Technical Report No. 32-158

Systems Engineering of a Nuclear-Electric Spacecraft

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ABSTRACT

Studies have shown that nuclear-electric propulsion systems will provide superior payload capability and unique advantages over chemical systems for high-energy deep-space missions.

Conceptual design studies of unmanned spacecraft employing nuclear-electric propulsion systems have been undertaken to determine some of the major integration problems. Early recognition of these problems will help to stimulate the development effort that will be required to bring these systems into fruitful utilization.

Typical designs under consideration for interplanetary missions for the next decade employ a nuclear reactor providing thermal energy to a turbogeneration system which, in turn, supplies electrical power to an ion engine for primary propulsion and additional utility power for guidance and control, powered-flight radio transmission, instrumentation, et cetera.

The major systems and components which form a complete spacecraft are listed in this Report, and a review of the significant physical and operational characteristics of these various systems and components which affect spacecraft integration is made. Conceptual configurations and detailed weight studies for a 60-kwe Venus-capture spacecraft and a 1-Mwe Jupiter-capture spacecraft are shown to illustrate typical physical arrangements based on the various hardware constraints. From these configurations, the major development goals are ascertained and summarized.

I. INTRODUCTION

Since the end of World War II, chemical propulsion, both liquid and solid fueled, has done an admirable job of providing the necessary propulsive capability for the initial phases of space exploration. Now, however, the desire to boost greater payloads into space and to accomplish higher-energy missions to the more distant major planets has created requirements which cannot be met entirely by chemical rocket systems.

The use of nuclear systems, both nuclear rockets and nuclear-electric devices, in place of or in conjunction with various chemical systems, holds promise in meeting these challenging new requirements. Figure 1 illustrates this point graphically by showing a comparison between some typical chemical systems and various conceptual nuclear systems.

The additional desire of providing relatively large amounts of electrical power in space for communication and scientific exploration makes the use of the nuclear-electric systems of particular interest to those concerned with deep-space investigations. The attainment, for the first time, of high-resolution video pictures and radar probing of planetary surface and atmospheric details appears feasible through the use of this power.

Progress in the development of various types of electric-thrust devices and nuclear-electric power sources indicates that serious consideration must be given now to the systems engineering or integration aspects of nuclear-electric systems if the utilization of such devices is to proceed without interruption. Given sufficient backing, nuclear-electric spacecraft could be performing useful missions within this decade.

It is the intent of this Report to review briefly some of the major integration problems and constraints which must be considered in the design of unmanned nuclear-electric deep-space instrumented scientific probes and to illustrate these considerations with a description of two spacecraft concepts. In addition, the major hardware-development goals which must be accomplished in bringing these designs to realization are discussed. Early recognition of these goals will help to stimulate the necessary development effort.

