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SENSE II

SPACE APPLICATIONS OF NUCLEAR POWER

VOLUME I
COMMERCIAL COMMUNICATIONS SATELLITE

PREPARED FOR
SPACE NUCLEAR SYSTEMS DIVISION
ATOMIC ENERGY COMMISSION

CONTRACT NO. AT (49-15) - 3063

MAY, 1974

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SENSE II -- SPACE APPLICATIONS OF NUCLEAR POWER
VOLUME I - COMMERCIAL COMMUNICATIONS SATELLITES

TABLE OF CONTENTS

<u>Section</u>	<u>Page</u>
1. Introduction	1
1.1 Scope of the Study	1
1.2 Conclusions	2
1.2.1 Conclusions Relating to Commercial Communications Satellites	3
1.2.2 Conclusions Relating to Military Missions (Volume II)	4
2. Commercial Communications Satellites	5
2.1 Introduction	5
2.2 Baseline Design -- Solar Powered	13
2.2.1 Overview	13
2.2.2 Electric Power System	16
2.2.3 Thermal Control System	16
3. Review of Power Systems	21
3.1 Nuclear Systems	21
3.1.1 Static Systems	21
3.1.2 Dynamic Systems	22
3.2 Solar Systems	27
3.3 Direct Comparisons	27
4. Economic Impact of Spacecraft Design Alternatives	31
4.1 Introduction	31
4.2 Factors Affecting Satellite Revenue	32
4.3 Redundancy, Reliability and Revenue	35
4.3.1 Redundancy	35
4.3.2 Reliability Calculations	39
4.3.3 Revenue	41
4.4 Conclusions	44

TABLE OF CONTENTS (cont'd)

<u>Section</u>	<u>Page</u>
5. Nuclear-Powered Spacecraft	48
5.1 Design Approaches	48
5.2 Alternative Approaches	49
5.2.1 Integrated	49
5.2.2 Semi-Integrated	49
5.2.3 Separated	51
5.3 Nuclear-Powered Satellite Reference Designs	52
5.3.1 Separated Design	52
5.3.2 Integrated Design	56
5.3.2.1 Configuration	56
5.3.2.2 Radiator Design	58
5.3.2.3 Thermal Control	63
5.3.3 Shielding Requirements for Curium Heat Source	64
5.3.3.1 Boundary Conditions	64
5.3.3.2 Conclusions	66
5.3.3.3 Use of the Hydrazine as a Neutron Shield	67
6. Solar vs. Nuclear Spacecraft	71
6.1 Power and Weight Breakdown	71
6.2 Revenue vs. Cost	75

APPENDIX

- A. Domestic Communications Satellites -- History & Technology
- B. Reliability of Redundant Configurations
- C. Thermal Design
- D. Effects of Nuclear Radiation on Typical Communications Satellite Components
- E. Structural Design and Detailed Weight Estimates

TABLE OF ILLUSTRATIONS

<u>Figure No.</u>		<u>Page</u>
2.1	U.S. Domestic Communications Traffic Projections	10
2.2	Bandwidth Allocations for Commercial Communications Satellites	12
2.3	Solar Powered Domestic Communications Satellites	14
3.1	Selenide RTG (Ref. 6)	23
3.2	Organic Rankine System Reference Design (Ref. 6)	24
3.3	Power System Weight Comparisons	29
3.4	Power System Recurring Cost Comparisons	30
4.1	Baseline Satellite Subsystem Reliability (excluding TWTA's)	34
4.2	TWTA Reliability Values	36
4.3	Single Redundancy Switching Arrangement	38
4.4	Semi-Redundancy Switching Arrangement	38
4.5	Expected Transponder Failure Times	42
4.6	Value of Weight in Orbit	45
5.1	Integrated Approach	50
5.2	Semi-Integrated	50
5.3	Separated Approach	50
5.4	Separated Design Configuration	54
5.5	Equipment Mounting Arrangement Separated Design	55
5.6	Separated Design; Stowed Configuration	57
5.7	Integrated Design Configuration	59
5.8	Equipment Mounting Arrangement; Integrated Design	60
5.9	Integrated Design; Basic Truss Structure	61
5.10	Integrated Radiator Details	62

TABLE OF ILLUSTRATIONS

<u>Figure No.</u>		<u>Page</u>
5.11	Radiator Temperature Variation; Integrated Design	65
5.12	Relative Effectiveness of a Linearly Consumed Shield (Compared to a Permanent Shield) as a Function of Shield Attenuation Ratio (Cm-244 Source)	70
6.1	24-Transponder/Delta 3914 Satellite, Nuclear-vs. Solar-Powered Versions	76
A-1	4&6 GHz Channel/Frequency Plan	A-13
A-2	Weight & Power of C-Band & K-Band Transponders	A-24
A-3	Atlas/Centaur Class 48-Transponder Satellite Weight Breakdown and Prime Power Requirements.	A-27
B-1	Single Redundancy	B-1
B-2	50% Redundancy	B-3
C-1	Four Alternative Spacecraft Configuration	C-2
C-2	Radiator Temperature vs. Heat Dissipation for the Integrated Configuration	C-5
C-3	Radiator Temperature vs. Equipment Dissipation for the Integrated Configuration in all Orbits	C-6
C-4	Payload Temperature vs. Dissipation as a Function of Internal Radiative Coupling	C-9
C-5	Condenser/Radiator Temperature vs. Equipment as a Function of Internal Radiative Coupling	C-10
C-6	Payload Radiator Temperature vs Dissipation for the Separated Configuration with Thermal Coupling	C-11
C-7	Radiator Temperature vs Sun Angles for the Separated Configuration with no AKM	C-13
C-8	Temperature Rise of Condenser/ Radiator During Ascent Separated Configuration	C-16

TABLE OF ILLUSTRATIONS

<u>Figure No.</u>		<u>Page</u>
C-9	Ground Cooling of Condenser/Radiator	C-17
C-10	Temperature Rise of Condenser/Radiator During Ascent; Integrated Configuration	C-18
D-1	D.C. to D.C. Converter	D-21
D-2	Power Switch	D-21
E-1	Exploded View	E-2
E-2	Panel Cross Section	E-4
E-3	Equipment Panels Weight Distribution	E-7

LIST OF TABLES

<u>Table Number</u>		<u>Page</u>
2-1	Summary of Intelsat Satellites (International Service)	8
2-2	Summary of Domestic Satellites	9
2-3	Major Spacecraft Parameters	14
2-4	Electrical Power Requirements; Solar Powered	17
2-5	Electric Power Subsystem Equipment Summary	18
2-6	Heater Locations	20
3-1	Nuclear Power Systems Major Parameters (400-WATT(E) (EOM)	26
3-2	Solar Power System Parameters	28
4-1	Present Value of Expected Revenue for 24-Transponder Satellite	43
5-1	Advantages and Disadvantages	53
5-2	Radiation Hardness Summary	68
6-1	Relative Power Budgets	72
6-2	Spacecraft Weight Comparisons	73
6-3	Revenue & Cost Comparisons: Nuclear vs. Solar Satellites Nuclear Minus Solar Values, Millions of Dollars	79
A-1	Frequency Allocations for Fixed-Satellite Service Operations in the United States	A-8
A-2	Comparison of Current Domestic Spacecraft Characteristics	A-10
A-3	Estimated Cost of Payload in Geosynchronous Orbit (September, 1973) ¹	A-15
A-4	Revenue Requirements	A-29
C-1	Properties of Thermal Coatings	C-1
C-2	Thermal Control System Weights, lbs.	C-8
C-3	Comparison of the Delta Spacecraft Configurations	C-14
D-1	Representative Semiconductor Components of Communications Satellite Subsystems with Associated Radiation Damage Information	D-3

LIST OF TABLES

<u>Table Number</u>		<u>Page</u>
D-2	Candidate Transistors for Frequency Multiplication	D-9
D-3	Radiation-Hardened Signal-Level Transistors (Fairchild Semiconductor Co.)	D-13
D-4	Radiation-Hardened Power Transistors	D-14
D-5	Radiation-Hardened Digital Integrated Circuits	D-16
E-1	Radiator Component Weights	E-4
E-2	Equipment Panel Weights	E-6
E-3	Weight of Structural Framing Members	E-14

1.0 INTRODUCTION

1.1 SCOPE OF THE STUDY

This report presents the results of the second phase of the SENSE Program, a continuing study of space applications for nuclear power.

In the first phase of the study (SENSE I), the applicability of standardized nuclear-powered spacecraft for performing NASA unmanned missions launched on the Space Shuttle was investigated. The present study concerns the use of nuclear radioisotope power systems for commercial communications satellites and various military space missions, launched on present-day booster vehicles. Also, the present study is predicated on the use of advanced or next-generation nuclear radioisotope power systems. The study was intended both as an evaluation of the competitiveness of such systems for missions which have heretofore been performed almost exclusively by solar power, and as a guide to the power system designers as to which characteristics of the systems are likely to be of major importance to this segment of the user community.

These advanced nuclear-radioisotope systems have been the subjects of recent studies for the Space Nuclear Systems Division of USAEC, which were performed partially concurrent with the present study. Two types of conversion systems were studied: static and dynamic. The studies were based primarily on the use of Curium-244 as the radioisotope fuel; however, the present study considers both ^{244}Cm as well as the current space-standard nuclear fuel: ^{238}Pu .

The present study was originally specified to include only the high-efficiency dynamic-conversion systems: the Mini-Brayton- and Organic-Rankine-Cycle Systems, which are now believed to be capable of long-term unattended space operation. As the economic portion of the study progressed, it became clear that low weight would be a highly leveraged economic factor in the design

of commercial as well as certain military spacecraft on existing boosters. The concurrent power system studies revealed a distinct weight advantage for the Organic-Rankine system as compared to the Brayton system in the power-range of major present interest (500-2000 watt(e)). Therefore, the former received major attention in the study. At the same time, it became apparent that static generator designs based on the new selenide thermoelectric materials would be capable of achieving specific power values approximately equal to the Organic Rankine system. Therefore, a representative selenide RTG was included for further consideration.

In comparing nuclear-powered with solar-powered spacecraft, advances in solar converters and in storage battery systems were also postulated, insofar as these are now foreseeable.

This report is contained in two volumes: Volume I presents the results of the study of commercial communications satellites, and Volume II, classified Secret, the results of the DoD mission study, including the preliminary evaluation of a nuclear-powered GPS (Global Positioning System) Satellite.

1.2 CONCLUSIONS

The major general conclusion of the study is the following:

- Comparisons of nuclear vs. solar power must be based on complete systems evaluations in order to achieve a true and complete comparison.

Based on the most optimistic future projections of both solar and nuclear technology, a comparison of power subsystems alone would continue to favor solar power. However, reduction in the power consumption and weight of other spacecraft subsystems as a direct result of using nuclear power, can result in a totally competitive or superior nuclear spacecraft.

1.2.1

CONCLUSIONS RELATING TO COMMERCIAL COMMUNICATIONS SATELLITES

- Organic Rankine- and Selenide-RTG-powered spacecraft are commercially superior to state-of-the-art solar-array/battery-powered spacecraft, and potentially competitive with the most advanced solar-array/battery systems.

This is due to:

- 1.) spacecraft net-weight reductions of $\sim 5\%$.
  - 2.) "housekeeping" power reductions of 35-to-55 percent.
- The power requirements and weight goals for nuclear power to be competitive with solar-array power are:

Sat. System	Power Range	Wt. Goal
Current C-Band	500-550 W(e)	≥ 3 W(e)/lb.
Replacement C-Band	500-1000	≥ 4
Future C+K Band	1500-2000	≥ 5

- Organic-Rankine-System power permits a fully-integrated satellite configuration resulting in a minimum weight design of optimum thermal characteristics.
- Selenide-RTG power permits low-weight design of superior fabrication, integration and start-up characteristics.
- The economics of commercial satellite operations provide that an increased satellite weight capability, if used to extend on-station lifetime of the spacecraft, can be translated into a substantial revenue increase per satellite. In terms of present value, the revenue value of incremental weight is 1/2-to-1 million dollars per pound. Relative cost differences between solar and nuclear power can be easily over-shadowed by the

relative revenue differences resulting from a net weight advantage of one of the systems.

- Nuclear power systems can match the flat power requirements of commercial communications satellites without need for potentially life-limiting batteries.
- With proper design and component selection, a curium heat source can probably be used without in-flight shielding for this mission.

2.0 COMMERCIAL COMMUNICATIONS SATELLITES

2.1 INTRODUCTION

Commercial communications satellites were selected as a primary objective of the study, because this is a basically new class of mission, and one likely to generate a continuing requirement for satellites over an extended period of time.

Also, relative value could be more readily established on a quantitative basis because of the commercial nature of the mission, permitting comparisons to be based on clear economic factors.

However, in choosing this mission, a built-in handicap to nuclear power was recognized. The geosynchronous orbit is among the most favorable for solar power; due primarily to three factors:

1. In the equatorial orbit the sun angle is confined within ± 23.5 -degrees of the spacecraft axis at all times.

2. Eclipse occurs relatively rarely during the year at this altitude. During maximum eclipse periods, which occur only near solar equinox, solar-array output is maximum for an array oriented along the north-south axis, and about 23 hours is available for battery charging between successive occultations. This tends to minimize both array and battery size. In general, an array oriented along the north-south axis and sized for full mission power at solar solstice will automatically produce enough additional power at solar equinox to charge the batteries, when, fortuitously, these are most needed.

3. The beam-pointing requirements of this mission requires a three-axis control system for the antennas. Therefore, the use of an oriented solar array introduces little additional attitude-control requirements to the mission.

There are basically two components to the commercial satellite business: international and domestic. Although the international sector has been provided with satellite service since 1965 via the Intelsat Consortium, U.S. domestic satellite operations are only now beginning. The first U.S. domestic satellite (Westar) was launched on April 13, 1974.

The major difference between the two types of service is a political-economic one which, however, has ultimate impact on technological approach. International Telecommunications Satellite Consortium (Intelsat) is a government-supported monopoly organization. By agreement among the governments which are signatories to the Intelsat treaty, it has sole operating authority for international communications satellites, with the U.S. Comsat Corporation currently acting as manager of this program. As such, it has some opportunity to investigate and experiment with new technological approaches without severe competitive pressure. However, its design decisions need not be based on free-market economic factors.

On the other hand, the Federal Communications Commission has elected to allow (regulated) free-market competition among domestic satellite operators who meet FCC standards. Although this may result in a less experimental tendency among the operating companies, the value of various design decisions are amenable, at least in theory, to economic comparisons. The present study was based on a domestic satellite configuration.

Perhaps the most significant technological difference between the two approaches at the present time is in the selection of launch vehicles: The Intelsat IV and IVA are launched on the Atlas/Centaur (Table 2-1), while the proposed domestic satellites of equivalent transponder capacity are, with one prominent exception, launched on the

lower-cost Delta (Table 2-2). The effect on satellite weight can be seen from the tables. One result of this severe weight limitation is to force the domestic satellite designers into carefully weight-optimized approaches.

Tables 2-1 and 2-2 summarize the technical characteristics of the international and domestic satellites known or announced to date. A review of the history and technology of the domestic satellite business is contained in Appendix A.

Although any prediction of the growth rate of domestic satellite service is highly speculative at this time, it should be noted that, historically, domestic telephone and message traffic has exceeded international traffic by a substantial margin, even excluding purely local and short-range service. For example, if the Bell System were to rearrange its trunk lines for satellite transmission of all long-range telephone calls, its present needs would be for 130,000 satellite voice circuits (Ref. 1). This compares with an under-utilized present Trans-Atlantic satellite capacity of approximately 21,000 voice circuits.

AT&T has estimated the growth rate of this requirement, as shown by the dashed curve in Figure 2.1. An independent analysis of requirements for telephone as well as various other services has been made, as shown by the solid curves of Figure 2.1. Of course, current telephone requirements are now satisfied by land lines and terrestrial microwave links while the "other" services are, to a large extent, unsatisfied (and undeveloped) at present.

Figure 2.1 also shows the transponder capacity in primary and on-orbit back-up circuits now expected to be available from the first-generation domestic satellites, based on applications made to the FCC to date (Appendix A). Replacement spacecraft are not shown; these will to some extent, await the results of the first generation spacecraft shown.

Table 2-1

SUMMARY OF INTELSAT SATELLITES
(INTERNATIONAL SERVICE)

	OPERATIONAL		COMMITTED
	IS III	IS IV	IS IV A
SPACECRAFT COST		16 M	24 M
TRANSPONDER CONFIG.			
C-BAND	2 x 240 - MHz	12 x 36 - MHz	24 x 36 - MHz
K-BAND	0	0	0
NO. SPACECRAFT AND COVERAGE	3 ATL. 2 PAC.	2 ATL. 1 PAC. 1 IND.	
EOM PRIME POWER, W	130	478	
CAPACITY			
VIDEO	4	12	24
ONE-WAY VOICE	1200	3000/9000	28,800
SPACECRAFT WEIGHT	335	1600	
LAUNCH VEHICLE	DELTA	ATLAS- CENTAUR	ATLAS- CENTAUR
OPERATIONAL TIME FRAME	1968 - 1973	1971 - 1978	1975 - 1982

Table 2-2

SUMMARY OF DOMESTIC SATELLITES

	OPERAT.	COMMITTED					PLANNED	
	ANIK-1	WESTAR	AMERSAT-I	ATT/COMSAT	RCA GLOBCOM	GTE/HUGHES	CML	AMERSAT-II
SPACECRAFT COST	8 M	8 M	8 M	16 M	11 - 15 M*	8 M	11 - 15 M*	
TRANSPONDER CONFIG. C-BAND K-BAND	12 x 36 - MHz 0	12 x 36 - MHz 0	12 x 36 - MHz 0	24 x 36 - MHz EXP	24 x 36 - MHz 0	12 x 36 - MHz 0	0 2 - 4 x 72 MHz	24 x 36 - MHz
NO. SPACECRAFT AND COVERAGE	1 + 1 CANADA + NORTHERN US	1 + 1 CONUS	1 + 1 CONUS	2 + 1 50 STATES + P. RICO	2 + 1 50 STATES	1 + 1 CONUS	1 + 1 (SPOT BEAMS) ?	
EOM PRIME POWER, W	250	250	250		620	250		
CAPACITY VIDEO ONE-WAY VOICE	12 14,400	12 14,400	12 14,400	24 28,800	24 28,800	12 14,400	12,000	24 28,800
SPACECRAFT WEIGHT	655	655	655	1600	925	655		925
LAUNCH VEHICLE	DELTA 2914	DELTA 2914	DELTA 2914	ATLAS- CENTAUR	DELTA 3914	DELTA 2914	DELTA 2914	DELTA 3914
OPERATIONAL TIME FRAME	1972 - 1979	1974 - 1981	1974 - 1981	1975 - 1983	1975 - 1983	1975 - 1983	1976 - 1984	1976 - 1986

*ESTIMATED RANGE

* BASED ON BOOZ, ALLEN, HAMILTON SURVEY FOR CML; ASSUMING 1000 SIMPLEX VOICE CIRCUITS PER TRANSPONDER, 1 TV CHANNEL PER TRANSPONDER, 1.38 BPS 1 Hz BANDWIDTH FOR ELECTRONIC MAIL AND DATA

Figure 2.1 U. S. DOMESTIC COMMUNICATIONS TRAFFIC PROJECTIONS

The total capacity of orbital slots for full continental U.S. coverage is also shown in Figure 2.1, based on single and double utilization of the assigned frequency bands (Figure 2.2), with orbital spacings as recommended by FCC (Appendix A). Clearly, pressure for frequency re-use (e.g., 24 C-band transponders per satellite) will be intense if the traffic projections are even approximately accurate.

Beyond C-band usage, different operators may prefer either Ku-band or Ka-band transmission (Figure 2.2). Although it involves greater signal-propagation problems, the Bell System has expressed a clear interest in the high-capacity Ka-band as more in keeping with its long-range growth requirements. For the present and probably next generation as well, domestic operators, with the possible exception of AT&T, will tend to concentrate on C-Band satellites incorporating frequency re-use by dualpolarization. This approach provided the baseline mission for this study.

The study was based on a detailed design for a 24-transponder solar-powered satellite, designed by a Fairchild-TRW team for launch on the uprated Delta 3914. This design, briefly described in Section 2.2, was used as the basis for competing nuclear-powered designs, described in Section 5. Although various uncertainties can be expected to exist in any purely paper design, however detailed, the comparative evaluation of the two approaches based on the same assumptions is believed to be more significant than the absolute values of either of the designs themselves.

AS PER 1971 INTERNATIONAL AGREEMENT (WORLD ADMINISTRATIVE RADIO CONFERENCE)

Figure 2.2 Bandwidth Allocations for Commercial Communications Satellites

2.2 BASELINE DESIGN -- SOLAR POWERED

2.2.1 OVERVIEW

The baseline solar-powered satellite is typical of the new generation of three-axis-stabilized spacecraft designed for launch on the up-rated Delta 3914. This booster lifts approximately 2000 pounds into synchronous-transfer orbit. The spacecraft is designed with an internal apogee-kick-motor (AKM) weighing approximately 1000 pounds including connecting structure, leaving some 1000 pounds for useful payload. The system is designed for spin-up prior to firing the Delta upper stage (perigee kick motor) and for continued spinning until AKM firing for final injection. Following successful injection, ground control then commands despin, final orientation, and solar array deployment. Because of its stationary orbit, the spacecraft is under continuous ground control.

Table 2-3 lists the major performance parameters of the baseline spacecraft.

As shown in Figure 2.3, the spacecraft is hexagonal in shape. The outer structure consists of honeycomb sandwich panels and framing, connected by GFRP trusses to a beryllium thrust cone. This, in turn, attaches to the Delta Launch adapter to form the primary structural load path. The internal faces of the honeycomb panels serve as both structural and heat transfer surfaces for equipment mounting. Therefore, aluminum sheet with thicknesses which vary between 0.010 to 0.080 inch is used for these internal faces; the latter thickness where heat dissipation is concentrated. The outer facing material of the honeycomb sandwich is .040-inch beryllium.

The two large deployable solar array panels are wrapped around the satellite body while the satellite is in a spin-stabilized mode, prior to synchronous orbit injection. The deployed arrays project on booms from

Table 2-3
MAJOR SPACECRAFT PARAMETERS

PARAMETER	CHARACTERISTIC
SIZE (MAIN BODY)	50-INCH HEIGHT; 86-IN. ACROSS CORNERS
WEIGHT, AT LAUNCH IN ORBIT	2,000 LBS. 1,015 LBS.
ELECTRIC POWER, SOLSTICE	833 W(e) (BOL), 580 W(e) (EOM) 86.2 SQ. FT. ARRAY, 28 A-HR BATTERIES
ATTITUDE CONTROL	± 0.15 DEGREE; 3-AXIS WITH BIAS-MOMENTUM AND SOLAR-ARRAY DRIVE ASSEMBLY
REACTION CONTROL	HYDRAZINE MONOPROPELLANT; 0.1 AND 0.5-LB. THRUSTERS
TELEMETRY & CONTROL	C-BAND
POWER DISTRIBUTION	REDUNDANT 20-65 VOLT UNREGULATED BUSES; 20-33 VOLT REGULATED BUSES
THERMAL CONTROL	SELECTIVE COATINGS PLUS HEATER/ THERMOSTAT UNITS FOR COLD EXTREMES
COMMUNICATIONS	24 40-MHz C-BAND TRANSPONDERS, DUAL POLARIZATION ANTENNAS

SOLAR-POWERED DOMESTIC COMMUNICATIONS SATELLITE

Figure 2.3 Solar Powered Domestic Communications Satellites

the north and south faces. The array panels are clock-driven by a solar-array drive assembly to maintain a near normal sun-line for maximum solar energy input, while the antenna mounting surface is continuously earth-directed through the gyroscopic action of the momentum wheel.

The hydrazine-fueled Reaction Control System is periodically used to unload the momentum wheel, realign attitude in roll, and correct and maintain orbit station.

The Tracking, Telemetry and Command sub-system operates on C-band with a nearly-omnidirectional antenna shown extended from the center of the polarization grid (Figure 2.3). It functions throughout the transfer orbit and apogee injection maneuver as well as at all times on station.

The Attitude Control System operates in two modes: 1) spin stabilized passively damped with thruster control during transfer, and 2) as a 3-axis control system after apogee injection and despin. The satellite has a favorable moment of inertia for spin stability in addition to being statically and dynamically balanced for the orbit transfer and injection sequence.

The 960-pound AKM which protrudes from the aft end (not visible in Figure 2.3), provides a velocity increment sufficient to perform plane change and orbit circularization. The burnout weight of the satellite including the spent AKM case is approximately 1015 pounds.

The Electric Power and Thermal Control Systems are of particular interest to this study since these are subsystems undergoing major change in converting to nuclear power. Therefore, these are discussed separately below:

2.2.2 ELECTRIC POWER SYSTEM

This system provides adequate electric power for 24-transponder operation for 7 years, including eclipse periods, and employs both non-regulated, as well as regulated redundant buses (Table 2-3). The regulated bus supplies all "housekeeping" functions, whereas the communications payload is supplied with unregulated power which is then conditioned within the individual transponder power supplies. A separate 28-volt regulator is provided for the thermal-control heaters. Also, most housekeeping modules include additional voltage regulation for their particular requirements.

The solar array uses standard n-on- 2 x 4 cm silicon cells with .010-inch fused-silica cover glass. These are mounted to fiberglass-faced 1/2-inch thick aluminum honeycomb substrate. There are two batteries, each with 14 amp-hour nickel-cadmium cells.

The electric power budget for solar solstice and equinox is shown in Table 2-4.

Table 2-5 gives the detailed EPS weight summary. Figure 2-4 shows the location of EPS elements relative to the major system elements.

