military flight community, and with the advances in power conditioning made with the advent of power electronics high-energy density permanent magnets in actuators, motors, and generators and the availability of 270-V dc systems, the day of the all electric might have arrived.

The 270-V dc generator is available—think of it as the dc link system without the dc-to-ac inverter added—and is in military use. The need for higher voltages on military aircraft is clear, and because the electrical loads are compatible with these higher voltages high-voltage dc systems probably will continue to expand in the military market. Migration of 270-V dc into commercial airliners remains an issue. Although studies have been done regarding hazards related to the dc system, issues related to safety, in addition to the recurring issues of weight and efficiency, will need to be resolved.

Aircraft designers are also keen to eliminate the main engine starters because they represent weight. Research continues on designs to start the engines by using the generators (powered backwards) as motors. This work is proceeding with the 270-V dc systems as well as other variable- and constant-frequency architectures. This migration from starters and generators to integrated starter-generator sets will continue because the integrated package will certainly be lighter than the individual systems. An old technology—switched reluctance machines (SRM)—made fresh with the advent of power electronics and digital signal processing holds promise for the integrated starter/generator concept. Switched reluctance machines are actually synchronous machines but with a significant difference from conventional design. In these machines torque is produced by the tendency of the rotor to move into a position where the inductance is a minimum. As the rotor moves past this point because of its rotational inertia, the current on the stator with which it was aligned is switched off. In a motor mode the stator windings are excited as the appropriate rotor pole approaches and then switched off, and in doing so, torque is continually produced. In the generator mode the stator windings are excited as the rotor pole separates (rather than approaches), and a braking action is produced. The Lockheed–Martin Joint Strike Fighter design employed SRM. In the generate mode that design produced 160 kW at 270-V dc and can be used for engine starting. The SRM offers advantages in simplified construction, relatively inexpensive magnetic materials, fault tolerance, and seamless transition from motor to generator modes all lead to its future consideration as an MEA candidate.

Conclusion

It has been a century that has seen the development of modern aircraft with wing spans greater than the total length of the first powered flight, of onboard power generation capacity matching that of a small village of 50 years ago, of the marriage of all branches of engineering into a single disciple. Where the next 100 years will lead us is certainly unknown, but equally certain, unimaginable.

Spacecraft Electrical Power

Beginnings

The first artificial satellite, the 184-lb Sputnik I (Fig. 18), was launched on 4 October 1957 and carried a silver-zinc primary battery as its only power source. The battery provided only a single watt to power the two transmitters that ceased broadcasting three weeks later. The satellite itself remained in orbit for an additional three months, until January 1958, but because the transmitters had ceased functioning the electrical power system, not orbital mechanics, effectively defined the lifetime of the satellite. This satellite was followed soon afterwards by Vanguard I, the first satellite to carry solar cells coupled to secondary (i.e., rechargeable) batteries. The batteries were necessary to provide electrical power during periods of eclipse, an event that takes place for more than 30 min in each orbit of a satellite in LEO (low Earth orbit) and up to 75 min in GEO (geosynchronous) orbit. The solar array contained eight panels of six p/n cells each. The battery-powered transmitter on Vanguard lasted less that three weeks, but the solar array continued to provide 1 W of power for six years. These first solar panels were easy to build and use, but they suffered from low (~10%) conversion efficiency.

Since those days, the sophistication of the payloads and the corresponding demands for electrical power to make them functional have increased by many orders of magnitude. Figure 19 shows the growth in electrical power needed for specific spacecraft over the past 40 years. These power requirements have been driven in large part by GEO communications satellites, which require 10–20 kW of power.

Early on, aircraft engineering practices were adapted for spacecraft, but those practices were not sufficient to accommodate the more stringent constraints imposed by the space environment. Although mass, reliability, and cost were considerations in aircraft power systems, those systems had the advantage of a very large prime power source, the aircraft engine, for which the added burden of supplying electrical power was incidental: about 1000 lb of thrust equates to about 1 mW of electrical power on an aircraft (and, for example, each of the engines on the Boeing 777 generates about 80,000 lb of thrust). Further, flight times for aircraft were measured in hours rather than years so that maintenance and refueling were more easily accommodated.

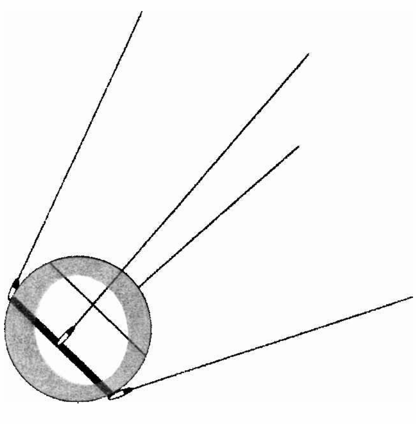

Fig. 18 Sputnik I, the world's first artificial satellite. It electrical system provided a single watt to power a transmitter.

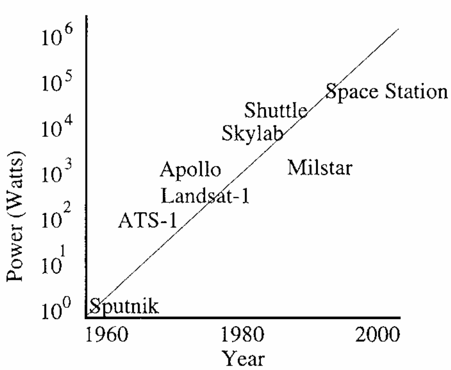

Fig. 19 Growth in demand for spacecraft electrical power over the past 40 years.[1]

Spacecraft Architectures and the Electrical Power System

Over the last 50 years spacecraft designers have succeeded in improving the lifetime, efficiency, reliability, and compactness of each of the subsystems aboard the satellite. What began as a simple design centered on a transmitter and a readily available power source has become a complex interrelationship among a number of subsystems, each requiring electrical power. The general architecture of a spacecraft is shown in Fig. 20. The mission payloads are specific to each satellite and define the overall design. The support systems tend to be functionally replicated from satellite to satellite. Continuing efforts to make the support systems modular are aimed at reducing costs. This was difficult from the power perspective because each system on the spacecraft might require electrical power at differing peak and average power levels, voltages, and duty cycles. Consider, for example, the attitude control system (ACS), which maintains the satellite pointing in the proper direction. A failure of the ACS on Galaxy IV in May 1998 caused the loss of that communications satellite carrying 90% of the electronic-pager traffic in the United States. The various ACS subsystems such as the accelerometers, sensors, and computers for data manipulation require low voltages and currents, whereas the drives and electromagnetic actuators for the solar arrays aboard the same satellite require very high peak powers.

As suggested in Fig. 21, the duration of the mission is a key factor in the selection of the prime power source. For short-duration missions chemical systems such as primary batteries, fuel cells, or chemical dynamic conversion, might be the appropriate choice, depending on the maximum power required. Primary batteries are used in meeting the high-power and high-energy demands of the launch vehicle itself as well as in the activation of pyrotechnic devices related to explosive stage separation. If the mission extends for more than a few days, the choices might be restricted to solar arrays with some storage system to accommodate eclipse or to a nuclear system. Operational issues will also influence the choice of prime sources: the survivability of solar arrays in certain orbits, the restricted maneuverability of large solar arrays, the infrared signature of nuclear systems, or compatibility with mission-related sensors are examples of issues that could eliminate options that otherwise would have been logical choices.

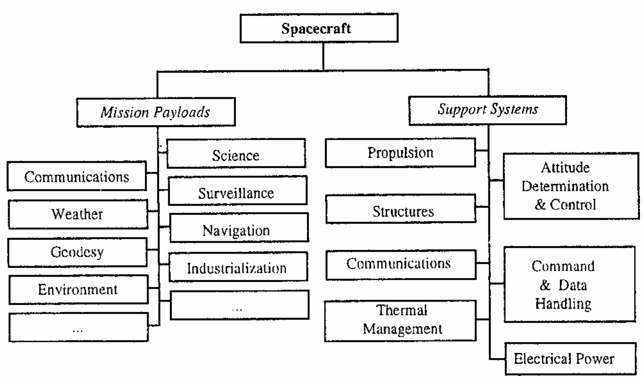

Fig. 20 General architecture of a spacecraft.[1]

The electrical power system (EPS) must do several things: it must be a continuous and reliable source of peak and average electrical power for the life of the mission; it must control, distribute, regulate, and condition the power provided to the various loads; it must be capable of providing data regarding the health and status of its operation; and it must protect itself and its loads from electrical faults anywhere within the spacecraft.

The heart of the spacecraft EPS is the prime power source, shown on the left side of the Fig. 22. Power from those sources can be stored or converted into electricity. The stored energy need not be in the form of electricity. For example, solar energy could be used to heat a thermal reservoir, or it might be converted in solar cells into electricity for charging a battery, spinning a flywheel, or regenerating a fuel cell.

Conversion of solar, chemical, or nuclear energy into electricity can occur through a variety of methods, depending often on the mission needs. For example, pointing-and-tracking requirements might demand that the conversion be done through a static process to reduce vibrations. The electricity must then be managed, regulated, monitored, and conditioned to match the electrical needs of the individual spacecraft subsystems.

The choice of three categories of prime power sources is arbitrary. Solar energy is clearly a prime source of electrical power for satellites, either through direct conversion in solar cells, or indirectly through alternative processes.

In the case of chemical prime power, batteries and fuel cells have long been considered power sources, although some would argue that they are actually energy storage devices. Other uses of chemical sources might be combustion used to drive dynamic converters such as the Brayton-cycle turbine. Fuel supply would be a critical consideration in such applications of chemical sources, of course.

For long-duration missions nuclear power is an alternative to solar. Nuclear power sources produce heat that must be then converted to electricity through either a dynamic or static conversion process.

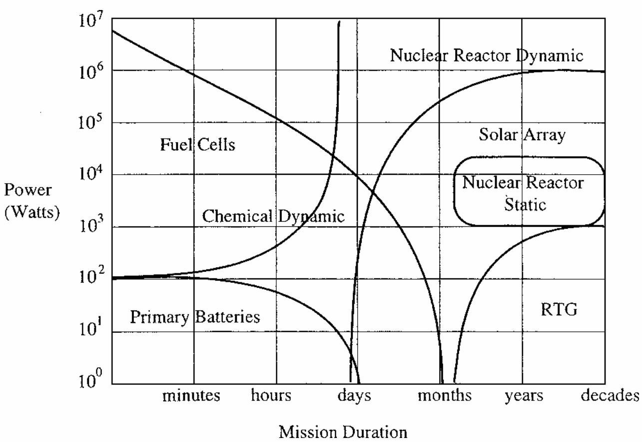

Fig. 21 Options for various prime power sources are governed by the mission duration.[1]

Significant improvements have taken place in power technologies, especially over the past two decades. A snapshot of some of these improvements is shown in Table 4, which presents a comparison of the state of the art in several key technology areas from the mid-1980s to the beginning of this decade. Historically, the solar-chemical option in the form of photovoltaic arrays and rechargeable batteries has been, by far, the most used, and so will discussed in some detail.