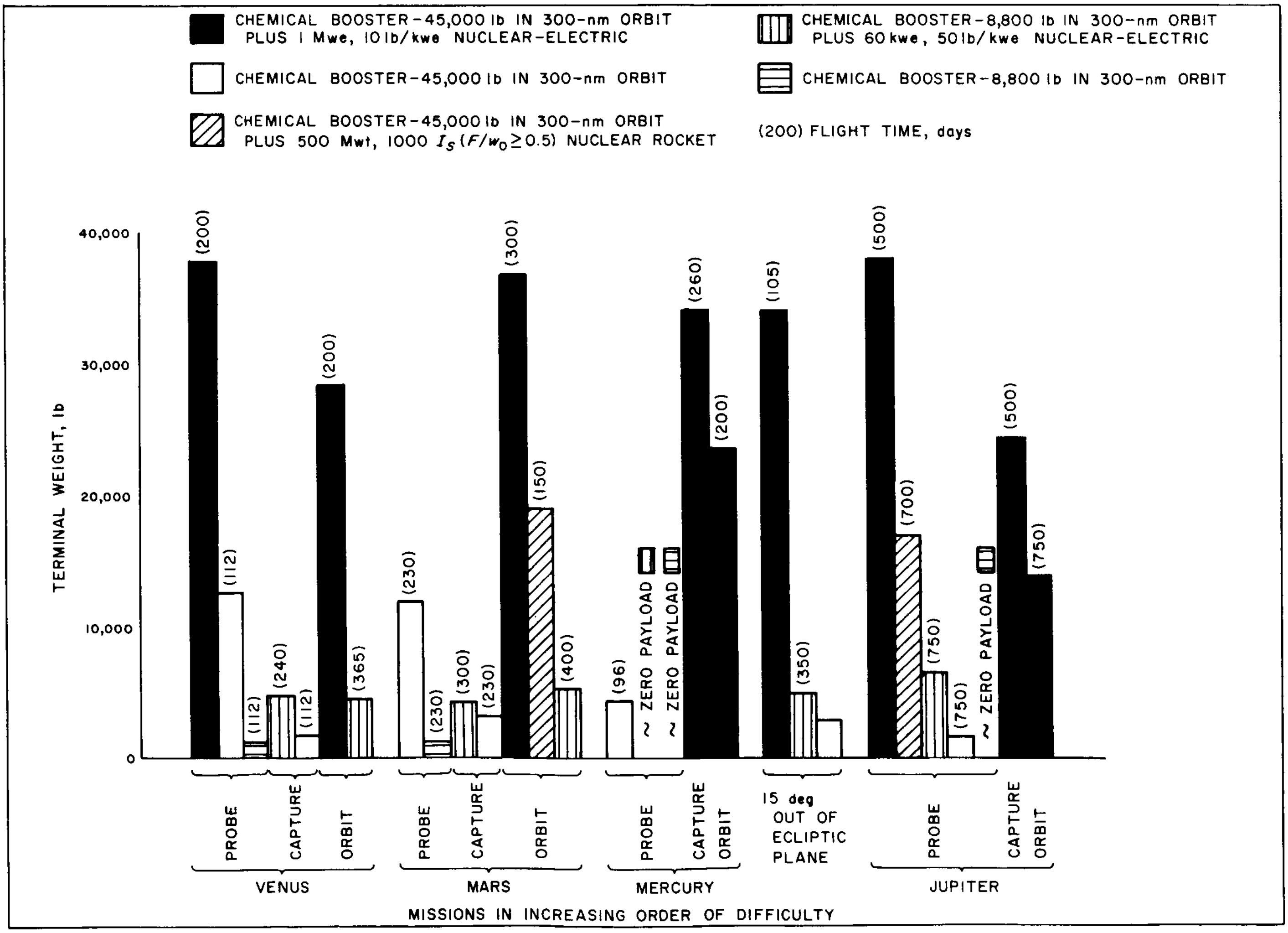

Fig. 1. Spacecraft missions comparison

II. CONFIGURATION REQUIREMENTS

The designing of an optimum spacecraft configuration obviously must begin with defining, in a gross manner, such initial inputs as booster capabilities and availability, the development status of major long-lead components, mission requirements, available monetary funding levels, schedule requirements, et cetera. From these inputs may be derived a conceptual design which, by a series of successively more sophisticated optimization analyses involving the interrelation of each system with another, will ultimately result in an optimum hardware design. In this Report, the initial inputs for the two illustrative configurations discussed are simply assumed in order that the review may proceed directly to the conceptual phase.

Investigation of booster capabilities and the state of the art of nuclear-electric power sources leads to the conclusion that propulsion-power levels of a nominal 60 kwe to 1 Mwe should be considered initially with total spacecraft weights ranging from about 8000 to 50,000 lb, respectively. It appears that, at least for the immediate future, the nuclear-electric power sources for deep-space propulsion will probably employ a nuclear reactor providing thermal energy to a turbogenerator system, which in turn would supply electrical power to an ion engine for primary propulsion and additional utility power for guidance and control, powered-flight radio transmission, instrumentation, et cetera. Later systems will undoubtedly employ more advanced direct-conversion cycles which will eliminate many of the problems associated with rotating machinery. Powerplant specific weights of about 50 lb/kwe at first, gradually improving to 10 lb/kwe, appear feasible.

The booster requirements to lift the electrically propelled spacecraft initially into space can be met adequately by chemical systems, at least in the immediate future.

Mission considerations indicate that useful scientific measurements could be performed on almost any type of nuclear-electric spacecraft flight, but it appears desirable to concentrate on those missions which are most difficult, or impossible, to accomplish with pure chemical systems. Of particular interest for early flights are planetary orbiters for the near-Earth planets, Mars and Venus;[1] fly-bys and planetary-capture missions for the more distant planets, Mercury, Jupiter, Saturn, et cetera;[2] and flights out of the plane of the ecliptic to enable various instruments to look "down" upon the solar system. Later, more ambitious missions may include separable landing capsules for both the near and distant planets, planetary orbiters about the distant planets, and solar probes making close approaches to the Sun. Figure 2 illustrates the trajectories for these various types of missions. In this Report, only simple planetary-capture missions will be discussed.

A booster capability having been chosen, the characteristics of the nuclear-power system established, and a selection made of a mission requirement, various over-all

[1]Planetary orbiters may be defined as missions in which the spacecraft establishes a near-circular orbit of relatively low altitude about the target planet.

[2]Capture missions may be defined as missions in which the spacecraft matches the planet's orbital velocity such that the spacecraft is barely caught in a highly elliptical orbit about the planet. Capture missions are easier to accomplish than corresponding planetary-orbiter missions.

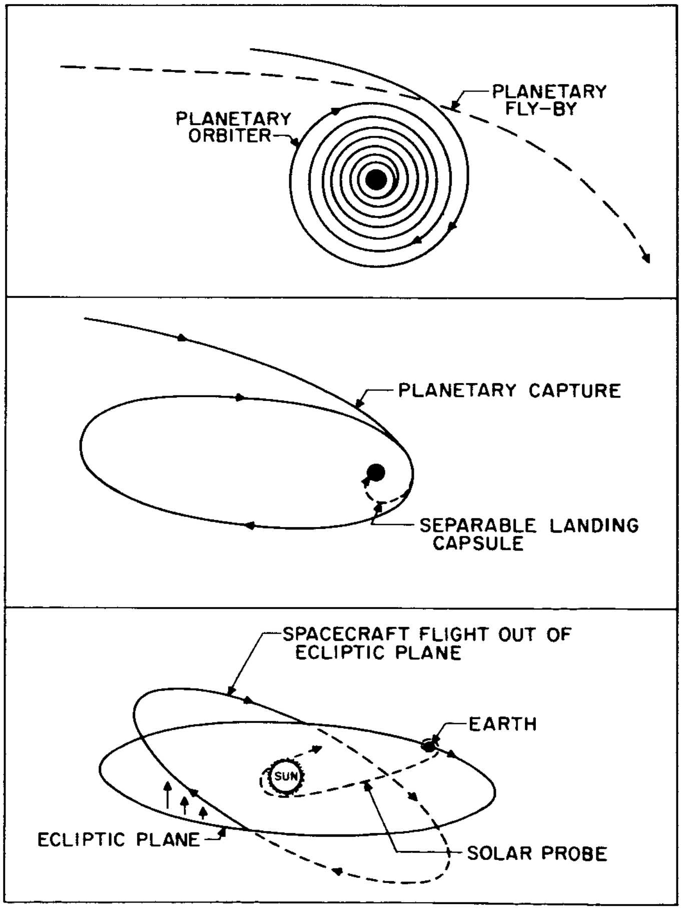

Fig. 2. Typical mission trajectories

spacecraft-performance parameters may be calculated. As an aid to understanding the significance of these parameters, it is worthwhile to look first at a typical nuclear-electric spacecraft flight trajectory.

It is generally agreed, at least for the immediate future, that a nuclear-electric spacecraft would first be boosted into a long-lifetime Earth orbit prior to initiation of the nuclear-electric propulsion phase of flight. (A 300-nm orbit may be considered typical.) The spacecraft would then be separated from the booster, the power generation initiated, and the electric-propulsion period of the flight begun.