2.2.3 THERMAL CONTROL SYSTEM

The basic approach to equipment thermal control is the use of a variety of surface coatings and heater/thermostat sets. An insulation blanket of alternating layers of Mylar and Kapton covers most external surfaces. Cutouts in the blanket expose second-surface fused-silica mirrors wherever dissipative equipment is mounted; including those portions of the north and south walls on which are mounted the traveling wave tube amplifiers (TWTA's) and their power supplies. Heat-load variations on the TWTA radiators result from two major sources: amplifier operation either at saturation (full signal transfer) or zero drive (no signal transfer), resulting in a 30% variation in local heat load; and variations of solar incidence angle of $\pm 23 - 1/2$ degrees. These produce gross temperature extremes of 20-to-120 F

Table 2-4. Electrical Power Requirements; Solar Powered

SUBSYSTEM	POWER CONSUMPTION	
	SOLSTICE	EQUINOX
COMMUNICATIONS	424	424
TELEM. & CONTROL	21	21
ATTITUDE CONTROL	28	28
REACTION CONTROL	0.5	0.5
ELECTRICAL POWER & DIST.	36	68
POWER DIST. ASSY	24	24
HARNESS	12	12
BATTERY CHARGING	-	32
THERMAL CONTROL	69.5	39.5
BATTERY HEATERS	30	
OTHER HEATERS*	39.5	39.5
TOTAL	579	581
ARRAY OUTPUT, 7 YRS. EOM	580	620

* see Table 2-6

Table 2-5

ELECTRIC POWER SUBSYSTEM EQUIPMENT SUMMARY

EQUIPMENT	QUANTITY	TOTAL WGT. LB.
SOLAR PANELS	6	55.9
BATTERY ASSEMBLY	2	95.6
POWER DISTRIBUTION	1	25.4
28 VOLT REGULATOR	1	7.0
ORDNANCE CONTROL	1	2.0
BUS TIE	1	1.0
OTHER COMPONENTS		15.4
INTERCONNECTION HARNESS		48.0

on these radiators. Surface heaters are provided to maintain the lower limit, if necessary.

Batteries have tighter temperature restrictions than the other components; 10-to-40 F. They are insulated from the interior and given an isolated heat sink radiator on two of the east-west walls. A charge-rate-regulating thermal switch and heater combination provide active thermal control for the batteries. The east-west panels undergo maximum sun-angle variation, but these support the relatively low-dissipation house-keeping modules, and are provided with heaters to maintain the lower temperature limit. Table 2-6 shows the heater allocations for various regions of the satellite.

Thermal considerations have also influenced the design of satellite appendages. To prevent thermal distortion of the polarizing screen, the antenna array and screen is covered with a .003-inch Kapton film coated internally with thermatrol white paint. In addition, the solar array panels must be extended from the north and south faces with booms at least 4 feet in length to avoid blockage of the critical north-and south-face radiators.

Table 2-6. Heater Locations

LOCATION	NUMBER	(WATTS, EACH)
TWT*	24	9.3
TWT-POWER SUPPLY*	8	7.5
TRANSPONDERS RECEIVER	2	3.0
TDPA	2	1.5
EAST/WEST PANELS	8	2.5, 3.5, 6.0, 6.5 (2 OF EACH)
BATTERIES	2	15
AKM	1	12
RCS TANKS	2	2
RCS LINES	2	1.5
RCS VALVES	18	0.25
RCS THRUSTERS	18	1.5

* Used only in the event of transponder failure

3.0 REVIEW OF POWER SYSTEMS

3.1 NUCLEAR SYSTEMS

As explained in Section 1, the study was to be based on radio-isotope dynamic system under concurrent study (Refs. 2, 3). Toward the later stages of the study, consideration of a selenide ERTG was included when it became apparent that the specific power of this system can be comparable to or greater than that of the dynamic systems. However, this consideration is mostly reflected in Volume II of this report; Military Missions.

3.1.1 STATIC SYSTEMS

State-of-the-art radioisotope generators all employ thermoelectric solid-state converters made of doped lead-telluride or silicon-germanium, and use plutonium-238 radioisotope fuel in blast-, fire-, and re-entry-proof containers. Despite the rigid fuel containment requirements, current versions of these generators achieve up to 2 watts(e) per pound in the range of 150-to-200 watts per generator (Ref. 4). Converter efficiencies fall in the range of 5-to-7 percent. Because the plutonium fuel, at 650 dollars per thermal watt, has been the most expensive part of these RTG's, converter efficiency has a strong influence on the unit cost of these devices. Therefore, a strong incentive has existed to increase conversion efficiency.

Increased conversion efficiency is now believed to be achievable in either of two ways: 1.) new high-efficiency thermoelectric materials
2.) turbine-generator conversion.

The first approach has resulted in the development, under AEC sponsorship, of a new class of thermoelectric materials based on selenide alloys (Ref. 5). Current "standard" selenide materials are believed to be capable of achieving 10-to-11 percent conversion efficiencies reproducibly, with an ultimate efficiency of 13-to-15 percent expected from this material. Lifetesting indicates non-degrading performance with time, promising a long life potential for these selenide materials. Two parallel studies on

generator designs utilizing selenide converters have recently been completed, in which overall generator efficiencies of 10-to-11 percent were predicted based on thermoelectric efficiencies of approximately 12 percent (Ref. 6, 7). One of these designs is shown in Figure 3.1.

3.1.2 DYNAMIC SYSTEMS

The other approach now believed to be feasible in the several-hundred-watt range, uses miniature turbine-generator units. The turbines are driven by gas which has been heated by the nuclear fuel, and which is then returned to the heater in a closed cycle. Low- and fractional-horsepower turbines and a new form of non-contacting hydrodynamic gas-film bearing for the rotating unit, have been developed for aircraft APU and other applications. Successful long-term operation under multiple start-stop conditions has imparted confidence in the readiness of this approach.

Design studies on each of two different turbine-generator systems were recently completed. One uses a Brayton conversion cycle, in which the working fluid is an inert gas (Ref. 8); the other a Rankine cycle, in which an organic hydrocarbon working fluid is used (Ref. 9). A number of these organic fluids have thermodynamic properties which result in high cycle efficiencies at lower peak cycle temperatures than, for example, steam or metal-vapor working fluids, which were proposed for space applications in former years. A satisfactory body of experience with the applicable organic fluids has been acquired in recent years in experimental automotive, artificial heart, and other engines in a variety of sizes.

Both Brayton and organic-Rankine systems promise overall system conversion efficiencies in excess of 20%. Each has its advantages and particular virtues. However, in the size range of interest to the reference mission, the organic-Rankine system is significantly lighter and was, therefore, selected as the dynamic-system representative in this study.

Figure 3.2 shows the basic components of the organic-Rankine system. In the nuclear spacecraft the components are re-arranged to fit within the available volume. In particular, the largest single component,

Figure 3.1 Selenide RTG (Ref. 6)

ORGANIC RANKINE CYCLE REFERENCE DESIGN

Figure 3.2 Organic Rankine System Reference Design (Ref. 9)

the condenser-radiator is specially-designed for the spacecraft, as is discussed in Section 5.

Two different radioisotope heat sources were considered in these studies; plutonium-238 and curium-244. The former has been the standard space nuclear fuel; the latter is potentially available in larger quantities and at lower cost, from the growing supply of nuclear-reactor waste products.

The heat source secondary radiator shown in Figure 3.2 is required with a curium heat source, and serves two functions: it power-flattens the heat source by radiating excess isotope heat to compensate for natural decay of the curium-244 (18.1 year half-life); it provides for standby cooling of the heat source prior to system start-up or in case of system failure. The longer half-life of plutonium-238 (86.4 years) and the use of a vented heat source, obviate the need for this secondary radiator with plutonium fuel.

The major features of each of these power systems is summarized in Table 3-1, for a nominal power output of 400 watts(e) after 5 years in orbit.

In general, the weight numbers shown are useful for scoping purposes; in calculating overall satellite weight (Section 6), the manner in which the power system was integrated into the spacecraft ultimately determined the power-system weight.

Concerning costs, it is obvious that the availability of curium-244 fuel at 100 dollars per watt(t) would result in sharply reduced unit recurring costs. However, facilities for the separation of this material from reactor wastes are not now available, and no commitment for the construction of such facilities has been made to date. As an alternative, the possibility of reducing plutonium cost while increasing its production rate is being studied under AEC sponsorship. However, the conclusions presented in Section 7 do not rely on such possible cost reductions.

Table 3-1

NUCLEAR POWER SYSTEMS MAJOR PARAMETERS
(400 - WATT(E) (EOM))

PARAMETER	SELENIDE RTG		ORG. RANKINE	
EFFICIENCY, % (EOM)	10.6		20.9	
MAX. CYCLE TEMPERATURE, F	1472		600	
RADIATOR TEMP., F (BOL/EOM)	300/200		108/108	
ENVELOPE, INCHES	BOX; 60 x 36 x 8		CYLINDER; D36 x L60 ^a	
WORKING MATERIAL	N&P - TYPE SELENIDES		HEXAFLUOROBENZENE	
FUEL	Pu-238	Cm-244	Pu-238	Cm-244
BOL POWER, WATTS (E)	421	519	400	400
NOMINAL WEIGHT, LBS.	134	104	161 ^b	154 ^b
LOW WEIGHT VERSION, LBS.	110 ^d (80) ^e	(N/A)	90 - 100 ^d	120 ^b
UNIT RECURRING COST ^c , K\$	3182	1288	1671	665
EXCLUDING FUEL, K\$	822		429	

a. determined by radiator surface area requirement

b. based on unvented heat sources

c. contractor estimates; based on Pu@ 600 \$/W(t) and Cm@ 100 \$/W(t), nominal generator version

d. The nominal designs of both power systems were based on minimum cost rather than minimum weight. These were then investigated for variations which could reduce weight. The low-weight version of the organic-Rankine system is based on the use of a beryllium radiator and several other changes of lesser significance. For the plutonium heat source, the fuel must be vented, allowing a sizeable reduction in the weight of this heat source. This reduction could possibly be achieved in curium as well. At present, however, only plutonium heat sources may be said to have flight-proven vents. (All recent RTG's have contained vented plutonium heat sources.) The selenide RTG design was based on a vented heat source in both curium and plutonium versions. The low-weight version is based mainly on the achievement of a smaller envelope at some slight sacrifice in efficiency ($\sim 1/2\%$) by operating at a higher radiator temperature.

e. The 80-pound selenide RTG is based on a Fairchild study (Ref. 8) which was initiated with the specific intention of designing a minimum-weight RTG. However, this design was not far enough along at the time of the present study to permit its inclusion in detail.

3.2 SOLAR SYSTEMS

The reference-design solar spacecraft was based on conventional current technology. In order to permit longer-ranged comparisons, the possible improvements in solar-array and battery technology have been estimated, as shown in Table 3-2.

3.3 DIRECT COMPARISONS

Based on the information presented above, direct comparisons of solar and nuclear power system weights can be made on a watts-per-pound basis (Figure 3.3). Some historical information is also shown for perspective.

From Figure 3.3 it can be seen that the advanced nuclear systems can compete on a weight basis with state-of-the-art solar-battery systems, even including violet cells. However, the appearance of nickel-hydrogen batteries in the same time frame would reestablish the weight advantage of the solar-battery systems.

The situation with regard to cost is hardly more encouraging (Figure 3.4). However, there is some justification for including non-recurring costs for solar systems while not doing so for nuclear systems. In general, a new solar-battery system development-test program accompanies each new spacecraft design, while the same nuclear system may be adapted to a variety of different spacecraft without redevelopment. In the past, this has been only approximately true for nuclear systems.

Although the advanced nuclear systems make a better showing in both cost and weight than the current systems, one would clearly not be justified in considering nuclear power for synchronous-orbit missions based on these direct comparisons alone.

However, compared on a systems bases (Section 6) quite the opposite conclusions can be drawn.

Table 3-2. SOLAR POWER SYSTEM PARAMETERS

SOLAR ARRAY POWER, 7-7R, SOLSTICE/EQUINOX: 580/620 WATT(E)			
TECHNOLOGY	CONVENTIONAL HARD-MOUNT Si	FLEXIBLE ROLL-UP Si	VIOLET CELLS
AREA, FT ²	86.2	98.2	71.8
WEIGHT, LBS.	55.9	55.6	44.5

BATTERY CAPACITY, 24 TRANSPONDERS PLUS HOUSEKEEPING: 618 W-HRS.		
TECHNOLOGY	CONVENTIONAL NICKEL-CADMIUM	POTENTIAL NICKEL-HYDROGEN
NO. CELLS PER BATTERY	22	6
DEPTH OF DISCHARGE, %	65	85
SPECIFIC CAPACITY, W-HR/LB	6.5	18.3
WEIGHT, LB.	95.6	33.7

Figure 3.3 Power System Weight Comparisons

POWER SYSTEMS RECURRING COST COMPARISONS 400-800 WATT(E) RANGE (EOM)

BASED ON USAF COST MODEL -- 1974 UPDATE

* INCLUDING NON-RECURRING COSTS AMORTIZED OVER 3 UNITS

Figure 3.4 Power System Recurring Cost Comparisons

4.0 ECONOMIC IMPACT OF SPACECRAFT DESIGN ALTERNATIVES

4.1 INTRODUCTION

The first generation of communications satellites to be launched and operated in a commercially competitive environment is now being built. Under the circumstances, the emphasis on minimizing the costs of procuring and installing these systems is natural. Once this initial hurdle is overcome, however, and assuming the user volume develops as anticipated, concern is bound to turn from minimizing cost to maximizing profit. The missing factor in this equation, now largely ignored as a technical concern, is income, or revenue.

In this sense, commercial spacecraft clearly differ from their NASA and DoD counterparts, which are almost always designed for minimum cost within the set of minimum performance requirements established by the user. In a commercial spacecraft, if the goal of increasing revenue conflicts with that of reducing cost, the guiding principle should clearly be the maximization of profit, rather than the minimization of cost.

The calculation of profit in any commercial enterprise is a complex exercise involving matters transcending purely systems design. However, the design factors which most strongly influence revenue can be identified and treated as an engineering problem, subject to design trade-offs. Such information can then be used as input data by a financial analyst to derive the profit-optimized approach for a particular operating system.

This section addresses the design factors involved in revenue production by a commercial communications satellite, within the context of the baseline-design 24-transponder spacecraft. As with most spacecraft, little weight margin is available for optional usage. If a few pounds could be saved in one or another sub-system by redesign, how should this weight savings be used to increase system profitability? It is important to try to quantify this potential benefit, for it is then possible to estimate the price that one can afford to pay for the weight savings.

Although the quantitative conclusions derived herein only apply to the specific spacecraft in question, these should be generally applicable to a wider class of commercial spacecraft. The main purpose of this discussion, however, is to illuminate the need for such an analysis as part of the system design process, and to indicate the order of magnitude of revenue increments which are potentially achievable as a result of subsystem weight reductions. This, as will be seen, turns out to be quite a bit larger than the magnitude one generally associates with the value of weight in space.

4.2 FACTORS AFFECTING SATELLITE REVENUE

A commercial communications satellite system can maximize revenue by maximizing any or all of the following:

- a. transponder load factor, i.e., customer usage of the available transponder capacity
- b. the number of circuits per transponder
- c. the number of operating transponder years (OTY) in synchronous orbit

The first of these is essentially a sales or marketing problem. The second is mainly an earth-station and network design problem. The third is a satellite engineering problem, and the subject of our current concern.

The value of OTY can be limited by three major factors:

- a. Total satellite failure -- due to random failure of a key component or subsystem, e.g., attitude control.
- b. Transponder failure -- due mainly to random failure of Traveling Wave Tube amplifiers (TWTA).
- c. Systematic Failure -- due mainly to battery failure or, more predictably, depletion of stationkeeping and attitude-control propellant by normal (or abnormal) usage. This marks the end of useful satellite operations.

Clearly, each of these factors can be improved by the availability of added weight capacity in orbit; the first two by the addition of redundancy to the key components, and the third by adding propellant. Within the usual weight-limited context of existing booster vehicles, therefore, the design problem boils down to a question of choice; i.e., how best to distribute the available weight capacity.

At this point the question of component reliability comes into play. In order to make this choice in a systematic and logically-supportable way, one needs to know the component reliability values to a consistent, and hopefully high, level of accuracy and confidence. Ideally, one would like to adjust the expected lifetime of the satellite due to random component failure to equal the expected satellite lifetime due to propellant depletion. In this way, overall system lifetime can be increased by trading system redundancy for increased propellant loading, or vice-versa, depending on the relative values of these quantities. However, component reliability values are rarely reported with the detail, and backed by the data, needed to make this optimization in a rigorous manner. The usual approach is to attempt to eliminate all "single-point" failure modes by providing redundancy wherever this seems feasible within the weight limits, while allowing at least five and perhaps six or seven years of propellant loading.

For the reference-design satellite, this approach results in the subsystem reliability values shown in Figure 4.1 as a function of time to 10 years. (The communications subsystem includes everything except the 24 parallel TWTA's, which are discussed separately in Section 4.3.) Each of these curves results from a combination of serial and redundant elements of various kinds and are based on the assumption that no wearout failures occur in the time interval under consideration. The overall system reliability is the product of the separate subsystem reliability values. Therefore, the mean-time-to-failure for the entire system (except the TWTA's), which is the time-integral of the total system reliability function from time zero to infinity, can be computer-calculated. For our purposes, this value was

REFERENCE SATELLITE SUBSYSTEM RELIABILITY (EXCLUDING TWTA'S)

Figure 4.1 Baseline Satellite Subsystem Reliability (excluding TWTA's)

estimated by extrapolating this function and integrating graphically. From this approximation, the expected value of system lifetime is estimated to be 12 years.

In this design, propellant loading is calculated as adequate to approximately 6 years. It would appear, therefore, that as wearout failure modes are eliminated in this time frame, and as more confidence is gained in the published reliability values for various components, the optimization of subsystems redundancy level can yield an extension of overall system operating lifetime within the same total weight figure. The potential revenue-value of such lifetime extension is discussed in the next section.

For the present, satellite lifetime extensions to perhaps ten years would appear to be a reasonable goal. It should be remembered, however that submarine-cable repeaters were designed for 20-year lifetime in the 1940's, and have already exceeded this figure in operation.

4.3 REDUNDANCY, RELIABILITY AND REVENUE

4.3.1 REDUNDANCY

The problem of the TWTAs is somewhat unique. Their role is central to the operation of the satellite; yet they are the most failure prone of the electronic components in the system. Typical failure rates of other electronic components are in the range of 1-200 failures per 10^9 hours. A quoted failure rate for a typical TWT is 1800 failures per 10^9 hours, based on space-acquired experience. (Ref. 9).

The inevitable result is the expectation of some TWT failure during the lifetime of the satellite, unless redundancy is provided. The probability of survival of all 24 transponders is shown in Figure 4.2 for the non-redundant case as well as for the cases of semi- and full-redundancy. In the semi-redundant case, one TWT is in standby mode for each two operating units.

Figure 4.2 TWTA Reliability Values

The improvement in reliability which can be achieved with TWTA redundancy is apparent. Undoubtedly, such consideration led to the decision to include full redundancy in the Intelsat IV series. On the other hand, various domestic-satellite designs, no doubt cost-constrained, do not include such redundancy. Which is the ultimately correct approach from the viewpoint of system profitability? To approach the answer to this question, we first examine the weight increments involved.

The original complement of 24 TWTA's weigh 78 pounds in the reference design (tubes plus power supplies), over half the weight of the communications subsystem (excluding antennas and feeds). Therefore, in the case of full (or single) redundancy, where a stand-by TWTA is provided for each operating unit, this weight would have to be duplicated. In addition, a pair of latching ferrite circulators would be required for each pair of tubes to switch the RF signal, on command, from the one TWTA to the other following a failure (Figure 4.3). Each such circulator and the associated length of additional waveguide hardware is estimated to weigh 1/2 pound, resulting in a total weight increment for full redundancy of 90 pounds.

This substantial weight (as well as mounting-surface and volume) penalty leads naturally to a consideration of semi-redundancy, as well as possibly lower orders of redundancy. Unfortunately orders less than one-for-two redundancy do not appear practical. This is due mainly to the resulting complexity of switching and waveguide routing, although various other factors, such as TWTA bandpass efficiency, may also play a role.

The switching arrangement required for the semi-redundant configuration, shown in Figure 4.4, requires six latching circulator switches per pair of operating units. This results in a weight increment for this arrangement of 75 pounds, with almost half this weight in the switches. (Replacement of some of the circulators with simpler and lighter "Y" branches results in approximately 3-db power loss through each Y. The cost in power-system weight to replace the power loss would exceed the weight thus saved in circulators.)

Figure 4.3 Single Redundancy Switching Arrangement

Figure 4.4 Semi-Redundancy Switching Arrangement

Considering the weight as well as cost penalty associated with TWTA redundancy, at what point might such redundancy be economically justifiable? And if justifiable, should one consider semi-redundancy at all, when for just another 15 pounds one can have full redundancy?

Clearly, these questions ought to be answered in the context of profit optimization, since a minimum-cost design principle is of no value whatsoever in this regard. Also, the answers must of necessity be based on probabilistic predictions of failure, and will only be as good as the data on which these are based.

4.3.2 RELIABILITY CALCULATIONS

Using well-known probability principles, the expected value of the time to first and subsequent TWTA failures can be calculated. The mean time to failure, θ , is defined as

$$\theta = \int_0^{\infty} t f(t) dt \quad (1)$$

where $f(t)$ is the probability density function of the time to failure. It can be shown that, in general, this reduces to

$$\theta = \int_0^{\infty} R_i(t) dt \quad (2)$$

where $R_i(t)$ is the reliability of the i -th event. If $i = k/N$, then equation (2) expresses the mean of the elapsed time during which exactly k out of N original units are operational, until a failure causes $k-1$ units to be operational. The reliability for k of N discrete units is the well-known summation:

$$R_{k/N}(t) = \sum_{i=k}^N C_i^N R_1^i (1-R_1)^{N-i} \quad (3)$$

where C_i^N are the binomial coefficients, and R_1 is the reliability of any one typical unit.

For the non-redundant case, $R_1 = \exp(-\lambda t)$, where λ is the failure rate for identical units.

For the case of full redundancy, an equivalent single unit value, R_{1f} , can be derived (Appendix B):

$$R_{1f} = R_1 \left[1 + \left(\frac{\lambda}{\lambda_s + r\lambda} \right) (1 - R_1' R_s) \right] \quad (4)$$

where λ_s is the failure rate of the switch, r is the ratio of TWTA failure rate in the standby mode to that in the operating mode, $R_1' = \exp(-r\lambda t)$, the standby reliability of the TWTA, and $R_s = \exp(-\lambda_s t)$, the switch reliability.

A useful approximation to equation (4) can be derived by assuming the reliability of both switch and standby TWTA's are unity ($\lambda_s = r = 0$).

Applying L'Hôpital's rule to equation (4), this results in:

$$R_{1f} = R_1 (1 - \lambda t) \quad (5)$$

The case of semi-redundancy is best treated by assuming that the transponders consist of twelve sub-modules, each containing one stand-by and two operating units. In this case, "failure" is defined as failure of any two of the three units. It can be shown (Appendix B) that the reliability against such failure is:

$$R_{1s} = 3 R_1^2 - 2 R_1^3 \quad (6)$$

assuming no switching or stand-by failures. In this case θ of equation (2)

must be calculated separately for k of 12 redundant units and $(12-k)$ non-redundant units, since if k of the redundant units have failed, then there are still $(12-k)$ non-redundant units available, until failures occur in this population.

With these formulations as a basis, it is possible to calculate the mean-time-to-failure for transponders in the various configurations. This result is illustrated in Figure 4.5, which shows that for the first ten year period, three transponder failures can be expected in the non-redundant case, while none can be expected in the semi- and full-redundancy cases. Therefore, semi-redundancy at least, would appear to be worthy of further consideration. Full redundancy would not be a profitable option at the indicated value of failure rate, and the current mission lifetime goal of up to ten years. (This conclusion, it should be remembered, is drawn from data without variances or confidence limits indicated, and should be re-evaluated with this information before final design decisions are made.)

4.3.3 REVENUE

With the assumption that transponder failure will actually occur at the calculated values of time, it is possible to calculate the normalized revenue of a 24-transponder system, reduced to "present value" at time zero. This is shown in Table 4-1 for the non-redundant and semi-redundant cases, assuming a nominal discount rate of 10 percent compounded annually, and normalized to a revenue of one-dollar per operating transponder per year.

Implicit in this calculation is the assumption that satellite revenue will be a fixed proportion of the number of operating transponders, which implies a system with a high transponder load factor.

Clearly, the revenue lost due to failed transponders is somewhat mitigated by the effect of discounting future revenue. Nevertheless, a potential revenue loss of several million dollars is involved; probably more than enough, in present value, to pay for the expense of adding the redundant TWTA's.

Figure 4.5 Expected Transponder Failure Times

Table 4-1.

PRESENT VALUE OF EXPECTED REVENUE FOR
24-TRANSPONDER SATELLITE

	NORMALIZED REVENUE*, TOTAL DOLLARS	
YEAR	NON-REDUNDANT	SEMI-REDUNDANT
1	21.818	21.818
2	41.653	41.653
3	59.415	59.684
4	75.125	76.077
5	89.406	90.979
6	102.056	104.526
7	113.346	116.842
8	123.609	128.038
9	131.814	138.217
10	139.910	147.469

*NORMALIZED TO 1-DOLLAR PER OPERATING-TRANSPONDER-YEAR

However, the desirability of doing so should be evaluated against the possibility of investing all excess weight capacity into stationkeeping propellant, without TWTA redundancy.

For the reference spacecraft, propellant increments varying between 22 and 29 pounds are required for each additional year in orbit, based on a total fueled spacecraft weight of 842 pounds, a total velocity correction requirement of 185 feet per year, and a propellant specific impulse of 220 seconds (Hydrazine monopropellant).

