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Preliminary Investigations to Determine Nuclear-Electric Spacecraft Configurations for High-Energy Missions

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Abstract

This study determines how the mass of spacecraft systems is affected by changes in spacecraft configuration for a conceptual 300-kwe nuclear-electric spacecraft designed for a Jupiter orbital mission. The power plant employed a reactor-turbo-generator system. The nuclear-radiation shield, heat-rejection radiators, spacecraft structure, booster-spacecraft adapter, and aerodynamic shroud are considered as subsystems most likely to vary appreciably in mass with changes of spacecraft configuration. The two selected spacecraft configuration concepts feature the heat-rejection radiator systems assembled in the flat-plane configuration and the cylindrical configuration. The flat-plane radiator configuration was found to be the more favorable concept.

Preliminary Investigations to Determine Nuclear-Electric Spacecraft Configurations for High-Energy Missions

I. Introduction

The most popular configuration concepts for unmanned nuclear-electric ion-propelled spacecraft are those which have the heat-rejection radiator systems assembled either in the flat plane or the cylindrical form. This preliminary study compares the related subsystems of the spacecraft in terms of mass for these two configuration concepts on an equal design basis and shows how the configuration constraints affect the subsystems of each concept. The study attempts to show which concept is the more desirable and economical in terms of total spacecraft mass, and points out how significantly the related subsystems' masses are affected by spacecraft configuration changes.

The subsystems whose masses are affected by spacecraft configuration are the nuclear-radiation shield, spacecraft supporting structural elements, aerodynamic shroud, and heat-rejection radiators. The other subsystems and components have masses and dimensions which are considered to be independent of changes in spacecraft configuration.

The two most fundamental spacecraft parameters influencing the general spacecraft configuration are the shroud semi-vertex angle and spacecraft length. In this study, these two parameters are treated as variables for the two selected configuration concepts.

II. Design Constraints

The design constraints used to evaluate the spacecraft are established by the mission, launch vehicle, spacecraft power plant, and configuration concept. These must be selected prior to design and analysis of the various related subsystems.

A. Selected Mission

A Jupiter orbital mission is selected for this study for two reasons: (1) the mission is representative of high-energy missions most suited for nuclear-electric propulsion, and (2) for increasing mission time, the solar thermal-irradiance has less effect upon thermally sensitive components.

There are three distinct phases of the spacecraft's variable-thrust optimum trajectory: (1) the spacecraft spirals out from the initial orbit (700 nmi minimum) until Earth escape energy is reached; this occurs at about 75 days; (2) following Earth escape is the heliocentric transfer to Jupiter rendezvous; and (3) the spacecraft spirals into a desirable elliptical orbit around Jupiter for the remaining period of the 833-day mission. The trajectory is based on an assumed initial spacecraft acceleration of 10^{-3} m/sec^2 and an initial spacecraft weight of 20,800 lb.

The mission trajectory constraint is necessary to determine requirements for propellant, telecommunications, power-profile, and scientific payload. Also by integrating the mission trajectory with the selected meteoroid and penetration models, the protection requirements for vulnerable components to resist impact damage by meteoroidal particles may be evaluated.

B. Launch Vehicle

The two-stage *Saturn IB* is assumed as the launch vehicle for each of the spacecraft configuration concepts considered. It is expected that for the time period in which a 20,800-lb nuclear-electric spacecraft might be considered for a mission flight (late 1970's or later), this launch vehicle would have the capability of placing the spacecraft into a minimum 700-nmi Earth orbit. The total booster payload would be a nominal 28,000 lb, of which $\sim$ 7,200 lb is allowed for aerodynamic shroud, booster-spacecraft adapter, and startup equipment requirements. It is assumed that the aerodynamic shroud is ejected at the end of first-stage burn.

Results from a preliminary investigation[1] to determine the structural compatibility of the two-stage *Saturn IB* in combination with a 20,000-lb, 500-kwe nuclear-electric spacecraft (Ref. 1) indicated that, with primarily minor structural modifications to the booster, such a combination would be compatible. Since the spacecraft configurations considered in this paper are similar in physical size and weight to that reported in Ref. 1, the aerodynamic shroud design was based on the maximum dynamic pressure conditions reported in footnote No. 1. This condition is:

Maximum dynamic pressure = 4.28 psi

Mach number = 1.22

Angle of attack = 7.10 deg

The maximum shroud diameter for all configurations was taken to be 260 in.; the clearance between any point on the inside surface of the shroud and the nearest point of the spacecraft was assumed to be no less than 6 in.

The structural design was based on the maximum inertia loads expected at first-stage cutoff. It is estimated that the maximum vertical acceleration at this condition would be $\sim$ 5 g, which includes a factor of 1.25 to account for vibration. If a safety factor of 2.0 is used, then the maximum vertical inertia loading becomes 10 g. This, in combination with an assumed 1.5-g lateral loading through the spacecraft center of gravity, was used to design the spacecraft structure and booster-spacecraft adapter.

C. Power Plant

The power plant assumed for all configurations was one employing a nuclear reactor as the heat source, a turboalternator system for converting the heat energy to electrical energy, and noncondensing heat-rejection radiators to radiate the waste heat resulting from the thermal-to-electrical energy conversion. It was assumed that the power plant would provide 300-kwe unconditioned power to the electrical thrustors and have a cycle efficiency of $\sim$ 17%.

There are two radiator systems required using NaK as the coolant: (1) a primary radiator system which rejects $\sim$ 1.75 Mwt due to the inefficiency of the thermal-to-electric energy conversion and (2) an intermediate radiator system to reject $\sim$ 120 kwt picked up in cooling pump motors, bearings, etc. The pressure differences between the inlet and exit conditions for the two radiator systems were assumed to be 9 and 7 psi, respectively.

D. Configuration Concepts

The two basic configuration concepts selected for this study are the flat plane and cylindrical. As shown in Fig. 1, the outermost dimensions and the spacecraft length can be varied within the design constraints set by the shroud and booster. The systems integration philosophy may be generalized as follows: (1) the fixed reactor is placed as far as possible from the payload to decrease the shield mass requirements; (2) the fixed heat-rejection radiator systems should be arranged for minimum heat-exchange to thermally sensitive components; (3) the telecommunications antenna is placed near the payload to minimize interference problems; and (4) the propellant and feed systems are located near the thrust engines at the payload region, primarily to lower the center of gravity location of the spacecraft.

The radiators in the flat-plane configuration are placed within the spacecraft's outer and central structures to obtain the minimum spacecraft length (Fig. 1). Using optimum radiation systems, only one spacecraft length for the flat plane configuration would result for each

[1]Personal communication from authors to L. Blomeyer, Jet Propulsion Laboratory, *Structural Analyses, Saturn IB Launch Vehicle/JPL Navigator Spacecraft*, July 2, 1964.