Solar Cells

The workhorse of space electrical power is the solar array-battery system. The solar energy available outside the atmosphere (referred to as air mass zero, AM0) is 1367W/m^2, about 40% greater than solar power seen at the surface of the Earth (air mass one, AM1) at noon on a clear day. This difference is caused by atmospheric absorption, primarily at the shorter wavelengths. Until the appearance of multiband gap solar cells, that part of the spectrum was not effectively used because earlier solar cells did not respond well to these short wavelengths. Research on space photovoltaics has been a continuing struggle to increase the efficiency of conversion and to improve the resistance of the cells to the space radiation environment while reducing the cost and mass per watt.

Solar cells convert photons directly to electricity, albeit with modest efficiencies. Becquerel first noted the photoelectric effect in 1839, but it was more than 100 years later before significant progress in the efficiency of conversion was made. As with developments in aircraft power, it was the stimulus of World War II that led to the availability of germanium and silicon semiconductor materials appropriate for solar cell applications. The early cells made by the high-temperature diffusion of p-type impurities on n-type silicon. Silicon cells now made for space applications are n-on-p devices. The reason for the change was the recognition that minority lifetimes were greater in p type when the cells were subjected to radiation environment of space.

The earlier satellites were limited in power also because of their basic design. These satellites were powered with arrays mounted on the outside of the spacecraft structure, so called body-mounted arrays. The overall size of the satellite then determined how many solar cells it could accommodate, and this directly limited the total power that could be generated. Typical early satellites were spherical or cylindrical with small panels distributed evenly over their external surface to ensure continuous power generation as the spacecraft slowly spun about its axis. The availability of larger launch vehicles allowed growth in the diameter and length of these spacecraft and so allowed this configuration to be used into the 1980s and even today in limited applications. The growth in power demands soon required the entire spacecraft body to be covered with solar array panels, and finally, in order to provide even more power the satellites were outfitted with small paddles mounted on hinged arms that swung out from the body of the spacecraft. Explorer 6, the first satellite to use a paddle array system, was launched in August 1959 and carried four 51-cm^2 hinged paddles. The paddles were oriented to provide continuous power as the spacecraft rotated. Folded and hinged rigid panel arrays became the standard configuration for spacecraft that followed. The rigid panels, ranging in thickness from

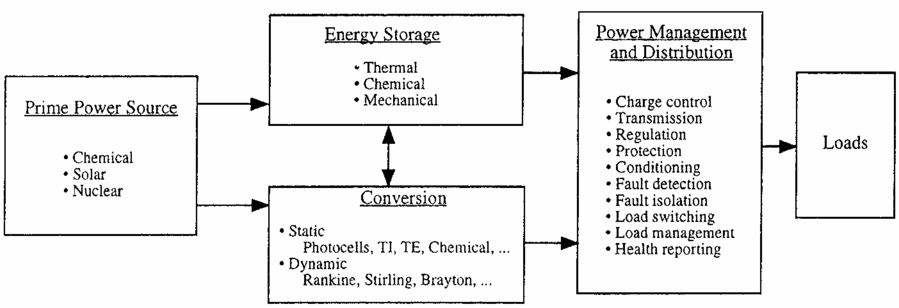

Fig. 22 Generalized overview of the electrical power system.[1]

Table 4 Evolution of space power technologies (after Ref. 1)

System or component	Parameter	Circa 1985	Estimated 2002
Solar-battery systems			
	Power output	5 kW	>100 kW
	Specific power	10 W/kg	>50 W/kg
	Array-battery costs	$3000/W	<$1000/W
Solar cells			
	Power	5 kW	>100 kW
	Efficiency	14%	>25%
Solar arrays			
	Specific power	35 W/kg	>150 W/kg
	Design life (LEO/GEO)	5yr/7yr	10yr/15yr
	Specific cost	$1500/W	<$500/W
Batteries			
Primary			
AgZn	Energy density	150 W-h/kg	——
	Design life	2 yr	——
$LiSOCl_2$	Energy density	200 W-h/kg	700 W-h/kg
	Design life	3 yr	5 yr
Secondary			
NiCd (LEO/GEO)	Energy density	10/15 W-h/kg	——
NiCd (LEO/GEO)	Design life	5yr/10yr	——
NiH_2 (LEOGEO)	Energy density	25/30 W-h/kg	——
NiH_2 (LEO/GEO)	Design life	2yr/3yr	——
Fuel Cells			
	Power load	7 kW	50 kW
	Specific power	100 W/kg	150 W/kg
	Specific cost	$40/W	$25/W
	Design life	~2000 h	4000 h
Nuclear power			
Reactors	Power level	10 kW	10 kW
	Specific power	10 W/kg	10 W/kg
	Efficiency	10%	10%
RTG	Power level	1 kW	2 kW
	Specific power	6 W/kg	10 W/kg
	Efficiency	8%	12%
Typical overall system parameters	Power	12 kW	25 kW
	Voltage	28 V	50 V
	Frequency	DC	DC/AC
	Cost on orbit	~$1000/kW-h	
	Radiator specific mass	20 kg/kW	

6 to 25 mm, were initially made of two thin sheets of aluminum glued to a honeycomb core. Later, graphite-epoxy sheets replaced the aluminum. The panels were deployed using springs to open and lock the panels in place, and once deployed these rigid arrays could not be refolded.

The International Space Station (ISS) (Fig. 23) does not use rigid panels but rather represents the first use of large, flexible blankets. This change was necessary for several reasons, all related to the large amount of electrical power needed aboard the ISS. The complexity of deploying such large rigid structures, of controlling them once on orbit, and the limited volume of the shuttle all argued against the rigid panel configuration.

Flexible arrays cannot replace rigid ones in lower power level applications because the mass and volume advantages of the flexible array do not scale well below several kilowatts. Specific power for the rigid arrays range from about 25 to 60 W/kg (depending on the type of solar cell used), as compared to an array specific power of about 40 W/kg for the ISS.

Silicon Cells

Until about 25 years ago, silicon solar cells were the only ones available for spacecraft power systems, and they have been the predominant power source in space for over 50 years. Silicon single-crystal cell efficiency reached 6% in the 1950s, and even though silicon cells and arrays have been flown since the first days of spaceflight efforts to enhance their performance are still underway.

Single-crystal silicon solar cells have demonstrated AM0 efficiencies in the laboratory of almost 22% but continued improvements in that efficiency are limited by the 1.1-eV band gap of the material. Figure 24, a plot of the variation of solar-cell efficiency with the semiconductor band gap energy, shows a maximum in efficiency at a band gap of about 1.55 eV. The maximum is understood by considering the extremes in band gaps: for a very small band gap electrons are excited from the valence band to unfilled states far above the conduction minimum. As the electrons relax to the bottom of the conduction band, they give their excess energy to the lattice as heat and efficiency suffers. For a very large band gap the energy required to move an electron from the valence band to the conduction band exceeds the photon energy available, and so again the incoming energy is deposited as heat.

Even if the efficiency of the silicon solar cell is limited, the area of the cell is not. Early silicon cells had an area of about 2 cm^2, and because a major cost of constructing an array is that of assembling and interconnecting the cells larger cell sizes offered both cost and weight advantages. For the International Space Station, an 8 $\times$ 8-cm cell, 300 μm thick, with an efficiency of greater than 14%, was developed to reduce the cost and weight of the overall assembly. But as the cells are made larger, their series resistance increases because of the increased length of the contacts needed to reach across the larger area. This increased resistance reduces conversion efficiency. Increasing the size of the contacts on the cell surface would reduce the resistance, but would mask a larger percentage of the light-absorbing surface. An ISS compromise resulted in a new, circular geometry for the contacts to replace the traditional linear design used on smaller-area cells. The ISS has the largest photovoltaic system ever sent to space. More than a quarter of a million cells, with an efficiency of 14.2%, generate an average power of 110 kW.

In 1958 measurements from Explorer I showed that satellites in the near-Earth environment are subjected to bombardment by

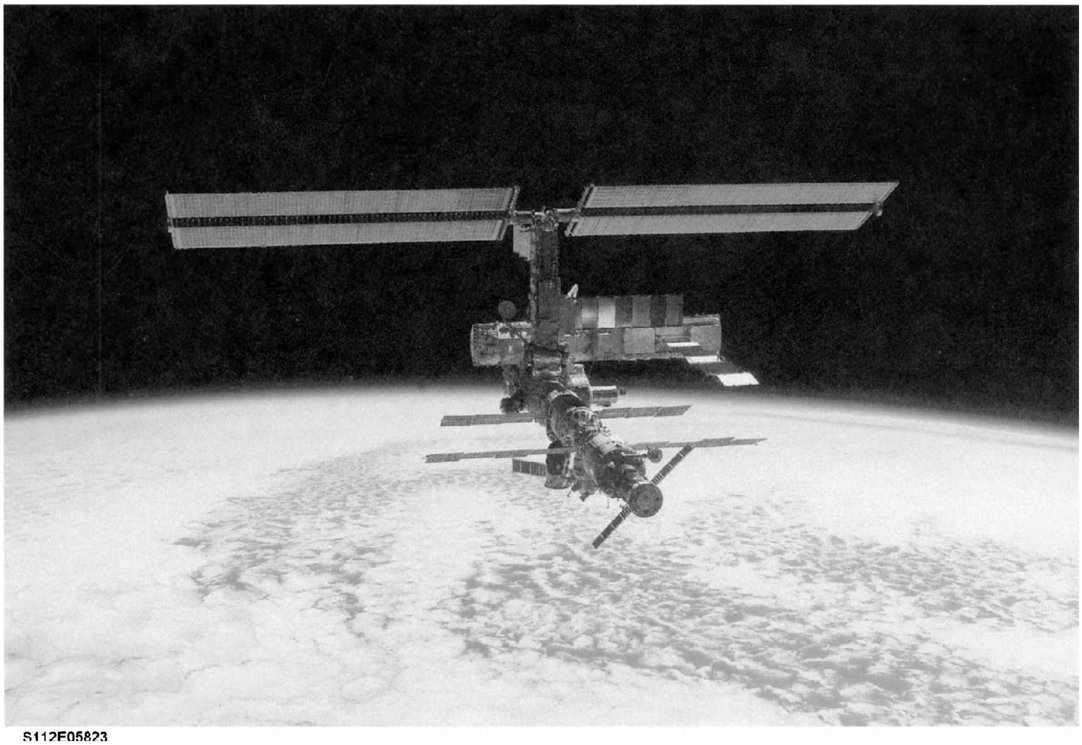

Fig. 23 International Space Station.