With this basis, the present value of expected satellite revenue can be calculated as a function of weight for the non-redundant and redundant cases, shown in Figure 4.6.

4.4 CONCLUSIONS

A number of conclusions can be drawn from Figure 4.6:

- Within the overall weight scope of the reference spacecraft and the predicted failure rate of TWTA's, the case of non-redundancy results in a superior revenue potential throughout the range of possible weight availability.
- The addition of TWTA semi-redundancy might be justified if the propellant weight loading were to result in a lifetime exceeding the expected satellite lifetime due to other subsystem failure, or beyond the crossing point of the two curves. This could occur if various other redundancies were eliminated in favor of extended satellite lifetime as suggested previously, or if TWTA failure rates were greater than the assumed value.
- The slope of the revenue curve at the seven-year point is 0.42 normalized dollars per pound, or 420,000 dollars per pound for one-million dollars of revenue per operating-transponder-year. Therefore, considering typical transponder revenue factors, the present value of incremental weight in orbit can be at least 1/2 - to - 1 million dollars of revenue per pound. This contrasts with the well-known value of 10 - to - 30 thousand dollars of cost per pound of weight in orbit.

Figure 4.6 Value of Weight in Orbit

This last point has various ramifications, some perhaps only dimly perceived at present. For one, design changes which can reduce weight at increased cost, should be evaluated against the revenue potential of this weight savings. For example, a saving of ten pounds at a cost of, say, one million dollars, may be a rather good investment on this basis. The general effect of this conclusion is to suggest a reevaluation of many long-standing design decisions which might originally have been based on the principle of minimizing cost. Clearly, one such decision concerns spacecraft prime power; the principal subject of this study.

This reevaluation should include the previously-discussed question of subsystem redundancy. The potential revenue benefit of even small amounts of weight saved by a more careful optimization of the level of subsystem redundancy can be substantial.

The impact of the revenue factor on booster selection should be the subject of a similar reevaluation. The cost of a larger booster may, in fact, be well-justified on a profitability basis. Of course, implicit in most of these conclusions is the assumption that satellite communications will prove to be a commercially viable venture, able to attract the capital needed to implement such higher initial-cost decisions, based on predictions of greater ultimate system profitability.

The introduction of ion propulsion or other high-specific-impulse systems for stationkeeping purposes in future systems, would clearly change these conclusions quantitatively and perhaps qualitatively as well. However, the value of weight is likely to be amplified in such an event, requiring even closer evaluations of sub-system redundancy levels and achievable system lifetime values.

Finally, it should be recognized that questions of revenue and cost are not always decisive. Factors of availability, risk, customer satisfaction, and perhaps government regulation will enter into many decisions. For example, a desire to provide guaranteed-service channels may dictate a need for TWTA redundancy even though this may not be justifiable on an

overall satellite profitability basis. However, the impact of revenue should be recognized along with that of cost as primary inputs to a profit-optimized design.

5.0 NUCLEAR-POWERED SPACECRAFT

5.1 DESIGN APPROACHES

In designing nuclear-powered spacecraft, the nuclear systems have traditionally been appended to or cantilevered from one face or one end of the vehicle. This is the so-called "separated" approach in which little or no benefit from the waste nuclear heat is usually realized. (An exception is the Viking-75 lander vehicle which employs active thermal switches to bleed waste heat from the RTG's into the lander during the Martian night.)

The nuclear Brayton and Organic-Rankine systems, however, offer a combination of characteristics which makes a so-called "integrated" design approach possible, wherein the power system radiator is used as the external skin of the spacecraft. The combination of high efficiency, >20 percent, and low radiator temperature, < 120 F, must be simultaneously achieved in order for the integrated approach to be feasible.

The low radiator temperature makes it practicable to mount the spacecraft equipment within the radiator, thereby achieving a controlled and nearly constant thermal environment for the equipment. This eliminates almost all thermal control problems normally generated by varying sun angles, and equipment-dissipation variations. The high power conversion efficiency limits the total thermal power which must be radiated for a given electrical power, and hence limits the spacecraft in size and weight to competitively acceptable values.

The integrated design approach was first investigated for the Brayton system in an earlier study for space-shuttle-launched spacecraft, in which weight was not as critical a parameter.[10] The Organic Rankine system is even more readily adapted to an integrated design, because the condenser-radiator is nearly isothermal, varying only a few degrees between the inlet and outlet ends. (The Brayton-system radiator, not normally isothermal,

can be deliberately isothermalized, as described in Reference 10. The helically-wound radiator tubes (Figure 3.2) serve to effectively distribute solar heat input over the entire radiator, hence spacecraft, surface, making all interior surfaces equally available for equipment mounting.

5.2 ALTERNATIVE APPROACHES

5.2.1 INTEGRATED

The basic fully-integrated approach would assume the configuration shown in Figure 5.1. The external radiator shell would be configured into a number of flat panels so that dissipative equipment can be mounted to the interior walls with good heat conduction. Both top and bottom surfaces are thermally insulated, so that all electrically-generated heat is radiated through the power system radiator, an approach which is seen to result in the least equipment temperature variation. The only part of the total heat load which is not radiated by the power system radiator is that which is radiated as RF energy by the antennas.

In this approach, the main communications-system antennas can be oriented in just about any direction permitted by packaging and viewing considerations, since the spacecraft itself has no preferred orientation with respect to the ecliptic.

5.2.2 SEMI-INTEGRATED

The so-called "semi-integrated" approach is to use the side-walls of the spacecraft as the power system radiator, while placing equipment on the "top" surface only (Figure 5.2). This surface, if north- or south-facing, contains adequate surface radiating area for the full equipment complement.

The major advantage to this approach is the ease with which the equipment modules can be separated from the power system, and subsequently installed as a unit. Although equipment temperature can vary through wider limits than in the fully-integrated approach, this variation will typically be less than that experienced with the solar-powered or separated designs. The radiative coupling between the power-system radiator and the equipment shelf is generally adequate to smooth out the major low-temperature excursions.

Figure 5.1 Integrated Approach

Figure 5.2 Semi-Integrated

Figure 5.3 Separated Approach

sions without requiring heaters or other active thermal-control devices in most cases.(Appendix C)

The major problem with this approach results from the use of a polarizing screen above the antenna reflectors in the reference design. This screen, and its associated thermal blanket, results in almost complete blockage of the view to space of the equipment panel radiator. Therefore, this approach did not receive major consideration in this study. However, it deserves more detailed consideration when other polarizing devices are used, or where this blockage is otherwise reduced.

5.2.3 SEPARATED

The separated design approach is the more traditional, except that in this case the power-system radiator is custom-shaped and sized to fit best within the shroud diameter (Figure 5.3). Essentially, this approach uses the solar-powered spacecraft with a minimum of change, replacing the solar-battery system with the nuclear system.

An area of concern with this approach is the possible upsetting effect of the lengthened configuration on the spin stability of the spacecraft, needed for injection. However, this was seen ultimately to be a solvable problem.

An apparent penalty for this approach is the interior heat shield required to protect the turbo-generator unit and radiator from the plume of the AKM.

The separated design approach was also viewed as representative of the RTG-powered spacecraft, since no integrated or semi-integrated versions of an RTG-powered spacecraft appear feasible for this mission. In this case, the cylindrical radiator and other Organic-Rankine system components would be replaced by two selenide RTG's each approximately 280 W(e), on opposite sides of the AKM, with their extending fins shaped to conform to the shroud envelope, if necessary.

The advantages and disadvantages of each of these approaches is summarized in the Table 5-1.

Both integrated and separated designs were selected for more detailed study.

5.3 NUCLEAR-POWERED SATELLITE REFERENCE DESIGNS

5.3.1 SEPARATED DESIGN

The basic arrangement of the separated configuration is shown in Figure 5.4. Except for the heat source, the power system is contained within the cylindrical radiator, which is cantilevered from the bottom of the basic hexagonal satellite structure.

The heat source itself protrudes upward into the satellite equipment bay. A local cutout in the face panels provides a direct thermal view to space for the secondary radiator. Dynamic balancing of the spacecraft is maintained by rearrangement of several of the housekeeping equipment modules onto panels opposite the heat source. Figure 5.5 shows the major-equipment mounting positions.

This location for the heat source provides a number of benefits:

1) it does not upset the spin-stability of the spacecraft, 2) it permits ready access to the heat source mounting structure both before and after integration of the power-system module, 3) it removes some of the more radiation-sensitive components from the direct vicinity of the heat source.

The use of a curium heat source would produce somewhat higher levels of external neutron and gamma radiation than would a plutonium heat source of the same thermal power. This leads naturally to a concern about long term radiation damage to electronic equipment and the possible need for shielding. A study revealed that a proper choice of components and materials should eliminate the need for shielding in this mission. (see Section 5.3.3., and Appendix D). Location of the more sensitive components on the sides of the spacecraft opposite the heat source is helpful in this regard. Of course, the use of a plutonium heat source completely obviates the problem.

TABLE 5-1. DESIGN APPROACHES

APPROACH	ADVANTAGES	DISADVANTAGES
INTEGRATED	- o RADIATOR USED AS PART OF PRIMARY STRUCTURE - o POWER SYSTEM RADIATOR DISTRIBUTES HEAT LOAD OVER ENTIRE SURFACE - o EQUIPMENT TEMPERATURE AUTOMATICALLY HELD WITHIN VERY NARROW BOUNDS - o SUN ANGLE AND ORBITAL ALTITUDE LESS CRITICAL TO THERMAL DESIGN - o LITTLE OR NO HEATER POWER REQUIRED FOR THERMAL CONTROL	- o MORE COMPLEX RADIATOR DESIGN - o LESS FLEXIBILITY IN FUNCTIONALLY TESTING PAYLOAD AND POWER SYSTEM INDEPENDENTLY - o DIFFICULT PAYLOAD INTEGRATION - o ELECTRIC POWER LEVEL FIXED TO SATELLITE SIZE
SEMI-INTEGRATED	- o PAYLOAD AND POWER SUBSYSTEM ARE MECHANICALLY AND FUNCTIONALLY INDEPENDENT - o PAYLOAD RADIATOR RADIATIVELY COUPLED TO CONDENSOR/RADIATOR TO REDUCE TEMPERATURE VARIATION OF EQUIPMENT - o THERMAL CONTROL PROBLEM REDUCED	- o CONDENSOR/RADIATOR STRUCTURE NOT UTILIZED FOR LOAD CARRYING. - o RADIATION BLOCKAGE DUE TO ANTENNAS AND POLARIZING SCREEN DESIGN
SEPARATED	- o PAYLOAD & POWER SUBSYSTEMS ARE MECHANICALLY AND FUNCTIONALLY INDEPENDENT - o MAXIMUM VARIABILITY OF EACH WITHOUT CROSS-IMPACT - o SIMPLER FABRICATION AND ASSEMBLY	- o NO THERMAL DESIGN BENEFITS OF POWER SYSTEM WASTE HEAT - o AKM PLUME PROTECTION SHIELD ON POWER SUBSYSTEM REQUIRED - o POSSIBLE SPIN INSTABILITY DUE TO EXTENDED LENGTH - o POSSIBLE EXCESSIVE LENGTH FOR SHROUD

Figure 5.4 Separated Design Configuration

Figure 5.5 Equipment Mounting Arrangement Separated Design

For this design, the Organic Rankine system radiator is slightly reduced in size by using a different working fluid (Flourozene-M) which produces peak efficiency at a slightly higher temperature (140 F). All other external power system parameters remain unchanged.

The 140 F radiator is a cylindrical shell, 85-inches in diameter and 25 inches high, for an area of 46 square feet. The shell is a 0.047-inch beryllium sheet bonded to .035-inch wall aluminum condensor tubes, as required by meteorite penetration criteria. The outer surface is covered by a silver-on-teflon thermal control coating. The entire assembly displays satisfactory margins of safety for Delta launch loads.

The interface with the Delta third stage and payload envelope is shown in Figure 5.6. The radiator extension protrudes somewhat below the strictly-defined limits of the payload envelope. However, this is not expected to present a problem in assembly of the vehicle to the third-stage adapter.

In this configuration, the satellite thermal control is identical to that of the solar-powered design, with the exception of a thermal-insulation barrier between the 140 F power-system radiator and the 100 F equipment radiators.

Overall system weight and power comparisons will follow discussion of the integrated design.

5.3.2 INTEGRATED DESIGN

5.3.2.1 Configuration

In this version, the Organic Rankine system is used with hexafluorobenzene as the working fluid, at a nominal radiator temperature of 104 F. Because of the lower radiating temperature, and the inclusion of the full equipment heat load, the total radiator area requirement is 76 square feet (Appendix C). However, this is less than the total external area of the

Figure 5.6 Separated Design ; Stowed Configuration

hexagonal solar-powered satellite; approximately 83 sq. ft. In this design all of the internal surface is available for equipment mounting, while in the solar-powered version only the two north-south panels could be feasibly used for mounting the TWTAs.

The arrangement of power system within the spacecraft envelope is shown in Figure 5.7. In this case, the heat source fits inside the radiator, insertable from below. The secondary radiator, if required, faces downward from below the radiator.

Figure 5.8 shows the basic equipment mounting arrangement. The weight of the heat source and converter equipment is balanced by most of the housekeeping electronic equipment on the opposite faces, similar to the arrangement used in the separated design.

The basic truss structure of the integrated design is shown in Figure 5.9.

5.3.2.2 Radiator Design

The radiator consists of spirally wound aluminum condenser tubes adhesively bonded to external beryllium skins, forming a hexagonal box structure, as shown in Figure 5.10. Where equipment support panels are required, aluminum honeycomb is bonded to the external skin between the tubes, and internal aluminum skins are bonded to the composite honeycomb/tube core to provide the internal equipment mounting surfaces (Detail A). Threaded inserts are potted in the honeycomb, where required, to create equipment attachment points, and the internal skins are slotted near the edges to accommodate generous radiator-tube bend-radii in the six corners of the hexagonal structure. Vertical longerons are provided in the corners of the box to tie the panels together (Detail B). The condenser tubes are welded to the liquid and vapor manifolds at the top and bottom surfaces of the hex (Detail C).

The basic hexagonal structure is assembled using a male-type mandril with an outside surface contour which has the shape of the condenser tube

Figure 5.7 Integrated Design Configuration

Figure 5.8 Equipment Mounting Arrangement; Integrated Design

Figure 5.9 Integrated Design; Basic Truss Structure

Figure 5.10 Integrated Radiator Details

configuration. The internal skins required would be positioned on the mandril and then the condenser tubes wrapped individually around the mandril in their predetermined locations, formed to the proper shape, and bonded to the internal skins where they exist. The aluminum honeycomb is then bonded to the internal skin where required. The vertical edges of the honeycomb are stiffened with corfil and all required inserts potted into the honeycomb, matching predrilled holes in the internal aluminum skins. The strips of honeycomb and internal skin required along the vertical edges of the non-equipment-bearing panels are prepared in an off-line operation, and the inserts required to attach these panels to the vertical stringers are installed. Then the external skins are bonded to the honeycomb/tube core or simply to the radiator tubes forming the basic hexagonal structure. The vertical stringers are then inserted into each corner of the hex between the condenser tubes and the panel corners and attached to the pick-up points provided along the panel edges. Only a limited number of these attaching screws will be installed at this time to stiffen the panel corners. The final step for which the mandril fixture is utilized is the welding of the top and bottom manifolds to the condenser tubes. The structure is then removed from the mandril and the installation of the vertical stringers completed.

The entire structure is quite similar to the honeycomb panels of the solar-powered design, except for the included radiator tubing and manifolds. The top and bottom manifolds add rigidity to the structure and contribute to its strength. Where equipment panels are not required, the combined tube/external skin structure is sufficiently strong, by itself, to withstand the boost flight loading environments.

5.3.2.3 Thermal Control

A major benefit of the integrated design is the achievement of automatic temperature regulation for any equipment contained within the radiator. Equipment mounted to the radiator interior walls will be constrained to follow the wall temperature closely. The spirally-wound radiator tubes serve

to distribute sun load or other unbalanced heat load uniformly over the radiator surface.

The maximum radiator temperature occurs with full sun load with a normal angle of incidence, and a fully degraded thermal-control coating. This is the upper curve of Figure 5.11; the maximum point on this curve is achieved at zero RF output, when the full heat source thermal power is dissipated through the radiator (Appendix C).

The minimum radiator temperature occurs with the solar-solstice sun load, non-degraded coatings, and full RF radiated power. As seen from Figure 5.11 the maximum temperature variation is from the 80-to-104 F between these extremes. By contrast, the solar-powered (and nuclear separated) designs undergo a gross temperature variation between 20 and 110 F, with the assistance of heaters at various locations.

5.3.3 SHIELDING REQUIREMENTS FOR CURIUM HEAT SOURCE

5.3.3.1 Boundary Conditions

The use of radioisotope fuel in a spacecraft inevitably results in some exposure of the included equipment to neutron and gamma radiation. For the fuel in common use today, plutonium-238, it is generally agreed that the radiation levels are too low to have any significant damaging effect on equipment (with the possible exception of photographic film nearby). However, the possible use of the lower cost curium-244, with its greater level of external neutron radiation particularly, reopened the question of the possible need for shielding. With the quantities of fuel of interest, the radiation exposure over a period of five years to equipment in the vicinity (order of inches) of the (unshielded) fuel canister has been determined to be in the range of 1-to-2x 10^{14} neutrons/cm². [11] At a distance of 60 inches this would be down to approximately 10^{13} n/cm². To reduce exposure below this level would probably require either greater separation distances, or some shielding around the fuel. Either would introduce a weight penalty not well tolerated by this reference mission nor by many potential

Figure 5.11 Radiator Temperature Variation; Integrated Design

missions. Note that approximately 10 cm. of LiH would be required to reduce neutron fluence by an additional order of magnitude.

At a level of 10^{12} n/cm², damage effects to certain components are noted. Items such as photographic film, solid-state display devices, and radiation-measuring instruments would be seriously affected. However, in communications satellites, such devices are generally not present. Therefore, a closer look was taken at the actual radiation levels which can be tolerated by communications satellites.

In communications satellites the most radiation-sensitive items are likely to be the semiconductor devices. In the past, it has been commonly assumed that the threshold damage level for transistors is $10^{11} - 10^{12}$ n/cm².

A cursory review of the literature indicates that while 10^{12} n/cm² may be a threshold level for some damage to be detectable to some devices, a good deal more exposure might be tolerable if circuit effects of property changes were considered and if either or both of these actions were taken:

- a.) design circuits around less sensitive or specially radiation-hardened devices.
- b.) design circuits to tolerate the property changes that are predicted over a wider range than is normally done.

If a.) is not possible in totality, then identifying and limiting the more sensitive devices in number and location would perhaps permit shielding them locally, rather than shielding the entire spacecraft.

5.3.3.2 Conclusions

The most significant result of this study is that no radiation shielding against neutron fluences as high as 10^{14} n/cm² is likely to be required for any component if proper radiation hardening procedures are followed in the selection and design of all components.

A survey of all of the components of the spacecraft was made in order

to identify those with the greatest potential sensitivity to nuclear radiation (Appendix D). Table 5-2 summarizes the results of this survey, as detailed in Appendix D. The eleven most sensitive components to be identified are listed. The comments indicate the action which should be taken to minimize radiation damage susceptibility. Note that in the majority of cases proper selection of parts and materials rather than circuit design change is the action required. This minimizes the potential spacecraft design perturbations which would be required with a curium heat source.

The single major component which has been identified as preferably requiring replacement are the step-junction diodes, used for frequency multiplication in a number of circuits in the spacecraft (Table 5-2). An alternative design approach using more radiation-resistant transistors appears to be feasible. However, this must be fully confirmed by a more detailed investigation than was possible in this study.

The selection of radiation-hardened components where indicated will generally result in increased cost and possibly schedule delays to any spacecraft program. The magnitude of these costs has not been estimated. However, based on the analysis presented in Section 4, this approach is likely to be preferred to the use of neutron shielding. However, the increased cost will act to reduce the cost differential between curium and plutonium fuels.

5.3.3.3 Use of the Hydrazine as a Neutron Shield

Approximately 170 pounds of hydrazine propellant is loaded in the baseline spacecraft as on-orbit propellant. Approximately 10 pounds are expended in initial orbit and attitude acquisition, with the remainder expended approximately uniformly over the life of the spacecraft, mainly for north-south stationkeeping. The hydrogen density per unit weight of hydrazine, N_2H_4 , is approximately equal to that of lithium hydride, LiH , leading to the possibility of using the available hydrazine as a neutron shield, albeit disappearing with time. As a neutron shield, hydrazine should be somewhere between lithium-hydride and water in effectiveness.

Table 5-2

RADIATION HARDNESS SUMMARY

COMPONENT	USED FOR	HARDENED DAMAGE THRESHOLD (EST.) n/cm^2	ACTION REQUIRED
STEP-JUNCTION DIODE	FREQ. MULTIPLEX	10^{12}	ELIMINATE; DESIGN WITH TRANSIST. HARM. GEN.
ELECTROLYTIC STORAGE CAPS	LATCHING RELAY SWITCHING	10^{14}	1. PRE-IRRADIATE TO ELIM. WEAK UNITS 2. DISCONNECT FROM BUS IN QUIESCENT MODE
GFRP	MICROWAVE CAVITIES AND STRUCTURAL STRUTS	$\sim 10^{14}$	1. PRE-IRRADIATE GRAPHITE 2. CHOOSE EPOXY PROPERLY (E. G. 91-LD RESIN)
TRANSISTORS	POWER & SWITCHING	10^{14}	SELECTION OF HARDENED DESIGNS
LOGIC IC'S	TT&C & POWER REG.	10^{14}	SELECT HARDENED VERSIONS
FERRITE ISOLATOR	MICROWAVE LOAD MATCHING	$\sim 10^{14}$	1. USE THIN-FILM 50-OHM TERMINATION 2. USE PROPER EPOXY
MICROWAVE TRANSISTOR	DRIVER AMP., FREQ. GEN.	$\sim 10^{14}$	
TD AND ZENER DIODES	PRE-AMP., VOLTAGE REF.	$\sim 10^{14}$	
TRANSFORMERS REED VIBRATORS	DC-DC CONV., TONE DETECTORS, POWER DIVIDERS, ETC.	$\sim 10^{14}$	USE PROPER EPOXIES & POTTING COMPOUNDS
QUARTZ CRYSTALS	OSCILLATORS & FILTERS	10^{15}	SELECT UNSWEPT OPTICAL GRADE SYNTHETIC QUARTZ AT
TWTA	OUTPUT STAGE AMPLIFIER	10^{16}	1. DESIGN WITH GROUNDED HELIX INSTEAD OF CATHODE 2. TUBE BARREL IN ELECTRICAL CONTACT WITH RF CAPSULE

An investigation into the radiation stability of hydrazine was conducted. In Reference 12, the decomposition rate of hydrazine due to nuclear radiation exposure is concluded to be 0.003% per megarad. Estimating the dose due to a 7kW(t) ^{244}Cm heat source to be of the order of 2 megarad per year, at a distance of the order of 10 cm, leads to a decomposition rate of 6×10^{-3} percent per year. Compared to a use rate of the order of 10 percent per year, this is totally insignificant.

In order to estimate the effect of the continuous consumption of the shield relative to a permanent shield, the quantity Relative Shield Effectiveness (RSE) was defined as the ratio of the time-averaged dose rate over the life of the shield to the dose rate at $t=0$. The fact of radioisotope decay serves to increase RSE somewhat. However, RSE is most sensitive to the desired shield attenuation factor. This is shown in Figure 5.12, calculated by assuming exponential attenuation and a ^{244}Cm source (18.1 year half-life).

For the moderate attenuation factors which may be of value for use with a curium heat source, RSE values of 0.3-to-0.5 can be achieved. Thus, for a shield attenuation factor of 10, a lithium hydride shield of 8 gm/cm^2 ($\sim 10 \text{ cm}$ thick) could be replaced by a linearly-consumed hydrazine shield of approximately 30 gm/cm^2 ($\text{RSE} \approx 0.3$), or a thickness of 30 cm ($\rho_{\text{N}_2\text{H}_4} \approx 1 \text{ gm/cm}^3$).

The diameter of each of the two hydrazine tanks in the baseline spacecraft is almost 50 cm. To make most effective use of this hydrazine, however, would probably require repositioning and reconfiguring the tanks. In view of the other system requirements of the spacecraft propulsion system, and the possibility that shielding would not be required even for a ^{244}Cm heat source, the use of hydrazine for neutron shielding was not implemented in this system. However, this possibility is recorded here for possible future consideration.

FIGURE 5. 12

RELATIVE EFFECTIVENESS OF A LINEARLY CONSUMED SHIELD
(COMPARED TO A PERMANENT SHIELD) AS A FUNCTION OF
SHIELD ATTENUATION RATIO (Cm-244 SOURCE)

6.0 SOLAR VS. NUCLEAR SPACECRAFT

6.1 POWER AND WEIGHT BREAKDOWN

The power requirements and weight of the nuclear spacecraft have been determined mainly by comparison with the solar design, adding or subtracting components where indicated, and generally accepting the same assumptions. Therefore, the comparative values are likely to be valid to a greater accuracy than the absolute values of subsystem weights or power consumption in either of the systems. Nevertheless, the absolute values are based on quite detailed considerations, and are the best estimates of technical specialists in each of the applicable disciplines. The weight calculations and structural-adequacy estimates of those spacecraft components which undergo major revision in the nuclear-spacecraft designs are given in Appendix E, except for the thermal control components, which are detailed in Appendix C.