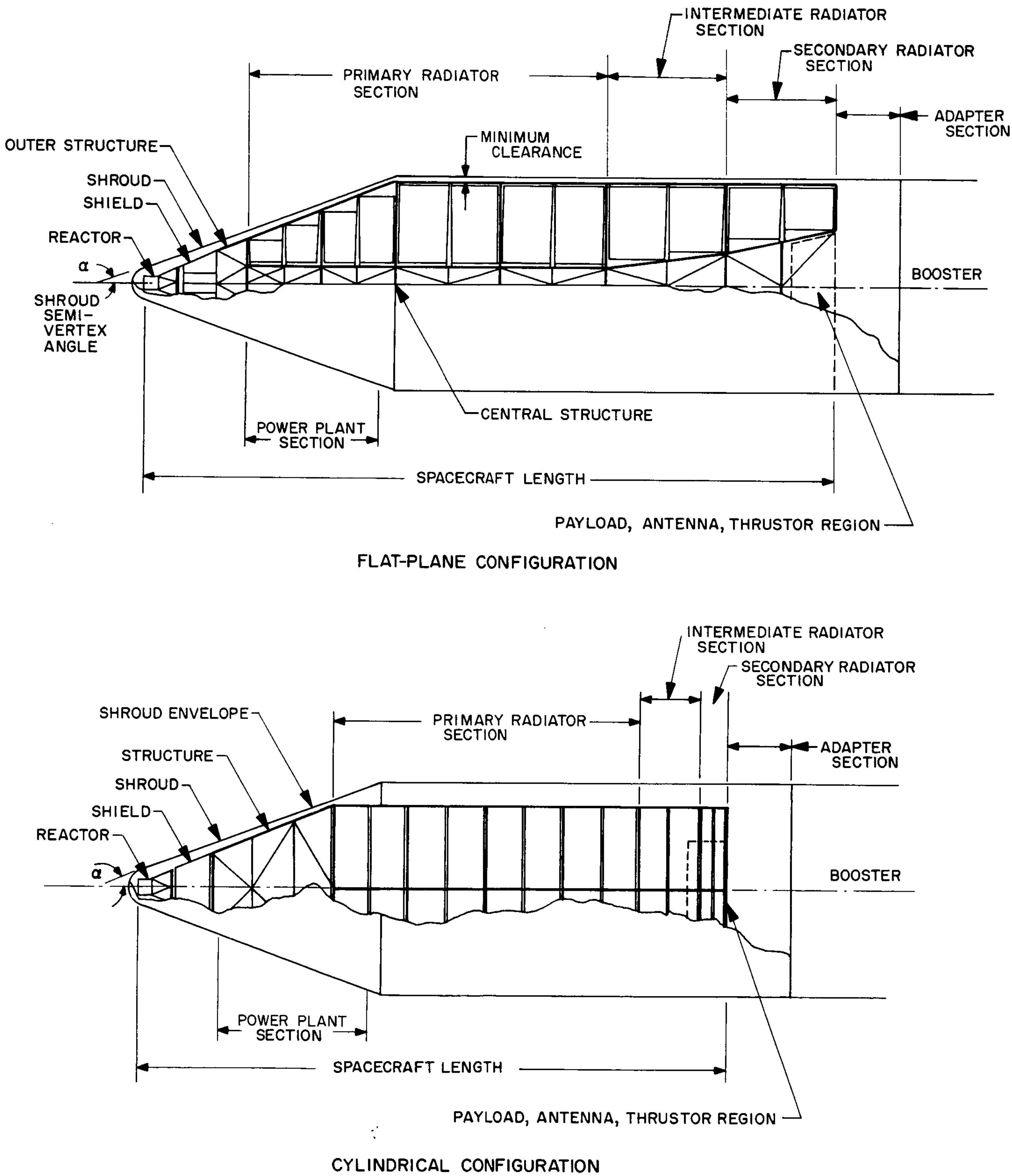

Fig. 1. Spacecraft configuration concepts

shroud semi-vertex angle. However, the radiators in the cylindrical configuration are attached to the shrouded spacecraft structure, whose diameter may vary, which causes the spacecraft length to vary. When this diameter is decreased, the spacecraft length increases due to the radiator requirements. Thus, the design of the radiators becomes a function of length. The design of the other subsystems can be shown to vary also with respect to spacecraft length. For example, the shield thickness and dimensions are a function of the payload location. The masses of the spacecraft structure and shroud are also a function of spacecraft configuration. However, other spacecraft components such as the propellant required for a specified mission, ion-engines, telecommunication antenna, power conditioning, boiler, and power plant less radiators, will not be significantly altered or affected by space configuration.

III. Spacecraft Subsystem Analysis

The subsystems whose masses vary significantly as a function of spacecraft configuration are the nuclear radiation shield, spacecraft structure, booster-spacecraft adapter, spacecraft shroud, and the heat-rejection radiators. This section describes the analyses used to determine these mass changes.

A. Nuclear Radiation Shield

The radiation sensitive elements of the power conditioning and scientific payload equipment must be protected from an excessive dose of nuclear radiation emitted by the nuclear reactor. This protection is obtained by a nuclear radiation shield located near the nuclear reactor. The shield and reactor systems are located as far as practicable from the payload zone so that maximum attenuation of radiation occurs at the payload. The allowed dose rates at the payload location are: the integrated dose over a 20,000-four full-power operating period is 10^{13} nvt (neutrons-cm^{-2}) for neutron energies greater than 0.1 Mev, and 10^7 rad for gamma radiation.

The simplified calculational procedure selected to determine the required shield configuration and weight is such that the comparative results should be reasonably correct. The results are similar to those obtained by other investigators (Ref. 2).

The geometric shape of the shield is dependent upon the spacecraft configuration and the allowed integrated dose. The spacecraft systems such as the structure, radiators, and power plant must lie in the shadow of the solid cone angle of the shield to reduce the scattered neutron dose at the payload.

A 2-in. thickness of tungsten placed close to the reactor is considered sufficient to reduce the gamma ray induced heating, produced in the selected LiH fast-neutron attenuator, to tolerable temperature levels (Ref. 3). A tungsten shield weight of about 750 lb is required for both spacecraft configurations.

The fast-neutron flux leaving the LiH shield surface in the payload direction was estimated from one-dimensional diffusion calculations. The angular flux distribution from this surface, which acts as a uniform plane source, was assumed to vary as the cosine of the angle θ to the normal of the surface (Fig. 2). The direct flux as a function of dose point distance is given in Fig. 3. The results of equivalent LiH thickness are based on the radiation which has passed through the tungsten shield.

The scattered flux contribution to the total dose at the payload location was calculated using a scattering probability coefficient (albedo), β. This coefficient is the probability that a neutron will scatter upon striking a surface such as the radiator. The angular distribution of the scattered neutrons is assumed to be isotropic. From calculations based on the scattering cross-section and radiator thickness, an albedo range from 0.10 to 0.20 was estimated for the plane radiators. For the cylindrical radiator whose thickness is about 0.5 of the flat radiator, the albedo range is from 0.05 to 0.10. The total shield weight is found to be relatively unaffected by a large range of albedo values.

