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Nuclear Electric Spacecraft for Unmanned Planetary and Interplanetary Missions

Dwain F. Spencer
Leonard D. Jaffe
John W. Lucas
Owen S. Merrill
John I. Shafer

JET PROPULSION LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
PASADENA, CALIFORNIA

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ABSTRACT

Advanced electric propulsion spacecraft are shown to exhibit a unique capability in performing planetary and interplanetary missions. In particular, performance analyses indicate that an electric propulsion spacecraft of 45,000 lb initial weight can perform all fifteen high-energy missions which are of interest to space scientists. Comparable chemical and nuclear heat exchanger spacecraft can perform only seven and nine of these missions, respectively.

Based on what are believed to be realistic estimates of system weight, the two powerplant types considered, thermionic and turbogenerator, appear to have comparable specific weights (12 to 14 lb/kwe) at the 0.3 to 1.5 Mwe power level. This provides a strong incentive to adequately support both concepts, at least in the early phases of research and development.

Systems considerations regarding the utilization of these powerplants indicate a preference for the static (thermionic) type. For example, attitude control requirements of the spacecraft are minimized with the thermionic system by (1) eliminating rotating mechanical devices and (2) providing a smaller radiator area, thus a less severe dynamic stability problem at the same power level.

Probably the most important factor is the inherent reliability associated with a static system. In addition, demonstration of this reliability is feasible in ground testing. Because low-thrust propulsion units must operate for unusually long times, this factor will be extremely important.

It is recognized that many problems exist in developing a flyable thermionic or turbogenerator powerplant; however, based on the preceding arguments, a highly accelerated research and development program for both systems is deemed warranted.

I. INTRODUCTION AND PERFORMANCE ESTIMATES

For nearly two years, the Jet Propulsion Laboratory has been investigating the feasibility of utilizing electric propulsion systems in spacecraft for deep-space missions. Early studies were oriented toward an understanding of their unique characteristics and spacecraft integration problems as well as a preliminary evaluation of the SNAP 8 propulsion system for a spacecraft with a Mars orbiter mission or a probe out of the plane of the ecliptic. In October 1960, an internal interim report (Ref. 1) was issued to summarize the status of the studies.

It quickly became apparent that the *Centaur*-Double SNAP 8 vehicle would be limited in its capability to perform scientific missions in deep space. Consequently, more recent studies (Ref. 2) have focused on the characteristics and time schedule for more advanced electric systems.

The approach to the problem has consisted of the following:

1. Determine the space-science missions that are desirable for the next two decades. Include, if possible, today's estimate of the relative desirability and magnitude of those missions, i.e., the fraction of available vehicles that would be devoted to each type of mission.
2. Estimate the net or scientific payload capability of existing and planned vehicles for those missions.
3. Determine the relative advantages of appropriate vehicles and their state of development, and estimate the time schedule for availability.
4. Narrow the field of vehicle systems and recommend one course, or possibly alternate courses, of action.

This report covers a less ambitious area than that indicated, but studies have progressed to the point that preliminary information can be provided and pertinent conclusions can be drawn.

Before discussing performance capabilities, it seems desirable to define several terms and indicate the assumptions made. Gross payload will refer to the weight of the spacecraft at its destination less the weight of the propulsion powerplant:

$$W_{pa} = W_{sc} - W_{pp}$$

Thus, the gross payload normally includes the weight of the scientific instruments, telecommunications, guidance and control, and their associated structures.

For the electric systems it is convenient to refer to the powerplant weight in terms of its electrical gross power level and specific weight α; thus,

$$W_{pp} = \alpha P_g$$

Discussions will exclude secondary power and will be confined to an electric propulsion system that consists basically of the following: (1) a nuclear reactor which feeds, (2) a power conversion system, either dynamic or static, which converts the thermal energy of the reactor to electric energy, and (3) an electric thrust unit with its associated propellant tank and controls. The performance calculations are based on ion-thrust units, but subsequent discussions on the power-conversion systems are applicable to magnetohydrodynamic and other electric thrust units as well.