The power requirements and weight breakdowns for the solar, nuclear-separated, and nuclear-integrated designs are shown in Tables 6-1 and 6-2 respectively. The Communications, T, T & C and Reaction Control Systems are unchanged in each of the designs. The Attitude Control System is affected by removal of the 14-pound solar-array drive assembly, which consumes 0.8 watt(e) .

A significant reduction in the power consumption of the nuclear satellites is achieved by elimination of the batteries, even though no battery charging power is budgeted at solar solstice. This comes about from elimination of the battery heaters (Table 2-6). In the integrated version, most other heaters are dispensed with as well (Table 6-1).

Although the power consumption of the Thermal Control System is reduced by elimination of heaters, the weight of these heaters is small, and compensated by the weight of added thermal-control surface coatings in the nuclear versions. In addition, the separated design contains an added AKM plume shield, which was included in the weight of this subsystem.

RELATIVE POWER BUDGETS

SUBSYSTEM	POWER, WATTS (e)		
	SOLAR (SOLSTICE)	NUCLEAR SEPARATED	NUCLEAR INTEGRATED
COMMUNICATIONS	424.1	424.1	424.1
TELEM. & CONTROL	21.3	21.3	21.3
ATTITUDE CONTROL	27.8	27.0	27.0
REACTION CONTROL	0.5	0.5	0.5
ELECT. PWR. & DIST.	35.8	11.0	11.0
PWR. DISTRIBUTION	23.6	5.0	5.0
HARNESS	12.2	6.0	6.0
THERMAL CONTROL HEATERS	69.5	39.5	7.0
E-W PANELS	18.5	18.5	0
BATTERIES	30.0	0	0
RCS TANKS	4.0	4.0	0
THRUSTERS & LINES	12.5	12.5	7.0
TRANSPONDERS	4.5	4.5	0
TOTAL	579.0	523.4	490.9

Table 6-1 Relative Power Budgets

SPACECRAFT WEIGHT COMPARISONS

SUBSYSTEM	SOLAR	NUCLEAR SEPARATED	NUCLEAR INTEGRATED
COMMUNICATIONS	186	186	186
TT&C	27	27	27
REACTION CONTROL (DRY)	58	58	58
AKM (DRY)	56	56	56
ATTITUDE CONTROL	114	100	100
THERMAL CONTROL	42	53	42
STRUCTURE & INTEGRATION	169	162	208
THRUST CONE & TRUSSES	82	75	75
EQUIPMENT PANELS & FRAMES	87	87	133*
ELECTRIC POWER SYSTEM	225	197	149
ARRAY/CONVERTER	56	113	105
BATTERY/RADIATOR	96	40	*
POWER/DIST. & CONTROL	25	15	15
HARNESS	48	29	29
SUBTOTAL	877	839	826
BALANCE FOR RCS EXPENDABLES	138	176	189
TOTAL	1,015	1,015	1,015

*RADIATOR IS INTEGRATED INTO STRUCTURAL FRAME

Table 6-2 Spacecraft Weight Comparisons

In the Power Distribution Assembly, most power is consumed by the voltage regulators needed for the heaters, battery charge controllers, and the regulated bus. However, the output of the Organic Rankine generator is naturally regulated by the speed controller included in the electric power system. Therefore, most of the power consumption and a good deal of the weight of the PDA system is eliminated by either of the nuclear designs. Conservatively, however, only the weight of the 28-volt battery-charging regulator and ordnance controls used for solar-array deployment (see Table 2-5) were removed from this subsystem in Table 6-2.

Further reduction in power consumption and weight is achieved by a reduction in the harness length which results from elimination of the deployed solar arrays and the battery bus (Appendix E).

The net result is to establish the significantly different power consumption requirements for the nuclear and solar designs, as shown in Table 6-1, which were then used to determine the weights of the respective electric power systems. These are based on the conventional solar array/battery system for the solar-powered spacecraft, and on the Organic Rankine system with unvented fuel for the nuclear designs.

The structural and integration hardware is broken down into two weight segments. The basic truss structure is reduced by the weight of the solar array support trusses (13 lbs.) and increased by the radiator support trusses (6 lbs.). (The supporting structure for converter components is included within the converter weights.) The equipment panels are essentially unchanged for the solar and nuclear-separated designs. The panels of the integrated version contain the radiator hardware, leading to this apparent weight penalty. Clearly, this is accompanied by a concomitant reduction in power-system weight.

Finally, these establish the overall system weight totals shown in Table 6-2. This illustrates a net weight advantage for the nuclear separated and integrated designs of 38 and 51 pounds respectively. This added weight

margin can be used to extend system lifetime by adding propellant for station-keeping and attitude control, as shown. The economic impact of this lifetime extension is discussed in the next section, where the comparison is extended to include the more advanced solar-battery technology and other versions of the nuclear systems.

6.2 REVENUE VS. COST

In Section 4, it is concluded that for this reference spacecraft, the most economically effective utilization of excess weight is in additional propellant for stationkeeping and attitude control. Propellant is consumed at a rate varying from 29 to 22 pounds per year for the first ten years, mainly for north-south stationkeeping. Further weight savings in the power systems can, therefore, be effectively used by adding to the propellant supply. This permits a direct comparison of the systems effect of these power system variations, assuming that none of the other spacecraft subsystems will be further affected by changes to the power systems within each of the designs.

Figure 6.1 illustrates this comparison, in which mission lifetime is plotted as a function of power system specific power, assuming uniform propellant consumption of 25 pounds per year. The three curves, for solar, nuclear-integrated, and nuclear-separated systems, express the different weight balances and power requirements of the three spacecraft designs. Thus, as was already illustrated in Table 6-2, a nuclear system of a lower specific power can result in a significantly longer mission lifetime than that achievable by a comparable solar-powered spacecraft. For example, the reference Organic Rankine system (unvented heat source, beryllium radiator), although approximately 15% heavier than a state-of-the-art solar system of the same power level (Figure 3.3) can result in a net spacecraft weight reduction of approximately 50 pounds, and therefore, a mission lifetime extension of about 2 years.

A similar conclusion is seen to obtain in comparing the Organic Rankine

24-TRANSPONDER/DELTA 3914 SATELLITE NUCLEAR - VS. SOLAR-POWERED VERSIONS

Figure 6.1

system with a vented heat source (either curium or plutonium-fueled) with the most advanced solar system combination now envisioned; violet-cell array with Ni-H₂ batteries. In this case the solar system is over 40% higher in specific power; yet the nuclear system can achieve comparable lifetime in the separated configuration, and increased lifetime in the integrated configuration. (It has been assumed for the purpose of this comparison that specific power will not change significantly with lifetime for the period between five and nine years. This is clearly only an approximation both for solar-battery systems as well as for radioisotope-fueled nuclear systems.)

This increased lifetime capability can be directly translated into increased revenue produced by the satellite, against which subsystem cost increments can be compared. The right-hand scale of Figure 6.1 shows the present value of expected satellite revenue, derived from Figure 4.6. This includes the effects of predicted transponder failure and a 10-percent per annum discount rate. The scale reads directly in millions of dollars of total satellite revenue if it is conservatively assumed that the average income per operating transponder is 1-million dollars per year.

From Figure 6.1 therefore, we can derive a number of conclusions:

- Either of the advanced nuclear systems would be clearly preferable to current solar systems, including violet cells.
- The revenue impact of increased lifetime can justify costly design changes which reduce weight. For example, the transition from an aluminum to a beryllium radiator skin in the Organic-Rankine system can result in a revenue increase of approximately 15 million dollars; orders of magnitude more than the cost differential between these radiators.
- The potential revenue advantage of the best of the nuclear systems is such as to exceed the predicted cost disadvantage, when compared to any of the solar systems.

This last point is illustrated in Table 6-3 by combining the information contained in Figures 3.4 and 6.1.

For this table, only recurring costs from Figure 3.4 are used for the solar as well as nuclear systems. It is further assumed that the cost per watt of solar power will remain unchanged with the development of the advanced solar systems, and that no reduction in the cost of plutonium fuel will be achieved.

In summary, it is seen that a comparison on an overall systems basis can lead to a wholly different conclusion than that which would be drawn from a cost and weight comparison of the power systems alone. Savings in the weight and power requirements of other satellite subsystems can result in a net weight advantage for the nuclear-powered design, even compared to the lowest - weight solar-battery systems now envisioned. This weight advantage, if used to extend satellite useful lifetime, can result in a revenue advantage which exceeds the possible cost disadvantage of the nuclear systems.

Although the nuclear-integrated design appears slightly superior to the separated version, the difference in weight is small and within the uncertainties. The advantage of separability for development and testing, and the greater ease of assembly of a separated design, could well result in a practical preference for this design. On the other hand, the tighter and simpler temperature control of the integrated version could tip the choice that way for many designers.

REVENUE & COST COMPARISONS: NUCLEAR VS. SOLAR SATELLITES
NUCLEAR MINUS SOLAR VALUES, MILLIONS OF DOLLARS

SOLAR \ NUCLEAR	SELENIDE RTG		ORGANIC RANKINE (VENTED; BE RAD.)	
	Cm-FUEL	Pu-FUEL		
	Δ -REVENUE	Δ -COST	Δ -REVENUE	Δ -COST
Si ARRAY/Ni-Cd BATT.	+25 +21	+1.4	+33	+0.4
VIOLET/Ni-Cd	+20 +16	+3.9	+28	+1.1
VIOLET/Ni-H2	-3 -7		+5	

Table 6-3

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APPENDIX A

DOMESTIC COMMUNICATIONS SATELLITES -- HISTORY & TECHNOLOGY

A-1 Introduction [1]

International satellite communications-relay service via the Intelsat Consortium has been available since the launch of the first commercial satellite, Early Bird, in 1965. Domestic satellite service, however, is just getting started. While the United States has made significant contributions to the success of Intelsat, attempts to realize benefits from satellite communications in the domestic area have been delayed by the regulatory review, and the necessity for approval of individual applications for domestic commercial satellites by the Federal Communications Commission.

In September 1965, the American Broadcasting Company (ABC) requested a satellite authorization for network television broadcast distribution.[2] Because the ABC application presented the Commission with novel questions of law and policy, it was returned in March 1966 pending the outcome of a public inquiry (Docket 16495), which the FCC then instituted.[3] The inquiry solicited views and comments of interested parties on questions relating to the establishment of domestic satellite systems by non-common carrier, non-governmental entities. Common carriers were not precluded from making applications under the inquiry.

Western Union, as a common carrier, made application for a domestic satellite system in early November 1966, following studies made with Western Union by General Electric and ITT and a June 1966 briefing to Commission staff. [4] However, in late October 1966 the Commission changed the scope of its inquiry to exclude the establishment of domestic satellite systems by common carriers. This Commission action was virtually coincident with Western Union's filing, which was then returned as premature under the new scope of the inquiry.

The FCC pursued its inquiry until issuing the March 1970 Report & Order inviting applications for the establishment of domestic satellite service.[5]

In the inquiry years, comments were voluminous and several organizations outlined systems; namely AT&T, Comsat, General Electric and the Ford Foundation.

Both legislative and executive branches of the government showed increased interest in domestic satellite systems during the inquiry period. Testimony before legislative committees focused largely on public benefits. Interest in satellite systems was voiced by the Alaskan congressional delegation, which saw such systems as capable of providing a unique opportunity to reach the people of that sparsely populated state with a communication system it so urgently required.

In 1967, President Johnson announced the appointment of a Task Force on Communication Policy and its final 450-page report, commonly called the "Rostow Report", was issued in December 1968.[6] It concluded that establishment of full-scale domestic satellite systems was premature and recommended a limited pilot program with Comsat as overall manager of the system and owner of the space segment as trustee. However, this report was delivered at the end of the Johnson Administration.

The Nixon Administration, elected in November of 1968, decided to take a fresh look at domestic satellite systems and it fell to them also to undertake preparation for the World Administrative Radio Conference (WARC) for Space Telecommunications which would take place in June 1971. SPACE WARC would reach agreements on frequency bands for communications satellites which would have far reaching effects upon developments throughout the decade to follow. The Commission provided industry inputs for the United States proposal through a docket instituted expressly for that purpose.[7] Ultimately the Department of State, with both government and industry inputs reconciled, submitted the United States proposals to SPACE WARC in December 1970.[8] The significance of this activity, and the dates, is that the availability of new frequency bands for communications satellites was in a

constant state of flux throughout 1970, and this period overlapped the Commission's invitation to submit applications for domestic satellite systems and the final closing date for applications.

Finally, in late January 1970, the executive branch submitted to the Commission its policy recommendations relating to the establishment of domestic satellite systems. In a brief memorandum, commonly called the "Whitehead Report", a public policy objective of open competition was set forth and it was clearly stated that "we find no public interest ground for establishing a monopoly in domestic satellite communications". [9]

The FCC, with the policy guidance of the Whitehead Report, issued a Report & Order in Docket 16495 on March 24, 1970 inviting applications for domestic satellite systems. In doing so, it established detailed procedures for filing and identified both technical and economic contents of applications. The Commission noted that the only frequency bands available for commercial domestic satellite systems were at 4 and 6 GHz (C band) to be shared with the existing common carrier microwave plant. It also noted that additional allocations were anticipated following the 1971 SPACE WARC but, "the precise nature of such allocations is not now known". [10] Western Union filed the first proposal and applications in July 1970. [11] When the Commission accepted the Western Union application for filing it set into motion the events which would lead to the closing date for initial consideration [12], ultimately set at March 1971, by which date the applicants had filed.

The seven other applicants were Comsat, Comsat/AT&T, Fairchild Industries, Hughes/GTE, MCI/Lockheed, RCA, and Western Telecommunications. The types of satellites proposed ranged from the 12 transponder C-band satellites of the Hughes HS 333 type to a 120-channel satellite with transponders in the C and K_u bands. A transponder in current domestic satellite designs is nominally 36 MHz wide and is capable of supporting one color television channel

or up to 1800 voice circuits depending on the size of the earth station.

The FCC issued an initial policy order on June 16, 1972. The final FCC policy on US domestic satellite systems was released on December 22, 1972. In summary, the policy is that any technically, financially, and legally qualified private organization may become a satellite common carrier with certain restrictions placed on Comsat and AT&T. The history of each of the applicants and their systems is as follows:

Comsat- Comsat applied for two separate systems, one for general use and one committed to AT&T as the sole user. AT&T simultaneously filed for a number of earth stations as part of the Comsat/AT&T system. The FCC decision of December 22, 1972 gave Comsat the choice of implementing one system or the other but not both. Comsat chose to implement the system committed to AT&T. This system consists of three 24 transponder satellites in orbit, two active and one spare, and one in ground reserve. These are large spin stabilized satellites which are an extension of Intelsat IV technology. The satellites are currently being manufactured by Hughes. The FCC ruled that satellite users could own their earth stations which further cleared the way for the Comsat/AT&T system. AT&T, however, had to divest itself of its 29% ownership of Comsat and to terminate its membership on the Comsat board of directors. AT&T is further barred from offering private line service via satellite for three years from the date on which its satellite service begins.

The second Comsat application, which Comsat chose not to implement, would have consisted of a space segment similar to the Comsat/AT&T space segment and approximately 150 earth stations. Comsat has, with FCC approval, joined the MCI/Lockheed applicant to form the CML Corporation of which Comsat is 1/3 owner. CML is currently performing system tradeoffs in the definition of its system configuration.

Fairchild Industries - Fairchild's initial filing was for two satellites in orbit and one in ground reserve as well as six earth stations within CONUS.* The satellites were to be three axis stabilized, and were an extension of the ATS-F technology. They would have had a 30-ft parabolic reflector which would be used to form six spot beams aimed at each of the six CONUS Stations. Three other beams would serve Alaska, Puerto Rico, and Hawaii. A total of 96 C-band transponders were accommodated through frequency reuse by polarization, frequency interleaf and spatial isolation. The design also included a 24 transponder system with broadbeam coverage of CONUS for providing service to the television networks. This latter system was to operate with a 14 GHz uplink and a 7 GHz downlink.

Fairchild and Western Union International (WUI) formed American Satellite Corporation (ASC) with WUI as 50% owner. WUI later changed its ownership to 20% and finally sold its share back to Fairchild.

While the 120 transponder satellite was shown to have the lowest per channel cost, the initial investment required for the space segment and the risk involved in potentially losing a large portion of the initial investment in a launch accident tend to rule against it. ASC consequently decided to buy three Hughes HS 333 satellites for its system. The HS 333 is a twelve transponder satellite which was developed for the Canadian Telesat system and can be adapted to US domestic systems with only very minor changes. ASC was to initially operate via leased channels on the Telesat ANIK satellites pending the launch of its own HS 333's. ASC has since decided not to buy the HS 333's but rather to lease transponders on the Western Union satellites pending the results of further tradeoffs in defining its ultimate system configurations. Current projections are for service to start July 1, 1974.

Hughes/GTE - This application is for Hughes to launch two HS 333's and partially assemble a third while GTE would install its own earth stations and lease channels on the Hughes satellite. Hughes would also serve a number of CATV customers via its satellites. Recently, GTE announced

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* Contiguous United States

an agreement with AT&T to share the satellites of the Comsat/AT&T system.

MCI/Lockheed - MCI/Lockheed initially planned a system consisting of two 43-transponder satellites in space with a third in ground reserve. MCI filed applications for a number of earth stations to complete the system. MCI/Lockheed has since formed the CML Corporation with Comsat as mentioned earlier.

RCA - RCA Global Communications Inc. initially filed for a system consisting of two satellites in orbit and one in ground reserve as well as a number of earth stations. The satellites were to be of a new design and the number of channels was not yet defined. RCA has since opted for a 24-transponder satellite design, the contract for which was awarded to RCA Astro after competitive bidding. Fairchild and TRW bid as a team in this competition. RCA has already commenced service among stations in New York, California, and Alaska via the Canadian Telesat satellites.

Western Telecommunications Inc. (WTCI) - This applicant initially filed for two operational satellites in orbit and one in ground reserve. The satellites were defined to the extent that they were to be small spin stabilized satellites to be launched on the Delta launch vehicle. These were to be 12-transponder satellites with some of these potentially in the K_u band. The satellites were to be built by Rockwell International. WTCI has not made any further effort to develop its system and is presumably not currently planning to implement a domestic satellite system in the near future.

Western Union - This application is for two operational HS 333 satellites with one in-orbit spare. The system includes six earth stations located in CONUS. Western Union has followed its initial plan and launched the first of its satellites on April 13, 1974. The Western Union satellite system is expected to begin commercial operation in August 1974.

For all these systems except the Hughes and WTCI, one of the in-orbit satellites is a backup. All satellites operate primarily in the C-band (3.7 to 4.2 GHz down and 5.925 to 6.425 GHz up). Fairchild, MCI/Lockheed, and WTCI were to operate partially in the K_u band, proposing the use of the 12, 13, and 14 GHz bands. As was anticipated, the WARC allocated more bandwidth at frequencies higher than C-band for the fixed satellite service. In the K_u band, a 500 MHz bandwidth was allocated for both up and down paths (11.7 to 12.2 GHz down and 14.0 to 14.5 GHz up.) A 3500 MHz bandwidth has been assigned to the fixed satellite service in the K_a band (17.7 to 21.2 GHz down and 27.5 to 31.0 GHz up.) Table A-1 shows the frequency bands which have been allocated in the US for domestic non-government fixed satellite service. The upper 1 GHz of the two K_a bands have been reserved by the FCC for government use.

The current generation of domestic satellites for the US all operate in the C-band. The only domestic satellite currently being manufactured to operate at the higher frequencies in the Japanese broadcast satellite being built by GE at Valley Forge, Pa., which will use the K_u bands. As a broadcast satellite, the only other downlink band available would be a 190 MHz band between 2.5 and 2.69 GHz. This band, however, has a power density flux limit and is further limited in application to community reception services such as educational TV.

A-2 Current US Domestic Satellite Technology

The first generation of domestic communications satellites will fall in three categories: small spin-stabilized satellites, large spin-stabilized satellites, and body-stabilized designs capable of being launched on the Delta family of launch vehicles.

The precursor of synchronous communications satellites was the NASA-Hughes Syncom launched in mid 1963. This was a small spin-stabilized satellite which demonstrated the feasibility of synchronous satellite communications. This was contemporary with the early years of Intelsat and Comsat. Two years later, in mid 1965, Comsat launched Intelsat I which was not much different from Syncom. The Intelsat family of satellites

Table A-1

FREQUENCY ALLOCATIONS FOR FIXED-SATELLITE SERVICE OPERATIONS IN THE UNITED STATES
(Domestic Non-Government)

<u>Frequency Band</u>	<u>Allocation Status</u>	<u>Other Space Services Using Band</u>	<u>Terrestrial Services Using Band</u>	<u>Power and Power Flux Density Limitations</u>	
3700-4200 space-to-earth	Primary/shared	None	Fixed; Mobile	$-152 \text{ dBw/m}^2/4 \text{ kHz}$ $-152 + \frac{\theta - 5}{2} \text{ dBw/m}^2/4 \text{ kHz}$ $-142 \text{ dBw/m}^2/4 \text{ kHz}$	$0^\circ \leq \theta \leq 5^\circ$ $5^\circ \leq \theta \leq 25^\circ$ $25^\circ \leq \theta \leq 90^\circ$
5925-6425 earth-to-space	Primary/shared	None	Fixed; Mobile	$+40 \text{ dBw}/4 \text{ kHz}$ $+40 + 3\theta \text{ dBw}/4 \text{ kHz}$	$\theta \leq 0^\circ$ $0^\circ \leq \theta \leq 5^\circ$
11.7 - 12.2 space-to-earth	Primary	Broadcasting Satellite	Mobile except aero- nautical (secondary allocation)	No limit	
14.0 - 14.5 earth-to-space	Primary	Space research (13.25-14.2) (14.4-15.35) Radio navigation (14.3-14.4)	Radio navigation (14.0-14.3)	$+40 \text{ dBw}/4 \text{ kHz}$ $+40 + 3\theta \text{ dBw}/4 \text{ kHz}$	$\theta < 0^\circ$ $0^\circ \leq \theta \leq 5^\circ$
17.7 - 20.2 space-to-earth	Primary & exclusive	None	Fixed; Mobile (17.7-19.7)	$-115 \text{ dBw/m}^2/\text{MHz}$ $-115 + \frac{\theta - 5}{2} \text{ dBw/m}^2/\text{MHz}$ $-105 \text{ dBw/m}^2/\text{MHz}$	$0^\circ \leq \theta \leq 5^\circ$ $5^\circ \leq \theta \leq 25^\circ$ $25^\circ \leq \theta \leq 90^\circ$
27.5-30.0 earth-to-space	Primary & exclusive	None	Fixed; Mobile (27.5-29.5)	$+64 \text{ dBw}/1 \text{ MHz}$ $+64 + 30 \text{ dBw}/1 \text{ MHz}$	$\theta \leq 0^\circ$ $0^\circ \leq \theta \leq 5^\circ$

developed in size, capacity, and lifetime capability through Intelsat IV which is designed for seven year life and is capable of handling 12 TV channels or 4000 voice circuits (or combinations of these).

The Intelsat technology forms the basis for the spin-stabilized domestic satellites, whereas the three-axis-stabilized vehicles for synchronous missions are essentially new development. The system characteristics of the three types of domestic satellites being manufactured for the US market are summarized in Table A-2. The three-axis-stable design characteristics are those of the Fairchild/TRW satellite proposed for domestic applications.

A-2.1 HS 333 Satellite

This satellite was originally developed for the Canadian Telesat domestic system. So far three copies have been launched. These are the ANIK's F-1 and F-2 of the Telsat system now operational at 109° and 114° West longitude, respectively, and the Westar F-1 of the Western Union system which will ultimately be operational at 99° West longitude. To date this has been a very successful series of satellites. This is the lightest of all the high capacity synchronous communications satellites and is designed to be launched on the low cost Delta series of launch vehicles.

The HS 333 has 12 transponders with essentially the same capacity as the much larger and heavier Intelsat IV satellites. The satellite has all the advantages of the simplicity and reliability of dual-body spinners which have been well-demonstrated in synchronous orbit by Intelsat III and IV satellites.

The satellite consists of a cylindrical, spinning main body in which all the active equipment is mounted and a despun platform which supports the antennas. The power transfer across the rotating joint is all at RG.

Table A-2. Comparison of Current Domestic Spacecraft Characteristics

	HS 333 (Western Union)	Large Spinner (Comsat)	Fairchild/TRW Three Axis
Weight on Station (lb)	655	1600	1225
Weight on Pad (lb)	1242	3100	2000
Launch Vehicle	Delta 2914	Atlas/Centaur	Delta 3914
Number of transponders (36 MHz)	12	24	24
Power per transponder (watts)	5	5.7	5
EIRP at beam edge (dBW)	33	35	32
Stabilization	Spinning	Spinning	3-axis *
Reaction Control	4 Hydrazine thrusters	6 Hydrazine thrusters	18 Hydrazine thrusters
Attitude stability	$\pm 0.2^{\circ}$	$\pm 0.6^{\circ}$	$\pm 0.14^{\circ}$
Stationkeeping Accuracy	$\pm 0.1^{\circ}$	$\pm 0.1^{\circ}$	$\pm 0.1^{\circ}$
End-of-life power (watts)	220	610	580
Solar array type	Spinning Cylinder	Spinning Cylinder	Flat-sun oriented
Power transfer	RF across despin joint	DC across despin joint	DC across solar array drive

* During transfer orbit, this satellite is spin stabilized and has a separate attitude control system for this period. After the AKM is fired, the satellite is despun and the three axis attitude control system takes over.

The outside of the cylinder serves as a support for the solar arrays and the axis of the cylinder is nominally perpendicular to the orbit plane. This optimizes the solar incidence angle on the solar array. Sufficient battery power is provided to maintain 10 transponders operational during the longest eclipse period of 72 minutes. Eclipses occur for approximately 45 days twice a year centered around the equinoxes. Fortuitously, this is the time when the sun incidence angle on the solar array is close to 90° . Therefore, the extra power required to charge the batteries is available since the solar arrays are operating at their maximum output.