The expressions used to compute the radiation flux at the payload location are as follows:

The direct beam

$$\Phi_d = \Phi \frac{R^2}{4L^2} \tag{1}$$

where $L/R \leq 0.01$.

The scattered flux for flat-plane radiators

$$\Phi_{s_F} = \frac{R_F^2 \Phi \beta_F}{8\pi} \int_{x=0}^{L} \int_{c}^{D/2} \frac{y\,dy\,dx}{(x^2 + y^2)^{3/2}\,[(L-x)^2 + y^2]} \tag{2}$$

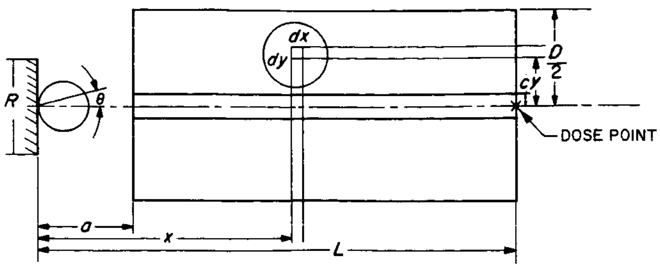

Fig. 2. Radiation scatter geometry

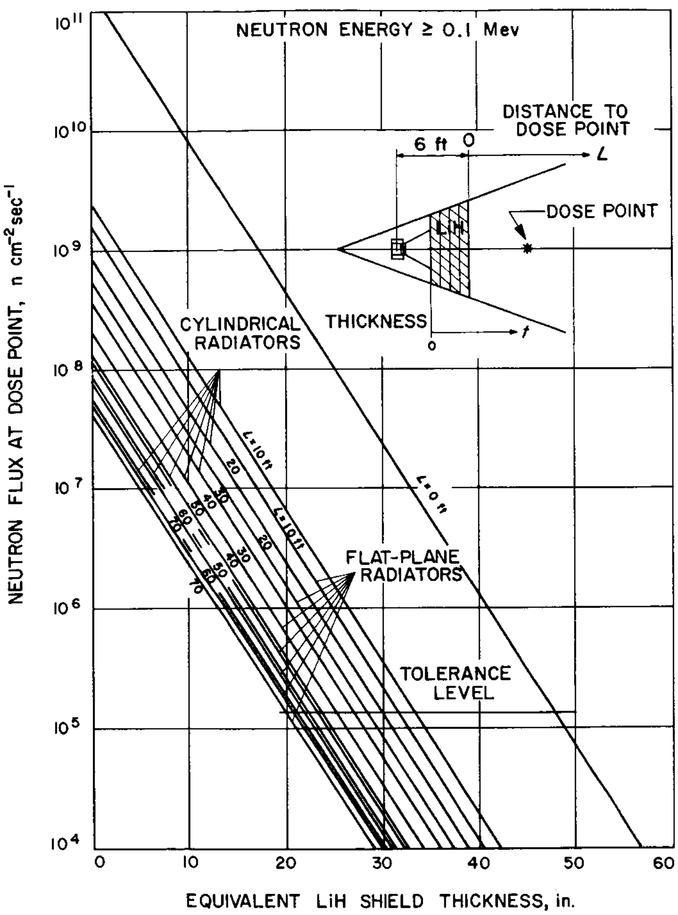

Fig. 3. Direct radiation flux vs dose point distance

The scattered flux for cylindrical radiators

$$\Phi_{s_C} = \frac{R_C^2 \Phi \beta_C D^2}{2} \int_{a=0}^{L} \frac{dx}{(4x^2 + D^2)^{3/2}\,[4(L-x)^2 + D^2]} \tag{3}$$

Figure 2 defines the geometric notation used in Eqs. (1) through (3). The scattering to direct flux ratios are given

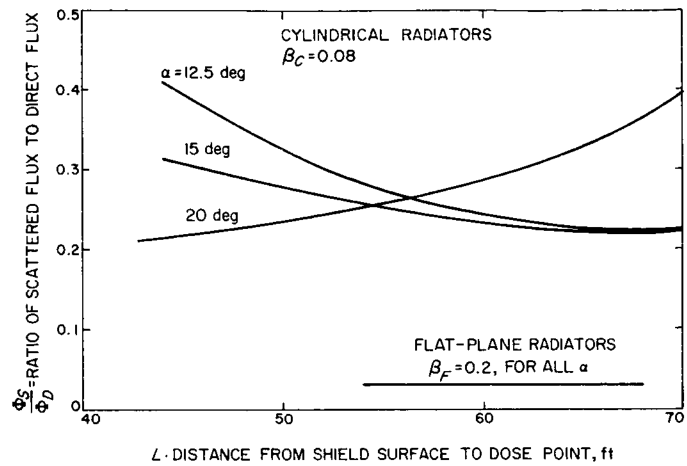

Fig. 4. Scatter to direct flux ratio as a function of dose point distance for flat-plane and cylindrical configurations

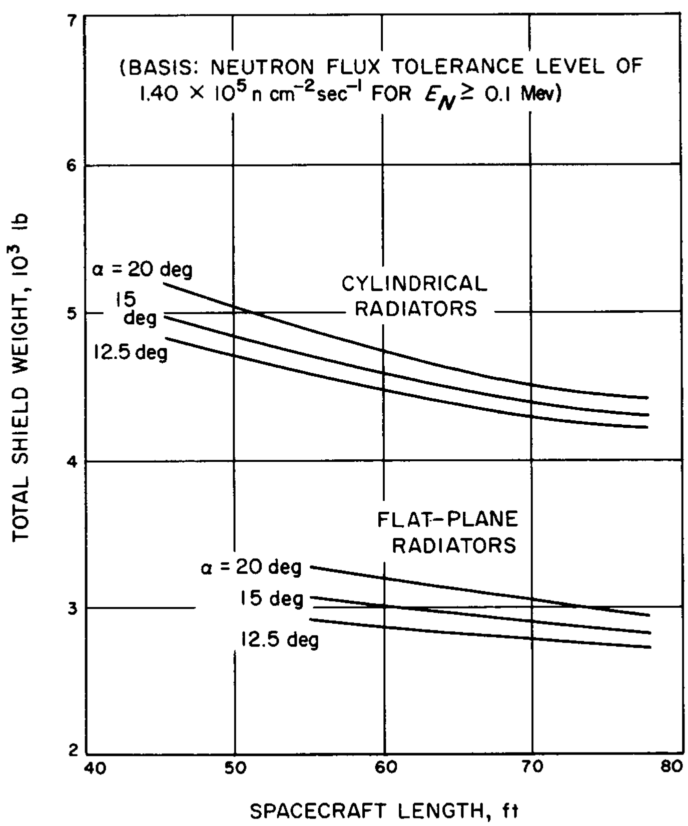

Fig. 5. Total radiation shield weight vs spacecraft length

in Fig. 4; the total shield weights for the allowed tolerance flux rate (1.40×10^5 n cm^{-2} sec^{-1}) are given in Fig. 5. The results of Fig. 5 are believed to be reliable to within -10% to $+30\%$ of those weights necessary to meet the design requirements.