The *Nova* was conceived as a first stage, with six *F-1* engines and liquid hydrogen-liquid oxygen in the second and third stage. Such a vehicle would have a launch weight of approximately 7,000,000 lb and a payload capability of about 300,000 lb in a 300-nm Earth orbit. A fourth stage with LOX-H_2 would be used to achieve Earth escape. In the case of the orbiter missions, solid-propellant retrorockets with a specific impulse of 300 sec and propellant-to-gross-propulsion-weight ratio of 0.9 would be utilized.

The electric missions were computed using constant thrust directed tangentially in the spiral trajectories and an Irving and Blum optimum-thrust program in the heliocentric trajectory (Ref. 3). Consequently, payload weights are slightly optimistic for future practical missions—probably 5 to 10%.

Table 1 indicates the specific weight assumed for the electric propulsion systems; additional details are given in Sec. II, which compares thermionic and turbogenerator systems. The specific weights are compared for 300- and 1500-kwe power levels.

Table 1. Specific weight of turbogenerator and thermionic electric propulsion systems

Component	Specific weight, lb/kwe			
	Turbogenerator		Thermionic	
	300 kwe	1500 kwe	300 kwe	1500 kwe
Reactor and conversion system (described in Sec. II)	11.3	10.8	10.1	9.9
Thrust unit, propellant tank, associated plumbing, and controls	1.5	1.0	1.5	1.0
Contingency	1.0	1.0	1.0	1.0
Total	13.8	12.8	12.6	11.9

Figure 1 compares the gross-payload estimates for the *Nova* chemical vehicle, a direct heat-exchanger type nuclear system sized for the *Saturn S-1* booster, and the 300 and 1500 kwe electric systems for fifteen planetary and interplanetary missions of interest. They consist of seven probes or flybys (to the Sun, Mercury, Venus, Mars, Jupiter, Saturn, and Pluto), five planetary circular orbiters for Mercury, Venus, Mars, Jupiter, and Saturn, and extra-ecliptic probes with trajectories inclined at 15, 30, and 45 deg to the plane of the ecliptic. In the case of the orbiter missions, performance was calculated for circular orbits at an altitude of 500 mi for Mercury, Venus, and Mars and 2000 mi for Jupiter and Saturn. Reference 1 has pointed out the unusual scientific interest and value in experiments conducted well outside the plane of the ecliptic. It should be noted that, in the case of solar probes, scientific interest rises very rapidly as the spacecraft becomes capable of reaching or penetrating the inner solar corona (20 solar radii or 0.094 AU).

One other class of missions especially interesting to space scientists, the planetary orbiter-lander, has not been computed as yet. However, the orbiter-lander will be a logical extension of orbiters; presently shown orbiter payloads should give some indication as to whether sufficient weight is available in them to permit construction of a capsule that would be ejected from the orbiting spacecraft for a landing. Note, therefore, that the ability to perform orbiter missions may be doubly valuable if it appears that the spacecraft would be capable also of the orbiter-lander missions at a later date.

Missions have been listed in Fig. 1 approximately in the order of increasing difficulty. The number at the top of each bar graph indicates total transit time from launch to final mission destination. Vehicles studied are shown in the legend:

1. The chemical *Nova* is estimated to have a 300-nm Earth orbit capability of 300,000 lb.
2. The *Saturn* nuclear vehicle utilizes a RIFT type nuclear stage on top of the *S-1 Saturn* first stage and is estimated to have a payload capability of 79,000 lb in Earth orbit. The spacecraft stage utilizes an advanced nuclear heat exchanger system with an initial thrust to weight ratio of 0.3 *g*. The dead weight of the final nuclear spacecraft has been based on a compilation of weight estimates for a reactor of the advanced type.
3. The electric vehicles have an 8500-lb initial weight in 300-nm orbit with a 300-kwe powerplant, and a 45,000-lb initial weight in orbit with a 1500-kwe powerplant.

The designation "no mission" signifies either that the propulsion system is incapable of delivering any weight to its destination or that the gross weight was less than that required for spacecraft guidance, control, telecommunications, structure, and a minimum of scientific instruments.