For a representative 1-Mwe-propulsion Jupiter-capture mission, the electric-propulsion flight period would consist of about 70 days of powered flight in a spiral path of approximately 295 turns, starting in the Earth orbit and gradually increasing in radius until escape (as shown in Fig. 3). This would be followed by a heliocentric transfer (Fig. 4) of about 145 days of powered flight, 400 days of coasting, and another 60 days of powered flight, ending up in the vicinity of the planet Jupiter. During the spiral portion of the flight, the thrust vector would be directed

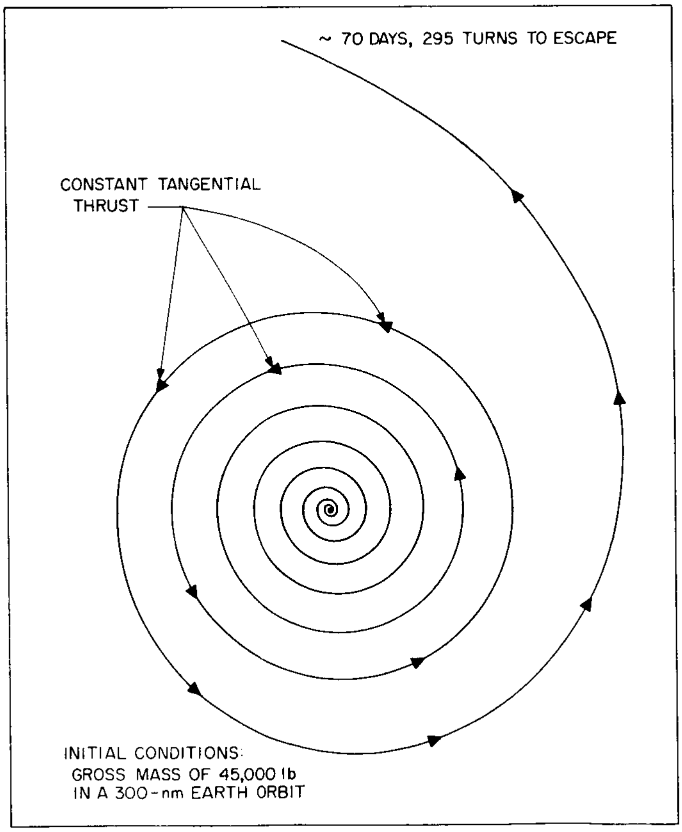

Fig. 3. 1-Mwe-propulsion Earth-escape spiral

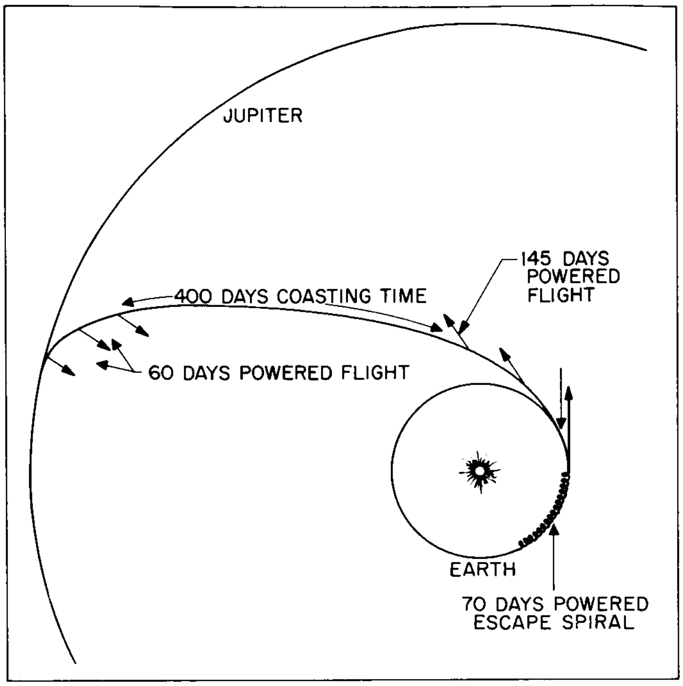

Fig. 4. Jupiter capture; 1-Mwe-propulsion heliocentric transfer trajectory

tangentially to the flight path, as shown in Fig. 3. The heliocentric portion of the flight would be characterized by a constantly changing thrust-vector orientation, as shown in Fig. 4. If guidance constraints were properly met, then the spacecraft would be caught in a highly elliptical orbit about Jupiter approximately 675 flight days after initiation of the electric-propulsion portion of the flight.

During the electric-propulsion thrust periods, various space measurements and functional data from the spacecraft systems could be transmitted back to Earth by a moderately powered transmitter.

For the coasting period, several modes of operation are feasible, depending upon the degree of sophistication of the particular power system employed. As examples, the power-generation system could be throttled back to the 10-to-20% level to provide only enough power to meet spacecraft utility requirements; or, if throttling capability is too ambitious, as will probably be the case for early-generation systems, the unused ion-engine power could be dumped into a resistive load and radiated into space as waste heat; or, finally, the power could be utilized to operate a high-powered transmitter and some major mid-flight space experiments.

Upon arrival at the target planet following the last thrust phase, the major portion of the power would be utilized for the primary planetary experiments and the operation of a high-powered transmitter. The spacecraft would remain in this mode until the power source expired or was commanded off.

Referring again to the Jupiter-capture spacecraft as an example, a requirement to produce power to operate the scientific experiments upon arrival at the planet for anywhere from a few months to a year, in addition to the 675-day flight period, implies that the power system should have a total lifetime of 2 to 3 years. Early systems, however, will probably be capable of only about one year of operation and will be limited to much less ambitious missions.

With the preceding picture of a representative trajectory in mind, some of the performance parameters can be reviewed and some of the major optimization trade-offs which can be made can be investigated. Figure 5 represents a typical performance estimate for the example of a Jupiter-capture spacecraft.[3] Illustrated in the Figure are the following significant points:

Assuming that the booster vehicle has a capability of boosting a 45,000-lb spacecraft into a 300-nm Earth orbit,

[3]The performance values shown in this illustration are based on a variable-thrust heliocentric transfer analysis. Propellant-consumption values for the constant-thrust plus coast-period transfer discussed in this Report will be 10 to 15% higher than those implied in the illustration.