The scattered radiation introduces only a very small shield-weight requirement. The cylindrical radiators scatter about ten times more radiation than the flat-plane radiators to the payload location. The shield-weight requirement for the cylindrical configuration as a function of spacecraft length can be seen in Fig. 5 to be about 60% greater than the shield-weight requirement for the flat-plane configuration. This is because the shield attenuating the scatter radiation for the flat-plane configuration assumes a form of slim rectangular lobes attached to the direct-beam cylindrical shield, whereas the scatter shield for the cylindrical configuration assumes an annular form which surrounds the direct-beam shield.

B. Spacecraft Structure

Structure is required to provide secure attachments to support the various spacecraft components under the various loads to which the spacecraft might be subjected during its lifetime. The design of the structure must, of course, be based on the most critical combination of loading. Past experience indicates that the most critical combination occurs during the boost phase. The primary loadings which the spacecraft, and its components, experience during the phase are the high inertia loads resulting from the acceleration of the launch vehicle, shock loads, and vibratory loads imposed by the rocket-engine sound field, and also by the rocket engine directly. For this preliminary investigation, the inertia loadings were considered to be 10 *g* longitudinal and 1.5 *g* lateral. These were assumed to be maximum and were considered to act simultaneously. It should be pointed out that the 10-g load factor includes an assumed factor of 1.25 to account specifically for the longitudinal vibratory loads. It is recognized that this factor alone may not ensure the adequacy of the structure from the standpoint of vibration, but it is felt that it is sufficient within the scope and requirements of this investigation.

As pointed out in Section II-D, the two spacecraft-configuration concepts considered were the flat-plane and cylindrical-radiator configurations. For each of the concepts the shroud half-angle varied from 12.5 to 20 deg; in the case of the cylindrical-radiator configurations, the diameter of the radiator varied from 8 to 16 ft for each shroud half-angle considered.

Figure 6 illustrates conceptually the structural models used in the analysis. Truss structures of tubular construction were assumed for each configuration. Longitudinal members form the main portion of the trusses.

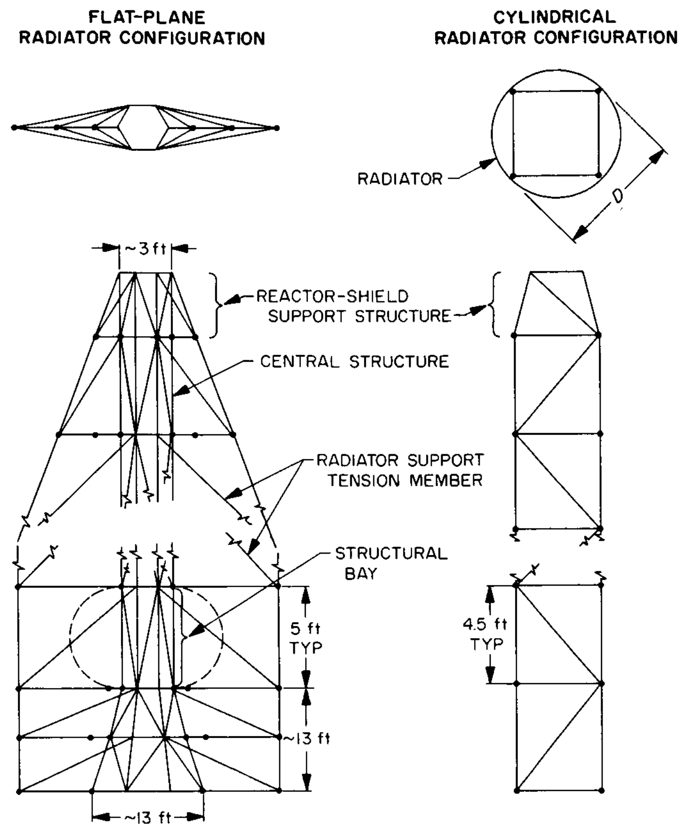

Fig. 6. Structure models for flat-plane and cylindrical configurations

The structure is divided into bays by a closed lattice of tubular cross members which provide relative rigidity between the main members. Diagonal members between adjacent main members serve to provide overall truss rigidity. A square geometry was found to be optimum for both configurations. However, to allow for sufficient volume to properly package the power system in the flat-plane radiator configuration, an off-optimum hexagonal geometry was assumed for the structure. This selection permits ~ 30% increased volume with an associated 10% increased weight.

For the flat-plane radiator configuration structure, tension members extending from the central structure to the outermost support points of the radiator panels provide panel rigidity, as well as panel support during launch. The radiator panels for the cylindrical configuration are wrapped around the structure and securely attached to the support points (Fig. 6).

Design of compression members was accomplished by using either Euler's formula (where the member was determined to be a long column) or Johnson's formula (for short column members).

Results of this analysis are presented in Fig. 7. For the flat-plane radiator configurations, spacecraft length and structure weight increase as the semi-vertex angle, α, is varied from 20 to 15 deg. This trend can be explained by noting from Fig. 1 that as α is decreased from 20 to 12.5 deg, the usable area in the vicinity of the power plant, for packaging a portion of the primary radiator area, is decreased and the primary radiator section length must be increased to accommodate the required area.

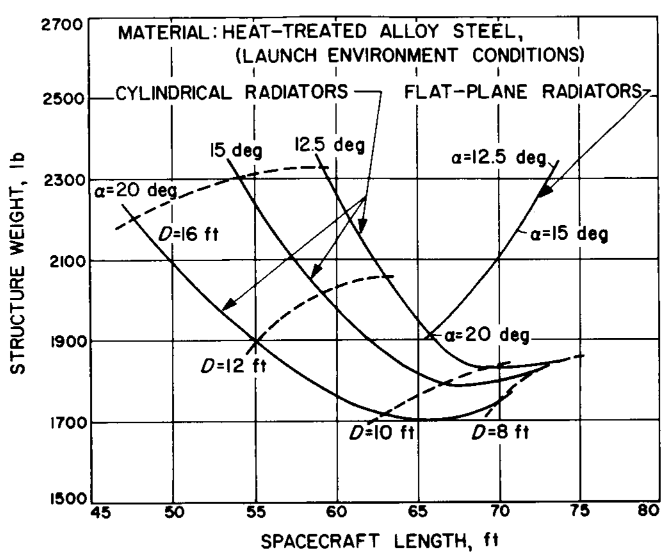

Fig. 7. Spacecraft structure weight vs spacecraft length

Structure weights for the cylindrical-radiator configuration depend on an additional variable, radiator diameter. For an assumed α, structure weights are maximum at the maximum considered radiator diameter (D = 16 ft). As radiator diameter is made smaller, structure weight decreases to a minimum value at some intermediate diameter between 10 and 8 ft, then increases as radiator diameter decreases to 8 ft. This trend can best be explained in terms of structural loads and geometry. As described elsewhere in this section, the structure is composed of (1) longitudinal, and (2) cross and diagonal members. Lengths of the longitudinal members remain relatively constant for all the range of radiator diameters considered, and as a result, changes in weights of these members are primarily a function of the imposed loads. These loads are a maximum at D = 8 deg, continually decreasing as radiator diameter increases. Conversely, loads imposed on the cross and diagonal members are maximum at D = 16 ft and continually decreasing for decreasing values of radiator diameter. Over the range of diameters assumed, changes in these loads were generally found to be small. However, lengths of these numbers, which are maximum at D = 16 ft, decrease significantly as radiator diameter decreases. Consequently, weight changes of these members are primarily a function of the change in their lengths.

