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METEOROLOGICAL SATELLITE STUDY
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SNAP Reactor Power System for the
Operational Meteorological Satellite

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ABSTRACT

A study has been conducted to determine the applicability of reactor-powered SNAP systems for advanced Operational Meteorological Satellites (OMS). The advanced OMS systems currently being planned for operation in the 1970's would carry highly redundant electronic payloads to permit satellite lifetimes of 3 to 5 yr. The complete meteorological payload weight would be in the range of 1300 to 1600 lb and the power requirements are estimated to be from 600 to 1600 watts. The orbital altitude for the OMS is 1500 n. mi, so the payloads will have to be protected by an aluminum shield and radiation "hardened" to withstand the Van Allen radiation encountered at this altitude.

The weight of advanced operational nuclear power systems, based on extrapolation of the SNAP 10A concept and using the SNAP 2 reactor would range from approximately 800 lb at 500 watts, to 1500 lb at 2.5 kwe, including sufficient shielding to reduce the fast neutron dose to one-seventh that of the high energy Van Allen protons. Operation of these systems with passive control for more than 3 yr is feasible for power levels up to 1500 watts. By using a more advanced reactor concept with a smaller shieldable volume, the power system weights could be reduced another 120 to 140 lb.

It is shown that SNAP 10A powered satellites could meet the advanced OMS system payload and power requirements using a standard Atlas/Agena launch vehicle, and with a floxed Atlas/Agena another 900 lb of payload could be accommodated. The optimum configuration for such a satellite is also ideal for a passive stabilization system based on the gravity gradient principle.

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I. INTRODUCTION

This report presents the results of an Applications Engineering study performed for the Atomic Energy Commission under Contract No. AI-11-1 Gen-8. The purpose of the study was to determine the applicability of a nuclear reactor-thermoelectric power system for advanced Operational Meteorological Satellite (OMS) systems. The technical and programmatic information pertaining to the OMS system was obtained from and coordinated with personnel at the National Weather Satellite Center, U. S. Weather Bureau, Department of Commerce, and from their contractor, the Aerospace Corporation.

The advanced Operational Meteorological Satellite (OMS) system is a concept intended to meet the long range requirements established by the U. S. Weather Bureau. As presently conceived, it would be an extension and application of the technology developed by the TIROS and NIMBUS programs and should be ready for operational use around 1970-71. An improved TIROS system is currently being developed to serve as an interim operational system until the advanced OMS can be developed.

Section III of this report contains a brief review of the TIROS and NIMBUS programs, a discussion of the operational requirements for the advanced OMS, details of the payload characteristics and power requirements, and an analysis of the Van Allen radiation shielding and payload hardening requirements.

The specific type SNAP system selected for this study is based on the SNAP 10A concept which employs a compact nuclear reactor as the heat source and a passive thermoelectric power conversion system. This choice was based on the advanced state of development of this type of system, the range of power levels required, and the general OMS requirement for a reliable, long-lived, passive power system. The first generation of flight-qualified SNAP 10A units are now being readied for flight demonstration tests in March 1965. A complete description and discussion of the developmental status of this 500-watt system is included in Section IV of this report.

The SNAP 10A-type systems which would be considered for use in an advanced OMS system 6 yr hence are based on an extrapolation of the technology developed for the flight test system. These extrapolations are consistent with the objectives for advanced SNAP development programs which are currently

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underway and with an availability date of 1968 to 1970. A description of these advanced operational SNAP 10A systems and a discussion of their performance characteristics for a wide range of power levels is included in Section IV.

Finally, a conceptual design of a SNAP 10A-powered OMS is developed for the purpose of evaluating its parametric performance capabilities. A description of this design and an analysis of the payload and electrical power available for a variety of launch systems is presented in Section V. The major results and conclusions of the study are summarized in Section II.

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II. SUMMARY AND CONCLUSIONS

A. OMS REQUIREMENTS

The Weather Bureau's requirements for an advanced Operational Meteorological Satellite system call for a system of two or more satellites operating in circular polar orbits at an altitude of 1500 n. mi. The orbital planes would be displaced to provide global coverage at least four times a day at 6-hr intervals. A payload composed initially of TV cameras, IR sensors, recorders, transmitters, etc., would be used to obtain cloud cover pictures, heat balance information and other useful meteorological data on a routine basis. Spacecraft flexibility and substantial power and payload margins are also desired to accommodate additional operational equipment and future payload variations, thus increasing the usefulness of the system and avoiding early absolescence.