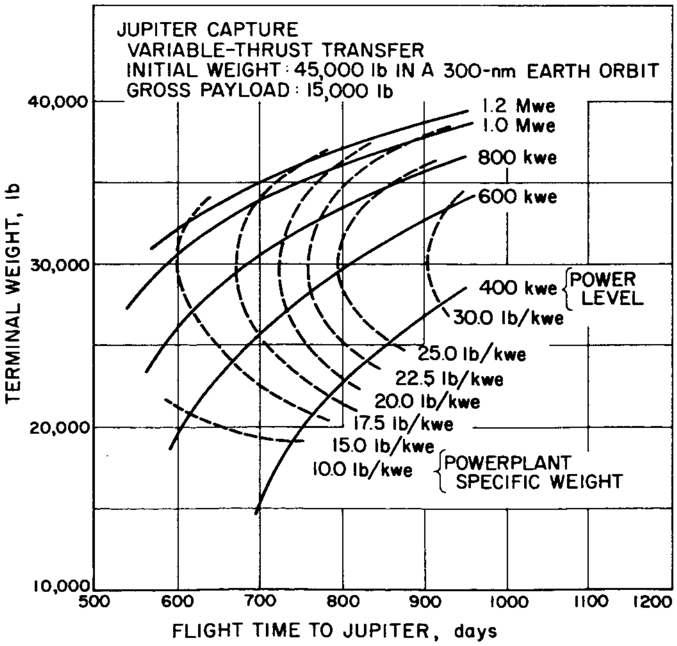

Fig. 5. Performance summary for a Jupiter-capture mission

and assuming that the gross payload[4] required by the mission is 15,000 lb, then it can be seen that, for any given powerplant-output level to the thrust device,[5] the longer the allowed flight time the heavier the powerplant may be, as indicated by the increasing specific-weight values. From another viewpoint, if the powerplant weight remains fixed for a given power-output level, then longer flight times will allow delivery of greater gross payloads to the target planet. In either case, the increase in allowed weight results from a decrease in required propellant load which follows from an increase in operating specific impulse at the longer flight times. For other booster capabilities, the relative proportion of powerplant, payload, and propellant weights will remain essentially the same.

Another interesting point which can be noted in Fig. 5 is that, for any given total flight time and gross payload weight, there is one power level which will allow for the highest powerplant specific weight. That is, for greater or smaller power-output levels about this point, the powerplant must have a lower specific weight to deliver the same gross payload in the same flight time. Again, as the flight time is increased with the gross payload still remaining fixed, the payload can be delivered with higher specific-weight powerplants at lower power-output levels.

To summarize this discussion of performance considerations, in selecting the powerplant from a hardware standpoint, it would probably be desirable to choose the lowest power-output, highest specific-weight powerplant to deliver the required payload in the shortest time. Unfortunately, these factors work against each other; therefore, one of the parameters must be arbitrarily fixed. As the range of flight times shown in Fig. 5 is already in the realm of system lifetimes which will offer a very severe challenge in terms of actual achievement with real hardware, it appears that the tendency would be toward selecting the shortest flight time possible employing a powerplant with a reasonable power-output level and specific weight.[6]

[4]The gross payload in this analysis is defined as the weight necessary for all of the spacecraft systems other than the power-generation system. This includes propellant tankage, the thrust device, structure, guidance and control, telecommunications, instrumentation, the scientific payload, etc.

[5]In this simple analysis, it is assumed that the entire output of the power source goes to the primary thrust device. In refining the analysis, with more involvement in the actual hardware design, it is found that an additional 10-to-30% power output must be made available to meet power-conditioning losses and utility requirements.

[6]For a further discussion of performance trade-offs see ARS Preprint No. 2224-61, "Performance of Nuclear-Electric Propulsion Systems in Space Explorations," by E. W. Speiser.

There are obviously other system inputs which must be considered in these trade-off studies; for example, the effect of thrust-device efficiency, specific impulse, and thrust level; power-conditioning system efficiency; utility-power requirements; et cetera. However, these are beyond the scope of this Report and will not be discussed here.

The next item to be investigated briefly comprises the actual systems which together make up a complete spacecraft.

Table 1 lists the major systems and subsystems or components of significance. For the range of power levels and spacecraft weights of concern for this Report, the following general estimates can be made about the sizes and characteristics of these systems and components:

1. A typical nuclear-reactor energy source may range in size up to about 20-in. diameter by 30-in. length, with coolant outlet temperatures up to 2000°F. The reactor, which is an intense source of nuclear and thermal radiation, should be placed away from sensitive equipment and be relatively free of surrounding structure.

2. The turbogenerator units probably will be installed as counter-rotating pairs to minimize gyroscopic effects upon the spacecraft's attitude-stability characteristics; one unit may range in size from 30-in. diameter by 40-in. length to 30 by 70 in., depending upon the power level required. High temperatures and some radiation are also characteristic of these devices.

3. The primary nuclear-radiation shield will generally be located adjacent to the reactor. It will require either active cooling or, at least, the ability to radiate heat to space. This heat is acquired both by thermal radiation from the reactor and by gamma-ray absorption by the shield material. A typical shield thickness may range from 2.5 to 3.5 ft, depending upon the particular reactor in use and the required operating lifetime of the spacecraft systems.

4. The primary heat-rejection radiators, which are required by the thermal-to-electrical conversion cycle, will probably operate at temperatures of about 700 to 1500°F and will involve the installation of large surfaces which, because of their vulnerability to micrometeorite damage, will require special protection considerations. Radiator areas may vary from 1000 sq ft to over 3500 sq ft, depending upon operating temperatures and power level.