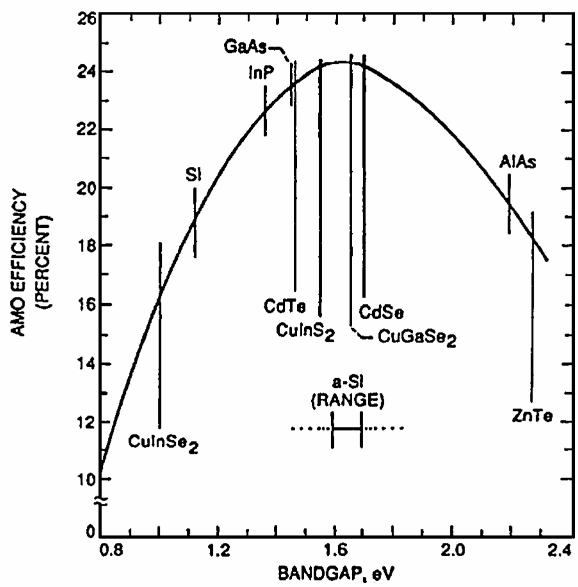

Fig. 24 Ideal photoconversion efficiencies as a function of band gap.

solar radiation and by high-energy electrons and protons trapped in the van Allen radiation belts, and into the 1960s much work was done to understand how extended exposure to this radiation degraded the performance of solar cells. The launch of Telstar in 1962 started the space-based communications market and indirectly reinforced the need for radiation-resistant solar cells. The Telstar was launched with a beginning-of-life power of 14 W but even that modest level was seriously degraded by exposure to radiation from an atmospheric nuclear weapons test.

Amorphous silicon and polycrystalline silicon, as well as thin single crystal cells, have been studied as replacements for the crystalline silicon cells that have been used for 40 years. Because of its low efficiency and costs comparable to single crystal silicon, polycrystalline silicon might not be viable for space applications. However, research is ongoing on crystalline thin-film silicon largely because of its potential radiation resistance.

Yet another important enhancement of space silicon cells is the development of high-efficiency, thin single crystal cells. Thin-film cell use in space is interesting for two reasons: potentially lower costs and radiation hardness. Although amorphous silicon single-junction cells can be made inexpensively, their current low efficiency ($<10\%$) poses a serious barrier to their use in space. These devices are 50 to 60 μm thick and allow weight savings at the array level. At 62 μm the cell is too thin to allow full light absorption, and additional enhancements are required to maintain cell efficiency. These include light trapping by creating a textured top surface and light reflection at the back surface. Many of the enabling features of these high-efficiency cell structures are extremely susceptible to radiation damage, however, and so will delay their use in space. The impact on array weight depends critically on the basic structure of the array and can be substantial, particularly for flexible array structures.

Substrates account for almost two-thirds of the cost of a solar cell, and so will always be the subject of research. Current programs focus on reducing the temperature for cell deposition or developing flexible lightweight substrates capable of withstanding the deposition temperatures because the present low-cost lightweight polymers are not compatible with deposition temperatures.

In the end, however, the end-of-life cell conversion efficiency remains the single most important discriminator in the search for the ideal solar cell. Although weight and cost are considerations, because of the infrastructure costs associated with integrating a solar array into the overall satellite design, only the most efficient cells available will be used, and even at 17% silicon cells might not be competitive.

Beyond Silicon

Semiconductor materials with large band gaps were studied as early as the mid-1950s, even before the launch of Sputnik. The first materials were those made from groups III and V of the periodic chart. The two that attracted the most attention were gallium arsenide (GaAs) and indium phosphide (InP). Although the early efficiencies of these cells were less than 2%, they had theoretical efficiencies significantly greater than silicon. Compared to silicon, they had

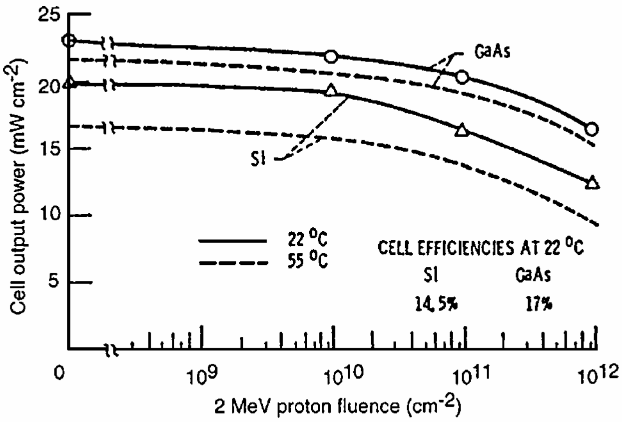

Fig. 25 Comparison of radiation damage in Si and GaAs photocells following irradiation by protons.

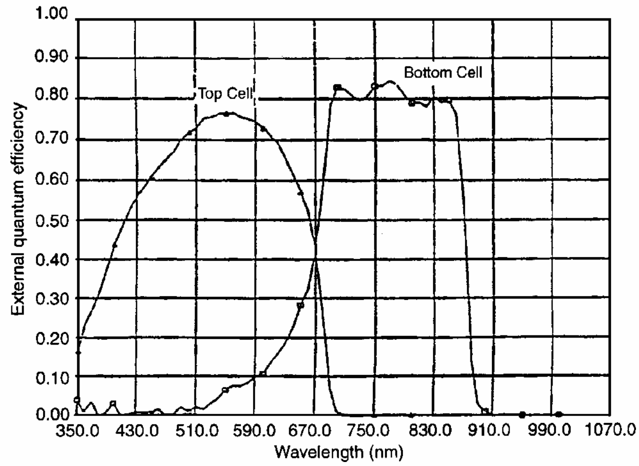

Fig. 26 Representative integrated response of a large band-gap solar cell placed over a lower band-gap material.

other potential advantages by virtue of their larger band gap: they could operate at higher temperatures, could be built with larger specific power, demonstrated a higher tolerance to radiation, but most important, had the potential for significantly higher conversion efficiency.

Two of the major advantages of GaAs cells over silicon are captured in Fig. 25: higher conversion efficiency and greater radiation resistance. In addition, because of their larger band gap the gallium-arsenide cells can operate at a higher temperature, easing thermal management difficulties.

The same is true of InP, which had even greater radiation damage resistance. However, the development of InP for solar-cell applications lagged that of GaAs because of the cost of InP single-crystal wafers. Although the cost of GaAs dropped through its widespread use in microelectronics industry, the same was not true for InP whose use was hampered because of difficulties in matching lattice constants.

Driven by the synergies with the semiconductor industry, GaAs cells gained favor and began to replace silicon in some applications in the late 1980s. GaAs cells are made on germanium substrates, and the cost of the germanium wafers at any time fixed the cost of the cell. Even though the cost of GaAs cells were at times almost 10 times the cost of silicon for comparable power levels, the higher efficiency and greater radiation resistance of the germanium-based cells showed a system-level benefit for the more costly alternative. GaAs cells now have demonstrated an efficiency of more than 22%, but even at that level are not competitive with the newer, multiple-band-gap (MBG) cells that have become available.

MBG solar cells are made by monolithically growing or mechanically stacking different materials into a single structure. Monolithic cells are produced as a single-crystal structure and require that all layers be closely lattice matched throughout the cell to minimize the introduction of crystalline defects, which will degrade cell performance. Current research is focused on mitigating the effects of lattice mismatch to permit a wider choice of band gaps for higher efficiency devices and also to permit development of MBG cells on inexpensive substrates. Mechanical stacking allows each subcell to be produced separately, but then requires careful joining using, for example, an optically transparent adhesive. Although radiation damage in MBG cells is a complex situation very difficult to predict, in general, the radiation tolerance of the overall cell is dictated by the least radiation-resistant subcell, although there are exceptions. The GaAs cell dominates the radiation degradation in the MBG cell, although Ge actually would be the least radiation resistant if it were a single-junction cell. The MBG cells are designed to give higher end-of-life efficiencies even though this generally means the beginning-of-life efficiencies might be lower.

Figure 26 shows the integrated response of a high-band-gap space solar cell, such as $GaInP_2$, in the AM0 spectral intensity distribution, along with that of a GaAs cell, which has been placed underneath it. Putting the $GaInP_2$ cell on top of the GaAs cell will clearly cut off some of the photons that would otherwise reach the GaAs cell, but the combined output of the two devices will still be higher than it would be from either cell separately. The MBG solar cell has been studied extensively to determine the optimum energy bands to use for a two-, three-, or four-junction device, and triple-junction devices are now commercial realities. The theoretical value for an optimum four-junction cell is 42%.

An example of the MBG cell is the triple-junction cell made three layers: the bottom substrate is germanium on which is placed a GaAs layer, which is then covered by a gallium-indium-phosphide ($GaInP_2$) layer. The top layer of $GaInP_2$with the large band gap absorbs the short-wavelength photons and allows the longer wavelengths to pass through to the layers underneath. The middle layer of GaAs then absorbs the middle wavelengths and allows the longer wavelengths to reach the Ga where its smaller band gap can convert the remaining portion of the spectrum. The GaInP cell is connected to the GaAs cell by a tunnel junction (as is the case between GaAs and Ge). These MBG cells have efficiencies of over 29%. The full MBG cell radiation hardness is determined by damage that first causes a departure from its current-matched condition. The resultant damage in GaAs has more to do with the penetration of the particle radiation in the vicinity of the GaAs junction than with the single-cell radiation degradation. The MBG cell actually has slightly better radiation resistance than a single-junction GaAs cell.

Array Enhancements

For multikilowatt solar arrays such as found on the ISS, it was necessary to move to higher operating voltages than those used in the past. The primary system operates at 160 V, but the regulated voltage for ISS operations is 120 V. The reason was straightforward: the mass of the wiring harness. The flexible blanket, flat-fold arrays originally developed for a solar electric propulsion spacecraft can have a wiring harness that will easily exceed 10% of the entire array mass depending on array size and total current. Low voltages create excessive ohmic losses. There are upper limits on the voltages imposed by interactions between the array and the space plasma in low Earth orbit (LEO) and spacecraft charging effects in geosynchronous orbit (GEO), and ways to overcome those limitations continued to be studied. A plasma contactor was developed for ISS to "ground" the array to the space plasma. This has significantly reduced the voltage difference between International Space Station (ISS) and the space plasma. Arcing was believed to be a significant hazard for extra vehicular activity without the plasma contactor.

Further gains in space solar-array performance can be achieved using solar concentrators, as seen in Fig. 27, which shows the effect of concentration on cell efficiency as a function of bandgap energy. It is readily apparent that a significant efficiency gain can be realized at modest concentration levels, that is, at 100X or below.