The combined weight of the temperature control, guidance, attitude control, and telecommunication systems is taken as 1000 lb, although the actual weights obviously depend on the mission to be performed. In the case of the chemical and direct nuclear systems, an additional weight for the telecommunication power supply must be allocated. For the purposes of this report this was selected as 500 lb.

In the case of the nuclear systems, allowance has been made for thrust and interstage structure in the dead weight, thus an additional 500 lb which is the estimated weight of spacecraft structure for the electric and chem-

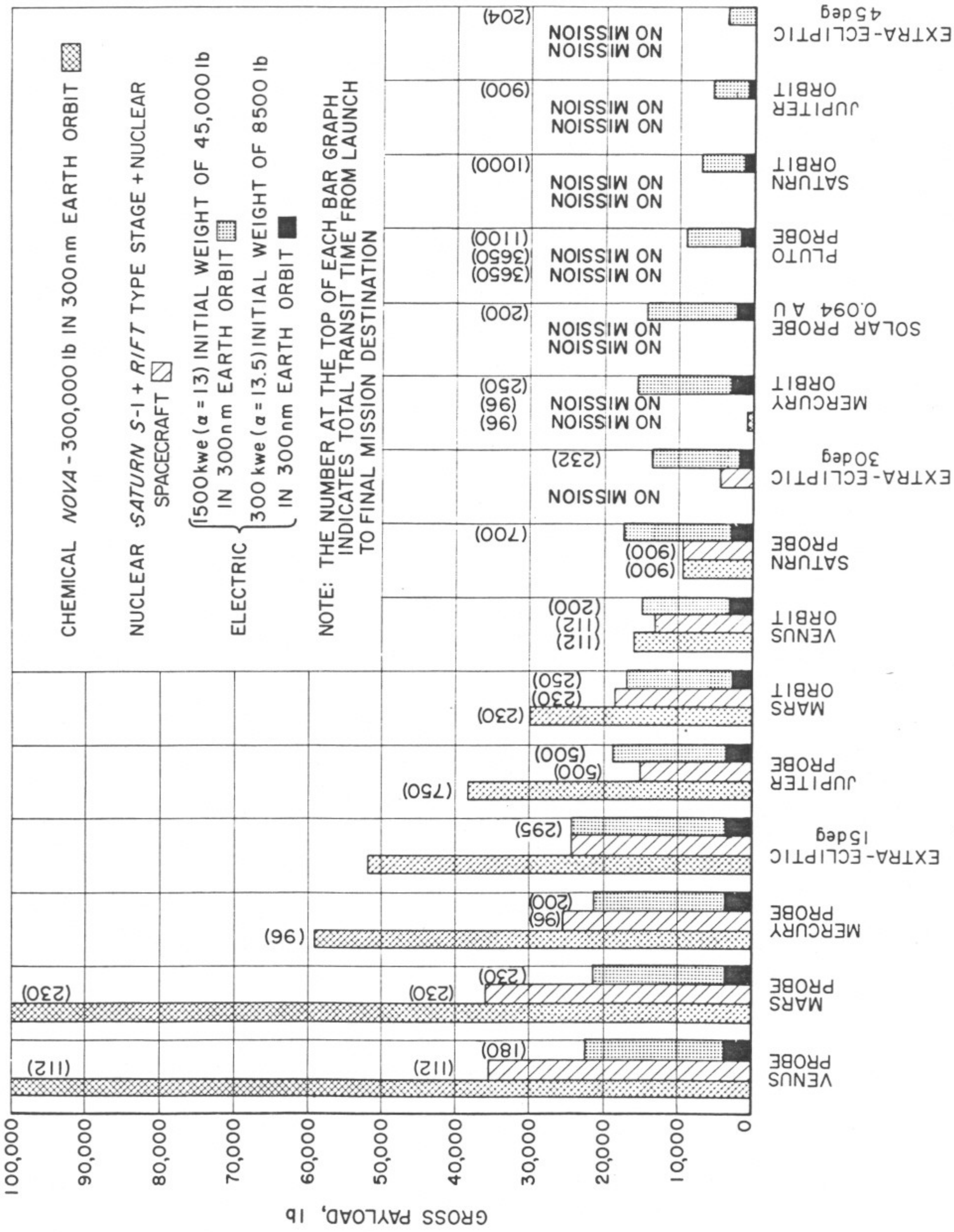

Fig. 1. Performance comparison of chemical, nuclear, and electric systems

ical systems must be deducted in their respective cases to present the results on a comparable basis. Thus in order to perform a useful mission with the chemical system a gross payload of at least 2000 lb must be obtained and with the nuclear or electric systems, 1500 lb. For example, the *Nova* can perform "no useful scientific mission" in a Mercury circular orbiter, even though its gross payload is approximately 500 lb.

Payloads for the chemical *Nova* are unusually high for the low-energy missions (about 50,000 to 100,000 lb) but rapidly drop off and become impractical or nonexistent for a Mercury circular orbiter, a Pluto probe, Saturn or Jupiter circular orbiters, a solar probe at 20 radii, and the 30- and 45-deg extra-ecliptic missions. The Mercury orbiter mission, as indicated, turns out to be much more difficult to perform than might be expected because of the extremely large excess spacecraft velocity that must be removed at encounter in order to go into orbit; the orbital velocity of Mercury itself, of course, is very high (29.75 mi/sec mean). If Mercury and Jupiter orbiters with large eccentricities and low perigees should prove to be acceptable to space scientists, *Nova* might provide meaningful missions in these cases also.

Although the three-stage chemical *Saturn C-1* is not shown, it is of interest to note that its performance would confine its use to Venus and Mars flybys (3000- and 3200-lb payloads, respectively).

The *Nova* has the greatest payload capability for low- and medium-energy missions, whereas the electric systems are superior for high-energy missions. The nuclear heat exchanger systems, on the *Saturn S-1* stage, nowhere play a singular role in the unmanned exploration of planetary and interplanetary space.