5. The power-conditioning equipment, like other electronic systems on board the spacecraft, will be both temperature- and nuclear-radiation-sensitive. This equipment should be both shielded and cooled. Power losses of up to 10% of 60 kwe to 1 Mwe, depending upon the spacecraft operating power level, indicate the magnitude of the cooling problem. Closely regulated multiple voltage output levels, ranging from kilovolts for the ion-engine supply down to 28 volts for utility functions, will be characteristic.

6. The heat released by the power-conditioning-system transformers and rectifiers, as well as other electronic components, may require secondary radiating surfaces at relatively low operating temperatures with active cooling loops. Temperatures of 200 to 600°F, with areas of 100 sq ft to over 1500 sq ft, might be expected for the power levels under discussion. As in the case of the primary radiators, micrometeorite protection must be considered for the secondary radiators.

7. The primary ion motor will probably consist of a cluster of smaller modular motor assemblies and may range in total frontal area up to 15 sq ft. The cluster will generally be located so as to facilitate thrust-vector control of the spacecraft.

8. Propellant tanks for most propellants under consideration will usually be operated at elevated temperatures to produce the desired feed pressures and must function in essentially a zero-*g* environment. The size of the propellant tank will obviously be dependent upon the propellant selected and the total quantity required by the mission. As a typical example, the Jupiter-capture spacecraft discussed previously would require a tank approximately 130 cu ft in volume operating at about 600°F.

9. The electronic equipment, which would include the guidance and control systems, functional instrumentation and telecommunication equipment, and the scientific payload, will be both radiation- and temperature-sensitive and must be appropriately located, cooled, and shielded, as in the case of the power-conditioning equipment. The cooling requirements for this equipment, as well as those of the power-conditioning components, would probably be met by a common cooling system. Typical volume requirements for the electronic equipment might run from a few up to 60 cu ft, depending upon mission and power levels. Small sections of secondary nuclear-radiation shielding might be

utilized within the electronic packages to provide additional protection for some highly sensitive components. A steerable high-gain parabolic antenna of about 10-ft diameter will generally be employed by the communication system.

10. The structure must, of course, be lightweight and strong and must maintain its integrity under high temperature gradients and intense nuclear-radiation fluxes. Associated with the structure may be some additional thermal-control equipment to handle varying external loads, such as the Sun's thermal radiation. Controllable shutters to vary total emissivity might be employed.

Table 1. Electric-propulsion spacecraft systems

Systems	Subsystems or major components
Power-production equipment	Reactor Turbogenerator units Nuclear-radiation shield Primary radiators Heat-transfer loops and controls
Power-conditioning equipment	Transformers Rectifiers Regulation controls Secondary radiators
Ion-motor cluster	Ion-motor modules Controls
Propellant system	Propellant tank Feed equipment Propellant
Guidance and control equipment	Sensors Information-processing unit Dynamic-compensation networks Control actuators
Functional instrumentation and telecommunication equipment	Instruments Transmitters Receivers Antennas
Structure and thermal-control equipment	
Scientific payload	

From the preceding initial definitions of the items to be integrated into an electrically propelled spacecraft, design rules resulting from the physical and operational characteristics of these items can be established which, in turn, lead to the various arrangements illustrated in this Report. The major rules adopted are as follows:

The elevated-temperature devices should be located together at one end of the spacecraft along with the primary high-temperature heat-rejection radiators. The primary sources of spacecraft-generated nuclear radiation are also high-temperature components and, hence, form a logical grouping at the high-temperature end.

The temperature-sensitive devices, which usually tend to be sensitive to nuclear radiation as well, such as the electronic equipment, power-rectification units, et cetera, should then be placed with a lower-temperature secondary heat-rejection radiator at the maximum practical separation distance from the higher-temperature items.

Those components which require nuclear-radiation shielding should be located in as small a shadow cone angle as possible in order to reduce primary-shield weight. In addition, other components which may not be particularly sensitive to radiation should also fall within the primary-shield shadow cone or should, at least, present the minimum possible cross-sectional area for radiation scattering. The primary shadow shield should be positioned at the hot end of the spacecraft, close to the major radiation source—the nuclear reactor—to provide the largest shadow-cone area in the vicinity of the sensitive equipment for the smallest possible total shield mass.

The primary-thrust cluster should be located near the ion-motor-control equipment on the cooler end of the spacecraft to minimize the length of the sensitive motor-control circuits. The thrust vector must obviously pass through the spacecraft center of mass.

The propellant load, which represents a significant changing quantity of mass during the flight, must be placed relative to the primary-thrust axis and the spacecraft center of mass so as to minimize changes in control stability as the propellant is consumed. Location of the propellant mass between the hot and the cold end of the spacecraft can provide some shielding aid and should be accomplished where possible.

The primary communication antenna must be positioned on the spacecraft to avoid interference from the thrust-unit beam (particularly in the case of an ion-engine beam); that is, the antenna should not be required to point through the beam, nor parallel to it, when communicating back toward Earth.

With respect to the radiator panels, to provide high radiation efficiency, the high-temperature and low-temperature radiator surfaces should not "see" each other. In addition, each radiator panel should be oriented to minimize the possibility of one portion of a given radiator "seeing" another part of the same radiator or other equipment on the spacecraft.

Radiator-panel orientation must also take into account the effect of the Sun's thermal-radiation load and the probability of micrometeorite impact. Ideally, the location of all of the radiator panels, both high- and low-temperature, in one plane (the plane of the ecliptic) will reduce the Sun load and may decrease the meteorite hazard.

Finally, the configurations should be capable of folding into a package of reasonable length and an appropriate diameter and must be capable of high acceleration loads in order to be compatible with the various booster vehicles which may be employed.

III. ILLUSTRATIVE SPACECRAFT CONFIGURATIONS

In order to illustrate the integration of the systems and components just described into a complete nuclear-electric spacecraft, taking note of the design rules previously summarized, two configurations are discussed.