The spinning cylindrical body of the satellite also forms the angular momentum vector which provides a very simple means of controlling satellite attitude. The angular momentum vector is nominally fixed in inertial space and is such that it is precessed only slightly under the action of the disturbance torques present at synchronous altitudes. A pair of axial jets parallel to and offset from the axis are used in a timed firing mode to return the angular momentum vector back to its nominal direction when required to cut down on antenna pointing errors. Another pair of jets are mounted in such a way that they fire through the center of mass of the spacecraft. These are used for East-West stationkeeping. North-South stationkeeping is performed by continuous firing of the axial jets so that no net torques are applied to the body.

The jets are part of the reaction control subsystem which uses hydrazine for its fuel. The hydrazine is stored under pressure and remains in the liquid state up to the thruster where it is decomposed by catalytic action. This decomposition is exothermic and the resultant gases are thus heated to achieve a specific impulse of the order of 220 seconds. Sufficient fuel is on board to handle the stationkeeping and attitude control requirements for seven years as well as for initial orbit correction.

The launch vehicle leaves the satellite in a synchronous transfer

orbit with a perigee of 100 miles altitude and an apogee at synchronous altitude. The inclination of this orbit is between 25 and 28 degrees. The satellite therefore requires an apogee kick motor (AKM) to circularize the orbit at synchronous altitude and to remove the inclination. Slightly less than half of the satellite pad weight is the AKM fuel. The satellite is essentially designed around the AKM and its spin, which is imparted by the Delta third stage, serves to stabilize the satellite during AKM burn.

The launch loads are carried to the Delta via a structural cylinder which spreads out into a conic section at the aft end of interface with the Delta third stage. The AKM, equipment shelf, and solar array cylinder are suspended from the structural cylinder. The despin motor and bearing and the antenna platform are supported at the forward end on the structural cylinder.

A-2.2 Twenty-Four Transponder Dual Spin Satellite

This satellite, to be used in the Comsat/AT&T system, is substantially an outgrowth of the Intelsat IV series. It has approximately the same weight and size characteristics but effectively doubles the useable bandwidth by reusing the available 500 MHz. This is done by using the frequency plan shown in Figure A-1. In this plan, the "A" transponders are 20 MHz offset from the "B" transponders. The two sets of transponders are also orthogonally polarized with respect to each other to further increase isolation. The HS 333, on the other hand, has a frequency plan which uses only the "A" frequencies of Figure A-1.

Except for size and capacity, the system-level differences between the HS 333 and the Comsat/AT&T satellite are minor.

The Comsat/AT&T satellite is designed for launch on the Atlas/Centaur vehicle which does not spin up the payload before separation. The Comsat/AT&T

D W -
L K

UP-LINK

Figure A-1 4&6 GHz Channel/Frequency Plan

satellite is thus provided with two extra hydrazine thrusters for the spin-up functions.

The 24-transponder satellite also has the transponders mounted on the despun platform rather than on the spinning equipment shelf. Thus, power is transferred across the rotating joint at DC rather than RF. The power transfer uses a slip ring mechanism. Another requirement of this division of equipment is that commands and telemetry must be transferred across the rotating joint. This is done via rotary pulse transformers which transfer command pulses to a remote decoder and telemetry pulses from a remote encoder on the spinning body.

The last difference between the Comsat/AT&T satellite and the HS 333 is in the number of antennas. The Comsat/AT&T satellite will have more antennas to accommodate the higher capacity.

A-2.3 Three-Axis Twenty-Four-Transponder Satellite

As the HS 333 has the capacity of the Atlas/Centaur-launched Intelsat IV but can be launched on the lower cost Delta vehicle, so the three-axis satellite has the capacity of the Comsat/AT&T satellite but can still be launched on the uprated, but still moderate-cost, Delta vehicle. (Table A-3). The result is increased complexity and possibly lower reliability of the satellite. However, the return is a lower cost per transponder in orbit.

The transition from a dual-spin design to a three-axis bias-momentum design for synchronous missions involves storing the angular momentum in an internal wheel instead of the satellite body. The attitude control function operates in substantially the same manner as with the dual-spin satellite except that the momentum stored in the wheel is less than can be stored in the large spinning body. Thus attitude update is more frequent. The next important aspect in the transition is to flatten out the solar array and orient it to the sun with a solar array drive mechanism. This necessitates the addition of solar array control loops and slip rings to transfer power across the rotating joints. This is where the greatest weight savings accrue since the solar array area is reduced by a factor of π .

Table A-3

ESTIMATED COST OF PAYLOAD IN GEOSYNCHRONOUS ORBIT
(SEPTEMBER, 1973)

LAUNCH VEHICLE	TOTAL PAYLOAD (TRANSFER)	EST. USEFUL PAYLOAD IN ORBIT, LB	LAUNCH COST (\$ MILLION)	COST/LB PAYLOAD
DELTA (1960)	100 pounds	46.	3.2	69,565
DELTA 2914 (1972)	1550	686.	9.1	13,265
DELTA 3914 (1975)	2000	895.	9.8	10,949
ATLAS-CENTAUR (TRANSFER)	4100	1856.	22.0	11,853
TITAN IIIC (GEOSYNCHRONOUS)	3100	3000	35.0*	11,666

* 1975 cost

The solar arrays are mounted to and deployed from the North and South faces so the antennas are mounted flat on the earth viewing face.

The attitude control system requires a control mode similar to that of the spinner for that period of time when the satellite is in transfer orbit. Since then AKM is still required, the satellite uses the spin imparted by the Delta third stage for stability during AKM firing. After AKM, the satellite is despun, the solar arrays are deployed, and a different set of attitude control sensors takes over. Pitch and roll attitude are sensed by a two-axis earth sensor and yaw attitude is sensed with an integrating rate gyro which is updated based on the earth-sensor roll error history. This is feasible because the roll and yaw body axes interchange in inertial space as the satellite rotates around the earth.

The reaction control subsystem is also more complex with the three-axis. With the dual-spin satellite, a single thruster can be used to perform all attitude control functions. The three-axis satellite requires at least two thrusters for momentum precession and two more for dumping momentum when the wheel speeds up or slows down beyond set limits. The precession thrusters can be fired simultaneously for North-South stationkeeping but East-West stationkeeping requires that more thrusters be added. The minimum number of thrusters required to perform the reaction control functions without redundancy is six compared with two for the dual spin satellites.

The three-axis satellite design with its increased complexity does lead to a 24 transponder design which is substantially lighter than the Comsat/AT&T dual-spin satellite, but not enough to allow the 24-transponder satellite to be launched on the existing Delta vehicle, rather than the Atlas/Centaur vehicle, which is currently approximately twice as expensive. RCA, therefore, is funding a development program to uprate the Delta vehicle from a 1560 lb capability to a 2000 lb capability as the launch payload for a synchronous transfer orbit mission. (Table A-3).

A-3 Second Generation Domestic Communications Satellite Technology (1980-?)

A-3.1 Future Trends

It is almost foolhardy to attempt to forecast the second generation of a system which has not yet been put into practice. The communications satellites with some operational history are those of INTELSAT. In the international area, no competing operational satellite systems have been involved and cable systems, as well as NASA satellites, have provided operational backup consistent with the INTELSAT mission. Emphasis appears to have been on the emerging technology of satellite communications. The prestige of participating in the international consortium during the emergence of this new telecommunications technology and the preservation of a delicate balance between government and commercial interests have tended to submerge many of the economic considerations that may be expected to surface in competing domestic satellite systems. The second generation of domestic satellites will undoubtedly be heavily influenced by the technical and economic experience acquired with the first generation systems. Therefore, the introduction of new technology in the second generation systems would be resisted unless a clear economic advantage could be predicted. Nevertheless, a great amount of new technology will inevitably be required. For example, AT&T, in its application to the FCC in 1971, stated that they were sponsoring research on a domestic system based on satellites with 100,000 voice circuits, or 10 times the capacity in their application to the FCC. They spoke of using the 18/30-GHz (now 20/30 GHz) frequency band. In their 1971 statement AT&T estimated the cost per circuit in the advanced 18/30 GHz system would be less than one third that of a 4/6 GHz system.

Cost of the advanced system was estimated at \$500-750 million, depending largely on whether one or two satellites were orbited to operate with one in-orbit backup.

This would include research and development costs estimated by AT&T at \$80-140 million: \$60-120 million for the satellites and \$20 million for the earth segment.

From its studies over the past five years, AT&T presented to FCC these four conclusions on frequencies for satellite communications:

- "For high reliability point-to-point service, 12/13 GHz has most of the rain attenuation and other loss disadvantages of 18/30 GHz, but lacks the compensating bandwidth and multiple frequency reuse advantages. At 12/13 GHz, the cost of diversity earth stations and their interconnecting links, and losses in efficiency would more than offset any saving in terrestrial end links over 4/6 GHz.

- "For reliable high-capacity point-to-point common carrier systems, use of 12/13 GHz compares unfavorably with use of 4/6 GHz in terms of both cost and the time required to provide a system.

- "Reliable high-capacity systems operating at 18/30 GHz could be developed which would have a lower cost per unit of capacity than would systems operating at 4/6 GHz. However, in order to bring the system into service at the earliest possible time, about six years of intensive effort would be required.

- "A domestic communications satellite system operating at 18/30 GHz, capable of serving the anticipated traffic loads of the 1980's, warrants further detailed study. Implementation of 4/6 GHz systems based on the current state of the art, as a means for evaluating the operational problems of satellite systems in the domestic environment and the propagation characteristics of advanced systems, is a logical step to take at this time."

Various other future trends can be discerned: [13]

- A design life goal beyond the currently attempted 7 years
- Antennas capable of switching transponder channels between antenna beams and the ability to steer antenna beams among various earth stations to accommodate changing traffic patterns
- Spacecraft flexibility, so that major changes in payload may be made with only minor modifications in supporting subsystems.
- Redundancy and back-up operating modes to preclude any abrupt termination of communications service, sometimes called single point failure
- One might also expect, where warranted, utilization of the improved communications technology: frequency re-use by polarization and by space diversity; multifrequency bands, adding the newly allocated WARC frequencies to the presently used 4/6 GHz bands; and transponder bandwidth tailored for the most efficient response to traffic demands

As a further indication of the incorporation of new technology, the following is based on reports of the considerations involved in the preliminary design of the next generation of INTELSAT satellites; INTELSAT V. [14]

A-3.2 Ground System Aspects

With the use of spot beams, orthogonal polarization, and higher frequency bands, the spectrum required for a high-capacity satellite is available. In order to provide the connectivity between the various antenna beams, two basic methods have been considered:

(1) Frequency Domain Multiple Access (FDMA) which is based on interconnections of the various beams by frequency filters; and

(2) Time Domain Multiple Access (TDMA) which is based on interconnecting pairs of ground stations for a fraction of a digital transmission frame

(d) Greater flexibility to adapt to changing mission requirements. (Antennas, communications equipment and other equipment can be readily modified or changed, and since there are fewer shroud restrictions for a deployable solar array, power can be adjusted accordingly.)

2) Power and Energy Storage: All INTELSAT satellites to date have used spinning, body-mounted, silicon solar cell arrays to achieve specific weights of about 2 watts per pound, including batteries. The use of single-axis, deployable, flexible substrate, oriented solar arrays should increase the specific power to approximately 4 watts per pound including deployment, orientation, power transfer, and power conditioning equipment.

INTELSAT satellites normally require full power capability during eclipse; nickel-cadmium batteries have been used to store the requisite energy. Specific weight up to 6 watt-hours per pound have been achieved and modest improvements are possible. Alternatively, rechargeable H_2O_2 fuel cells or similar devices can be used to reduce the energy storage subsystem weight. Laboratory devices (Nickel-Hydrogen batteries) have demonstrated specific weights of over 20 watt-hours per pound; however, cycle life remains a problem.

3) Propulsion: Propulsion system weight can also be reduced by using higher energy propellants. For example, catalytic hydrazine has a specific impulse of approximately 220 seconds; however, ion engines of the mercury type have specific impulses of 2200 seconds or more. In effect, the fuel weight is reduced. For a 1500-pound spacecraft (beginning of life, synchronous orbit) with full north-south and east-west station keeping for seven years, a catalytic hydrazine system requires about 207 pounds of fuel (230 pounds total for tanks, thrusters, etc.) A total weight savings of about 100 pounds may be realized assuming that an oriented solar array is used and that the portion of its weight needed (to power the ion engine) is charged to the ion engine.

The advantages of TDMA are as follows:

- (1) Each transmitter essentially operates in the single-carrier mode, i. e. at transmitter tube saturation (the point of maximum efficiency);
- (2) The weight and power of the switch are negligible; and
- (3) The allocation of time slots can be changed almost instantaneously to adapt the system to changing network requirements.

Points 1 and 2 permit the use of a satellite with the lowest mass and power that can be foreseen without sacrificing reliability; point 3 permits highly flexibly operation.

A-3.3 Spacecraft Technology

Improvements in the non-communications portions of the spacecraft were deemed necessary to reduce costs. According to Comsat estimates, the following improved technology should be available with reasonable probability when required for INTELSAT V.

1) 3-Axis Body-Stabilization . A body-stabilized satellite is one in which the principal body is constrained to lie along the earth local vertical. Either gimbaled or orthogonal momentum wheels can be used to provide the necessary pointing accuracy.

Features of body-stabilized satellites which are attractive for commercial communications satellites are:

- (a) Good pointing accuracy to facilitate the use of high-frequency spot beams;
- (b) Higher reliability achieved through the use of more redundancy (use of redundant stabilization wheels)* and less stringent mechanical and thermal design requirements on bearings and motors;
- (c) Lower weight than a spinning satellite achieved by using oriented solar arrays and other weight-saving devices which are applicable to satellites of this weight and power class; and

---

* No satisfactory method which provides bearing redundancy on spinning satellites is presently available.

Another advantage of the ion engine is its low thrust. Satellite torques caused by thruster misalignment are small and can be corrected as part of the "thrusting program". An orbit demonstration of an ion engine is thought to be necessary before a commitment to an operating system is made. Such a demonstration is part of the ATS-F satellite, in which two cesium bombardment ion engines will perform the north-south stationkeeping function.

A-3.4 Effects of Higher-Frequency Bands

As noted earlier the bandwidth available for satellite communication in the 20/30 GHz range is seven times as large as that allocated to the lower frequencies. But there should be a disproportionately larger number of channels available in the new bands because the higher frequencies provide narrower beams, permitting greater reuse of the same frequencies.

The per-circuit cost-advantage will depend on the specific application and system configuration. However, an order-of-magnitude appraisal can be obtained from figures recently submitted to the Federal Communications Commission by American Telephone & Telegraph Co. in connection with its proposed domestic satellite service, mentioned earlier. As indicated, a network designed to the Bell System's domestic needs, and operating at frequencies of 20/30 GHz would yield at least a 10-fold increase in channel capacity compared to a system operating at 4/6 GHz. Total cost of the spaceborne and terrestrial segments of the 20/30 GHz system would be 2-3 times the price of one operating at 4/6 GHz, AT&T estimated.

A system operating at 11/14 GHz would be less costly than at 20/30 GHz, because the technology is more available and less demanding. But the 11/14 GHz band lacks the channel capacity that Bell expects to need in the 1980s, especially to accommodate the growing number of Picture-Phone installations each of which requires 100 times the bandwidth of a voice circuit for the digital transmission envisioned.

Several factors will increase the cost of spacecraft and earth stations that operate at the higher frequencies. The most important is the greatly increased signal attenuation at the new frequencies when the satellite-to-earth link must penetrate a rainstorm.

The down-link is the critical one because of the limited prime electric power available in the spacecraft. For this reason, the lower of two frequencies in any assigned band normally is employed for the down-link.

At 11 GHz, the attenuation due to atmospheric absorption and rain attenuation may be as much as 40-50 times greater than at the present 4-GHz band, and at 20 GHz the figure can be 100 times greater. [15]

The problem of generating more RF power aboard the spacecraft is aggravated by the fact that the overall efficiency of satellite transmitter tubes decreases as operating frequency increases. This means that more prime power must be developed aboard the spacecraft simply to generate the same RF level at 11 GHz or 20 GHz compared to that required at 4 GHz.

An illustration of the weight and power of the transponders increase is given in Figure A-2 which shows a comparison between the transponders at C-band (4/6 GHz) and K_u band (11/14 GHz). [15]

There is one partially compensating factor. At higher frequencies, a satellite transmitting antenna of any given size produces a smaller beam, so that the effective radiated signal strength increases as the square (second-power) of frequency. However, this gain is only available if a smaller total coverage area is acceptable to the system. Similarly, the gain of an earth-station antenna also increases as the square of frequency for any given dish size. But the increased gains of satellite and earth-station antennas fall far short of compensating for the signal loss during a heavy rainstorm.

In those geographic areas where rain is a rarity, or for those applications where continuous service is not required, the higher frequencies pose no major problems. But for most areas within the U.S. and in the populated temperate zones around the earth, it now appears certain that the new bands will require that each earth station be equipped with two separate antennas,

ASSUMPTIONS; PER TRANSPONDER

	C-BAND	K-BAND
OUTPUT POWER	5-watts	10-watts
INPUT POWER	17-watts	45-watts
WEIGHT	4-lbs.	10-lbs.

Figure A-2 Weight & Power of C-Band & K-Band Transponders

separated by distances ranging from 5 to 10 miles in the hope that at least one of the two will be in communication with the satellite during a rainstorm. This is called "space diversity". The two antennas will need to be interconnected by a reliable terrestrial microwave link which will increase the overall cost of the facility.

Current and future experiments are intended to determine the optimum separation distance between the two antennas, called "space diversity distance," need to achieve a high level of continuous service.

The Bell System requirements for two-way transcontinental service call for a total outage time of no more than 2 hr. per year from all causes, or 1 hr. per year for each terminal, corresponding to a reliability of approximately 99.99%. If half of the 1-hr. outage at each terminal is allocated for equipment problems, a single severe storm could consume a large fraction of the remaining 1/2 hour total yearly outage time, unless effective space diversity techniques are available.

At least several more years of experimentation are needed before it will be possible to establish within close limits the performance characteristics that will be needed for satellites and earth stations operating in the new higher frequency bands to assure that outage times will be less than 0.01%.

A-3.5 Orbital Capacity

A plan suggested by the Canadian Domestic Satellite Communications Corporation (TELESAT) is to operate on 7° longitude orbital arc spacings with satellites having identically polarized transponder channels, and then inter-leave another set of satellites cross-polarized and separated by 3.5° from the first set. Whether or not a 3.5° spacing is optimum remains to be determined and is dependent upon other factors such as earth station antennas and performance objectives. Starting at 60° and progressing to 130° West longitude, there are 21 prime orbital locations from which satellites can

provide contiguous states coverage, if a 3.5° orbital spacing is adopted as standard. If a 24 transponder satellite is stationed in each orbital location, the 4&6 GHz capacity of that orbital arc is 504 transponders. If the same plan were used at 11/14 GHz another 504 transponders could occupy that same orbital arc. All of the foregoing assumes that broadbeam antenna coverage of the contiguous states is provided from all satellites. If spot-beam satellites are considered, the orbit/spectrum utilization will be increased even further. Without consideration of either spot-beam techniques or frequency bands other than 4&6 GHz, the availability of orbital locations to provide 504 transponders in orbit appears more than adequate to meet the service demands discussed earlier.

A 3.5° spacing plan is realistic, and the achievable spacing may be even less provided adjacent satellites have sensitive receivers and earth station antennas are 30 feet in diameter or larger. Even the minimum 5° spacing which the FCC used as a guideline provides 15 orbital locations in the same orbital arc and 360 transponders in just the 4&6 GHz frequency bands.

A-3.6 Power/Booster Limitations

Considering the added power demands of a K-band satellite (Figure A-2), it is not likely that a 24-transponder K-band system will fit on a Delta booster. Therefore, such a system would have to be launched on the next larger booster—the Atlas-Centaur. However, in order to be cost-competitive with a 24-Transponder Delta-launched satellite, the system operators would be pressured to incorporate 48 transponders, if possible, including 24 C-band with the 24 K-band transponders.

The weight and power requirements of both nuclear and solar-powered versions of such a satellite have been estimated, as shown in Figure A-3. This was based on the estimates shown in Figure A-2 as a starting point, with various other subsystems extrapolated from the Delta-class designs.

Figure A-3. ATLAS/CENTAUR CLASS 48- TRANSPONDER SATELLITE WEIGHT BREAKDOWN AND PRIME POWER REQUIREMENTS

The clear conclusion which can be drawn from Figure A-3 is that the 48-transponder system would be viable, i.e., capable of at least 8-years of on-orbit life, if either one of three advanced developments is available:

- 1.) Solar-battery systems of at least 7.5 W/lb. (Ni-H₂ batteries)
- 2.) Nuclear systems of at least 5 W/lb. (low-weight advanced RTG's)
- 3.) Electric propulsion systems of adequate lifetime and reliability

If none of these developments is available in time, the system would have to be launched on the Titan IIIC, (or the Shuttle-Tug if already developed). The added cost of the Titan IIIC, however, would make this a largely non-competitive approach, as seen in the following section.

A-3.7 Revenue Requirements

Revenue requirements generally are based upon maintenance and depreciation expense, general and administrative expense, federal income and other taxes, and the required return.

Based on satellite costs published in the FCC filings and for current procurements and launch vehicle costs in Table A-3, an estimate can be made of the revenue requirements of a basic satellite communications link, consisting of two in-orbit satellites and one ground spare. Only one of the two in-orbit satellites can be leased on a protected or priority basis, because of the "sun-outage" problem; i.e., the ground stations must be able to switch to the second satellite when the first are interrupted by sun-noise background. Conservatively, therefore, income from only one of the two in-orbit satellites will be assumed to provide all of the needed revenue. Clearly, if both satellites are assumed to be equally available, as may in time occur, then the required revenue per transponder would be reduced proportionately. Table A-4 summarizes the data on revenue requirements including insurance costs estimated at 25% of the hardware cost.

Recently announced leasing costs serve to illustrate the relative accuracy of these estimates:

Table A-4
REVENUE REQUIREMENTS

NO. OF TRANSPONDERS PER SATELLITE	12	24		48	
<u>INVESTMENT, M\$</u>					
SATELLITE PROCUREMENT COST, EA	8	13.5		30.7	
TOTAL FOR 3	24	40.5		144	
LAUNCH COSTS, EA.	9.1 ^a	22.0 ^b	9.8 ^c	35.0 ^d	22 ^b
TOTAL FOR 2	18.2	44.0	19.6	70.0	44
TOTAL HARDWARE	42.1	84.5	60.1	214	188
INSURANCE @25%	10.5	21.1	15.0	53.5	47
TOTAL INVESTMENT	52.6	105.6	75.1	267.5	235
<u>ANNUAL REVENUE REQUIRED, M\$</u>					
DEPRECIATION & AMORTIZATION (5 YEAR)	10.5	21.1	15.0	53.5	47.0
OPERATION & MAINTENANCE	1.0	1.0	1.0	1.0	1.0
RETURN ON INVESTMENT @12%	6.3	12.7	9.0	32.1	28.2
INCOME TAX	6.3	12.7	9.0	32.1	28.2
TOTAL	24.1	47.5	34.0	118.7	104.4
REVENUE REQUIRED PER TRANSPONDER	2.01	1.98	1.42	2.47	2.17

- a - DELTA 2914
b - ATLAS/CENTAUR
c - DELTA 3914
d - TITAN IIIC

American Satellite Corp. announced that it had contracted with Western Union to lease 3 Westar transponders at a cost of approximately 5 million dollars per year [16].

Western Union filed a rate of \$1000 per month for a private 2-way voice circuit using Westar I. At 500 two-way voice circuits per transponder, this is the equivalent of 6 million-dollars per year per transponder. This value may be viewed as a retail cost, while the cost to ASC is the wholesale rate [17].

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APPENDIX B

Reliability of Redundant Configurations

B-1. Single Redundancy

Figure B-1 - Single Redundancy

The case of single (100%) redundancy is depicted in Figure B-1. The probability that 100% redundancy will yield the desired operation (i.e., output within specification limits for a given output) over the time period, t , under consideration is defined as the

reliability, $R(t)$ and is as follows.

$$R_1(t) \Big|_{100\%}^{-100\%} = R_1(t) + \int_0^t p_1(u) R_2(t-u) R_s(u) R_2'(u) du \quad B-1$$

where

- R_1 is the reliability of the operating unit (unit 1),
- p_1 is the probability density for random failure of unit 1,
- R_2 is the reliability of the standby unit in operating mode,
- R_2' is the reliability of the standby unit in standby mode,
- R_s is the reliability of the switch

Equation B-1 states that the reliability of the group of Figure B-1 is equal to the probability that the active TWT-1 survives the entire time period under consideration plus the joint probability over the entire time period that TWT-1 fails between time $u-dt$ and $u+dt$, standby TWT-2 survives in standby mode up to that time and thereafter survives in the active state, and the switching mechanism survives up to the failure time t , after which it no longer needs to operate.