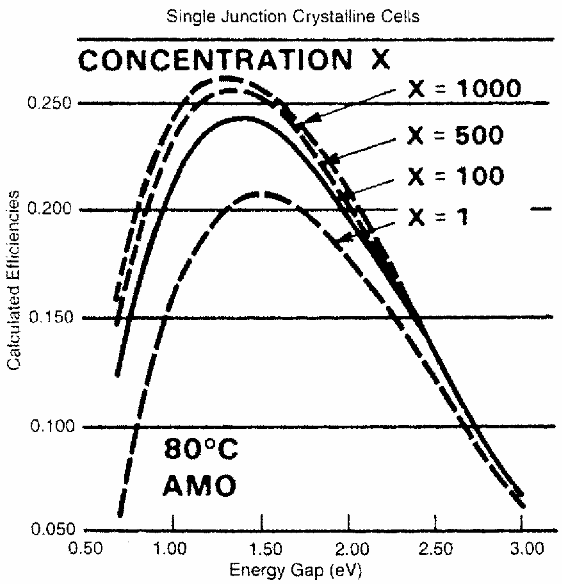

Fig. 27 Theoretical concentrator solar-cell efficiency as a function of band gap.

Although the idea of a concentrator space solar array is not new, none has yet made the significant impact that they will in the future. The first attempt to do so in 1995 ended when the launch vehicle failed to achieve orbit. That first array, known as the Space Concentrator Array with Refractive Linear Element Technology (SCARLET I), consisted of four concentrator panels mounted on a deployable array structure together with two planar silicon cell panels that constituted the original array design. The concentrator used domed linear Fresnel lenses that provided about a seven-fold concentration, in conjunction with GaAs/Ge cells designed for optimum performance at that concentration level. Deep Space 1 (launched in 1998) was powered by the SCARLET array and worked well. Scarlet is the highest technology solar-array flying today and is the result of a convergence of optical, photovoltaic, and structural technology breakthroughs.

The concentrator arrays have two major advantages over planar arrays. The active semiconductor area required for a given power output is reduced by the factor $1/X$, where X is the concentration ratio of the optical element. This has important impacts on array cost and mass. The reduced area allows the use of advanced, high-efficiency space solar cells at considerably lower cost than for the same output planar array. The reduced cell area also means that additional shielding against radiation damage can be provided without a major impact on total array mass, giving the concentrator array a potentially significant advantage in end-of-life performance compared to its planar counterpart. This comes at the expense of needing to maintain precise pointing, however. In the case of SCARLET II, the array output drops to zero if the misalignment is greater than 3 deg.

Other mechanisms of converting solar energy into electrical power have paralleled the work on photovoltaic cells. These alternate conversion schemes have focused on solar thermal systems, using solar energy simply as a heat source in conjunction with thermophotovoltaic (TPV) or solar-thermal dynamic processes

TPV energy conversion, although still in an early stage of development, is an attractive possibility for use in space power systems. Thermal energy in space is provided by concentrated sunlight, a nuclear reactor, or a radioisotope heat source of the sort now used by NASA in its deep-space missions. The conversion takes place in two steps: first, the conversion of thermal energy to radiant energy and then the conversion of the radiant energy to electric energy. At the heart of the TVP conversion process is a solar cell designed for optimal performance in the infrared region of the spectrum. The challenge is to match the energy of the radiated photons to the energy band gap of the cell. The incident photons can originate any heat source, solar photons or the general-purpose heat source (GPHS), for example. Incident energy is used to heat a material to a temperature generally in the 1000 K to 2000 K range, where it then radiates onto the cell material. In effect, rather than depend on the solar spectrum to determine photon wavelengths, a minisun is created with a spectrum tailored to the cell material being used. Another conversion system utilizes a selective emitter, which is then matched to the solar cell.

The thermal energy can be converted into electrical energy in a thermodynamic cycle, however, the TPV system can be simpler. In principle, a TPV system can be completely static, with no moving parts at all, although a pumped, cooling loop would probably be needed.

A major issue for photovoltaic space power systems is the size and mass of the energy storage subsystem, usually batteries. A space TPV system has the advantage of an integrated heat-receiver/thermal-energy storage system. As with solar dynamic systems, this integrated use has the potential to decrease the total weight of the TPV power system and improve the efficiency compared to a solar-array/battery system. On orbit, efficiencies approaching 20% are possible with STPV, compared to the 10% efficiencies of typical array/battery systems. The higher efficiency is the result of gains realized in storage process. TPV systems compete well with thermodynamic-cycle power systems on the basis of total orbital efficiency and should be much simpler to build and maintain.

Summary of Solar Cells

Silicon cells were the cells of choice in space solar arrays from the time of the first Vanguard array in 1958 until the commercial development of GaAs cells in about 1987, a period of nearly 30 years. During that time, space silicon cell efficiencies rose from about 10% to over 15% and grew in size from 2 to 64 cm^2. In the same time period the radiation resistance of the cells doubled. Advances in space solar cells using other materials have continued, and current arrays are being launched with >21% multiple-band-gap solar cells of $GaInP_2$/GaAs on germanium substrates. The promises of even higher efficiency and stronger radiation resistance have also been realized at the laboratory level with >30% MBG concentrator cells and the radiation-hard InP cells.

The thin film cells continue to show improvement and offer the promise of significantly lower cost solar arrays in the future. Fully encapsulated, monolithically integrated thin-film solar arrays could potentially operate with array voltages exceeding 1000 V. Array technology has progressed from the few watts on Vanguard I to over 100 kW on the ISS.

Batteries and Fuel Cells

Space photovoltaic systems have depended on batteries from the beginning, with primary batteries used on the very first spacecraft and both primary and secondary used ever since. Both fuel cells and primary batteries have been used for electrical power in manned missions.

The electrochemical cells in the battery are the basic source of the stored energy, and each electrochemical cell is a self-contained device that releases stored chemical energy as electrical energy on demand from an electrical load. The number and capacity of the connected cells in the battery determine the energy and power capability.

In a primary cell the reactions are irreversible, and therefore the chemical energy can be converted to electrical energy only once. In a rechargeable cells (secondary batteries) the reactions are reversible, and thus, by reversing the flow of electrons (e.g., from a solar array during the sunlight period), the reactions are reversed, restoring the potential energy difference of the electrodes as chemical energy. The ability to reverse the discharge-charge process thousands of times is a function of the cell chemistry.

The fuel cell has been used as the primary power source for the space shuttle. This system includes a number of fuel cells electrically assembled like the cells in a battery to form the fuel cell stack. The remainder of the system includes the external fuel and oxidant tanks, water collection apparatus, and the associated electrical and plumbing hardware.

The difference between the individual battery cell and the individual fuel cell is that in the former the chemical energy is stored and converted to electrical energy within each cell case. In the fuel cell the chemical energy is stored in the form of hydrogen gas (or more recently methanol) and oxygen in tanks external to the cells. The energy output of each fuel cell is the result of hydrogen or methanol reacting at one electrode releasing electrons on demand from the load and the spontaneous and simultaneous reaction of oxygen gas and the electrons at the other electrode. The circuit is closed within the cell by the flow between both electrodes of ions in the electrolyte. The product is water, which can be used for cooling or human consumption. Like the battery, the voltage of the fuel-cell stack is the sum of the individual fuel-cell voltages required for the spacecraft power. However, the fuel-cell system energy capacity is limited only by the quantity of H_2 and O_2 gases in the external storage tanks.

Batteries

The history of batteries is almost 250 years old, going back to 1786 when Galvani invented the copper-iron cell. It was inevitable that batteries were the first power source for space systems: they were available, inexpensive, reliable, and simple. Silver zinc was the battery of choice in the early days of space missions, and because of its long cycle life nickel-cadmium batteries became the major energy storage device over the next 25 years. The nickel-hydrogen battery started to play a role in the 1980s, and a fourth choice, the lithium-ion battery, is currently being used for planetary missions while undergoing qualifications to replace nickel-cadmium and nickel-hydrogen batteries in LEO and GEO applications. A chronological history of first uses of batteries in space applications appears in Table 5.

A number of factors are involved in the selection of a battery, and certainly the specific energy and energy density are key. The operational cell voltage, specific energy (watt-hours/kilogram), and energy density (watt-hours/liter) of major cell types are shown in Table 6. Equally important, but perhaps not as obvious, are factors such as the capacity, lifetime, rate at which power can be delivered, cycle life vs depth of discharge, voltage as a function of temperature, operating temperature range, and safety.

The capacity (expressed as ampere-hours) is related to the number of hours the required load current can be sustained, and the capacity is affected by the rate at which current is delivered, the temperature, as well as other factors. The depth of discharge (DoD) is the percent of capacity that is removed during a single discharge, and there is a strong relationship between the DoD and the number of charge-discharge cycles to which the battery is subjected: the greater the DoD on a recurring basis, the sooner the cell will fail to deliver the necessary voltage for the time required. Recall that for satellites in LEO the capacity must accommodate 5000 cycles a year and (about 30 min of eclipse per cycle), whereas in GEO there will be only about 100 cycles annually (up to about 70 min per cycle). The effects of temperature and DoD on lifetime for a nickel-cadmium (Ni-Cd) battery are shown in Fig. 28.