In summary, then, as radiator diameter is decreased over the range of 16 to 8 ft, weights of the cross and diagonal members vary from maximum to minimum values and weights of the longitudinal members vary from minimum to maximum values. The shape of the curves in Fig. 7 indicates that at the larger radiator diameters, weights of the cross and diagonal members have a significantly greater influence on the total structure weight than the longitudinal members have on the total structure weight at the smaller radiator diameters.

For this analysis, it was assumed that all members were of tubular geometry and constructed of heat-treated alloy steel rated at 180,000 psi ultimate tensile strength (UTS) at room temperature. Mechanical properties for the estimated launch-temperature environment were obtained from Ref. 4. Assuming the same material of construction, it has been determined that an alternate geometry, such as I-beam construction, could increase the structural weight by as much as 20%. Considering other materials, it has been estimated that if a titanium alloy (e.g., Ti-6Al-4V) were used in conjunction with the tubular construction, the structure weights indicated in Fig. 7 could be reduced by $\sim$ 10%.

It is estimated that the results of Fig. 7 are within 5% of the minimum weight requirement for the structure.

C. Booster-Spacecraft Adapter

The booster-spacecraft adapter is the interface structure required to mate the spacecraft to the launch vehicle for the launch phase of flight. At second-stage cutoff, the spacecraft is separated from the adapter; the adapter remains with the second stage.

Circular truss arrangements were assumed in the analysis for both the flat-plane and cylindrical-radiator configuration concepts. The lower attachment points of the adapter tie directly to the launch vehicle structure, whereas the upper attachment points are tied together with cross members separate from the spacecraft structure. For all configurations, a separation distance of 6 ft is maintained between the base of the spacecraft and the booster, so as to provide equipment access.

The design was based on the loads described in Section III-B. Heat-treated alloy steel tubing with 180,000-psi UTS was assumed as the material for all members of the adapter.

Results of the analysis for the two configurations are presented in Fig. 8. It has been estimated that if a titanium alloy (Ti-64-4V) rated at 160,000-psi UTS were used, a savings of about 13% of the total weight could be realized; using an aluminum alloy (6061) does not appear advantageous since the adapter weights would increase by ~ 10% over that of steel. Tubular construction appears to provide a 10% weight advantage over I-beam construction.

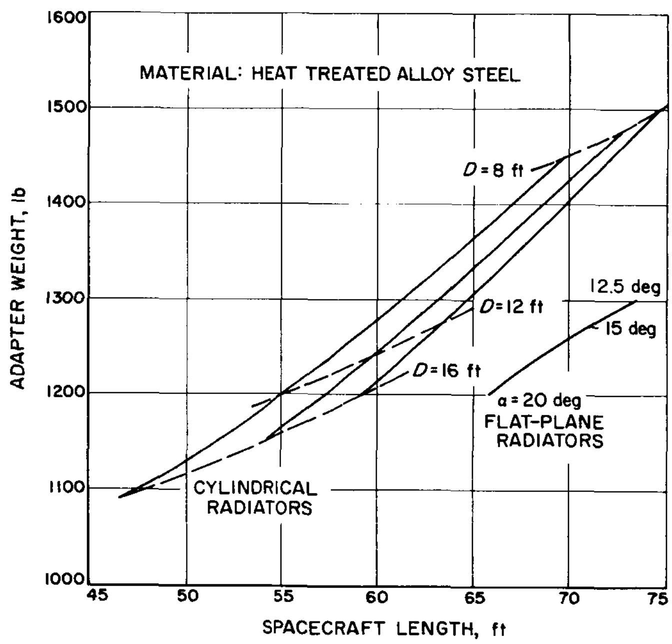

Fig. 8. Booster-spacecraft adapter weight vs spacecraft length

D. Aerodynamic Shroud

Figure 9 presents the results of a preliminary investigation of the aerodynamic shroud requirements for the flat-plane and cylindrical-radiator configurations. The conditions which the shroud must withstand are maximum dynamic pressure = 4.28 psi at angle of attack = 7.1 deg and Mach number = 1.22.

Aluminum alloy (2024-T4) semi-monocoque shell construction was assumed for both the conical and cylin-

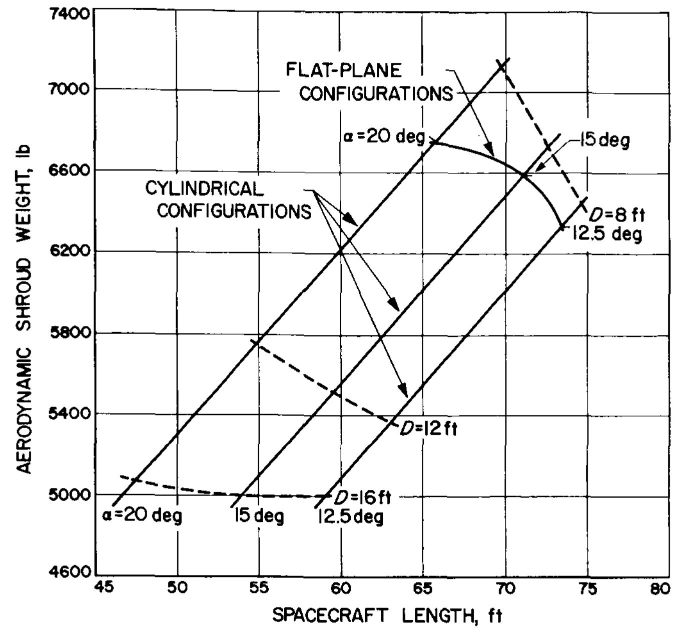

Fig. 9. Aerodynamic shroud weight vs spacecraft length

drical sections. Included in the design would be (1) a beryllium nose-tip cap (Ref. 5) to present an adequate barrier to the high stagnation temperatures expected during launch through the atmosphere and (2) a thermal coating such as thermolag resin (Ref. 6) over the upper portion of the conical section to preclude excessive shroud temperatures during launch. It is assumed that the shroud design would also be compatible with the temperature environment, resulting from power plant startup requirements. For example, it will be necessary to preheat (and maintain) portions of the power plant (including the radiators) to temperatures upwards to 1000°F. Some of this equipment will be located in close proximity to the shroud (e.g., the nuclear reactor), and could possibly produce an adverse environment for the shroud. If this were the case, either thermal insulation on the inside surface of the shroud could be considered or else some other material, such as Ti-6Al-4V heat-treated alloy, could be used in the critical area.