The first configuration, shown in Fig. 6, illustrates the 1-Mwe-propulsion Jupiter-capture spacecraft which has been referred to earlier in this Report. This configuration, although very conceptual in illustration, is nevertheless fairly representative of high-power-level designs. This particular packaging arrangement resulted in circular, disc-shaped radiator panels which fold up, umbrella-fashion, for the boost flight. Although other packaging schemes might result in different radiator shapes, the general arrangement of systems and components would probably still hold true as being a reasonable approach.

In the illustration, the nuclear reactor, primary shield, and two turbogenerator units are shown nestled together within the large primary heat-rejection radiator at the forward end of the spacecraft. The radiator provides approximately 3500 sq ft of radiating surface at a temperature of about 1000°F.

At the aft end of the spacecraft, the propellant tank and ion-engine cluster can be seen. Sandwiched between the two, and taking advantage of any shielding acquired from the propellant mass, are the electronic packages (including the scientific payload) and power-conditioning equipment. Surrounding this equipment lies the large secondary heat-rejection radiator which is required to cool the electronic and power-conditioning components. This radiator provides about 1700 sq ft of surface, radiating at 200°F. Hopefully, development effort will result in future electronic components having higher temperature tolerance, which, in turn, will allow for higher secondary radiator-operating temperatures and reduced radiator area.

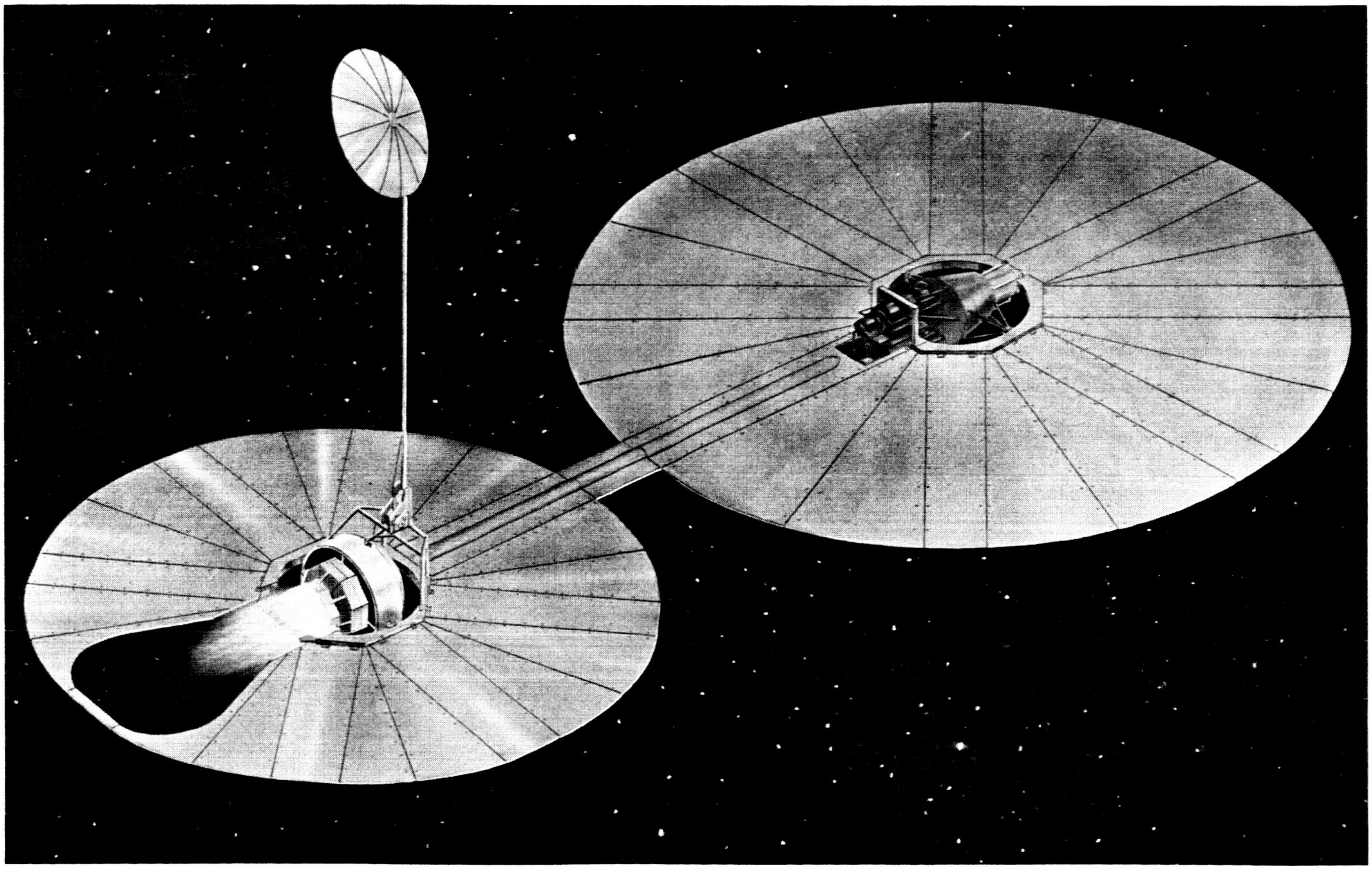

Fig. 6. Jupiter-capture spacecraft configuration

The ion-engine cluster is split in half to minimize interception of the ion beam with the adjacent radiator structure. Steering of the spacecraft would be accomplished either by gimballing or by thrust modulation of the ion-engine modules.

During the flight from the Earth to Jupiter, the spacecraft is oriented such that the radiator discs lie essentially in the plane of the ecliptic. The steerable communication antenna, shown above the propellant tank, is continually directed toward the Earth, whereas the main body of the spacecraft gradually changes direction to satisfy the thrust-vector trajectory requirements previously described.

Table 2 shows a preliminary weight summary[7] for this spacecraft; Fig. 7 illustrates how the power generated by the nuclear reactor is utilized by the spacecraft. It can be seen from Fig. 7 that 1 Mwe actually represents the power delivered to the ion engine, whereas the raw electrical output of the generating system is about 1.3 Mwe, with the difference going into losses and utility uses.