The silver-zink (Ag-Zn) primary battery used on the Russian Sputnik I had a reasonably high specific energy and was designed to power the spacecraft for up to three weeks. The second Sputnik, launched a month later, carrying the dog Laika, was six times larger, and lasted five months. It also depended on a Ag-Zn battery, although a much larger one. Explorer 1, the first of several Explorer

Table 5 First use of various battery types in space (after Ref. 1)

Date, m/d/y	Satellite	Type	Comments
10/4/56	SPUTNIK I	Ag/Zn	1W for 3 weeks
12/6/56	VANGUARD	Zn/HgO	First U.S. launch
2/1/58	EXPLORER 1	Zn/HgO	Van Allen Radiation Belt
8/6/59	EXPLORER 6	Cyl Ni/Cd	First Earth photos
3/13/61	IMP 1	Ag/Cd	Nonmagnetic
1/26/62	RANGER 3	Ag/Zn	Moon photos
4/26/62	ARIEL I	Pris Ni/Cd	First LEO mission
8/27/62	MARINER 2	Ag/Zn	Venus mission
6/23/63	SYNCOM-2	Cyl Ni/Cd	First GEO
5/20/65	APOLLO CM	Ag/Zn	LTD cycle life
6/23/66	NTS-2	Ni/H_2	12-hour polar
9/23/66	USAF	Ni/H_2	LEO
2/14/80	SOLAR MAX	Ni/Cd	Standard battery
5/19/83	INTELSAT V	Ni/H_2	GEO
4/4/83	STS-3	Li-BCX	Astronaut use
4/6/84	LDEF	LITHIUM	Exposure to space
10/18/89	GALILEO	$Li\text{-}SO_2$	Jupiter probe
4/25/90	HST	Ni/H_2	NASA LEO
6/10/90	LEASAT	Super Ni/Cd	GEO
1/25/94	CLEMENTINE SPV	Ni/H_2	Lunar mapping
1/25/94	TUBSAT-B	2 Cell CPV	Store messages
5/1995	CENTAUR	$Li\text{-}SOCl_2$	28V, 250AH battery
5/5/96	IRIDIUM-1	50Ah SPV	34 to date-LEO
12/4/96	Mars Lander	Ag/Zn	40AH rechargeable
12/4/96	Mars Rover	$Li\text{-}SOCl_2$	3 D-cell batteries
11/19/97	FLIGHT EXP	Na/S	Wakeshield platform

Table 6 State-of-the-art characteristics of operational cells (after Ref. 1)

Battery system	Anode	Cathode	Cell voltage, V	Specific energy, Wh/kg
		Primary cells		
LeClanche	Zn	MnO_2	1.5	85
Alkaline-MnO_2	Zn	MnO_2	1.5	125
Mercury	Zn	HgO	1.3	100
Silver oxide	Zn	Ag_2O	1.6	120
Zinc/Air	Zn	O_2(air)	1.5	340
Li/SO_2	Li	SO_2	3.0	260
$Li/SOCl_2$	Li	$SOCl_2$	3.6	320
Li/MnO_2	Li	MnO_2	3.0	230
$Li(CF)_n$	Li	$(CF)_n$	3.0	220
		Secondary cells		
Lead-acid	Pb	PbO_2	2.0	35
Nickel-cadmium	Cd	NiO_2	1.2	35
Nickel-metal hydride	(MH)	NiO_2	1.2	50
Nickel-hydrogen	H2	NiO_2	1.2	55
Silver-zinc	Zn	AgO	1.5	90
Silver-cadmium	Cd	AgO	1.1	55
Zinc-air	Zn	O_2 (air)	1.5	150
Lithium-ion	C	$LiCoO_2$	4.0	90
Lithium-organic	Li	Mn_2O_4	3.0	120
Lithium-polymer	Li	V_6O_{13}	3.0	200
Sodium-sulfur	Na	S	2.0	160
Zebra	Na	$NiCl_2$	2.3	120

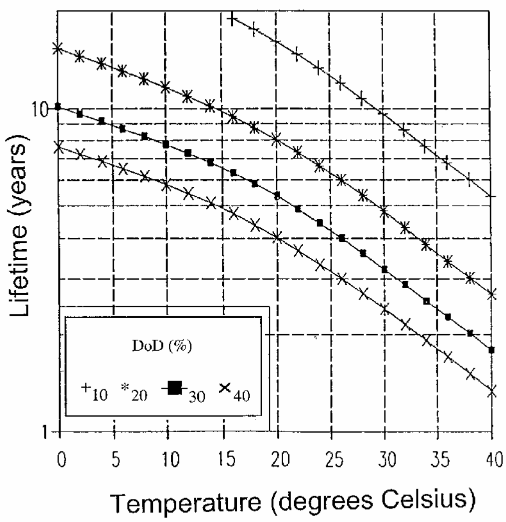

Fig. 28 Effects of temperature and depth of discharge on Ni-Cd battery cycle life.

spacecraft, followed the initial Russian launch on 1 February 1958. The spacecraft was a cylinder 80 in. in length and 6 in. in diameter containing 5 kg of instruments, batteries (primary mercury type), and a radio.

The first U.S. spacecraft to attempt was the Vanguard Test Vehicle 3, which relied on zinc-mercuric oxide primary batteries (in the form of D-sized cylindrical cells) and solar cells to provide power, but the power system was never tested because the satellite failed to orbit.

Vanguard, part of the U.S. contribution to the International Geophysical Year, carried a payload of seven mercury cell batteries (in a hermetically sealed container), a pair of tracking radio transmitters (to allow ground-based stations to track the flight), a temperature-sensitive crystal, and six clusters of solar cells on the surface of the spherical spacecraft. A second satellite in the series (Test Vehicle 4) was designated Vanguard I and launched on 17 March 1958. This satellite achieved an elliptical orbit (apogee 2466 miles, perigee 404 miles) that was estimated to remain in space for more than 200 years. Data obtained from Vanguard I showed that the Earth is not quite spherical—it is elevated at the North Pole and flattened at the South Pole. The radio continued to transmit until 1965.

Explorer 6, launched in August 1959, was the first successful launch of Ni-Cd cells. In August of that year, the Pioneer spacecraft, the first stage of the lunar probe, carried both Ni-Cd and Ag-Zn cells.

The first time Ni-Cd batteries were used for prime power was in February 1960. The spacecraft Transit 1, which contained two packs of 28, 5-A-h cylindrical Ni-Cd cells, failed to launch. Just two months later, however, in April 1960 the weather satellite TIROS I, also designed to be powered by Ni-Cd batteries, successfully launched. TIROS (television infrared observation satellite) was a test meteorological satellite information system also designed to test sun angle and horizon sensor systems for spacecraft orientation. TIROS I, operational for only 78 days, was a 42-in.-diam, 19-in.-high cylinder that weighed 270 lb. The aluminum-alloy, stainless-steel craft was covered by 9200 solar cells to charge its Ni-Cd batteries.

The rudimentary electrical system powered two television cameras, low and high resolution. Magnetic tape recorders stored the photographs while the satellite was not within view of a ground station. The three strings of 21 Ni-Cd cylindrical cells contained glass-to-metal seals to insulate the positive terminal from the metal case. These cells were also fitted with a threaded base, which was used to screw into threaded holes in the battery baseplate. The spacecraft operated in a 90–110 min orbit. The spacecraft electrical loads were designed to remove only a conservative 3% of the capacity from the battery during the 30-min eclipse period. Although the low depth of discharge is a significant factor in extending life, it also means a much larger battery capacity is required for the low DoD requirement and was a very high price to pay in inefficiency and cost.

In November 1961 the U.S. spacecraft Ranger 3, containing two 14-cell, 50-A-h batteries for the main power and two 22-cell, 50-A-h batteries for the TV camera power, was placed into solar orbit and took photographs of the Moon. Mariner 2, with one 18-cell, 40-A-h Ag-Zn battery launched in August 1962, was the first successful interplanetary mission to Venus.

In November 1964 the first prismatic Ni-Cd cells were flown on Explorer 23. Each cell was a box-like configuration, which provided a means for producing an efficient battery design in which the cells were lined up in a close-knit battery pack held together with end plates and metallic rods. The cells also were designed with two insulated terminals such that the cells were electrically floating.

Although not the first astronomical satellite project (that honor goes to the United Kingdom for the Ariel series which first launched in April 1962), the U.S. entry into space-based observational was the OAO series. OAO-1 was launched successfully on 8 April 1966, but battery failure after only three days in orbit ended the mission. OAO-2, which followed two-and-a-half years later carried 11 UV telescopes, detected the first UV radiation from the center of the Andromeda Galaxy, as well as a supernova in May 1972. In spite of its failure, the OAO-1 power system did usher in a new technology. The OAO series used three batteries of 20-A-h prismatic Gulton cells uniquely assembled into two battery frames. Pairs of cells were interspersed between the two assemblies to minimize temperature variation. This power system employed a VT (temperature-compensated voltage) charge control system that applied a constant voltage to batteries during charge. The preset VT limited the charge, resulting in a safe and reliable charge condition. This is a condition in which the batteries return to full charge but are not excessively overcharged. The selected voltage limit was based on a parallel set of temperature-compensated voltage curves. The VT-curve selection used to limit the charge voltage provided flexibility to account for unexpected high depths of discharge and/or imbalance between cells and/or batteries.

As part of the effort to create modular support systems, in the mid-1970s NASA started a program to develop standard cells and batteries. The result was the standard Ni-Cd cell and then the standard battery (Fig. 29). Four companies were given the opportunity to develop prismatic 20 A-h standard Ni-Cd cells, which would then be capable of being assembled into the standard battery structure in the Modular Power System. The battery was designed to meet all NASA mission and launch requirements including vibration and shock. Following evaluation, only the cells manufactured by General Electric met the performance requirements and were selected as NASA standard cells. It was becoming apparent that variations in the manufacturing processes were introducing unacceptable uncertainties in the performance of the batteries, and so the cells were accompanied by a Manufacturing Control Document intended to track process and improve reliability. The first lot of standard 20-A-h cells in standard 20-A-h batteries were flown successfully on the Solar Max Mission for more than eight years. Subsequently, the technology was extended to 50-A-h cells, which were used on several NASA spacecraft, such as Landsat, TOPEX, UARS, and GRO.

Also in the 1960s, a technology was discovered that made use of the NiOOH electrode from the Ni-Cd cell and the H_2 electrode from the fuel cell. The individual pressure vessel Ni/H_2 cell was contained in a pressure cylinder configuration caused by the buildup of hydrogen during charge, as much or greater than 400–800 psi (Fig. 30). The replacement of the cadmium electrode with a hydrogen electrode reduced weight and increased energy, significantly improving the specific energy over the Ni-Cd cell. The NASA Standard 50-A-h cell had an average specific energy of 36 W-h/kg and a 50-A-h aerospace design Ni-H_2 cell had 49 W-h/kg. Because of the cylindrical configuration and the wider spacing of the cells on the base plate, the energy density (watt-hours/liter) of the battery was much lower than that of the Ni-Cd battery. However, the Ni-H_2 system offered the capability of extended life at higher DoD (higher DoD further added to the weight advantage). Comsat was the first to develop this battery and use it in the Intelsat V spacecraft in a GEO mission in 1983. Two cells were successfully combined in the same cylinder in 1983. This common pressure vessel cell was first used in NASA Jet Propulsion Laboratory (JPL) Mars Global Surveyor mission in 1994. The next step resulted in the development of a rechargeable single pressure vessel Ni-H_2 battery in which 22 cells were mounted in the same structure. It was used for the first time in 1994 in Clementine, a Navy satellite that circled the Moon. These alternate designs for the nickel-hydrogen cells offered improvements in weight and volume over the independent pressure vessel (IPV) battery. Ni-MH cells using chemically bonded hydrogen in the form of a hydride have been used in a few rocket experiments but have not been used in any major flight program. However, because of the low pressure these cell cases do not require high-pressure cylinders.