Table 2 summarizes the mission capability of the four vehicles shown in Fig. 1 with the addition of a 600-kwe electric system for use with a *Saturn C-1*.

The Table is broken up into three categories regarding performance: (1) not feasible, i.e., the final delivered gross payload is less than 2000 lb for the chemical systems and 1500 lb for the nuclear or electric systems; (2) marginal, those systems which deliver a payload within 1000 lb of the minimum allowable; and (3) definitely feasible, those systems which deliver a gross payload greater than the minimal requirements of (1) and (2).

Table 2. Vehicle capability for the fifteen missions

Vehicle type	Number of missions		
	Not feasible	Marginal	Definitely feasible
Chemical *Nova*	7	0	8
S-1 + 2 nuclear stages	6	0	9
300 kwe electric $\alpha = 13.5$ lb/kwe 8500 lb initial weight	3	3	9
600 kwe electric $\alpha = 13.5$ lb/kwe *Saturn C-1* (20,000 lb initial weight	1	0	14
1500 kwe electric $\alpha = 13.0$ lb/kwe 45,000 lb initial weight	0	0	15

Table 2 indicates that the *S-1* nuclear vehicle capability is comparable to the 300-kwe system with an initial weight of 8500 lb if viewed from a standpoint of the total number of possible missions. Each system is capable of performing 9 missions with three more marginal for the 300-kwe electric system.

The 1500-kwe electric system of 45,000-lb initial weight is capable of performing all of the indicated missions. This indicates that there is a strong incentive for a 1500-kwe electric spacecraft which will allow all missions of interest to be performed with a single spacecraft propulsion system.

The *Saturn C-1* is included to indicate the type of performance which is possible for an intermediate power level. It is assumed to utilize a cluster of two 300-kwe powerplants. If it is impossible to obtain a vehicle with a payload in the 45,000-lb weight class, the *Saturn C-1* would permit thirteen and possibly fourteen of the missions to be performed.

II. SPACECRAFT-SYSTEM COMPARISON OF THERMIONIC AND TURBOGENERATOR NUCLEAR POWER SOURCES FOR ELECTRIC PROPULSION

This section contrasts the characteristics of the thermionic and turbogenerator power generation systems. It covers a brief description of the systems, a weight breakdown, a comparison of system problems, and a discussion of the compatibility of the power generation system with the remainder of the spacecraft.