Table 2. Jupiter capture; 1-Mwe-propulsion preliminary weight summary

System	Weight, lb
Scientific payload	5,000
Instrumentation–telecommunication	1,200
Wide-band transmitter	800
Antenna dish	100
Guidance and control	1,000
Ion engine	200
Propellant	15,000
Propellant tank	500
Power-production equipment (1 Mw for propulsion + 0.3 Mw for utility and losses)	13,000 (includes primary shield)
Power-conditioning equipment	5,200
Secondary shield	1,000
Structure and thermal-control system	2,000
Initial mass in Earth orbit	45,000

The scientific experiments which might be performed by the spacecraft upon arrival at Jupiter would be chosen to take advantage of the unique character of nuclear-electric propulsion systems; that is, the large amount of power which can be made available and the wide-bandwidth communication transmitter which can be operated on this power. A typical transmitter-radiated power of 50 kwe or more might be used for such a mission. Transmission at this level, with appropriate receiver equipment on Earth, would allow for the reception of high-quality video pictures from Jupiter. Some of the typical scientific experiments which might be performed on Jupiter are as follows:

1. Magnetic-field measurements in interplanetary space and adjacent to Jupiter
2. Spectrophotometric measurements of the Jovian atmosphere
3. Microwave-radiometry measurements of the Jovian radiation belts and ionosphere.
4. Radar probing of the Jovian atmosphere and planetary surface
5. High-resolution video pictures of the Jovian cloud structure

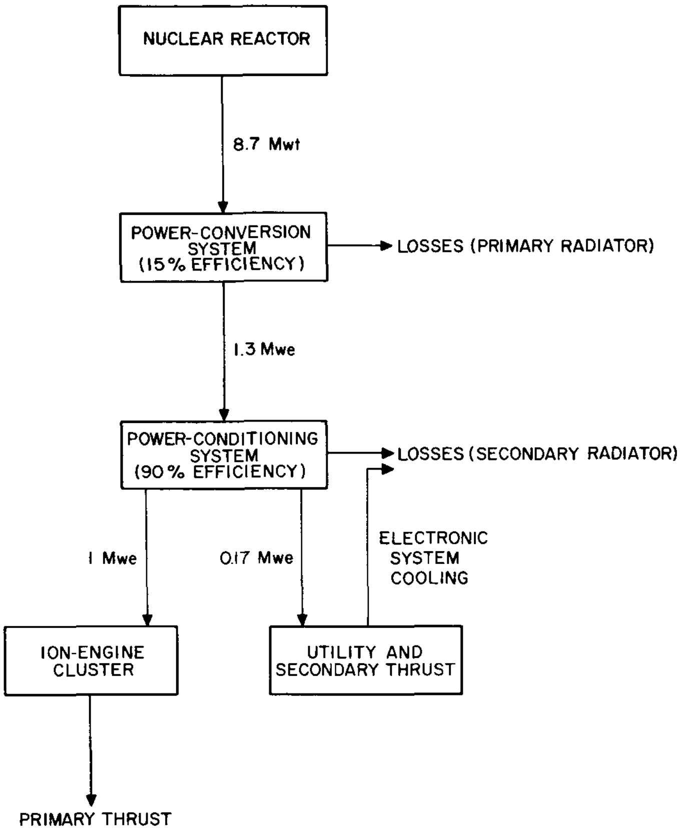

Fig. 7. Jupiter capture; 1-Mwe-propulsion preliminary power-utilization summary

[7]It may be noted from the weight summary that the power level selected and the corresponding system weights are not exactly optimum according to the argument presented in the discussion on performance. Additional refinement of the design is obviously necessary but is not carried out here, as it is not pertinent to the level of this review.

The second configuration, shown in Fig. 8, illustrates a lower-power spacecraft designed around a nominal 70-kwe *SNAP-8* type power-generation system but employing the same general design approach previously discussed. In this case, 60 kwe are utilized for primary propulsion, with 7 kwe going into power-conditioning losses and 3 kwe for utility. This spacecraft would have the capability of performing a Venus-capture mission.

Following establishment of a 300-nm Earth orbit, as in the case of the larger Jupiter craft, the power-generation system and ion engine would be started, and