E. Heat-Rejection Radiators

The three all-liquid heat-rejection radiator systems of the spacecraft are the primary radiator system (1200°F) required to reject the waste heat of the power generating plant, the intermediate radiator system (500°F) used for auxiliary equipment cooling, and the secondary radiator system (200°F) used to cool certain electronic components. The three design constraints, which each of these radiator systems must satisfy, are total heat rate to be rejected, fluid flow requirements, and protection requirement to resist meteoroid impact damage.

A radiator system is composed of supply and return headers and panels. Each radiator system is arranged in a two-panel configuration (Ref. 7). This panel configuration may be described as having one supply header distributing fluid to the two adjacent panels which connect to individual return headers (Fig. 10). The tapered cylindrical headers are arranged and sized to distribute fluid uniformly to the panel-tube elements. The selected panel is composed of an array of finned-tube elements. The diameter of the cylindrical tube is uniform. A minimum weight finned-tube element design for the panel defines the selected rectangular fin geometry. This radiator system has a total mass which is within ½% of a mass optimized radiator system (Ref. 7). The minimum weight finned-tube design is defined as that geometry whose heat rejection rate per unit element mass is maximum for a given tube diameter, specified tube temperature, and materials.

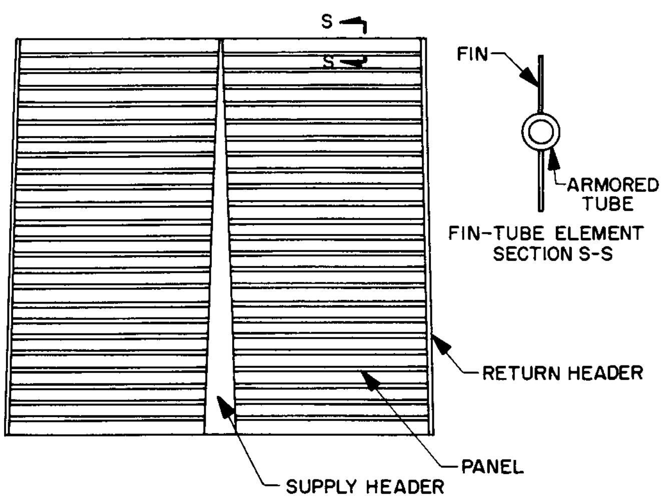

Fig. 10. Typical two-panel configuration

The radiator panels of the flat-plane configuration radiate heat to the space environment from both sides. However, the radiator panels of the cylindrical configuration radiate heat, theoretically, from only one side, the outer surface. Should the radiator element of the flat configuration be longitudinally divided in half and assembled in a cylindrical configuration, the radiator system for each configuration can be very similar in dimensions and mass. The tube element in this comparison must, however, include an additional enclosure for the cylindrical configuration as shown in Fig. 11. For this study, it is assumed that the inner wall of the tube enclosure for the cylindrical configuration is minimum and will not require any armor protection. To be compatible with the liquid-metal intube material, columbium is selected for the tubes and headers of the primary and intermediate systems. Meteoroid impact damage to vulnerable components is resisted by armor. Beryllium is selected for the

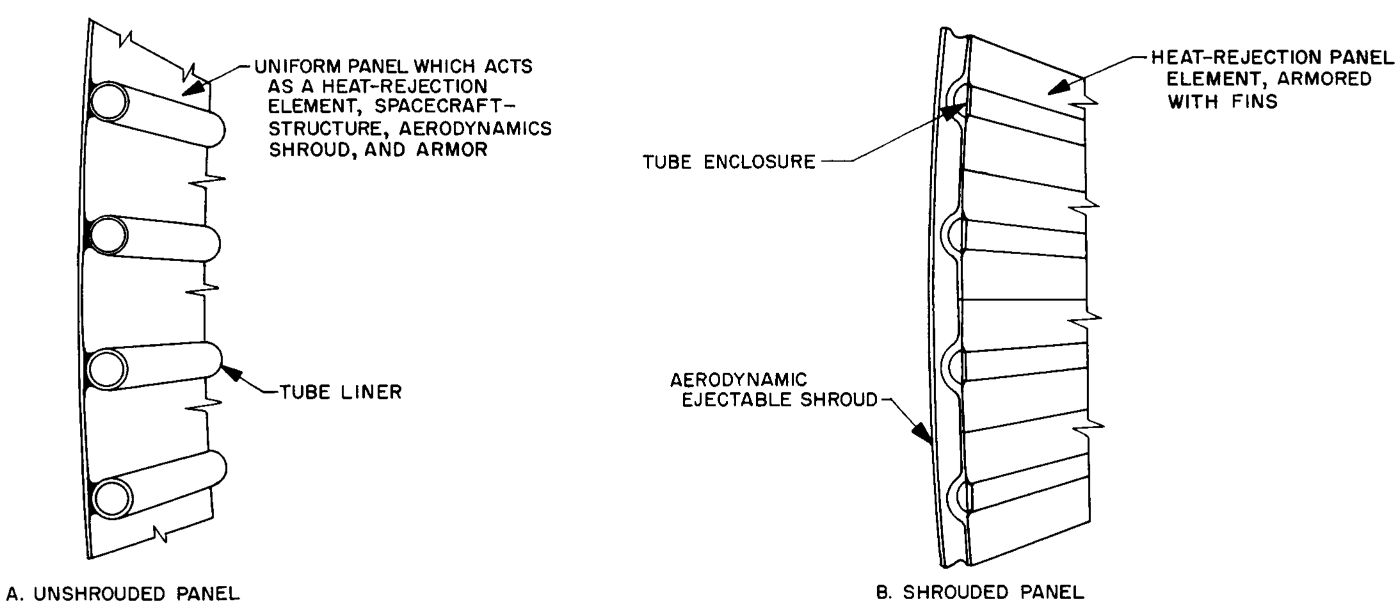

Fig. 11. Cylindrical-radiator configuration panel concepts

armor and fin material for smaller radiator mass requirements. The material used for the secondary radiator system is all aluminum.

The three design constraints can be satisfied with many radiator system sizes. However, to fit a particular planform dimension specified by the spacecraft's dimensional envelope, only one particular radiator system will satisfy both the design constraints and the planform dimension. This radiator system may not be optimum, resulting in a tradeoff between dimension constraints and mass optimization. A JPL computer program, M091 (Ref. 8), includes the three design constraints expressed as follows:

The total heat rejected by the radiator system of the flat-plane configuration is

$$Q_F = NZ\sigma\varepsilon_t\theta_t^4 \, [F_{t\pi}(d_i + 2\Delta_a + 2\Delta_l) + 4F_f l \, \eta] + \sum_{k=1}^{n} (F_h A_h \sigma \varepsilon_h \theta_h^4)_k \tag{4}$$

where the first term represents the heat rejected by the finned-tube elements and the second term represents the heat rejected by the headers.