Both power generation systems were examined on what is believed to be a comparable basis. Although each system requires a considerable extrapolation from the state of the art, it appeared justifiable to consider advanced versions to provide an incentive for their development.

A. Description of the Conversion Systems

The dynamic conversion system employs a two-loop Rankine cycle with a nuclear-reactor heat source, as shown in Fig. 2. The reactor is cooled by the primary coolant, lithium, which is used to minimize system pressure and primary pumping power. The fission energy is transferred by the primary coolant to sodium in a shell and tube heat exchanger. The sodium vapor is fed to a turbine assembly, condensed in a waste heat radiator, and returned to the boiler to complete the secondary loop.

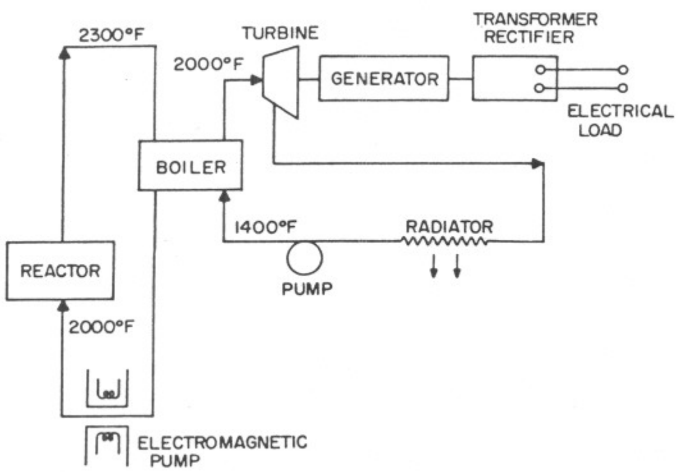

Fig. 2. Nuclear-turbogenerator power system

The turbogenerator delivers ac power to a transformer-rectifier which provides the desired output to drive the electric thrust unit. If an ion engine is employed, transformation and rectification will probably be required; however, if a magnetohydrodynamic thrust device is used, the generator may be capable of supplying the required electrical output directly.

The reactor exit temperature of 2300°F will provide superheated sodium vapor at a turbine inlet temperature of 2000°F. These temperatures are estimated to be the maximum allowable values commensurate with material limitations over the next decade. A radiator temperature of approximately 1400°F minimizes the required radiator area per unit power.

For comparison, a schematic of the thermionic reactor conversion system is shown in Fig. 3. In essence, the thermionic system utilizes the reactor fuel element as the electron emitter (cathode) and the inner surface of its cooling jacket as the collector (anode) to produce electrical power directly. The collector is electrically (and therefore, to some degree, thermally) insulated from the reactor coolant, lithium. This coolant is fed to a waste-heat radiator, where it rejects the excess thermal energy from the reactor and is recycled by use of an electromagnetic pump.

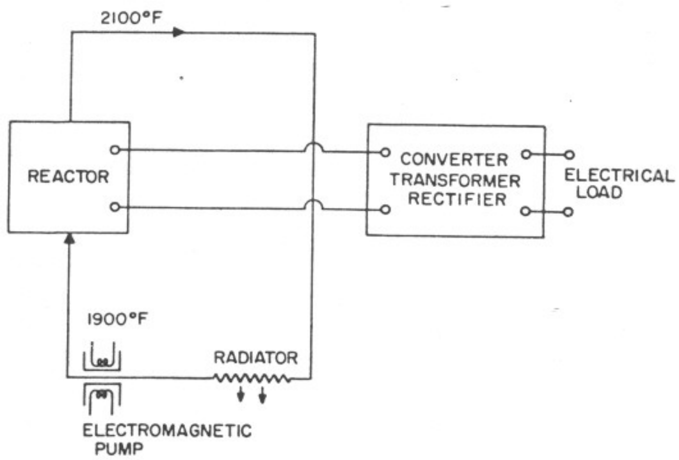

Fig. 3. Nuclear-thermionic power system

A reactor coolant temperature range of 1900 to 2100°F was assumed. Recent experimental tests indicate that operation of the anode at this temperature does not require a sacrifice in diode efficiency, and, of course, the high radiating temperature is desirable from a systems standpoint.