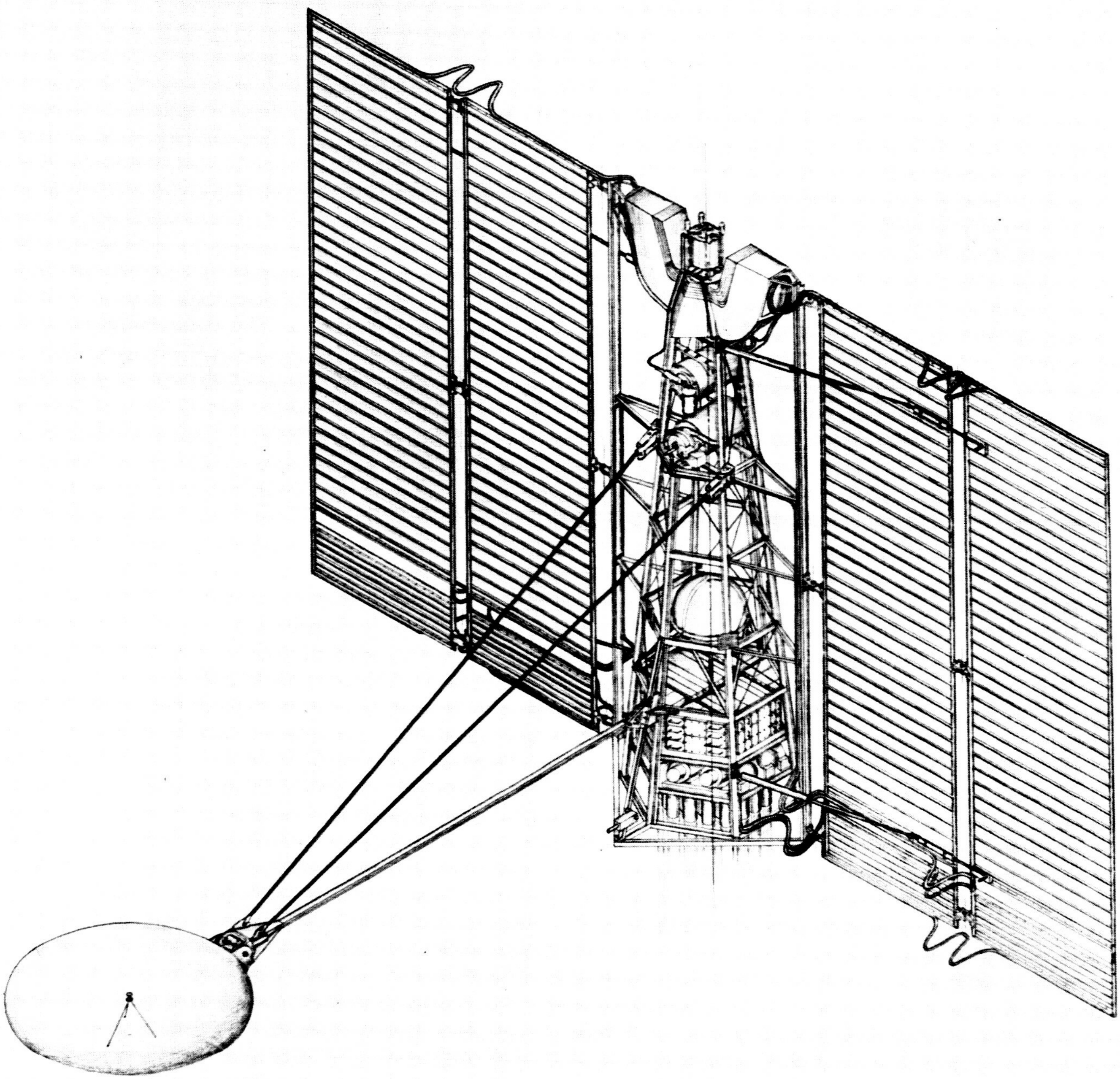

Fig. 8. Venus-capture spacecraft configuration; isometric presentation

the spacecraft would spiral to escape, pass through a heliocentric transfer, and finally be caught in an elliptical orbit about the planet Venus. Approximately 365 days would be required for this operation, which, in addition to a requirement to supply useful power at Venus for several months, would impose an over-all lifetime requirement of at least 10,000 hr on the spacecraft systems. This represents quite a goal for an early-generation spacecraft!

Again referring to Fig. 8, the power-production system can be seen located in the upper section of the spacecraft, the reactor being in the uppermost position. The nuclear shielding is below and around the sides of the reactor. This shield configuration represents an attempt to attenuate both the direct radiation and the radiation which would be scattered off the radiator panels.

The two radially mounted flat-panel high-temperature radiators required to reject the waste heat from the power-production system are shown extended as they would be in flight. They provide an area of about 1200 sq ft, radiating at 700°F. Attached to and below the primary radiators are the low-temperature secondary radiators required to reject the excess heat from the electronic and power-conditioning equipment. The secondary radiators provide an area of about 100 sq ft, radiating at a nominal 200°F.

Directly below and in the shadow of the radiation shield are located the two turbogenerator power-conversion units. Beneath these, in the lower section of the spacecraft, is a spherical tank which would contain the propellant for the thrust unit. Directly under the propellant tank are the electronic and power-conditioning systems, and below these is the 60-kwe ion engine.

During boost, the radiators would be folded around the main body of the spacecraft, so that the complete spacecraft could be enclosed by an aerodynamic shroud. For communication purposes, both an omnidirectional and a high-gain antenna system have been integrated into the spacecraft. Four omnidirectional antennas are located around the propellant tank. The high-gain antenna system utilizes a 9-ft parabolic antenna on a support mounted near the propellant tank.

As in the case of the Jupiter craft, the spacecraft would be oriented during flight such that the radiator panels would lie in the plane of the ecliptic.

Table 3 shows a preliminary weight summary for the Venus-capture spacecraft. The scientific experiments which could be performed by this spacecraft would probably be similar to those discussed for the Jupiter craft. For high-quality video transmission from Venus, a transmitter-radiated power of 10 kwe might be utilized. This would correspond to a transmitter input of about 50 kwe, which could be obtained from the nuclear-electric power-generation system upon arrival at Venus.

Table 3. Venus capture; 60-kwe-propulsion preliminary weight summary

System	Weight, lb
Scientific payload	500
Instrumentation–telecommunication	250
Wide-band transmitter	200
Antenna dish	100
Guidance and control	500
Ion engine	200
Propellant	2,000
Propellant tank	100
Power-production equipment (60 kwe for propulsion + 10 kwe for utility and losses)	2,500
Power-conditioning equipment	350
Primary nuclear shield	1,000
Structure and thermal-control system	800
Initial mass in Earth orbit	8,500

IV. MAJOR DEVELOPMENT GOALS

It should now be obvious, from the preceding review, that a number of major development goals must be met in working toward the ultimate mission utilization of nuclear-electric systems. Some of the most significant items which should receive early attention are the following:

1. Achievement of extremely long lifetime and high reliability for systems and components operating under extreme environmental conditions.
2. Determination of adequate ground test techniques to demonstrate reliability in reasonable time periods at reasonable costs.
3. Reduction in radiator surface-area and weight requirements through more efficient, higher-temperature radiators employing improved micrometeorite-protection techniques.
4. Improvement of radiator boost-phase packaging to minimize the complexity of unfolding following the boost phase and to reduce the dimensions of the aerodynamic shroud during boost.
5. Improvement of component tolerance to high temperature and nuclear-radiation levels to ease radiator requirements and reduce shield weights.
6. Improvement of over-all efficiency of the electric-power-generation, power-conditioning, and thrust-producing equipment to maximize scientific-payload capabilities for the spacecraft.

Only as a result of diligent effort to achieve these goals, and others which undoubtedly will come to light through continuing integration studies, will nuclear-electric spacecraft be ready to assume their proper role in space exploration.

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