The total heat rejected by the radiator system of the cylindrical configuration is

$$Q_C = \frac{1}{2} Q_F \tag{5}$$

The fluid flow can be expressed by a form of the Darcy equation as

$$\Delta P_t = \left[0.83 + \frac{(0.184)Z}{d_i Re^{0.2}} \right] \left(\frac{\rho V_t^2}{2g} \right) \tag{6}$$

The armor requirement is defined as (Ref. 9)

$$\Delta_a = \mathcal{R}(0.562)\,(10^{-5}) K_8 K_{10} \left[\frac{A_v T}{-\log_e P_{(0)}} \right]^{0.248} \tag{7}$$

where the vulnerable area, A_v, includes the headers as well as the tubes. The mission time, T, is 833 days and $P_{(0)}$ is selected as 0.995. The material factor, K_8, is 1.00 since beryllium is the material selected for the armor (Ref. 9). The unit conversion factor, K_{10}, is 1.82. For a Jupiter mission (Ref. 9), the value for $\mathcal{R}$ is 0.43. System redundancy of radiator segmented panels was not considered in this study for either spacecraft configuration.

The primary radiator system was designed to reject 6×10^6 Btu/hr of heat from the intube NaK-78 fluid. The overall pressure drop for the fluid is 9 psi. About 4×10^5 Btu/hr are rejected at a pressure drop of 7 psi. The secondary radiator system cooling water fluid at 5 psi rejects heat at the rate of 3.1×10^4 Btu/hr. For all these radiator systems, the total hemispherical emissivity of the heat-rejection surfaces is taken as 0.9.

For this preliminary investigation, the radiator system was not intended to serve as a structural member supporting other components of the spacecraft. However, should the aerodynamic shroud for the cylindrical-spacecraft configuration be used as the heat-rejection radiator and supporting structure by laminating the tube and header liners to its inside surface as shown in Fig. 11A (unshrouded panel), the fins and a portion of the armor as shown in Fig. 11B (shrouded panel) could be eliminated. This amounts to about half of the radiator system weight. However, this shroud would be on board the spacecraft for the entire mission which may demand unacceptable tradeoffs of propellant requirements, total spacecraft mass, mission time, and mission reliability. Therefore, for this study, an ejectable-shroud covering fixed-radiator panels is selected to yield a lighter spacecraft during the mission.

Of the flat-plane configuration, the primary radiator system was segmented into eight radiator systems, the intermediate and secondary radiators were each segmented into two radiator systems. The optimum number of radiator systems for the primary system of the cylindrical configuration was found to be sixteen, and four radiator systems each for the intermediate and secondary systems.

The radiator systems total weight for the two spacecraft configurations are shown in Fig. 12 as a function of spacecraft length and semi-vertex angle, α. These weights are based on the minimum weight design and include the three radiator systems in each case. In the flat configuration, as the spacecraft length decreases from the minimum weight design, the total radiator system becomes less optimum with decreasing fin lengths until an all-tube (isothermal) radiator occurs yielding the minimum spacecraft length shown by the dashed curves. In the cylindrical configuration, the curves shown in Fig. 12 are for a minimum weight design. The reason these weights are greater than those for the flat radiators

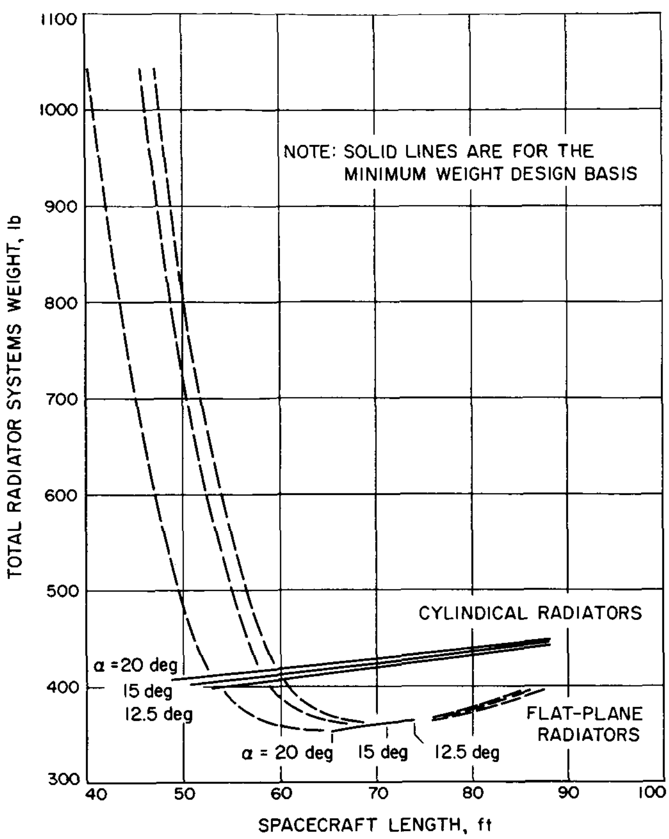

Fig. 12. Total radiator systems weight vs spacecraft length for various shroud semi-vertex angles

is due to tube enclosure requirements (Fig. 11B). It has been found that the changes in weight for the coolant inventory at the optimum design point did not vary significantly with changes in spacecraft configuration for either configuration concept.

The weight of the radiator systems will most likely increase for combinations of materials other than those selected. For example, if copper-stainless steel were used instead of the selected beryllium-columbium combination for the primary and intermediate radiator systems, the resulting weight of the two systems would increase by about 80% for either configuration. The results shown in Fig. 12, based on the accuracy of the mean values for the material properties, are believed reliable to within 2%.

IV. Summary and Conclusions

The total weights of the five subsystems as a function of shroud semi-vertex angle and spacecraft length are illustrated in Fig. 13A. These results were obtained by

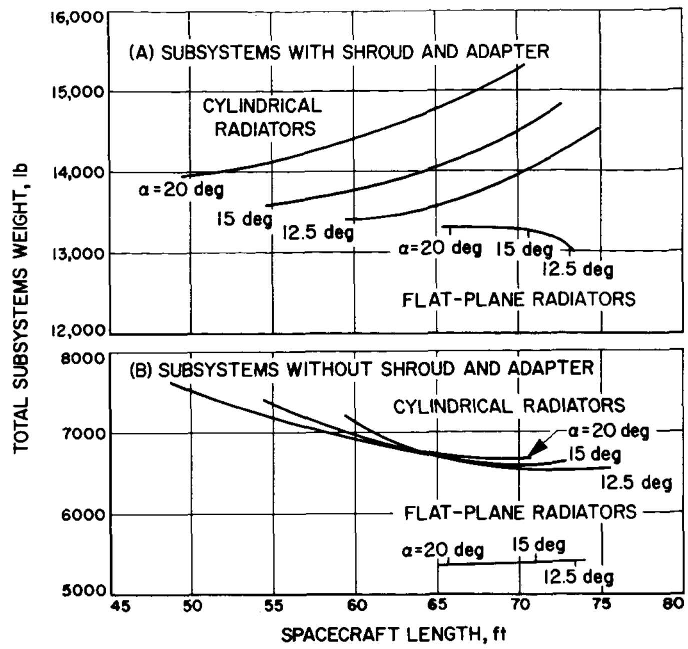

Fig. 13. Total subsystems weight vs spacecraft length

combining Figs. 5, 7, 8, 9, and 12. The upper family of curves represents the sum of all five subsystems as they would be used during the launch. The lower graph includes only the structure, radiators, and shield subsystems which are a permanent part of the spacecraft, and hence represents a flight condition. From Fig. 13A & B, the minimum weights of the subsystems for either concept for both launch and flight conditions show a weak dependence on the shroud semi-vertex angle.