The overall specific weights of both systems appear comparable at 300 or 1500 kwe. Even though the static system eliminates rotating machinery, the weight of the T/R which is usually neglected, makes its weight comparable to that of the dynamic system. If either operation of the T/R at higher temperatures or high voltage direct output from the reactor becomes feasible, the thermionic system weight could possibly be reduced by approximately 1 to 2 lb/kwe. A similar argument regarding radiator reliability is possible with the turbogenerator system if multiple turbogenerators and coolant loops are utilized. This could produce a reduction in primary radiator weight of 2 to 3 lb/kwe with little loss in reliability.

These considerations appear to balance, and the specific weights of the two systems would be still comparable. In addition, they allow contingency (or confidence) in system weights and indicate that the electric system performance estimates are realistic.

C. *Comparison of System Problems*

It is recognized that any systems comparison is dependent upon the solution of many complex and interrelated individual component and subsystem problems. A discussion of the major problems of each system that must be investigated and solved is presented in this section. It should be stressed that in the development and design of either system, an overall spacecraft viewpoint must be taken.

Some key problems of turbogenerator systems arise because the units must operate in an environment which is novel in several respects. The low thrust-to-mass ratio of electric propulsion spacecraft causes an essentially free-fall condition within the spacecraft during normal operation. Condensation of the working fluid which drives the turbine must take place under this condition. Either the condensation must take place directly in the radiator in which case the radiator design must be compromised, or condensation occurs in an intermediate heat exchanger whose weight must be added to the total system weight.

In free fall, rotating shafts are susceptible to a number of modes of bearing instability. Critical speed whip can probably be overcome much easier than translational whirling of the shaft which could cause vibration of the entire spacecraft. Bearing wear may, in turn, be accentuated by such a condition.

Associated with the gas phase in the dynamic system is a fluid circulation control problem. On similar systems in the past, this has given rise to development difficulties which required very substantial effort to rectify.

Because of the necessity to minimize specific weight, both the turbogenerator and thermionic systems must operate at high temperatures. The consequent use of high-temperature liquid metal working fluids leads to the problems of mass transfer of wall material in both systems. In the turbogenerator system, the high operating temperatures together with the high shaft speeds may produce severe creep problems in the turbine blades.

In a thermionic system which incorporates an electromagnetic pump, the above-mentioned problems associated with free fall are of no concern, however, there are problems peculiar to the thermionic system that will require investigation and solution. The major problems lie in the area of materials, converter technology, and thermionic reactor design and operation.

Foremost of these are (1) the development of a nuclear thermionic fuel element having a 2000 to 2500°K fuel center temperature, refractory clad fuel (or unclad) with desirable thermionic emission properties, structural stability, and suitable control of fission product gases, (2) the development of high-temperature cesium-resistant, radiation-resistant ceramic insulators and ceramic to metal seals, and (3) the design, fabrication, and testing of a prototype thermionic reactor, including consideration of such problems as integration of converters into a fast reactor, power density limitations, high void fraction, and non-uniform power and temperature distribution effect on converter operation.

D. *Compatibility of Power Generator System and Spacecraft*

Having summarized the major problems inherent in each of these systems, their compatibility with the spacecraft will now be discussed. The radiation environment produces one of the most difficult integration problems. Although shield weight has been allotted for transitorized electronics, it may not be sufficient to allow certain measurements of particle fluxes or the use of photographic equipment (for mapping the planets). This may require a weight penalty or, at least, an additional complexity in spacecraft or instrument design.

Another internal environment problem arises owing to the use of the ion engine. Stray magnetic fields from engine feed circuits or even the beam itself may make it difficult to perform magnetometer experiments.

Thermal control of such spacecraft with operating temperature extremes from 50 to 2000°F will certainly be unique. The prospect of utilizing some of the electrical power for refrigeration or heating may offset this problem somewhat.