The probability density for the random failure of the components is given by

$$p(t) = \lambda \exp(-\lambda t)$$

Hence

$$P(t) \equiv \int_0^t p(u) du = 1 - \exp(-\lambda t)$$

and

$$R(t) \equiv 1 - P(t) = \exp(-\lambda t)$$

Where:

t = Time of failure

λ = Failure rate (assumed constant)

Assume that λ for the switch is λ_s , and the λ for TWT-1 and TWT-2 in the active state are equal, with a value λ_a , and the standby failure rate for TWT-2 is $r\lambda_a$ with $r \leq 1$. Substituting in equation B-1 yields

$$\begin{aligned} R_1(t) \Big|_{100\%} &= R_1(t) + \int_0^t [\lambda_a \exp(-\lambda_a u)] [\exp(-\lambda_a(t-u))] [\exp(-r\lambda_a u)] [\exp(-\lambda_s u)] du \\ &= R_1(t) + \lambda_a \exp(-\lambda_a t) \int_0^t [\exp(-(r\lambda_a + \lambda_s) u)] du \\ &= R_1(t) + \frac{\lambda_a \exp(-\lambda_a t)}{r\lambda_a + \lambda_s} [1 - \exp(-(r\lambda_a + \lambda_s) t)] \end{aligned} \quad B-2$$

Let

$$R'_1 = \exp(-r\lambda_a t)$$

$$R_s = \exp(-\lambda_s t)$$

$$R_1 = \exp(-\lambda_a t)$$

Then

$$R_1(t) \Big|_{100\%} = R_1 + \frac{\lambda_a R_1}{r\lambda_s + \lambda_s} (1 - R'_1 R_s) \quad B-3$$

Consider the case now where the switch and TWT in standby approach perfection (i.e., $R'_1 = R_s \rightarrow 1$, $r = \lambda_s \rightarrow 0$).

Substituting in equation B-3 yields

$$R_1(t) \Big|_{100\%} = R_1 + R_1 \lambda_a \left[\frac{1 - \exp(-(r\lambda_a + \lambda_s) t)}{(r\lambda_a + \lambda_s)} \right] (r\lambda_a + \lambda_s) \rightarrow 0 \quad B-4$$

The second term is indeterminate. Hence, applying L'Hôpital's rule, B-4 becomes

$$R_1(t) \Big|_{100\%} = R_1 + R_1 \lambda_a \frac{t}{1}$$

or

$$R_1(t) \Big|_{100\%} = R_1 (1 + \lambda_a t)$$

B.2. Semi- Redundancy

The case of semi (50%)- redundancy is represented by Figure B-2.

Figure B-2, 50% Redundancy

B-2.1 Definition of Success

In terms of Boolean algebra, success (i.e., all the possible ways paths 1 and 2 will operate) is defined as,

$$\text{Success} = ABC + AB\bar{C} + \bar{A}BC + A\bar{B}C$$

B-5

where $A \equiv$ TWT-1 operating for the time period under consideration

$B \equiv$ TWT-2 operating for the time period under consideration

$C \equiv$ Standby operating for the time period under consideration

$- \equiv$ Negation (i.e., $\bar{A} =$ TWT - 1 failed)

$+ \equiv$ "or"

Product $\equiv$ "and"

$$\text{Therefore: Success} \equiv AB + C (\bar{A}B + A\bar{B})$$

B-6

B-2.2 Reliability

Reliability, $R_1(t)$ $\Big]_{50\%}$, is defined as the probability that the grouping shown in Figure B-2 will operate successfully during the time period, t , under consideration.

The failure of each TWTA can reasonably be considered an independent event, so that the joint probability of n events can be considered as the product

of the probability of the individual events. If the probability of successful operation during the time period (i.e., its reliability) is R_1 , then failure is $(1 - R_1)$. Assume the reliability of each TWTA equal, the standby failure rate, and the switching to be without failure. Then it is clear from equation B-6, that

$$R_1(t) \Big|_{50\%} = R_1^2 + R_1 \left[(1-R_1) R_1 + R_1 (1-R_1) \right] \quad B-7$$

which yields

$$R_1(t) \Big|_{50\%} = 3R_1^2 - 2R_1^3 \quad B-8$$

B-2.3 Definition of Failure

In terms of Boolean algebra, we may define failure as the negation of success. Hence, from equation B-6:

$$\text{Failure} = \left[AB + C(\bar{A}\bar{B} + A\bar{B}) \right]' \quad B-9$$

where $[ ]'$ denotes the negation of the quantity enclosed within the square brackets. By application of DeMorgan's theorem, and the theorem $(\bar{U} + U\bar{V}) = (\bar{U} + \bar{V})$ to equation B-7, one may obtain that,

$$\text{Failure} = \bar{A}\bar{C} + \bar{A}\bar{B} + \bar{B}\bar{C} \quad B-10$$

which is all the possible combinations of two of three TWTA failing, independent of the remaining TWTA.

Multiplying each product by unity yields,

$$\text{Failure} = \bar{A}\bar{C} (B+\bar{B}) + \bar{A}\bar{B} (C+\bar{C}) + \bar{B}\bar{C} (A+\bar{A})$$

$$\text{where: } (B+\bar{B}) = (C+\bar{C}) = (A+\bar{A}) = 1$$

Therefore:

$$\text{Failure} = \bar{A}\bar{B}\bar{C} + \bar{A}\bar{C}\bar{B} + \bar{A}\bar{B}\bar{C} + \bar{B}\bar{C}\bar{A} \quad B-11$$

which is all the possible ways paths 1 or 2 or both may fail.

As equations B-10 and B-11 are identical, it is clear that the case of the failure of all three TWTA is implicitly included in equation B-8 even though it need not be explicitly accounted for. From the Figure B-2 it is clear that the failure of 2 of 3 of the TWTA results in the loss of one of the two paths.

APPENDIX C
THERMAL DESIGN

C-1. Boundary Conditions

The thermal design was based on the following boundary conditions:

1. Three-axis-stabilized spacecraft in geosynchronous orbit
2. Organic-Rankine power sub-system; efficiency = 20%
3. Transponder operational mode requires full system power to be delivered at all times. However, a variation of up to 120 watts in heat dissipation over the entire transponder complement is possible as a result of transponder operation (maximum of 5-watts RF output per transponder)
4. The thermal properties of the possible surface coatings are given in Table C-1:

Table C-I

Properties of Thermal Coatings

Item	Use	Absorptivity (New)	Absorptivity (Degraded)	Hemispherical Emissivity
Z-93	Condenser/Radiator	0.18	0.28	0.88
Silver on Teflon	Payload Radiator	0.09	0.15	0.8
Second-Surface Mirror (fused silica)	Payload Radiator	0.07	0.14	0.8

C-2. Spacecraft Configurations

Four (4) spacecraft configurations have been considered. Each of the four (4) configurations delineated below and depicted schematically in Figure C-1 have different thermal design concepts.

INTEGRATED CONFIGURATION

SEPARATED CONFIGURATION,
INTERNAL AKM

SEMI-INTEGRATED CONF.

SEPARATED CONF.
EXTERNAL AKM

Figure C-1 Four Alternative Spacecraft Configurations

1. Integrated Configuration -- The spacecraft components are mounted directly to the condenser/radiator. The condenser tubes serve to distribute the generated heat around the radiator periphery. With this configuration, the condenser/radiator maximum operating temperature is 104°F , as in the reference power system with hexafluorobenzene as working fluid. From the thermal point of view, this configuration is the most flexible and has significant advantages over the other concepts.
2. Separated Configuration -- With this configuration, the condenser/radiator is cantilevered from the main spacecraft body. Some of the power subsystem components (heat source and speed control unit) are located within the main spacecraft body. This configuration requires a plume shield for the AKM, and extends somewhat beyond the DELTA payload envelope. No significant thermal benefits derive from this concept.
3. Semi-Integrated Configuration -- All payload components are located on a north-(or south-) facing plate surrounded by a cylindrical condenser/radiator. This configuration requires that the antennas be rotated outward or otherwise removed from blocking the payload radiator to maximize its effectiveness during on-orbit operation. Since the payload equipment and the relatively constant temperature condenser/radiator are radiatively coupled, some thermal benefit is realized from the concept.
4. Separated Configuration with External AKM-- All payload components are mounted to two opposing faces of a payload module. The power subsystem is basically separated. The two faces of the payload module containing equipment are radiatively (and to a degree conductively) coupled. This has certain thermal benefits but it has weight and power disadvantages for the Domsat mission. The weight and volume limitation of the Delta launch vehicle do not allow for this approach; however, it is included here for possible future application.

C-3 Thermal Design and Analysis

C-3.1 Integrated Configuration

The integrated configuration utilizes the condenser tubes to distribute the internally generated heat and the absorbed solar flux to achieve a benign environment for the payload components. Figure C-2 illustrates the variation of radiator temperature with internal dissipation. As can be seen, the predicted payload temperature range for the full range of possible mission conditions is 80 to 104°F. This contrasts with the temperature variation considered acceptable for all equipment (except batteries): 20-110°F.

Another possible advantage of the integrated configuration is its ability to operate at any altitude and solar orientation. This might permit its use for other missions without major redevelopment costs. Figure C-3 illustrates the radiator temperature variation with equipment dissipation two "worst case" orbits: A 200n.mi-altitude mission with full solar and earth-radiation input, and a synchronous-orbit mission with sun-angle parallel to the radiator surface. A larger condenser/radiator is required for the lower altitude orbits. However, payload temperature control can still be achieved even for the cold case orbit without any change in the radiator size. In the latter case, however, a portion of the condenser/radiator should probably be covered by a thermal blanket to avoid any possibility of falling below the freezing point of the hexafluorobenzene working fluid (42 F).

Construction of the integrated configuration complicates the integration of the payload into the spacecraft. A simplifying approach which is feasible is to mount and check out the payload on separate plates outside the radiator. These plates would then be attached to the inner radiator skin via thermal grease and an appropriate number of screws to insure good thermal contact. This approach would add approximately 10-pounds to the spacecraft.

Almost all of the heater-thermostats required on the solar-powered spacecraft can be eliminated with the integrated configuration (only RCS thruster and AKM still require heaters; the latter only operate until AKM burn is initiated).

RADIATOR TEMPERATURE VS HEAT DISSIPATION FOR
THE INTEGRATED CONFIGURATION

- A radiator = 76.2 ft.²
- T radiator = 104° F max

Figure C-2

RADIATOR TEMPERATURE VS. EQUIPMENT DISSIPATION
FOR THE INTEGRATED CONFIGURATION IN ALL ORBITS

second surface mirrors

A radiator = 90.3 ft.²

ORSEPS Waste heat = 2080 watts

Figure C-3

C-3.2 Separated Configuration

With this configuration, the payload thermal design is the same as that of the solar-powered spacecraft (passive radiator with thermostatically controlled heaters). The condenser/radiator temperature is based on the use of Fluorazene-M as working fluid, at a max. temperature of 140F. A plume shield is required to protect the condenser/radiator and payload from the AKM exhaust.

This configuration does not have the flexibility of the integrated version in that a new thermal design (and possibly a different thermal-control concept) is necessary for each different payload or mission. The detailed weight of the various thermal control components are shown in Table C-2 for the integrated and separated design.

C-3.3 Semi-Integrated Configuration

The thermal design concept selected for this configuration is that of a north-viewing payload radiator internally coupled to a warm (120F maximum) condenser/radiator. An analysis was performed to determine the degree of internal radiative coupling desired. Figure C-4 and C-5 present the results of the analysis, showing the temperature variation of the equipment shelf and condenser/radiator respectively for various values of internal coupling. It should be noted that the analysis did not consider any temperature gradients in the condenser/radiator due to environmental sources nor in the equipment radiator due to local heat concentration.

In Figures C-4 and C-5 it is assumed that heat dissipated on the equipment shelf is a fixed fraction of the heat source thermal power. Thus, the effect of the variation in heat dissipation is not included in the analysis. Figure C-6 illustrates the effect of the variation in equipment power dissipation for a fixed heat source output and internal coupling.

Table C-2. Thermal Control System Weights, lbs.

Component	Separated Design	Integrated Design
TWTA heat sinks	6.1	---
Panel insulation	5.0	---
Antenna radome	5.2	5.2
Top closeout	2.0	2.0
Bottom closeout	2.0	5.8
Tank & line insulation	.2	---
Second surface mirror	2.7	8.3
Temp. sensors	0.6	0.6
Paint	1.9	1.9
AKM insulation	3.7	3.7
AKM nozzle plug	.3	.3
Standoff	1.5	1.5
Heaters/thermostats	5.8	2.5
Plume shield	3.0	---
White paint cond./rad.	2.8	---
Insulation P.S.		
Min-K	9.0	9.0
Blanket	1.6	1.6
Total	53.4	42.4

PAYLOAD TEMPERATURE VS. DISSIPATION AS A
FUNCTION OF INTERNAL RADIATIVE COUPLING

Figure C-4

CONDENSER/RADIATOR TEMPERATURE VS. EQUIPMENT
AS A FUNCTION OF INTERNAL RADIATIVE COUPLING

Figure C-5

PAYLOAD RADIATOR TEMPERATURE VS DISSIPATION

FOR THE SEPARATED CONFIGURATION WITH THERMAL COUPLING

- Internal coupling = 30 ft.²
- Area (equipment radiator) = 20.5 ft.²
- Area (condenser/radiator) = 46.5 ft.²
- Temperature (condenser/radiator) = 120°F

Figure C-6

It is clear from this analysis that significant thermal-control benefits can accrue from the semi-integrated design in 3-axis controlled or equatorial-orbit applications, while still maintaining a clear separability between equipment and power system. However, a more detailed analysis, including the effects of local heat concentrations on the equipment shelf, must be performed to more clearly assess the possible requirements for heaters or variable shutter systems for the payload equipment.

This configuration does not have the flexibility of operating in any orbit or at any attitude, but it potentially could tolerate larger sun angle variations than the totally separated configuration.

C-3.4 Separated Configuration with External AKM

The internal AKM dictated by the payload volume limitation of the Delta effectively prevents internal heat exchange which could assist in flattening temperature variations of the equipment radiators. The possible magnitude of this effect was explored by assuming an external AKM, and calculating the equipment radiator temperature as a function of sun angle, with an identical equipment radiator mounted on the opposite face of a "suitcase"-shaped spacecraft. Both faces of the radiator and the back faces of the equipment modules were assumed to be coated with silver-on-teflon thermal control material. The result, shown in Figure C-7, indicates that the equipment radiator temperature would remain within acceptable limits with no active thermal control required. Again, the effects of local heat concentrations must be separately evaluated.

Table C-3 summarizes the thermal characteristics of the four configurations which were examined.

C-4. Ground-Cooling Requirements of the Organic Rankine System

It is very desirable to have the power system operating prior to and during lift-off both from the operational and thermal points of view. The maximum allowable temperature of the condenser/radiator is assumed to be 120F.

RADIATOR TEMPERATURE VS SUN ANGLE FOR
THE SEPARATED CONFIGURATION WITH NO AKM

Figure C-7

Table C-3

COMPARISON OF THE DELTA SPACECRAFT CONFIGURATIONS ⁽¹⁾

Configuration	Condenser/Radiator Temperature/Area (°F/ft ²)	Payload Power Req'ts. (watts)	Heater Power (watts max. on- orbit)	Weight of Thermal Subsystem (pounds)	Thermal Coatings	Comments
Integrated	104/73.5	498	14	42.4	Silver on Teflon or Second surface mirrors	Provides a benign thermal environ- ment for all payloads and missions.
Separated	140/47.1	498	39	53.4	a) Second surface mirrors for payload b) Z-93 on condenser/ radiator	Each payload must have its own unique thermal design.
Separated with Thermal Coupling	120/46.5	498	21	47.2	Same as above	Sun angle con- straints, Tight temperature control via radia- tive coupling.
Separated with no AKM (2 spacecraft)	140/47.1	548	42	58.0	Same as above	Independent of sun angle; not suitable for Delta

(1) Based upon FSEC's proposed Globcom design.

If this temperature is exceeded, the pump may be caused to cavitate, causing the power system to cease functioning, thereby necessitating a restart. A battery and various control provisions would be required to start the power system in space.

During ascent, the aerodynamically-heated fairing of the Delta vehicle reaches 400 F before it is ejected 290 seconds after lift-off. The emittance of the fairing interior is 0.3 as presently configured. The present Delta on-pad air-conditioning system is capable of supplying 100 lbs/min of 75 ± 5 F air. Figure C-8 illustrates the temperature rise of the separated-configuration condenser/radiator during ascent with an initial temperature of 60 F. As can be seen, the existing fairing will not keep the condenser/radiator below 120 F at fairing separation. A considerable improvement is obtained by insulating the interior of the fairing.

To obtain the 60 F initial temperature will require modification of the existing air conditioning system. From Figure C-9, it can be concluded that air at 40 F with a flow rate of 200 lbs/min must be provided.

The higher thermal mass of the integrated condenser/radiator configuration simplifies the ground cooling and ascent thermal requirements. Cooling air at 40 F must still be supplied, however, only 100 lbs/min are required. Figure C-10 shows that the condenser/radiator need only be sub-cooled to 80 F in order to survive the ascent without a change in the fairing internal emittance.

It should be noted that the assumed maximum radiator temperature could be exceeded without resulting in pump cavitation if the vehicle is continuously accelerated along the axis of the pump, in the direction which results in a net positive suction head at the pump. In the Thor/Delta launch profile, however, there is a zero-G coast period of 12 seconds between main engine cut-off and second-engine ignition beginning 48 seconds prior to fairing drop.

TEMPERATURE RISE OF CONDENSER/RADIATOR

DURING ASCENT

SEPARATED CONFIGURATION

- Fairing temperature peaks at 400°F at
- Heat dissipation = 2000 watts - condenser
500 watts - payload
- Radiator density = 1.2 lbs/ft.^2

Figure C-8

Figure C-9

GROUND COOLING OF CONDENSER/RADIATOR

TEMPERATURE RISE OF CONDENSER/RADIATOR
DURING ASCENT; INTEGRATED CONFIGURATION

Figure C-10

APPENDIX D

EFFECTS OF NUCLEAR RADIATION ON TYPICAL COMMUNICATIONS SATELLITE COMPONENTS

D-1 Neutron Effects on Semiconductors -- General

Below certain energy levels, incident neutrons collide elastically with the semiconductor atoms and relatively little damage is done by each neutron. Above some threshold of energy (0.1 Mev for Silicon) the effect of the colliding neutron and the resultant recoil reactions are not longer resisted by the restoring forces of the crystal lattice and physical displacement of the atoms of the semiconductor materials occurs. This in turn results in damage clusters or charged defects being formed.

Some of these clusters (approximately one half of those formed) quickly relax back into a more ordered state if the extent of the damage is not too severe. Many of them are not self-restoring and strongly effect the semiconductor device's electrical parameters.

These charged defects act as both recombination centers for minority carriers and trapping sites which reduce the fixed charges produced by the atoms added to "dope" the semiconductor. The former causes decreased transistor current gain (β) while the latter can increase the bulk resistivity of the collector until it eventually becomes intrinsic with increasing neutron dose.

Transistors with the most lightly doped collectors and with relatively wide base widths are most vulnerable to neutron irradiation. Both of these factors are required for the manufacture of high voltage transistors. In most systems applications these devices are found in power supplies.

The effect of β degradation is more severe at low operating current densities so that operating a relatively high voltage transistor at very low collector currents (as is usually the case in operational amplifiers, sense amplifiers, and many other linear circuits) should be avoided for devices which are to operate in an energetic neutron environment.

Reduced minority carrier lifetime contributes to increased voltage drop at high forward biases in both bipolar transistor and in rectifier diodes. As minority carrier diffusion length is decreased, the conductivity modulated region in the transistor's collector or in the diode's base region becomes smaller. As more of the high resistivity region is exposed, parasitic IR drops can increase enormously.

Neutron-hardened bipolar transistor design attempts, by proper geometry and profile choice, to diminish the effects of all the current gain degradation mechanisms.

Table D-1 summarizes the representative semiconductors used on communication spacecraft along with the references which define the data on neutron fluence susceptibility. The last page of the table defines the legend used in this table.

The semiconductors and other neutron-sensitive components to be discussed further in this appendix are those which reveal possible damage susceptibility in neutron fluences of 10^{15} n/cm² or less.

D-2. Frequency Multipliers

Step Recovery Diodes (SRD) are extensively used for high order harmonic frequency multiplication, such as in frequency generators and TLM Transmitters. Where design considerations preclude the use of transistor harmonic generators, SRD multipliers will require LiH shielding to reduce neutron fluences on an acceptable 10^{12} n/cm² level.

No conclusive evidence is available (Ref. 1, p. F-76) to substantiate any general trends in step junction diode capacitance and switching time changes under nuclear radiation.

Table D-1 Representative Semiconductor Components of Communications Satellite
Subsystems with Associated Radiation Damage Information

	TYPE	RAD-DAMAGE CHARACTERISTICS REF.	DESCRIPTION
<u>ENERGY STORAGE ASSEMBLY</u>			
Linear I.C.'s:	1) MA741	7.15	Hi performance OP amp; high open loop gain internal compensation. Monolithic construction on single Si chip using Fairchild Planar, epitaxial and junction isolation process.
	2) MA723	7	Monolithic precision voltage regulator; contains TC reference amplifier, error amplifier, series pass transistor, and current limiter.
Transistors:	1) 2N2979	III-402	Si-N-AM
	2) 2N2905A	II-216	Si-P-SH
	3) 2N1132		PNP GP-Switching
	4) 2N5085		Si-NPN-SP
	5) 2N491A	$5 \times 10^{11} - 5 \times 10^{12} \text{ n/cm}^2$	Si U.J. P- Ngrown U.J.
	6) 2N2222A	II-152	Si-NPN-SH OIF Si Planar Epitaxial
	7) 2N2907A	III-382	Si-PNP-SH
	8) 2N5754	II-218	
Diodes:	1) 1N944B	III-86	Si DR Vz-12.28V
	2) 1N4565		Si DR Vz-6.4V
	3) 1N914	III-79	Si DS
	4) 1N3890		Si R* $V_r = 100V$, $V_f = 1.5V$
	5) 1N645	III-9	Si DS

Table D-1 (Continued)

TLM CENTRAL
ENCODER

I. C.'s:	1) SNC 5493W	7	4-BiT Binary Counter
	2) SNC 54154W	7	4L to 16 Line Dec.
	3) SNC 54107	RSN -7	Duel J-K Master Slave Flip-Flop
	4) SNC 54L00T	or MCE -7 RSN	Quad 2-input Pos NAND GATE
	5) SNC 54104T	or MCE -7 RSN	Hex inverter
	6) SNC 54L30T	or [RSN54L31] MCE (11-input Pos NAND GATE)	8-Input Pos NAND GATE
	7) SNC 54165	7	Parallel Load 8-Bit Shift Register
	8) SNC 5408T	7	Quad 2 input Pos and GATE
	9) LM 108A	7	OP. Amp. (Hi-Performance)
Transistors:	1) 2N3820	$1-3 \times 10^{15} \text{ n/cm}^2$	PNP (2N5460-2)J-FET
	2) 2N2222A	III-305	Si-N-SH Dif. Planar Epitaxial

CENTRAL CMD
DECODER

D-4	I. C.'s:	1) SNC 54L00T	RS W- 7	
		2) SNC 54L04	RS W- 7	
		3) SNC 54L02T	7	Quad 2-input Pos Wur GATG
		4) SNC 54L30T	7	
		5) SNC 54164W	7	8-Bit Parallel-Out Shift Register

Table D-1 (continued)

6) SNC 54154W	7	4L to 16L (1 of 16) Decoder/Demultiplexer)
7) SNC 54180	7	8 Bit Odd/Even Parity Generator/Checker
8) SE 567T	7	Time and Frequency Decoder
9) MA 741	7, 15	Hi Perf.OP. Amp
10) SN 5496S	7	5-Bit Shift Register

Transistors:

1) 2N2979A

Si-N-AM

PDA REMOTE CD

I. C.'s

1) SNC 54L00T	RSN
2) SNC 54L04T	RSN
3) SNC 54L02	
4) SNC 54154	
5) SNC 5496W	
6) SNC 54180	

Transistors:

1) 2N2979

Si-N-AM

PDA

I. C.:

1) MA741

OP. AMP

Transistors:

1) 2N491A	$5 \times 10^{11} - 5 \times 10^{12} \text{ n/cm}^2$
2) 2N3019	
3) 2N2905A	
4) 2N4863	
5) 1763-1220	

UNIJECT
S - N-AHF
S - P-SW-H.SP
S - N -AP
Westinghouse

Table D-1 (continued)

Diodes:	1) 1N3910	III-9	S-R* - $V_R = 100V$, $V_Z = 1.5V$
	2) 1N645		S-DS
	3) 1N4938		S-DS
	4) 1N823		S-DR
	5) 1N3024B		S-DZ $V_Z = 15V$
	6) 1N754A	III-45	S-DZ $V_Z = 6.8V$
	7) 1N5415		S-R*
	8) 1N3611		S-DS

HEATER RELAY IN PDA

OP. AMP.'s	1) MA741	7	Lin F.C. OP. AMP.
Transistors:	1) 2N3792		
	2) 2N2222	II-152	
Diodes:	1) N 67B	III-103	

REMOTE TLM BOX

I. C.'s:	1) SNC 54L30T	7	
	2) 3705-2DL	7	8 Channel Multiplex Switch
	3) SNC 54165	7	Parallel-Load 8-Bit Shift Register
	4) SNC 54L04	7	Hex Inverter
	5) SNC 5408W	7	Quad 2-input Pos. and GATE
	6) LM 108A	7	OD. AMP (SN52770)
	7) SNC54L00T	RSN	

Table D-1 (continued)

Transistors:	1) 2N3820	$1 - 3 \times 10^{15} \text{ n/cm}^2$	P-J-FET
	2) LW2222A	II-152	

Legend: Roman numeral refers to book number and number to page of computer read out of Ref. 9.

DZ = Zener Diode

DR = Reference Diode

DS = Signal Diode

R* = Rectifiers

SP = Switch Power

SH = Switch Hi-speed

AM = Amplifier

Since a number of frequency multipliers are used throughout the spacecraft, and between 10 and 20 cm of LiH shielding would be required to shield each component depending on distance from the source, it is expeditious to look for different techniques to perform the frequency multiplication function.