In the flat-plane configuration, the spacecraft weights for both launch and flight conditions remain nearly constant as shown by Fig. 13A & B. For launch conditions, the longest spacecraft will be lighter than the shortest spacecraft by only 2.3%. For flight conditions, the spacecraft weight remains nearly constant and is lightest for the shortest spacecraft. To determine the optimum configuration for the spacecraft, a flight dynamics study must be made. However, it is believed that the shorter spacecraft is favored for booster requirements.

In Table 1, the subsystems masses for the minimum-weight spacecraft are expressed as a percentage of the minimum total spacecraft launch-weight of 26,900 lb for both concepts. Also shown are the maximum weight deviations from the minimum total spacecraft weight of the various subsystems, due to spacecraft configuration changes. Group I lists the subsystems whose individual masses vary for changes in spacecraft configuration

Table 1. Optimum spacecraft mass summary

Group	Subsystem or component	Flat-plane configuration		Cylindrical configuration	
		Weight percentage of total spacecraft, %	Maximum weight deviation[a] of total spacecraft weight, %	Weight percentage of total spacecraft, %	Maximum weight deviation[a] of total spacecraft weight, %
I	Nuclear-radiation shield	10.3	1.3	16.6	+0.2
	Spacecraft structure	8.0	−0.9	8.4	−2.0
	Booster-spacecraft adapter	4.7	−0.2	4.5	+0.8
	Aerodynamic shroud	24.6	+0.5	18.9	+7.2
	Heat-rejection radiators	1.3	+0.03	1.5	+0.05
	Coolant inventory	0.7	Negligible	0.7	Negligible
	Total of Group I[b]	49.6	+0.73	50.6	+6.25
II	Power plant components and startup equipment less radiators and coolant inventory	16.8	Negligible	17.4	Negligible
	Power conditioning equipment	2.2	Negligible	2.2	Negligible
	Ion motors	1.9	Negligible	1.9	Negligible
	Propellant and tankage	21.9	Negligible	21.9	Negligible
	Communications	1.5	Negligible	1.5	Negligible
	Guidance and control	2.4	Negligible	2.4	Negligible
	Scientific instrumentation	3.7	Negligible	2.1	Negligible
	Total of Group II[b]	50.4	Negligible	49.4	Negligible

[a]Deviation is due to a change in spacecraft configuration parameters.

[b]Based on an optimum total spacecraft weight of 26,900 lb at launch.

parameters. Group II includes the remaining spacecraft subsystems whose masses are not significantly affected by configuration changes and therefore remain constant. It can be seen by the summation of the Group I subsystems that the flat-plane configuration weight is quite insensitive to configuration changes, varying by only 0.73%, while the conical-cylindrical configuration weight can vary as much as 6.25% from the minimum weight.

The propellant masses shown in Table 1 were assumed to remain constant for all the configurations considered. However, the flat-plane configuration spacecraft would carry about 75% more scientific instrumentation mass than the cylindrical-configuration spacecraft. The altitude of the parking orbit was also assumed to vary from the minimum 700-nmi distance for the various booster payloads considered. Weight tradeoffs between the shroud, adapter, and propellant requirements can be made to evaluate the effect which these tradeoffs have on the scientific instrumentation payload capabilities of the spacecraft. However, it is estimated that the effect of such a tradeoff study on the results shown in Fig. 13A and B would be of second order or less, and consequently would not significantly change the results of this study.

If the shroud for the cylindrical concept is used both as a spacecraft structural member supporting other subsystems and as the radiator systems (Fig. 11A), the maximum weight saving would be about 9.9% of the total spacecraft weight. This percentage figure is the combined figures of the spacecraft structure and radiators (Table 1). For a total minimum spacecraft mass of 26,900 lb at launch, this weight saving would be 2,660 lb. Subtracting the weight saving from the minimum total weight of the five combined subsystems of 13,610 lb, shown in Fig. 13A, results in 10,750 lb. This result is the weight of the subsystems in flight conditions when the spacecraft employs the shroud, both as radiators and structural systems. However, only 6,500 lb are required for these three subsystems in flight conditions using an ejectable shroud system (Fig. 13B). Since these weights are included in the total spacecraft weight during the entire mission, using the shroud both as a radiator and structural system does not appear feasible for this mission.

The subsystems weight requirements for the flat-plane configuration during flight conditions shown in Fig. 13B are about 1,100 lb less than those for the cylindrical configuration, both having the same minimum flight weight. This difference can be translated into an increased payload capability or a shorter mission time. Therefore, the results shown in Fig. 13B and Table 1 indicate that, within the constraints and assumptions made in this preliminary study, the flat-plane radiator configuration is the more favorable configuration concept. It has also been found that the nuclear-radiation shield, aerodynamic shroud, and spacecraft structure are the predominant subsystems in establishing the optimum configuration of the spacecraft; other subsystems have only a second-order effect.

NOMENCLATURE

A surface area, ft^2

c dimension parameter, ft

D radiator system overall diameter, ft

d tube diameter, ft

F view factor, sterad

f friction factor

g gravitational constant, ft/sec^2

K constant

l fin width, ft

N number of tubes

L distance from shield surface to dose point; column length, ft

$P_{(0)}$ probability of no catastrophic impacts by meteoroidal particles

Q heat transfer rate, Btu/hr

$\mathcal{R}$ protection requirement ratio for an interplanetary mission relative to near Earth

R equivalent disc source radius (1.46 ft for flat-plane, 1.88 ft for cylindrical)

Re Reynolds number

T mission time, sec

t thickness, ft

V flow rate, ft/sec

x geometry parameters, ft

y geometry parameters, ft

Z length of tubes, ft

α shroud semi-vertex angle, deg

β albedo

ε surface net hemispherical emittance

η heat transfer effectiveness of the fin

Δ thickness, ft

σ Stefan-Boltzmann constant, 0.1712×10^{-8} Btu/hr ft^2 $°R^4$

ρ liquid density, lb/ft^3

θ temperature, °R

Φ neutron flux at equivalent disc source surface, n cm^{-2} sec^{-1}

Φ_d direct neutron flux at payload location n cm^{-2} sec^{-1}

Φ_s scattered neutron flux at payload location n cm^{-2} sec^{-1}

Subscripts

a armor

f fin

h header, return or supply

i inside

l liner

t tubes

v vulnerable

F flat plane

C cylindrical

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