Packaging problems are accentuated by the addition of the reactor and conversion system to the spacecraft, especially at the 1500-kwe power level. At this power level the total radiator area is approximately 1100 ft^2 (550 ft^2 of plan area) for the thermionic system and approximately 2100 ft^2 (1050 ft^2 of plan area) for the turbogenerator system.

If a cylindrical radiator structure is employed, it only radiates from one side. This requires a substantial increase in length over a flat plate structure, and may present a packaging problem within the boost vehicle shroud. Another drawback of the cylindrical radiator structure is the fact that the lower temperature T/R and generator radiators see the interior of the primary radiator, thus lowering their efficiencies. On the other hand, the erection of a flat plate radiator and the attendant attitude control problem due to low natural frequency and stiffness complicate the operational aspects of the spacecraft.

In comparing the thermionic and turbogenerator system influence on the spacecraft, the primary difference is the demand on attitude control of the latter system. The problem of damping startup torques from the rotating machinery either complicates the spacecraft design or the operations of separation from the boost vehicle stage. Fluctuations in machinery operation produce constant demand on the system which must be compensated. Should flat plate radiator structures be required, the turbogenerator system must operate with a larger aspect ratio radiator for a given power level. This increases meteoritic impact probability and produces a greater upsetting moment upon impact. Again this must be considered in the design of the attitude control system.

The prospect of partial power operation seems to favor the static system. However, if multiple loops are utilized in the turbogenerator system this apparent advantage of the static system may disappear. On-off operation offers problems of storing the working fluid in either system, especially during attempts to re-initiate operation after some dormant period of time. This problem, however, is more complex for the turbogenerator system because of the rotating machinery. Of course, operation at partial or zero power is important in order to conserve the usable lifetime of the system.

The most important criteria for comparison of the thermionic or turbogenerator system are reliability and lifetime. In order to successfully utilize these systems in an interplanetary program, they must perform as required for the duration of the flight.

Although reliability of any piece of equipment is difficult to evaluate until units have been constructed and tested, past experience has shown that properly designed static systems are more reliable than those which contain valves, pumps, and other moving machinery. Because low-thrust propulsion units must operate for unusually long times, this reliability factor will be extremely important in selecting system components.

III. SUMMARY AND CONCLUSIONS

Although electric propulsion systems will undoubtedly be used for other purposes, they will probably find their greatest area of application as primary propulsion for planetary and interplanetary missions. The studies described in this report show that 50% of the unmanned planetary and interplanetary missions of major scientific interest examined cannot be performed by chemical systems based on *Nova,* and approximately 40% cannot be performed by nuclear heat-exchanger systems boosted by *Saturn S-1.* On the other hand, almost all of these missions can be performed with an electric-powered spacecraft which uses a *Saturn C-1* booster, and most can be performed by systems utilizing an electric-powered spacecraft with an initial weight of 8500 lb to 300 nm Earth orbit. All of the missions can be carried out by an electric spacecraft starting with 45,000 lb in 300 nm Earth orbit provided the vehicle becomes available. As a result, for an electric propulsion system to be useful for planetary and interplanetary missions, a power level of about 300 to 1500 kwe delivered to the drive unit seems desirable during the next decade.

Of the possible electric propulsion systems, those employing thermionic conversion of reactor power appear at this time to offer some advantages in system integration and perhaps in performance. The system reliability of the thermionic plant will probably be greater because: (1) the propulsion subsystem will have no moving parts, and (2) a greater confidence level can be obtained from ground testing due to its relative insensitivity to free-fall conditions. Attitude-control problems will be simplified by the freedom from the large gyroscopic moments associated with turbogenerators and the smaller size of radiators associated with higher radiator temperatures. Both restarting and operation at reduced power levels for long-time communication also seem simpler with thermionic conversion.

Although the concept of thermionic power conversion has been established, additional research and development is needed on thermionic materials, converter technology, and design. Turbogenerator systems, on the other hand, have been in existence for a long time though the present need is for one which will operate in a very different environment. Although many of the problems are novel in the development of either of these systems, they all appear to be amenable to engineering solution.

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