By simultaneously operating a transistor as an amplifier and the base-collector junction capacitance as a varactor frequency multiplier, power may be extracted well above the transistor cut-off frequency. The mechanism of multiplication in the transistor parametric multiplier (TPM) is similar to that of a conventional varactor multiplier. The requirements for good multiplication efficiency are met in some high-frequency high-power transistors. Table D-2 represents a compilation of candidate devices commonly used for frequency multiplication with some radiation damage information. These devices are usually silicon Planar epitaxial transistors and of such a construction as to offer relative radiation resistance (Note the relatively small damage constant [9] listed in Table D-2.) TPM devices are proposed for the frequency generators to avoid the use of step recovery diodes.

D-3 Storage Capacitors

Switching of latching ferrite circulators is accomplished by capacitor discharge through a coil mounted on the circulator switch. Components or circuits using these switches include the redundancy switching circuitry of the tunnel diode preamplifiers and the switches which route the telemetry signals either to the omni antennas or the high gain antennas.

The energies required to switch the circulators are typically 12 millijoules at 6 GHz. Storage of this energy requires several hundred microfarads.

Liquid electrolytic tantalum oxide capacitors provide the greatest capacitance-to-volume ratio of any of the capacitor types now in general use. While these capacitors are not generally recommended for space use, General Electric manufactures a liquid tantalum capacitor with a special

DEVICE	BIAS. COND.		REPORTED DAMAGE FACTOR	h_{FEO}	$\Delta(h_{FE})$ $(\Phi = 10^{14} \text{ n/cm}^2)$	h_{FE}
	VCE(V)	I_c (mA)				
2N3375	10	200	1.92×10^{-16} $(\bar{t}_\Phi \sim 4 \times 10^3)$	4.0 (@f = 100 MHz)	.019	3.7
2N3866	15	60	1.88×10^{-16}	11.0 (100 MHz)	.0188	9.1
MT1038A	12	19.0	6.64×10^{-16}	11.0	.066	6.37
	12	10.0	1.12×10^{-15}	(100 MHz)	.112	
	12	3.0	1.16×10^{-15}		.116	
MT1062	10	20.0	$\sim 1.0 \times 10^{-16}$ $(\bar{t}_\Phi = 1.6 \times 10^4)$	3.0 (f = 500 MHz)	.01	2.9
AT-301	10	35.0	N/A (est.: 10^{-16}) $\bar{t}_\Phi = 5 \times 10^3 \left(\frac{\text{ns}}{\text{cm}^2} \right)$	2.24	.01	2.19

Table D-2 Candidate Transistors for Frequency Multiplication

double seal which has found general acceptance in spacecraft design. These capacitors are used in the circulator switching application since space and weight preclude the use of other types.

Neutron fluences of 10^{14} n/cm² increase leakage currents in tantalum oxide capacitors, but may cause failures (short circuits) in some units. An irradiation screening process is recommended to precipitate failure of those units which are susceptible to short circuit under neutron fluence and so select only capacitors that are known to withstand the expected neutron fluences.

Power losses due to leakage current can be minimized by isolating the storage capacitors from the spacecraft bus until a few seconds prior to switching the circulator.

D-4   Graphite Fiber Reinforced Plastic (GFRP)

This advanced composite material is used in lieu of invar as the structural material for the manufacture of certain waveguide components and filters used in communications spacecraft. This material is used because of a major weight advantage over invar and has near zero temperature coefficient of expansion as does invar. The interior surfaces which must be conductive are plated with copper.

GFRP exhibits differential linear expansion [4] under irradiation by neutrons and, of course, some of the plastic binders used in GFRP tend to deteriorate under neutron fluence. The linear expansion would tend to detune the band pass filters in the communications equipment.

The proper choice of epoxy binder and irradiation of the composite material prior to component construction will lead to acceptable expansion under the effect of further neutron fluence of the levels expected from the RTG. 91-LD resin does not deteriorate under fluence of 10^{14} n/cm².

Experimental and theoretical investigations have established the principal characteristics of radiation-induced parameter changes in transistors. The parameter changes of major importance are generally the decrease of forward current gain, h_{FE} or β ; and the increase in leakage current, I_{CBO} .

Simple theory for the silicon NPN bipolar transistor predicts a linear change in reciprocal low frequency gain, $1/\beta$, with fluence as shown by the relation:

$$\Delta(1/\beta) = \left(\frac{1}{\beta_{\Phi}} - \frac{1}{\beta_0} \right) = \frac{1.22}{2\pi} \left(\frac{K_{\tau}}{f_{\alpha}} \right) \Phi$$

where

- β_{Φ} = common emitter low frequency gain at fluence level
- β_0 = pre-irradiation common-emitter low frequency gain.
- Φ = radiation fluence (particles/cm²)
- K_{τ} = minority-carrier-lifetime damage constant for the material and type of radiation.
- f_{α} = alpha cut-off frequency defined as the frequency at which the magnitude of the common-base small-signal current gain is reduced to 70 percent of its initial value.

For uniform base transistors,

$f_{\alpha} = 1.22 f_{\tau}$ is the common emitter-gain-bandwidth product.

Transistor base transit time, t_b :

$$t_b = \frac{1.22}{2\pi} \frac{1}{f_{\alpha}} = \frac{0.195}{f_{\alpha}} = \frac{.159}{f_{\tau}}$$

Hence,

$$\Delta(1/\beta) = \frac{0.159}{f_{\tau}} K_{\tau} \Phi = t_b K_{\tau} \Phi = K_b \Phi$$

where K_b is the bulk-damage constant which is equal to

$$t_b K_T \text{ (cm}^2\text{/particle)}$$

There are two potential solutions to the transistor degradation problem. First, the circuits can be designed to operate for the range of parameters from pre-irradiation to post-irradiation levels or, second, the circuit designer can select parts which have already been tested to levels greater than 10^{14} n/cm². With the availability of a variety of transistor which have been radiation hardened, the second plan is more efficacious.

Small signal devices for a wide range of switching and amplifier applications have been hardened to maximum neutron fluences up to 10^{15} n/cm² (E>10 keV, fission spectrum) (e.g. MM4261H, 2N5763, 2N2857). Hardened bipolar devices are available in both NPN and PNP polarities, as well as n and p-channel JFETs.

Bipolar power transistor presently available are parts ranging to 25-ampere, 150 volt devices hardened to 1×10^{14} n/cm². The maximum power transistor requirement for the DOMSAT spacecraft calls for devices with capabilities of peak currents up to 10 amperes.

Radiation hardened devices have minimized base transit time through the reduction of transistor base width. Narrow base width structures are characteristic of normal epitaxial, Planar, double diffused devices. Additional hardening is achieved by further width reduction to the minimum value required to support the specified voltage.

Tables D-3 and D-4 are manufacturer's data sheets listing some choices of radiation hardened transistors for signal and power applications, respectively.

D-6 Digital and Linear Integrated Circuits

Integrated circuits, both digital and linear, are used throughout the

Table D-3. Radiation-Hardened Signal-Level Transistors
(Fairchild Semiconductor Co.)

$T_J = 25^\circ\text{C}$ unless otherwise indicated

EIA Registration	RADIATION DOSAGE	LV Min	$h_{FE} @ I_C/V_{CE}$ Min Max	$h_{FE} @ I_C/V_{CE}$ Min Max	$f_{\alpha} @ I_C/V_{CE}$ Min Max	$t_{\alpha} @ I_C/V_{CE}$ Min Max	$t_{\alpha} @ I_C/V_{CE}$ Min Max	$C_{\alpha} @ V_{CE}$ Min Max	$V_{\alpha} @ I_C/V_{CE}$ Min Max	$I_{\alpha} @ V_{CE}$ Min Max	I Range	APPLICATIONS
2N4872 (TO-18) PNP	3×10^4 nvt	PRE -12V	50 120 10mA/-0.3V	20 10mA -0.3V ($T_J = 55^\circ\text{C}$)	900MHz 10mA -10V	15ns 10/1mA	20ns 10/1.1mA	3pF 5V	-0.15V 10/1mA	0.01 A -0V	0.1-100mA	High Speed Saturated Switch
		POST 12V	15 10mA	5 10mA	800MHz	20ns	20ns	3pF	0.20V 10/1mA	0.01 A -0V		
2N4873 (TO-18) NPN	3×10^4 nvt	PRE 15V	110 150 10mA/1V	70 100mA/1V	700MHz 10mA/10V	12ns 10/3.3mA	18ns 10/3.3/3.3mA	4pF 5V	0.20V 10/1mA	0.40 A 20V	0.1-100mA	High Speed Saturated Switch
		POST 15V	19 10mA/1V	15 100mA/1V	600MHz 10mA/10V	13ns 10/3.3mA	18ns 10/3.3/3.3mA	4pF 5V	0.30V 10/1mA	0.40 A 0V		
2N5065 (Low Profile) (TO-5) NPN	3×10^4 nvt	PRE 15V	50 300mA/0.5V	50 100mA/0.5V	550MHz 100mA/5V	15ns 1000/200mA	35ns 1000/200/200mA	15pF 5V	0.70V 1000/200mA	100 A 15V	10-1000mA	High Speed High Current Saturated Switch
		POST 15V	10 300mA/0.5V	11 100mA/0.5V	500MHz 130mA/5V	15ns 1000/200mA	35ns 1000/200/200mA	15pF 5V	0.95V 1000/200mA	100 A 15V		
2N5106 (TO-39) 2N5107 (TO-18) NPN	3×10^4 nvt	PRE 30V	75 10mA/10V	100 300 150mA/0V	250MHz 50mA/10V	65ns 300/100mA	550ns 300/100/100mA	8pF 10V	0.22V 50/10mA	0.01 A 10V	0.1-500mA	General Purpose Amplifiers and Switches
		POST 30V	8 10mA/10V	8 50mA/10V	200MHz 50mA/10V	65ns 300/100mA	550ns 500/100/100mA	8pF 10V	0.50V 150/10mA	0.10 A 50V		
2N5144 (TO-18) 2N5145 (TO-39) NPN	3×10^4 nvt	PRE 30V	60 100mA/1V	35 500mA/1V	300MHz 50mA/10V	35ns 500/100mA	60ns 500/100/100mA	12pF 10V	0.18V 100/20mA	1.7 A 40V	10-1000mA	Core Drivers
		POST 30V	9 100mA/1V	5 500mA/1V	250MHz 50mA/10V	35ns 500/100mA	60ns 500/100/100mA	12pF 10V	0.40V 100/10mA	1.7 A 50V		
2N5200 (TO-46) NPN	3×10^4 nvt	PRE 20V	50 150 10mA/5V	40 100 A/5V	900MHz 10mA/5V			2.5pF 5V	0.50V 50/10mA	0.01 A 10V	0.1-50mA	General Purpose Amplifier
		POST 20V	15 10mA/5V	5 100 A/5V	800MHz 10mA/5V			2.5pF 5V	1.0V 50/10mA	0.05 A 10V		
2N5200 (TO-46) NPN	10 nvt	PRE 20V	50 150 10mA/5V		900MHz 10mA/5V			2.5pF 5V	0.15V 10/2mA	0.01 A 10V	0.1-50mA	General Purpose Amplifier
		POST 20V	5 10mA/5V		700MHz 10mA/5V			2.5pF 5V	0.70V 10/2mA	0.10 A 10V		
2N5201 (TO-46) NPN	3×10^4 nvt	PRE 20V	75 150 10mA/5V	65 100 A/5V	1100MHz 10mA/5V			2.5pF 5V	0.50V 50/10mA	0.01 A 10V	0.1-50mA	General Purpose Amplifier
		POST 20V	23 10mA/5V	7 100 A/5V	1000MHz 10mA/5V			2.5pF 5V	1.0V 50/10mA	0.05 A 0V		
2N5201 (TO-46) NPN	10 nvt	PRE 20V	75 150 10mA/5V	70 1mA/5V	1100MHz 10mA/5V			2.5pF 5V	0.25V 10/2mA	0.01 A 0V	0.1-50mA	General Purpose Amplifier
		POST 20V	10 10mA/5V	5 1mA/5V	700MHz 10mA/5V			2.5pF 5V	0.70V 10/2mA	0.10 A 0V		
2N5236 (TO-39) NPN	3×10^4 nvt	PRE 20V	30 120 50mA/5V	$G_m @ 250\text{MHz}$ 3dB min $V_{CE} = 20V$	500MHz 50mA/10V			3pF 10V	0.20V 50/5mA	1.0nA 10V	0.1-100mA	Class C RF Amplifier
		POST 20V	10 50mA/5V	5 10V	500MHz 50mA/10V			pF	0.2V	nA		
2N5244 (TO-18) PNP	3×10^4 nvt	PRE -40V	140 100 A/5V	150 300 10mA/1V	450MHz 10mA -0V	40ns 50/6 mA	200ns 50/16/16 mA	3.5pF 10V	-0.28V 50/10mA	0.015 A -30V	10-100mA	High Speed Saturated Switch and RF Amplifier
		POST -40V	40 100 A/5V	14 10mA/-1V	400MHz 10mA/20V	40ns 50/16/7mA	200ns 50/16/5 mA	3.5pF -0V	-1.0V 50/10mA	1.0 A 30V		
2N5292 (TO-18) PNP	3×10^4 nvt	PRE 12V	30 10mA/0.3V	40 30mA/0.3V	800MHz 10mA/10V	15ns 50/6mA	35ns 50/6mA	4.5pF 5V	-0.44V 100/10mA	1.0nA 10V	10-100mA	High Speed Saturated Switch
		POST 12V	60 10mA/0.3V	10 30mA/0.3V	600MHz 30mA/10V	25ns 50/6mA	35ns 50/6mA	4.5pF 5V	-0.65V 100/10mA	1.0nA 10V		

Table D-4. Radiation-Hardened Power Transistors

Device No.	Polarity	Max. Rating	Pre-Irradiation	Post Irradiation
2N5938	NPN	$V_{CBO} = 60 \text{ V}$ $V_{CEO} = 50 \text{ V}$ $I_{c(\text{CONT})} = 3 \text{ A}$ $I_{c(\text{PEAK})} = 5 \text{ A}$	$h_{FE} = 30 \text{ (min)}$ @ $I_c = 1 \text{ A}$	$h_{FE} = 10 \text{ (min)}$ @ $I_c = 1 \text{ A}$ @ $1 \times 10^{15} \text{ n/cm}^2$
2N5939	NPN	$V_{CBO} = 80 \text{ V}$ $V_{CEO} = 80 \text{ V}$ $I_{c(\text{CONT})} = 10 \text{ A}$ $I_{c(\text{PEAK})} = 20 \text{ A}$	$h_{FE} = 40 \text{ (min)}$ @ $I_c = 5 \text{ A}$	$h_{FE} = 10 \text{ (min)}$ @ $I_c = 5 \text{ A}$ @ $1 \times 10^{14} \text{ n/cm}^2$
2N5940	NPN	$V_{CBO} = 70 \text{ V}$ $V_{CEO} = 70 \text{ V}$ $I_{c(\text{CONT})} = 10 \text{ A}$ $I_{c(\text{PEAK})} = 20 \text{ A}$	$h_{FE} = 40 \text{ (min)}$ @ $I_c = 5 \text{ A}$	$h_{FE} = 10 \text{ (min)}$ @ $I_c = 5 \text{ A}$ @ $1 \times 10^{14} \text{ n/cm}^2$
SP5867	PNP	$V_{CBO} = -80 \text{ V}$ $V_{CEO} = -80 \text{ V}$ $I_{c(\text{CONT})} = -10 \text{ A}$ $I_{c(\text{PEAK})} = -20 \text{ A}$	$h_{FE} = 30 \text{ (min)}$ @ $I_c = -5 \text{ A}$	$h_{FE} = 10 \text{ (min)}$ @ $I_c = -5 \text{ A}$ @ $1 \times 10^{14} \text{ n/cm}^2$
SP5632	NPN	$V_{CBO} = 90 \text{ V}$ $V_{CEO} = 85 \text{ V}$ $I_{c(\text{CONT})} = 5 \text{ A}$ $I_{c(\text{PEAK})} = 10 \text{ A}$	$h_{FE} = 30 \text{ (min)}$ @ $I_c = 3 \text{ A}$	$h_{FE} = 10 \text{ (min)}$ @ $I_c = 1 \text{ A}$ @ $1 \times 10^{14} \text{ n/cm}^2$
SP5858	NPN	$V_{CBO} = 90 \text{ V}$ $V_{CEO} = 80 \text{ V}$ $I_{c(\text{CONT})} = 20 \text{ A}$ $I_{c(\text{PEAK})} = 50 \text{ A}$	$h_{FE} = 45 \text{ (min)}$ @ $I_c = 10 \text{ A}$	$h_{FE} = 10 \text{ (min)}$ @ $I_c = 10 \text{ A}$ @ $1 \times 10^{14} \text{ n/cm}^2$
SP5714	NPN	$V_{CBO} = 100 \text{ V}$ $V_{CEO} = 100 \text{ V}$ $I_{c(\text{CONT})} = 20 \text{ A}$	$h_{FE} = 30 \text{ (min)}$ @ $I_c = 10 \text{ A}$	$h_{FE} = 10 \text{ (min)}$ @ $I_c = 7 \text{ A}$ @ $1 \times 10^{13} \text{ n/cm}^2$
SP6617	NPN	$V_{CBO} = 60 \text{ V}$ $V_{CEO} = 50 \text{ V}$ $I_{c(\text{CONT})} = 5 \text{ A}$ $I_{c(\text{PEAK})} = 10 \text{ A}$	$h_{FE} = 70 \text{ (min)}$ @ $I_c = 5 \text{ A}$	$h_{FE} = 10 \text{ (min)}$ @ $I_c = 5 \text{ A}$ @ $2 \times 10^{14} \text{ n/cm}^2$
SP6618	NPN	$V_{CBO} = 65 \text{ V}$ $V_{CEO} = 65 \text{ V}$ $I_{c(\text{CONT})} = 10 \text{ A}$ $I_{c(\text{PEAK})} = 20 \text{ A}$	$h_{FE} = 40 \text{ (min)}$ @ $I_c = 10 \text{ A}$	$h_{FE} = 10 \text{ (min)}$ @ $I_c = 10 \text{ A}$ @ $2 \times 10^{14} \text{ n/cm}^2$
SP7008	NPN	$V_{CBO} = 55 \text{ V}$ $V_{CEO} = 45 \text{ V}$ $I_{c(\text{CONT})} = 3 \text{ A}$	$h_{FE} = 70 \text{ (min)}$ @ $I_c = 2 \text{ A}$	$h_{FE} = 10 \text{ (min)}$ @ $5 \times 10^{14} \text{ n/cm}^2$
SP7009	NPN	$V_{CBO} = 55 \text{ V}$ $V_{CEO} = 45 \text{ V}$ $I_{c(\text{CONT})} = 10 \text{ A}$	$h_{FE} = 70 \text{ (min)}$ @ $I_c = 10 \text{ A}$	$h_{FE} = 10 \text{ (min)}$ @ $5 \times 10^{14} \text{ n/cm}^2$
TIC35	SCR	$V_D = 15 \text{ V}$ On-State $I_{c(\text{CONT})} = 225 \text{ mA}$	$I_{GT} = 20 \mu\text{A (max)}$ $V_{TM} = 1.6 \text{ V (max)}$ @ $I_{TM} = 1 \text{ A}$	$I_{GT} = 5 \text{ mA (max)}$ @ $1 \times 10^{14} \text{ n/cm}^2$ $V_{TM} = 1.6 \text{ V (max)}$ @ $I_{TM} = 1 \text{ A}$ @ $1 \times 10^{14} \text{ n/cm}^2$
TIC36	SCR	$V_D = 30 \text{ V}$ On-State $I_{c(\text{CONT})} = 225 \text{ mA}$	$I_{GT} = 20 \mu\text{A (max)}$ $V_{TM} = 1.6 \text{ V (max)}$ @ $I_{TM} = 1 \text{ A}$	$I_{GT} = 5 \text{ mA (max)}$ @ $1 \times 10^{14} \text{ n/cm}^2$ $V_{TM} = 1.6 \text{ V (max)}$ @ $I_{TM} = 1 \text{ A}$ @ $1 \times 10^{14} \text{ n/cm}^2$
W2051	SCR	$V_{DRM} = 80 \text{ V}$ Anode Current = 20 A	$V_{TM} = 1.5 \text{ V}$ @ 20 A @ $I_{GT} = 40 \text{ mA}$	$V_{TM} = 1.5 \text{ V}$ @ 20 A @ $I_{GT} = 40 \text{ mA}$
W2052	SCR	$V_{DRM} = 80 \text{ V}$ Anode Current = 20 A	$V_{TM} = 1.5 \text{ V}$ @ 20 A @ $I_{GT} = 30 \text{ mA}$	$V_{TM} = 1.5 \text{ V}$ @ 20 A @ $I_{GT} = 50 \text{ mA}$

spacecraft. Digital IC's are used in the command decoders and telemetry encoders. Linear IC's are used in a large number of components in control loops, scaling multipliers, etc.

Various IC's of both types, by utilizing dielectric isolation, thin film NiCr resistors, and all-aluminum interconnections, have been radiation hardened and are commercially available.

Fabrication techniques [11] for integrated circuits including passive elements are similar to those for discrete components. However, when more than one transistor is incorporated on the same silicon chip, the transistor design is modified. For instance, the substrate is moderately doped p-type rather than heavily doped n-type silicon substrate. A p-region must be diffused from the surface to the substrate to provide electric isolation between two transistors. The junction formed between p-regions and n-type collector, called the isolation diode, causes additional unwanted current flow during transient radiation pulses. For this reason, IC's in which a buried silicon dioxide layer replaces the isolation diode have smaller transient damage effects. This dielectric isolation technique may not be a requirement for applications with maximum fluence of only 10^{14} n/cm² [12].

The radiation resistance of microcircuit systems has been investigated by testing separate microcircuits and using statistics based on known data of transistor damage coefficients. Logic gates with bipolar transistors can be characterized by a transistor base transit time, τ_b , defined as $\tau_b = 0.2/f_\alpha$, where f_α is the alpha cutoff frequency. When $\tau_b \Phi < 10^5$ n sec/cm², the separate gates have been found to almost certainly survive; failure is predicted with almost certainty when $\tau_b \Phi > 10^7$ n-sec/cm². The quantity Φ is the fast neutron exposure. Thus for fluence of 10^{14} n/cm², τ_b should be kept below one nanosecond to guarantee survival. However, statistically, devices with τ_b of 10 nanoseconds have been found to have a high probability of survival.

Table D-5 is a manufacturer's list of radiation hardened digital IC's. Linear IC's use the same manufacturing processes and are also available.

Table D-5. Radiation Hardened Digital Integrated Circuits

Function	Device No.			
	1 mW/30 ns TTL	10 mW/10 ns TTL	20 mW/7 ns TTL	DTL
Quad 2 NAND	RSN54L00	RSN5400	RSN54H00	--
Hex Inverter	--	RSN5404	RSN54H04	--
Triple 3 NAND	RSN54L10	RSN5410	RSN54H10	RSN15962
Dual 4 NAND	RSN54L20	RSN5420	RSN54H20	RSN15930
Single 11 NAND	--	RSN5431	RSN54H31	--
Dual 4 Buffer	--	RSN5440	RSN54H40	RSN15932
Dual 4 Power Gate	--	--	--	RSN15944
Dual 2 Wide AO1	--	RSN5456	RSN54H56	--
Single 4 Wide AO1	RSN54L57	RSN5457	RSN54H57	--
Single 2 Wide AO1	--	RSN5458	RSN54H58	--
RS Flip-Flop	RSN54L71	--	--	--
JK Flip-Flop	RSN54L72	--	--	RSN15945
Dual D Flip-Flop	RSN54L74	RSN5474	RSN54H74	--
Dual JK Flip-Flop	--	--	RSN54H103	--
One Shot	RSN54L122	--	--	--
Dual 3 NAND	RSN54L130	--	--	--
Dual 3 NAND w/Expander	RSN54L131	--	--	--
3 to 8 Decoder	--	--	RSN54H149	--

D-7 Coax Y or T-Junction Isolators

Devices are constructed utilizing non-standard transmission lines which have been adapted from strip transmission line techniques. Ferrites, biasing magnets (Alnico V), and dielectric materials required may all be made radiation resistant. The unused third part of the junctions are terminated in 50-ohm loads. These loads should be constructed from thin film resistive elements to insure spec. performance. Cementing compounds must be chosen to withstand 10^{14} n/cm² fluence levels.

D-8 Microwave Semiconductors

Microwave diodes are characterized by very low junction capacitance and low carrier storage. In general, microwave diodes are relatively radiation resistant with point contact diodes constructed of relatively high resistivity material being more sensitive to radiation than other microwave units. PIN junctions, however, are considered very sensitive.

The high-frequency transistor has the characteristic for high-frequency operation at low currents. They are generally thinner based and have less junction capacitance than power or switching transistors. As a result they should show a better resistance to radiation. These devices are not expected to show some severe effects until after about 2×10^{14} n/cm².

D-9 Diodes

Many radiation hardened diodes suitable for a wide variety of circuit functions are available. Low voltage Zener diodes and rectifiers are inherently relatively radiation resistant and can be designed into systems without further hardening effort.

Degradation occurs from irradiation in the form of increases in forward-voltage drop and reverse current. Changes in these parameters from exposure to radiation must be accounted for in the circuit design.

Units with larger than 6 volt breakdown voltages have exhibited +.25% changes at fluences of 2×10^{14} n/cm² (E>10 ke V). At high level, 10^{15} to 10^{16} n/cm², reference voltages have been measured to change about 5%. The reference voltage generally remains within 1 to 2 percent out to fluences of 10^{14} n/cm² (E>10 ke V).

Data available on reverse current behavior indicates that the minimum fluence at which the reverse current increases by a factor of 10 or more is approximately 2×10^{14} n/cm² (E>10 ke V). In most circuitry, this is not a problem.