An interesting story regarding Ni-H_2 batteries comes from the NASA Glenn Research Center. The Hubble Space Telescope (HST) was designed to be launched with Ni-Cd batteries for energy storage. However, the delay in the launch of the HST because of the space shuttle *Challenger* accident meant that the Ni-Cds were nearing the end of their shelf life. The Ni-H_2 batteries, which had been planned as replacements on orbit for the original Ni-Cds, could be made ready for the HST launch date. After a lengthy study the decision was made to replace the Ni-Cd with Ni-H_2, even though the Ni-H_2

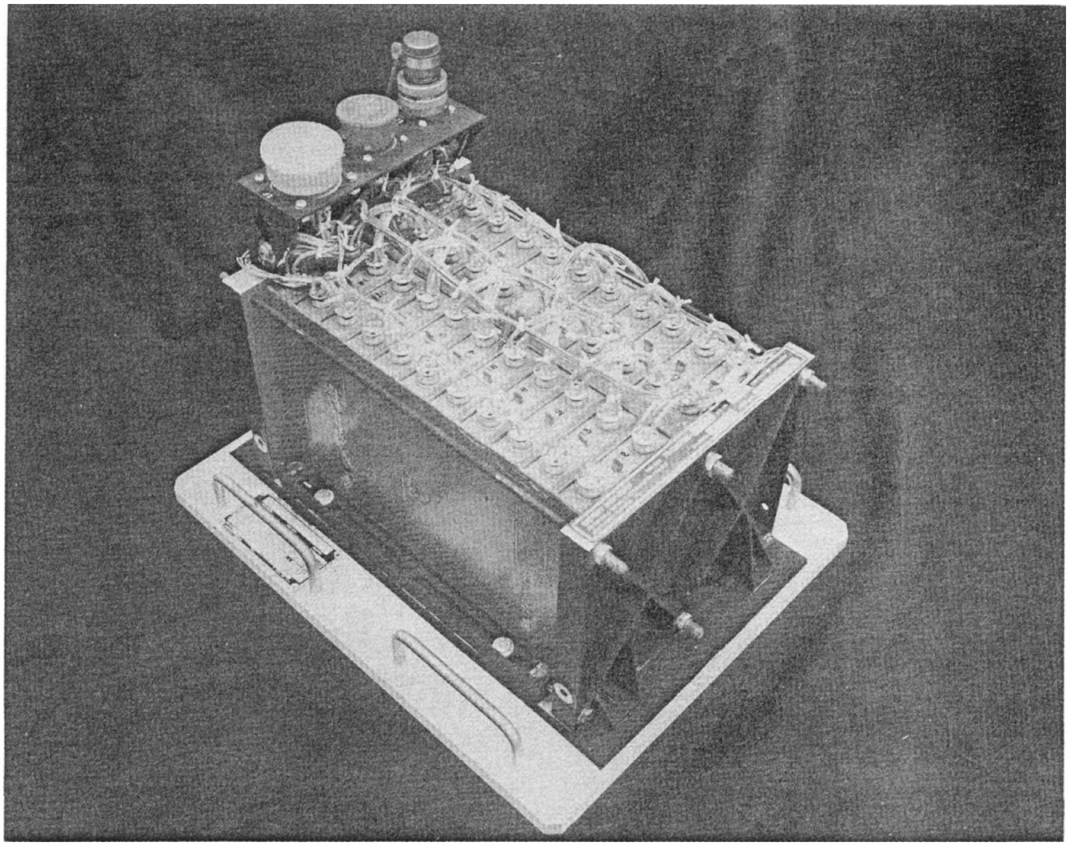

Fig. 29 NASA Standard 20-A-h battery.

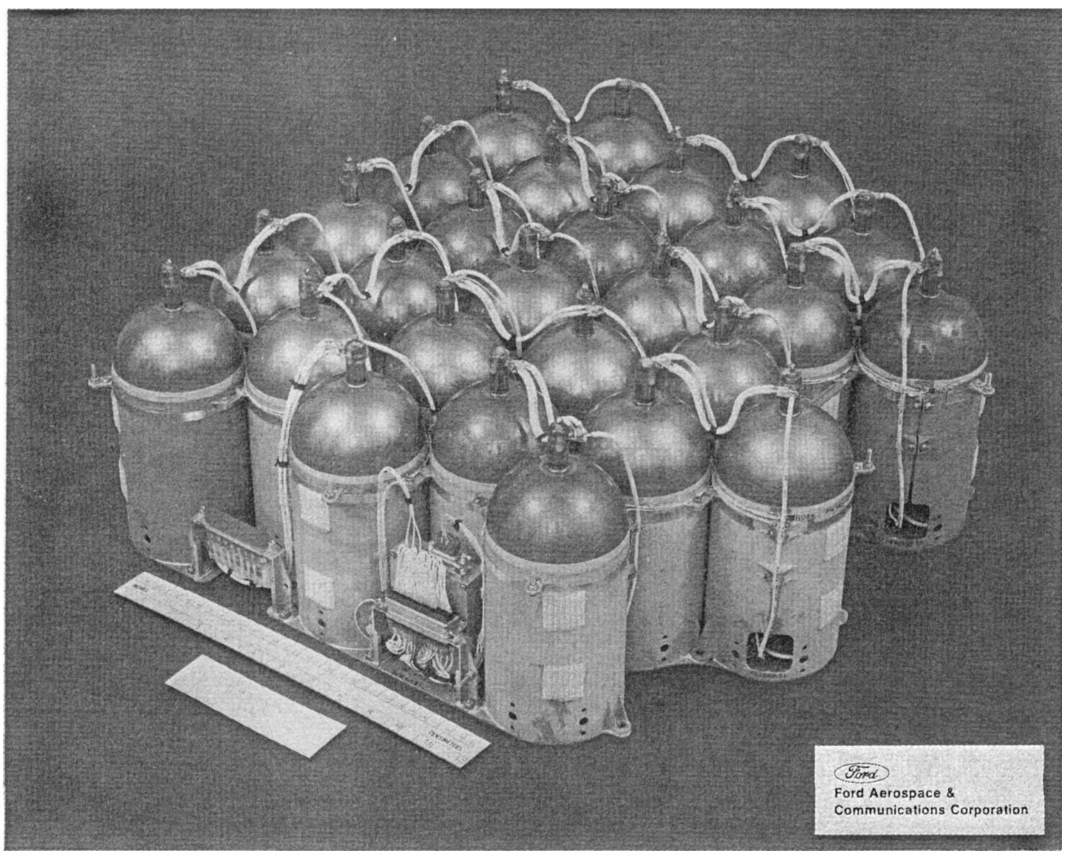

Fig. 30 Intelsat-V battery with 30-A-h individual pressure vessel cells.

had not previously been flown in LEO. The original Ni-H_2 batteries launched in 1990 are still in service.

Lithium primary cells were used in space during the 1980s. Li-$(CF)_x$ batteries were one of the first used for range safety on launch vehicles. Li-BCX (Li-$SOCl_2$ cells with bromine chloride additive) was selected by NASA for use in astronaut equipment, specifically the helmet lights and TV camera. Later, Li-SO_2 batteries were designed for the probe on the Galileo mission to Pluto, a mission profile that required nine years of storage before use. Seven kWh Li-$SOCl_2$ (250 A-h, 28 V) batteries were developed for the U.S. Air Force Centaur launch vehicles to replace the Ag-Zn batteries with an eye to extending the operating time in placing payloads in orbit. Smaller D-size cells based on this technology were used by JPL in the Mars Rover in 1997.

In the 1980s JPL initiated development of a rechargeable lithium cell in an in-house program. The lithium-titanium disulfide (Li-TiS_2) battery used pure lithium as the anode. The specific energy achieved 100 W-h/kg or twice that of the NiH_2 or Ni-MH system and was cycled more than 1000 times at a remarkable 50% DoD. However, environmental issues related to the metallic lithium foil concerned the users. In the follow-on lithium-ion cell development effort in the 1990s, coke or graphite replaced the lithium anode foil and several cathode materials, for example, cobalt, nickel, or manganese oxides, replaced the undesirable titanium disulfide. There was no lithium metal in the cell. This new system made use of the difference in concentration of lithium ions between anode and cathode. The potential of each cell is 4.0 V, and the specific energy is greater than 125 W-h/kg. Lithium-ion batteries will be flown as the energy storage subsystem on two Mars Exploration Rovers that are scheduled for launch later this year.

In 2001 Starshine 3 was launched, and that system marked two firsts: an integrated power system (solar cell integrated with a thin-film Li-ion battery) and the initial test of the triple junction Emcore solar cells. The system worked as predicted.

Fuel Cells

The (H_2-O_2) fuel cell has many attractive features including pollution-free operation of a direct-conversion process with no moving parts using high-specific-impulse hydrogen and oxygen fuels that are generally available on (manned) spacecraft. It also provides water and heat to the crew!

The first use of a fuel cell in space was in the Gemini program in August 1962. This first device was a proton exchange membrane electrolyte fuel cell (PEMFC), but at that time called the solid-polymer electrolyte-ion-exchange membrane fuel cell. First discovered over 150 years ago, fuel cells have been used on Gemini, Apollo, and the space shuttle.

The early Gemini fuel cells generated 350 W from each module, and three stacks of 32 cells were used in parallel to provide about 1 kW. Cryogenic hydrogen and oxygen were used, and conversion proceeded at about a 50% efficiency.

The Apollo flights (1968–1970) used the alkaline electrolyte fuel cell (AFC) with KOH as the electrolyte. The system developed 1.5 kW at 26 V and operated at 260°C. The platinum catalyst that was present in the PEMFC was not needed. The AFC delivered a peak power of 2.3 kW at 20.5 V, weighed 100 kg, and operated almost 700 h without failure.

The shuttle power is significantly greater. It uses three hydrogen–oxygen alkaline units each providing 12 kW at peak and is capable of 2000 h of operation. As an indication of the advances in fuel-cell technology, the shuttle fuel cell is 23 kg lighter and delivers eight times the power of the Apollo system.

A number of improvements to current fuel-cell technology involves the use of proton-exchange membranes, use of methanol in place of hydrogen, and the development of regenerative fuel cells for applications where charge and discharge operations are needed.

Nuclear Sources

Nuclear power sources might be the only choice for power in very long-duration missions, trajectories that carry the spacecraft far from the sun, or for payloads that require very high power levels. Cost, clear mission needs, and safety issues have been the main obstacles to the use of nuclear power in space.

Space nuclear power systems fall into two general categories: radioisotope systems generate heat by the natural decay of radioisotopes, and nuclear reactor systems that produce heat are generated by nuclear fission. Both the United States and the Former Soviet Union (FSU) have conducted major research and design programs for space reactors, but only the FSU has applied this technology to actual space missions.

Nuclear sources of either type are simply heat sources and must be married with an appropriate conversion technology to produce electricity. The conversion technologies, which will be discussed briefly, can be classified broadly as static or dynamic. Nuclear systems are not unique in their ability to produce heat—both solar and chemical sources can also serve as heat sources.

U.S. Space Nuclear Program

As early as a decade before successfully placing its first satellite in orbit, the United States began studies of reliable power systems for reconnaissance satellites, including nuclear power systems. By the mid-1950s U.S. Atomic Energy Commission (AEC) studies of space nuclear power led to the creation of the System for Nuclear Auxiliary Power (SNAP) program. In an interesting numbering scheme, the odd-numbered SNAP systems were based on radioisotopes and the even on reactors. An early AEC-U.S. Air Force collaboration gave way to AEC-NASA program in the mid-1960s. With subsequent name changes the AEC became the Energy Research and Development Administration and then the present U.S. Department of Energy (DoE), and throughout the process DoE and its predecessors retained responsibility for the nuclear program.

There has been a single launch of a U.S. nuclear reactor, the SNAP-10A, in 1965. It was placed in a 1300-km, circular polar orbit where it operated for only 43 days. Subsequent investigations identified an electronics problem that resulted in the reactor shutting down. The reactor was not designed to be restarted. A second SNAP-10A was ground tested for over 10,000 hours before the test, and the program was terminated.

The reactor used a UZrH fuel and produced 43 KW of thermal power. NaK was the coolant that carried the heat to a stack of SiGe thermoelectric converter elements that produced slightly less than 600 W of electrical power.

Interest in reactors was resurrected in the early 1980s primarily driven by the military missions and led to a design named the Space Power Advanced Reactor (SPAR). The SPAR became a baseline for the SP-100, a 100-MWe reactor. The SP-100 was to be a fast-neutron spectrum, lithium-cooled reactor using one of three conversion schemes: in-core thermionics, out-of-core thermoelectrics, or out-of-core Stirling. Even though the thermoelectric option was selected, work on the others continued. NASA maintains a program on Stirling engines that can be used in a solar dynamic system. As the Space Defense Initiative Organization mission began to look more to terrestrial-based missile defense, the SP-100 program lost its mission and its support. Presently, there are no mission requirements to support a renewed nuclear reactor power program for space.

The use of RTGs in space has been widespread by comparison. Although design specifics vary, all RTGs consist of two primary subsystems: a radioisotope heat source and a converter (usually thermoelectric)/radiator assembly. Any number of radioisotopes have been tested as candidates for the heat source, and thermionic as well as thermoelectric conversion has been used. All of the U.S. flight RTGs have used some form of plutonium-238 as the radioisotope source. Initially, several radioisotope fuels were considered, and some of the fuel characteristics of interest are shown in. The first unit, SNAP-1, used cerium-144 because it was readily available from reprocessing of defense reactor fuel. RTGs have powered navigational satellites, weather satellites, experimental communications satellites, five were left to power scientific packages on the Moon, and powered several planetary exploration spacecraft. RTGs deployed 30 years ago are still functioning. Table 7 lists the nuclear power systems launched by the United States.

Table 7 Nuclear power systems launched by the United States (after Ref. 1)

Date	Spacecraft	Power source[a]	Status
29 June 61	Transit 4A	SNAP-3B7	RTG operated for 15 years. Satellite now shut down but operational.
15 Nov. 61	Transit 4B	SNAP-3B8	RTG operated for 9 years. Satellite operation was intermittent after 1962 high-altitude test. Last reported signal in 1971.
28 Sept. 63	Transit 5-BN-1	SNAP-9A	RTG operated as planned. Non-RTG electrical problems on satellite caused satellite to fail after 9 months.
5 Dec. 63	Transit 5-BN-2	SNAP-9A	RTG operated for over 6 years. Satellite lost navigational capability after 1.5 years.
21 April 64	Transit 5-BN-3	SNAP-9A	Mission was aborted because of launch vehicle failure. RTG burned up on reentry as designed.
3 April 65	Snapshot	SNAP-10A	Successfully achieved orbit.
18 May 68	Nimbus-B-1	SNAP-19B2	Mission was aborted because of range safety destruct. RTG heat sources recovered and recycled.
14 April 69	Nimbus III	SNAP-19B3	RTGs operated for over 2.5 years (no data taken after that).
14 Nov. 69	Apollo 12	SNAP-27	RTG operated for about 8 years (until station was shut down).
11 April 70	Apollo 13	SNAP-27	Mission aborted on way to moon. Heat source returned to South Pacific Ocean.
31 Jan. 71	Apollo 14	SNAP-27	RTG operated for over 6.5 years (until station was shut down).
26 July 71	Apollo 15	SNAP-27	RTG operated for over 6 years (until station was shut down).
2 March 72	Pioneer 10	SNAP-19	RTGs still operating. Spacecraft is beyond solar system, 6.2 billion miles from Earth. NASA officially ended mission on 31 March 1997.
16 April 72	Apollo 16	SNAP-27	RTG operated for about 5.5 years (until station was shut down).
2 Sept. 72	"Transit"	Transit-RTG	RTG still operating (Triad-01-1X).
7 Dec. 72	Apollo 17	SNAP-27	RTG operated for almost 5 years (until station was shut down).
5 April 73	Pioneer 11	SNAP-19	RTGs still operating. Spacecraft successfully operated to Jupiter, Saturn, and beyond.
20 Aug. 75	Viking 1	SNAP-19	RTGs operated for over 6 years (until lander was shut down).
9 Sept. 75	Viking 2	SNAP-19	RTGs operated for over 4 years until relay link was lost.
14 March 76	LES 8*	MHW-RTG	RTGs still operating.
14 March 76	LES 9*	MHW-RTG	RTGs still operating.
20 Aug. 77	Voyager 2	MHW-RTG	RTGs still operating. Spacecraft successfully operated to Jupiter, Saturn, Uranus, Neptune, and beyond.
5 Sept. 77	Voyager 1	MHW-RTG	RTGs still operating. Spacecraft successfully operated to Jupiter, Saturn, and beyond.
18 Oct. 89	Galileo	GPHS-RTG	RTGs still operating. Spacecraft orbiting Jupiter.
6 Oct. 90	Ulysses	GPHS-RTG	RTG still operating. Spacecraft successfully measured environment over sun's poles.
Oct. 97	Cassini	GPHS-RTG	RTGs still operating. Mission to Saturn.

[a] All power sources are RTGs, except SNAP 10-A (reactor).

Several of the RTGs hold a special place in the history of nuclear power systems. One such is SNAP-3, which, ironically, was never launched. It was the first to use a static conversion process (thermoeloelectric) instead of the dynamic mercury-Rankine cycle used in SNAP-1. Another that must be mentioned is the SNAP-19 RTG that was modified to provide 120 W to the Pioneer 10 trip outside the solar system. In March 1997 NASA shut down the satellite after 25 years and 6.2 billion miles. Pioneer 11 continued to operate even after that.

RTGs launched into the mid-1970s all used devices developed under the original SNAP program. There were replaced by the Multi-Hundred Watt RTG (MHW-RTG) program that brought two technological advances: an oxide of plutonium-238 became the fuel, and SiGe unicouples were used as the thermoelectric converter. The MHW-RTG evolved to the GPHS, a module that has become the basic building block of RTG systems. Figure 31 depicts a GPHS module. Each module weighed 1.43 kg including 0.6 kg of 238PuO2 in four pressed fuel pellets generating about 250 W (thermal). The thermopile consisted of 572 SiGe unicouples that surround the heat source, and each RTG provides 300 W at 30 V. Any number of GPHS modules could be stacked to provide the power levels needed, generally less than 1 kW. A stack of 18 GPHS modules coupled with an appropriate number of thermocouples constituted a standard GPHS-RTG, two of which are seen on the Galileo spacecraft (Fig. 32).

Russian Space Nuclear Program

The FSU pursued a very aggressive development program in space nuclear power, including both RTGs and reactors, with an emphasis on reactors. Their development program consisted of four systems: Bouk, Romashka, TOPAZ, and Enisy. Bouk powered the Radar Ocean Reconnaissance Satellite (RORSAT) series of ocean surveillance spacecraft. Two of the newer TOPAZ units were flight tested in 1987 but have not been used on operational missions. Enisy and Romashka were ground tested, but neither flew in space.

Table 8 is an abbreviated list of the early Russian space missions that were powered by nuclear sources. The launches of nuclear-powered satellites continued until the end of 1996. Bouk was launched on 32 successful RORSAT missions plus one launch that failed to achieve orbit. The RORSAT satellites used a side-looking radar to track naval vessels and orbited at altitudes of less than 300 km. Because of the atmospheric drag at this low altitude, nuclear power was selected because of its much lower projected area in the direction of flight relative to solar arrays. At the end of these missions, Bouk reactors were moved to a disposal orbit of about 1000 km to preclude reentry of a "hot" reactor. Unfortunately, the reboost to the higher orbit failed on three occasions, most notably the Cosmos 954 reentry that spread radioactive debris over a remote region of Canada in 1978. After that failure Russia redesigned the safety system to extract the nuclear core from the Bouk reactor vessel for reboost to the higher orbit. If the reboost failed (as in Cosmos 1402 in 1983), the fuel elements would burn up in the atmosphere instead of impacting on Earth. For the two TOPAZ flights the operational orbits were in the range 800–1000 km, so that reentry was much less of an issue.

Beginning in 1970, Russia built and tested four developmental TOPAZ thermionic reactors, plus 10 thermal and structural mock-ups, leading to the final TOPAZ design. The first version operated successfully to a design life of 1000 h, and the second operated about 6000 h.

The two Russian in-core thermionic reactor systems, TOPAZ and Enisy, are moderated cores with highly enriched UO_2 fuel. Both produce about 5 kWe plus an additional 1 kWe to operate the electromagnetic pumps and weigh 1000–1200 kg. The primary difference is that TOPAZ employs a multicell TFE design, while the Enisy TFE is a single cell.