OTC Zener and very high current rectifiers have also been radiation hardened. Of the latter, devices (MBD 5300, 5400, and 5500) have been tested to neutron fluences of 3×10^{14} n/cm² (E>10 ke V, fission spectrum) and show negligible change in electrical performance.

Schottky Barrier (Hot Carrier) Diodes, used in microwave mixers and detectors [3] are affected by nuclear radiation as summarized below:

<u>Effects of Radiation</u>	<u>Effects on Schottky Barrier Characteristics</u>	<u>Hardening Method</u>
Carrier Removal	Increase in series resistance	Use of thin epi film with highest doping possible on n ⁺ substrate
Decrease in bulk lifetime	Increase in I _R and I _F due to generation-recombination	Not generally a problem

Barrier height of Schottky barrier diode is not affected by radiation. Radiation resistant ($\leq 10^{15}$ n/cm²) diodes require:

- (1) metal with a moderately low barrier height or n-Si, such as Al or Mo;
- (2) p-n junction guard ring, or passivation by the other dielectrics more radiation resistant than thorium-oxide;
- (3) fairly heavily doped ($\leq 10^{17}$ cm⁻³) thin epitaxial n-Si film on n⁺ substrate.

In applications where these conditions cannot be met, the extent of the degradation due to radiation can be estimated from existing data.

D-10 Transformers

Experimental data indicate no significant degradation of the electrical characteristics as a result of irradiation up to integrated fluxes of $5 \times 10^{17} \text{ n/cm}^{-2}$. Problems that are encountered are of a physical nature. Insulating material, especially Teflon and Teflon in combination with other materials has proven to be a troublesome insulator at levels of the order of 10^{16} n/cm^2 . Similar difficulty is encountered with potting material.

However, with the proper choices of insulator and potting compound fluence level of 10^{14} n/cm^2 are expected to offer no problems.

D-11 Quartz Crystals [6]

Quartz crystals are used extensively as reference oscillators throughout the spacecraft.

Permanent changes in the frequency vs temperature characteristics of unswept optical grade synthetic quartz AT resonators are caused by fast neutrons. Neutron damage shifts the frequency vs temperature characteristics of the unswept resonators upwards in frequency across the 0- to $+120^\circ\text{C}$. temperature range at a rate of $.56 \times 10^{-15} \text{ ppm per n/cm}^2$ (Ref. 6). For fluences of the order of 10^{14} n/cm^2 the total frequency shift is .06 ppm, well within the spec. requirement. Good quality, slowly grown Z growth quartz with a Li-doped solution has been found to be radiation resistant.

D-12 TWTA [2]

Experiments have shown that, when the external ground point is switched from the cathode to the helix, the sensitivity of the TWTA to radiation is reduced. Since the ground point can be chosen almost arbitrarily, this presents no problem. TWTA tube barrels which are in electrical contact with the RF capsule have also been shown to be unaffected by a 10^{16} n/cm^2 level.

D-13   Examples of Radiation Hardened Circuits

A combination of the above items, as well as a few other radiation hardening techniques can be used in designing radiation hardened circuits for spacecraft use. Here are a few examples of radiation hardened circuits which have been shown to operate after a neutron fluence of at least 10^{14} n/cm².

D-13.1   Radiation-Hardened DC-to-DC Converters

The hardened d.c. -to- d.c. converter (Figure D-1) will provide 2,000 volts d.c. at 50 ma when operated from a 28-volt d.c. supply and has a conversion efficiency of 75 to 80 percent at 100 watts output. The converter circuit is a single-stage two-transformer inverter. Two pairs of 120-watt high-frequency transistors provide the switching function. The compound connection of transistors assures adequate common emitter current gain, even after neutron-induced permanent damage occurs to the transistors. The common emitter current gain of the compound connection is equal to the product plus the sum of the individual transistor current gains.

The incorporation of magnetic circuitry minimizes the effects of pulses by insuring proper current and voltage time phasing. Since voltage and current are almost 90 degrees out of phase during the switching process, the power dissipation in the semiconductor devices is minimized. Resistors R6 and R10, and their counterparts R7 and R11, insure bias stability and prevent thermal runaway. Resistors R2, R3, and R5 provide the self-starting feature of the converter. Zener diode D7 and switching diodes D5 and D6 comprise a spike-clipping network which protects the transistors. The conversion frequency is determined by transformer T1 which has a ferrite core. The output voltage is determined by the turns ratio of the output transformer T2. Rectification of the a.c. to produce d.c. is accomplished by the diode bridge rectifier and C3.

The response of the d.c. -to-d.c. converter in a simulated gamma-radiation environment of 5×10^{11} rads (Si)/sec ($E \sim 1$ Mev) was prompt with no

Figure D-1. D.C. To D.C. Converter

Figure D-2. Power Switch

interruption of operation. After a neutron fluence of 10^{14} n/cm² (E>10 kev), the d.c.-to-d.c. converter suffered no loss in output d.c. voltage.

D-13.2 Radiation-Hardened Power Switch

The power switch (Figure D-2) was designed for voltages typically available in spacecraft systems and to be triggered by a low-level logic pulse. The switch can deliver a peak power of 250 watts (10 amps into 2.5 ohms) for 100 μ sec when operated from 28-volt positive and 5-volt negative power supplies. Energy output can be adapted to other applications if appropriate power supplies are available.

The power switch consists of two stages: a power inverter stage driven by a monostable multivibrator stage.

The inductors L1 and L2 limit current that would otherwise flow as a result of ionization-induced transistor saturation. Compensating transistors minimize this saturation time by providing sinks for primary photocurrent. Transistor Q2 is saturated prior to normal electrical operation which reduces its vulnerability to ionizing radiation. The compound connection of Q3 and Q4 affords sufficient overall transistor current gain to assure proper electrical operation after high neutron fluences. Q1 is a high-speed switching transistor while Q2 is a medium power RF device. Q3 and Q4 are both high-frequency power transistors. All of the transistors exhibit excellent tolerance to ionizing radiation and neutron-induced permanent damage.

The power switch has successfully operated after neutron fluences of 3×10^{14} n/cm² (E>10 kev). Exposure to peak gamma rates of 2×10^{11} rads (Si)/sec with total doses of 10,000 rads (Si) did not result in a prefire or a misfire.

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APPENDIX E

Structural Design and Detailed Weight Estimates

E-1 Basic Approach

The integrated and separate-design versions of the nuclear-powered domestic communication satellite designs are similar to the previously-designed solar-array-powered, Delta-launched, "Domsat" design, proposed by FSEC for commercial communication satellites. The same structural criteria were used to establish both the solar and nuclear designs. In addition, the same payload requirements, including antennas, were assumed. For both nuclear versions, the same Delta adapter and beryllium thrust cone was used as in the "Domsat" design. For the separated radiator design, the solar array power components were removed and the nuclear power equipment was added without changing the equipment panel or antenna support structure configuration, although equipment was rearranged within the structure in order to accommodate the heat source and power-conversion equipment. With the integrated-radiator design, the equipment panels were designed to mount to the interior wall of the specially configured radiator structure, and the antenna support structure was integrated with the radiator manifold to effect a weight reduction. The designs are discussed in detail below with specific weight breakdowns delineated.

E-2. Review of DOMSAT Design

The spacecraft consists of a hexagonal-shaped outer structure which is connected by struts to a beryllium thrust cone which, in turn, mates to a standard Delta adapter (Figure E-1). The Apogee Kick Motor (AKM) fits inside of, and is supported by, the thrust cone. The solar panels are wrapped around the hexagonal structure and are deployed in two steps to become flat, sun-tracking arrays. The antennas are located on the top surface of the spacecraft (as mounted on the booster) and supported by a structural framework which is attached to the top of the thrust cone. The antenna feeds are supported by a space truss which is mounted to the top of the hexagonal structure. The deployed arrays and the drive mechanisms are also supported from the top of the hexagonal structure. The hexagonal structure consists of two modules. The upper payload module is separately

1. ITC ANTENNA
2. POLARIZATION SCREEN
3. POLARIZATION SCREEN SUPPORT
4. ANTENNA FEEDS
5. ANTENNA
6. ANTENNA SUPPORT TRUSS
7. SOLAR ARRAY DRIVE
8. PAYLOAD MODULE
9. BUS MODULE
10. SOLAR ARRAY ASSEMBLY
11. TURNER GEAR
12. SPIN MOTOR
13. ADAPTER
14. SEPARATION CLAMP
15. PROPELLANT TANK
16. MOMENTUM WHEEL
17. ROTATION DAMPER
18. EARTH SENSOR
19. DUAL MODULATOR SWITCH
20. DIPLOMER
21. CONVERTER
22. DIGITAL SUN SENSOR ELECTRONICS
23. CENTRAL TELEMETRY ENCODER
24. CENTRAL COMMAND DECODER
25. CONTROL ELECTRONICS
26. COMMAND RECEIVER
27. BATTERY
28. POWER DISTRIBUTION ASSEMBLY
29. EARTH SENSOR ELECTRONICS
30. RATE CYRO
31. AUXILIARY ELECTRONICS
32. DIGITAL SUN SENSOR
33. ANALOG SUN SENSOR
34. SPIN SUN SENSOR

Figure E-1. Exploded View

assembled and then slides down over and attaches to the lower "bus" module. The entire north and south faces contain the communications payload, minimizing thermal control problems with these high-heat-dissipating components. The bus module houses the power, ACS, RCS, and TT&C subsystems. The propellant tanks are located midway between the two modules in the east and west corners of the hex. This modular approach allows fabrication and integration of the payload and housekeeping systems to proceed independently.

E-3. Nuclear-Integrated Version; Weight Estimates

A. Condenser Tubes and Radiating Outer Skin

The 11 aluminum condenser tubes, which have .090" thick walls and inside diameter of .32", are spirally wound to form a cylindrical shape. This shape is flattened on 6 sides to a hexagonal configuration but reasonable bend radii are maintained in the corners of the hexagon. The tubes are bonded to six 40" x 50" beryllium panels which are .047" thick and form the outer skin of the hexagonal spacecraft body. The weight of the tubes and panels are calculated as follows.

Total tube weight = (# of tubes) (tube area) (tube length) (density of aluminum)

$$\text{Total tube weight} = (11) \left(\frac{\pi}{4} (.5^2 - .32^2) \right) (224.) (.1) = \underline{31.2 \text{ lbs.}}$$

Beryllium panel weight = (# of panels) (length of panels) (width of panels)
(thickness of panels) (density of beryllium)

$$\text{Beryllium panel weight} = (6) (50.) (40.) (.047) (.067) = \underline{37.8 \text{ lbs.}}$$

The condenser tubes are connected to manifolds at the top and bottom of the hexagonal structure. The top manifold is a 3/4" outside diameter aluminum tube with a wall thickness of 0.090 in. The bottom manifold is a 1/2 inch outside diameter aluminum tube with a wall thickness of 0.090 in. Both manifolds are utilized as structural framing member to close out the top and bottom of the beryllium panels. The weight of the manifolds are calculated as follows.

Wt. of manifold = (tube area) (tube length) (density of aluminum)

$$\text{Wt. of top manifold} = \left[\frac{\pi}{4} (.75^2 - .57^2) \right] [240] [.1] = \underline{4.3 \text{ lbs.}}$$

$$\text{Wt. of bottom manifold} = \left[\frac{\pi}{4} (.5^2 - .32^2) \right] [240] [.1] = \underline{2.7 \text{ lbs.}}$$

Table E-1 summarizes the weights as calculated above.

Table E-1 - Radiator Component Weights

COMPONENT WT.	WEIGHT (LBS.)
Condenser Tubes	31.2
Beryllium Panels	37.8
Top Manifold	4.3
Bottom Manifold	2.7
Total Radiator	76.0

B. Equipment Panels

Where the hexagonal structure is to be utilized to carry equipment, aluminum honeycomb is bonded to the radiating skin between condenser tubes and a .040 in.-thick aluminum skin is bonded to the honeycomb/tube composite to form an inner skin, which serves as a heat transfer surface. Inserts are placed in the honeycomb in provide equipment mounting provisions. A typical cross section of an equipment panel is shown in Figure E-2.

Figure E-2. Panel Cross Section

The weight of the inner skins and honeycomb for a full 40 x 50-inch panel can be determined from the dimensions given:

inner skin	= 8.0 lbs.	
honeycomb	= 0.93	(Density = 1.6 lb/ft ³)
bond mat'l	= 0.83	(Density = 0.03 lb/ft ²)
<u>TOTAL</u>	= <u>9.76 lbs.</u>	

There are two such panels which form the north and south faces of the spacecraft. Two additional equipment mounting half panels are also required on the NE and SE faces of the spacecraft located on the upper half of the spacecraft body. These half panels are identical to the full panels on the north and south faces except that only .015 inch thick inner aluminum skins are required. The weight of these panels can be calculated as follows:

inner skin	= 1.5 lbs.
honeycomb	= 0.46
bond mat'l	= 0.42
<u>TOTAL</u>	= <u>2.38 lbs.</u>

One additional equipment mounting surface internal to the spacecraft body is required for mounting low heat dissipating equipment. The required panel is approximately 25 in. x 22 in. and is 0.5 in. thick with .015 in. aluminum face sheets mounted to an aluminum honeycomb core. The weight of this panel can be calculated as follows:

inner skin	= 1.65 lbs.
honeycomb	= 0.25
bond mat'l	= 0.23
<u>TOTAL</u>	= <u>0.48 lbs.</u>

All of the honeycomb panels described above must be closed out to allow attachments to structural framing members. The four external panels (2 full and 2 half) will be attached to vertical stringers and thus need to be stiffened along two edges. This is done by filling the honeycomb core along those panel edges with corfil and supplying inserts to allow attachment to the stringers. The weight of the corfil can be calculated as 1-lb per full panel and 1/2 lb per half panel (density = .02 lb/in³).

The interior panel is stiffened with a 0.5 inch width of corfil around the entire periphery. This weight of corfil can be calculated as 0.47 lb. Therefore, total corfil weight is 3.47 lbs.

C. Radiator Attachment

The sections of the external skin-tube radiator which are not converted into equipment panels must be stiffened along the edges with corfil to provide attachment points. This is accomplished by providing a strip of honeycomb and skin along the vertical tube-skin panels, filling the honeycomb with corfil and securing inserts within the stiffened core. The weight thus added is calculated to be 0.72 lb.

Table E-2 summarizes the weights of the equipment-panel components:

Table E-2. Equipment Panel Weights

	Skin Wt.	Honey- comb Wt.	Bond Mat'l Wt.	Corfil Wt.	Total Panel Wt.
North-Full Size	8.0	.93	.83	1.0	10.76
South-Full Size	8.0	.93	.83	1.0	10.76
NE-Top 1/2 Panel	1.5	.46	.42	.5	2.88
SE-Top 1/2 Panel	1.5	.46	.42	.5	2.88
Interior Panel (25"x22")	1.65	.25	.23	.47	2.60
Radiator Attachment	.45	.14	.13	---	.72
TOTALS	21.10	3.17	2.86	3.47	30.60

Total weight required for addition of equipment panel = 30.6 lbs.

E-3. Stress Analysis of Equipment Panels and Radiator Assembly

The weight distribution on the panels forming the hexagonal structure is depicted in Figure E-3.

Figure E-3. Equipment Panels Weight Distribution

The controlling environmental factor for the panels is acoustic noise. Each full equipment panel weight, excluding the top and bottom manifolds, can be calculated as 21.26 lb. (Tables E-1 and E-2). This does not include the manifolds, the mid-elevation stiffener (which will be discussed later) or the corfil used to stiffen the panel area.

E-3.1 North and South Panels

E-3.1.1 Mass Properties

The north and south equipment panels have a mass loading which can be calculated as follows:

$$p(\text{psi}) = \frac{Wt}{\text{area}} = \frac{21.26 + 93.0}{(40)(50)} = .05713 \text{ psi}$$

Using data developed for the design of honeycomb panels under acoustic loading, this pressure yields an "equivalent 'g'" load factor of 17.0 g's. The total design load can be calculated as follows:

$$\text{Design Load} = (17.0)(21.26 + 93.0) = 1,942 \text{ lbs.}$$

This load is carried out to the vertical stiffeners located at each corner of the hex, the design of which is discussed in Section E. The load is conservatively considered to be uniformly distributed over the panel with the value:

$$W = \frac{\text{Load}}{\text{Area}} = \frac{1942}{(40)(50)} = .97 \text{ lb/in}^2$$

Assuming the panel to be simply supported along vertical edges of the hex, the max. bending moment per unit panel length can be calculated as follows:

$$M_{\max} = \frac{WL^2}{8} = \frac{(.97)(40)^2}{8} = 194 \text{ in-lb/in}$$

The section properties of the panel cross section are calculated below:

ext. skin = .047 in thick beryllium
 11 tubes (1/2 in 0.0 x .090")
 int. skin (.040)" thick Al)

.50

Element	Area	y	Ay	Ay ²
Ext. Skin*	2.35	.5635	1.324	.746
Int. Skin	2.0	.02	.04	.0008
Condenser Tubes	1.275	.29	.37	.107
Totals	5.625		1.734	.854

* These stress calculations assume an aluminum external skin which is conservative. Thus the margins of safety will be greater than those quoted herein.

$$\bar{y} = \frac{Ay}{A} = \frac{1.734}{5.625} = .308$$

$$\bar{I} = Ay^2 - Ay^{-2} = .854 - (5.625)(.308)^2 = .32 \text{ in}^4$$

$$\text{Therefore } I = \frac{.32}{50} = .0064 \text{ in}^4/\text{in}$$

E-3.1.2 Stress Estimates

Shear Stress

If all the shear load is taken by the condenser tubes,

$$\text{tube average shear stress} = \frac{\text{Force}}{\text{Area}} = \frac{1942 \text{ lb.}}{5.625 \text{ in.}^2} = \text{very low}$$

If all the shear load is taken by the honeycomb-

$$\tau = \frac{\text{Force}}{\text{Area}} = \frac{1942}{(2)(50)(.5)} = 38.8 \text{ lb/in}^2$$

However, $\tau_{\text{allowable}} = 50 \text{ lb/in}^2$

Thus, shear stresses present no problem.

Bending Stresses

$$\sigma = \frac{Mc}{I} = \frac{(\text{in-lb./in})(\text{in})}{(\text{in}^4/\text{in})} = \frac{(194.)(.308)}{.0064} = 9336. \text{ psi}$$

Therefore, margin of safety is large.

E-3.2 Northeast top panel

E-3.2.1 Mass Properties

The north east upper equipment panel has a mass loading which can be calculated as follows:

$$1/2 \text{ panel wt} = 10.63 \text{ lbs.}$$

$$\text{With } 4.13 \text{ lbs of equipment, NE panel weight} = 51.93 \text{ lbs.}$$

This corresponds to a mass loading of:

$$\rho(\text{psi}) = \frac{Wt}{\text{Area}} = \frac{51.93}{25 \times 40} = .0519 \text{ psi}$$

$$\text{'g' equivalent} = 16.5 \text{ g's}$$

$$\text{design load} = (16.5)(51.93) = 857 \text{ lbs.}$$

$$\text{Max Moment} = \frac{WL^2}{8} = \frac{(857/25)(40)}{8} = 171.4 \text{ in-lb./in}$$

The section properties are similar to those of north and south panels except that the interior aluminum skin is only .015" thick.

Calculating the required properties:

Element	Area	Y	A _y	A _y ²
Ext. Skin	1.175	.5385	.633	.341
Int. Skin	.375	.007	.003	----
Condenser Tubes (51/2 eff)	.637	.265	.169	.045
Totals	2.187		.803	.386

$$\bar{y} = \frac{A_y}{A} = \frac{.803}{2.187} = .367$$

$$I = .386 - (2.187)(.367)^2 = .091 \text{ in}^4$$

E-3.2.2 Stress Estimates

$$\text{Max. bending stress} = \frac{(171.4)(.367)}{.0036} = 17500 \text{ psi}$$

$$\text{Margin of Safety} = \frac{\text{Allow Stress}}{\text{Stress}} - 1 = \frac{25000}{17500} - 1 = .43$$

$$\text{Shear Tube Stress} = \frac{857 \text{ lbs.}}{(2)(.637)} = (\text{low})$$

$$\text{Honeycomb Shear Stress} = \frac{857}{(.5)(25)(2)} = 34.3 \text{ lb/in}^2;$$

$$\text{M.S.} = \frac{50}{34.3} - 1 = .46$$

Either the tubes or the honeycomb can withstand the shear stress.

E-3.3 Southeast top 1/2 panel

The mass loading density of this half-panel is less than that of the northeast top half-panel. The panel construction is identical. Thus the margins of safety for acoustic loading of this panel is larger than was the case for the northeast top half-panel.

E-3.4 Bare Radiator Panels (Northwest, Southwest, Northeast Lower Half-panel, Southeast Lower Half-Panel)

These panels are not loaded by equipment and a load factor of 100 g's has been conservatively assumed for design purposes.

$$\text{Full Panel Wt} = \left(\frac{31.2 + 37.8}{6} \right) = 11.5 \text{ lb.}$$

$$\text{Max Shear Load} = \frac{(11.5)(100)}{2} = 575 \text{ lb.}$$

$$\text{Max Bending Moment} = \frac{(11.5)(100)}{8} (40) = 115 \text{ in-lb/in}$$

Calculating the Section Properties:

Element	Area	y	A _y	A _y ²	I _O
Ext Skin	2.35	.5235	1.23	.644	
Condenser Tubes (11)	1.275	.25	.319	.080	.0282
Totals	3.625		1.549	.724	

$$\bar{y} = \frac{A_y}{A} = \frac{1.549}{3.625} = .427$$

$$I = .724 - (3.625)(.427)^2 + .0282 = .063 + .0282 = 0.0912 \text{ in}^4$$

$$\text{Max Bending Stress} = \frac{(115)(.427)}{.001824} = 26,920 \text{ psi}$$

$$\text{Margin of Safety} = \frac{28000}{26920} - 1 = .04 \text{ which is satisfactory}$$

The shear stress is clearly low.

Thus the equipment panels and the radiator panels are sufficiently strong to withstand the acoustic environment which would be experienced during the launch phase of flight.

E-4 Frame Structure

The six hexagonal sides of the spacecraft are connected along their long sides by vertical framing members. The top and bottom of these panels are stiffened by the manifold structures and the panels are further stiffened by a horizontal framing member located midway between the top and bottom of the hexagonal spacecraft. The configuration is sketched schematically below:

The loading on the horizontal framing members, the manifolds and the midpoint stiffener, is essentially the same as the loading on the framing members in the solar-powered design. The manifolds are capable of withstanding more load than the corresponding framing members in the solar-powered design and are thus satisfactory. Two midpoint stiffeners identical to the one used in this design are used in the solar-powered design. This is due to the modular assembly procedure. One stiffener is more than adequate in strength. Since the manifolds were included in the radiator weight calculation, a substantial reduction in framing member weight plus one midpoint stiffener weight which amounts to a total of 16.3 lbs.

E-4.1 Vertical Longerons

The vertical longerons join each of the six panels at their vertical edges. The critical, i.e., most highly loaded, member is the one connecting the north and northeast panel over the upper half of the spacecraft. The member is sketched in cross section below:

The Properties of this Section are Calculated below:

Element	A	y	Ay	Ay ²	I _O
①	.048	1.07	.05136	.055	
②	.048	1.22	.05856	.0714	
③	.048	1.07	.05136	.055	
④	.088	.63	.05544	.0349	.011
⑤	.060	.04	.00240	.0001	
Totals	.292		.219	.2164	.011

$$\bar{y} = \frac{Ay}{A} = \frac{.219}{.292} = .75$$

$$I = .2164 + .011 - (.292)(.75)^2 = .063 \text{ in}^4$$

The loading on this member is the result of acoustic loading of the adjacent panels, as shown:

This distribution loading, R , must be bridged by the stiffeners at the top and midpoint of the hexagonal structure as shown below:

$$\text{Max moment} = \frac{wl^2}{8} = \frac{(31.66)(25)^2}{8} = 2473 \text{ in-lb.}$$

$$\sigma_{\text{bonding}} = \frac{Mc}{I} = \frac{(2473)(.75)}{.063} = 29440 \text{ psi}$$

$$\sigma_{\text{allow}} = 37,500 \text{ psi (Alum. 2024-T4)}$$

$$\text{Margin of Safety} = \frac{37,500}{29,440} - 1 = .27$$

Calculating the weight of the six vertical framing members,

$$W = (\text{\# of stringers})(\text{area})(\text{length})(\text{density}) = (6)(.292)(50)(.1) = 8.76 \text{ lbs.}$$

The rest of the structural frames contained in the solar-powered design are adequate and will be utilized on the integrated radiator design without change. The weight of structural framing members used on the integrated radiator design are compared to those of the solar-powered design in Table E-3:

Table E-3 Weight of Structural Framing Members

Item	Integrated Radiator Weight	Solar-System Weight
6 vertical stringers	8.76	5.70
Top, bottom & midpoint panel stiffeners		10.46
Other structural framing members (retained)	17.34	17.34
Totals	26.1 lbs.	33.5 lbs.

Therefore, the total weight of radiator, equipment panels, and frames in the nuclear integrated design is: $76.0 + 30.6 + 26.1 = 132.7$ lbs.

E-5. Harness Weight Estimate

The electrical harness consists of two segments in the solar-powered design. The segment servicing the solar arrays and batteries weighs 25.3 lb. It is conservatively estimated that 75% of this harness will not be required in the nuclear spacecraft, leaving 6.3 lbs.

The rest of the harness weight, 22.7 lb., services housekeeping and payload equipment and will be essentially unaffected by the nuclear design. Thus the total harness weight for the integrated radiator nuclear design is estimated at 29 lbs.