Conversion Technologies

To date, other than solar, only thermoelectrics and thermionics have produced electricity in space. Several other conversion

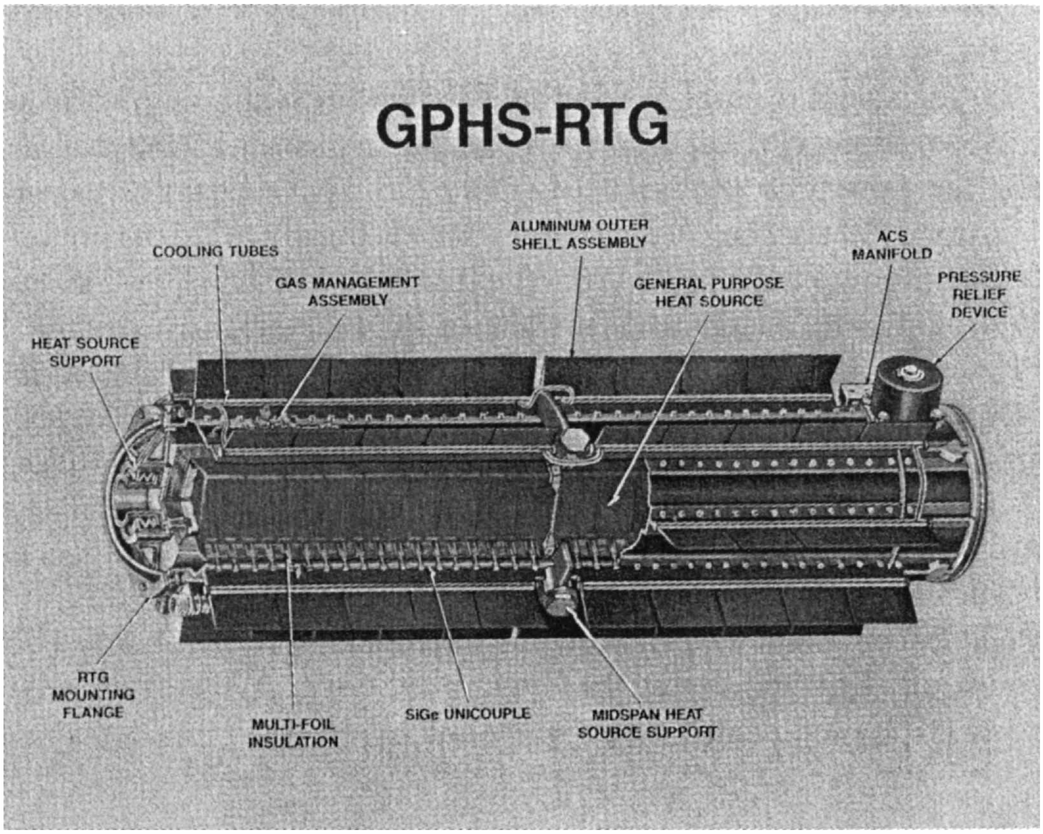

Fig. 31 GPHS radioisotope thermoelectric generator.

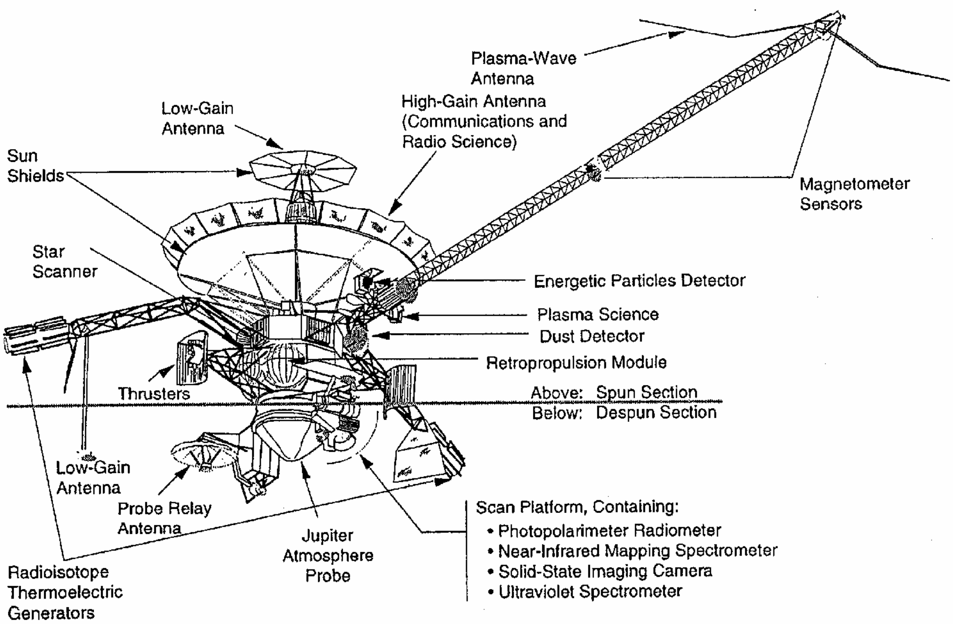

Fig. 32 Spacecraft Galileo with the radioisotope thermoelectric generators shown.

schemes have been intensively studied and might one day become a part of the history of space power. Rankine and Brayton thermodynamic-cycle converters have been built and tested as part of a nuclear reactor system, but not flown. Two static conversion processes are under study and should be mentioned as candidates for the future: AMTEC and TPV.

The operation of an AMTEC converter is tied to a unique property of the Beta-alumina solid electrolyte, a transparent crystalline ceramic in which alumina is stabilized with lithia or magnesia. The material is prepared as a tube into which a high-temperature working fluid, usually sodium, is placed. On the inside surface of the tube, Na^+ ions form, and migrate then through the Beta-alumina to the outside. The Beta-alumina has a high conductivity for these ions but essentially no conductivity for either electrons or neutral sodium. The excess electrons are produced at the anode (inside surface) and are collected and delivered to the external load, producing power.

A third conversion process that should be mentioned is the Stirling engine, invented by a Scottish minister in the early 19th century.

Table 8 Abbreviated list of Russian (FSU) missions powered by nuclear sources (after Ref. 1)

Launch date	Spacecraft	Power source[1]	Status/lifetime
3 Sept. 65	Cosmos 84	Orion 1 RTG	In orbit
18 Sept. 65	Cosmos 90	Orion 1 RTG	In orbit
27 Dec. 67	Cosmos 198	Reactor	1 day
22 March 68	Cosmos 209	Reactor	1 day
25 Jan. 69	RORSAT launch	——	——
23 Sept. 69	Cosmos 300	^{210}Po heater	——
22 Oct. 69	Cosmos 305	^{210}Po heater	——
3 Oct. 70	Cosmos 367	Reactor	1 day
1 April 71	Cosmos 402	Reactor	1 day
25 Dec. 71	Cosmos 469	Reactor	9 days
21 Aug. 72	Cosmos 516	Reactor	32 days
25 April 73	RORSAT launch	——	——
27 Dec. 73	Cosmos 626	Reactor	45 days
15 May 74	Cosmos 651	Reactor	71 days
17 May 74	Cosmos 654	Reactor	74 days
2 April 75	Cosmos 723	Reactor	43 days
7 April 75	Cosmos 724	Reactor	65 days
12 Dec. 75	Cosmos 785	Reactor	1 day
17 Oct. 76	Cosmos 860	Reactor	24 days
21 Oct. 76	Cosmos 861	Reactor	60 days
16 Sept. 77	Cosmos 952	Reactor	21 days
18 Sept. 77	Cosmos 954	Reactor	~43 days
29 April 80	Cosmos 1176	Reactor	134 days
5 March 81	Cosmos 1249	Reactor	105 days
21 April 81	Cosmos 1266	Reactor	8 days
24 Aug. 81	Cosmos 1299	Reactor	12 days
14 May 82	Cosmos 1365	Reactor	135 days
1 June 82	Cosmos 1372	Reactor	70 days
30 Aug. 82	Cosmos 1402	Reactor	120 days
2 Oct. 82	Cosmos 1412	Reactor	39 days
29 June 84	Cosmos 1579	Reactor	39 days
31 Oct. 84	Cosmos 1607	Reactor	93 days
1 Aug. 85	Cosmos 1670	Reactor	83 days
23 Aug. 85	Cosmos 1677	Reactor	60 days
21 March 86	Cosmos 1736	Reactor	92 days
20 Aug. 86	Cosmos 1771	Reactor	56 days
1 Feb. 87	Cosmos 1818	Reactor	~6 months
18 June 87	Cosmos 1860	Reactor	40 days
10 July 87	Cosmos 1867	Reactor	~1 year
12 Dec. 87	Cosmos 1900	Reactor	~124 days
14 March 88	Cosmos 1932	Reactor	66 days
16 Nov. 96	Mars 96	Angel RTG	——

Interest in the Stirling engine began in the 1970s because of its potential for operation at moderate temperatures with relatively high efficiency. One version of the Stirling, the free piston Stirling engine (FPSE), is potentially light enough for space use. For electrical power generation the power piston of the FPSE is connected directly to the armature of a linear alternator. Current designs of the free-piston Stirling engine at the NASA Glenn Research Center (GRC) require three to four times less fuel than an RTG at comparable power levels.

The GRC also has an active program in flywheel technology. Flywheels offer several advantages as an energy storage mechanism including combining storage with attitude control, operations at very high depth of discharge with no lifetime penalty, high efficiency, a broad temperature operating range, and a well-defined state-of-charge. Flywheels can operate with depth of discharge approaching 90%, an attribute that directly impacts the size of a system needed to meet a specific energy requirement. For example, to deliver 1 kW-h to a load, a battery limited to a nominal 35% DoD would require almost a 3-kW-h system. A comparable flywheel could be sized at only 1.1 kW-h.

A UT-CEM flywheel replacement for the Ni-H_2 batteries on the ISS has been proposed with the following operating parameters: energy (3660 W-h), power (3.6 kW), design life (>350,000 cycles), energy density (25 W-h/kg), efficiency (>90%), and maximum operating speed (53,000 rpm). GRC predictions project this system to an energy density of 100 W-h/kg, a power density of 2000 W/kg, and a 25-year life in GEO.

The future of space electrical power will continue to be driven by requirements related to power demand, weight, and reliability. Technologies emerging from programs such as the NASA nuclear initiative will doubtless present opportunities for growth to the levels needed to open space exploration, colonization, and commercialization to their fullest.

Summary

Aerospace electrical power technologies have made incredible advances in the first hundred years of powered flight. One can only marvel at the progress from the earliest days during which there was essentially no requirement for electricity to today's hundred-kilowatt power system aboard the International Space Station and thousand kVA generating capacity aboard the E-4B Command Post aircraft. Some of the progress certainly was the result of incremental improvements in existing technologies. Much, however, was the result of innovative genius drawn from the spectrum of engineering disciplines, driven by the ever increasing need for electrical power. The aircraft and spacecraft designers' appetite for electrical will continue to promote evolution and demand revolution. Where will the next one hundred years lead? Stay tuned.

Acknowledgments

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A number of resources have been used in trying to gain an historical perspective on electric power systems. I have attempted to list all of these sources, but have not attempted to indicate within the paper specific references. This in no way is meant to imply that the information was original to this paper but rather is a reflection of the interrelationships that mark the evolution of a technology as comprehensive as aerospace electrical power. I have referred freely to several of these resources and must acknowledge their primary roles in the preparation of this paper. These include the works of Levoy and Boice, Haag, and Hutton, and the text by Hyder et al.

Errors of omission and errors in fact are the sole responsibility of the author for which I apologize.

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