A useful spacecraft lifetime of 3 to 5 yr has been set as a design objective for the OMS in order to achieve a reasonable operating cost for the complete system (25 to 40 million dollars/yr). Studies performed for the Weather Bureau by the Aerospace Corporation indicate that this can be accomplished only if the electronic payload is made highly redundant and if semipassive power and stabilization systems are used. Specifically, a four to five-fold replication of the 320-lb nonredundant meteorological payload is required, which puts the minimum payload weight in the range of 1300 to 1600 lb. Payloads of this size require launch systems in the Atlas/Agena class but the increased costs of the larger boosters are more than compensated by the reduced launch rates necessary to maintain the system of two or more operational satellites. The power requirements for these highly redundant payloads are estimated to be from 600 to 1600 watts, depending on the techniques used to achieve the required redundancy and the extent to which microelectronic circuits are employed.

Operation of the OMS at an altitude of 1500 n. mi is near optimum for meteorological observations, but exposes the satellite to a region of high Van Allen radiation. Assuming that the high-energy "Starfish" electrons decay as expected and that no further experiments are conducted with nuclear weapons in space between now and 1970, the high energy proton flux would appear to be the biggest problem. The radiation dose due to electrons and low energy protons can be effectively reduced by surrounding the payload with an aluminum

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shell of suitable thickness. Aluminum is relatively ineffective as a fast proton shield, however, and the dose that would be accumulated in a 5-yr lifetime would be equivalent to about 4×10^{12} nvt of fission spectrum neutrons. Thus, the advanced OMS payload must be radiation "hardened" to this level regardless of the type power system used and in addition must be protected by an aluminum "environmental" shield at least 0.1 in. thick. Experience with other radiation hardening programs indicates that satisfactory operation of transistorized circuits at these dose levels is feasible, but requires that careful attention be given to circuit design and to the selection of radiation resistant components and materials. If such a program is undertaken, then an ionization dose tolerance of approximately 10^7 rads should also be achievable.

B. SNAP 10A SYSTEMS

The major improvements that would be realized with the advanced operational SNAP 10A systems intended for use in the late sixties are substantial reductions in both the shielded system weights and the neutron and gamma radiation dose rates to the payload. The weight reductions are achieved by raising the average operating temperature about 300°F, by eliminating most of the diagnostic instrumentation used in the current system, and through miscellaneous detailed, mechanical and metallurgical improvements. The improved shielding is obtained through geometrical modifications of the shield and (for the OMS system) the use of an extendible boom to increase the separation distance between the reactor and the payload. Using this approach, it is possible to obtain a 20 to 30% net decrease in the shielded system weight of a 500-watt unit and decrease the payload radiation dose to a level which is small compared to that due to Van Allen radiation.

The weights for advanced, operational SNAP 10A systems based on using both the SNAP 2 reactor and a more advanced reactor with a smaller shieldable envelope are shown in Figure 1 for a wide range of power levels. The fast neutron dose rate for all of these systems has been reduced to 10^{11} nvt/yr and the gamma dose rate varies from 10^5 to 5×10^5 rads/yr, depending on the power level. The weight of an unoriented solar cell-battery system designed for 3-yr operation in the specified orbit and altitude is included for comparison. The values shown for the latter include a 48% degradation allowance for Van Allen Radiation effects and between 10 and 15% discharge of the batteries. It will be noted that above approximately 500 watts, the SNAP systems have a significant weight advantage.

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Figure 1. Weight and Power Comparison

Since long-term reliable operation is particularly important for the advanced OMS system, the operating lifetime that could be achieved using the SNAP 2 reactor with inherent, passive control was investigated over a range of power levels. By increasing the amount of burnable, neutron-absorbing material in the core and overdesigning the power conversion system 20% to allow for long-term temperature changes and thermoelectric element degradation, it was found that the electrical power would not drop below the minimum rated power for more than 5 yr at 500 watts, 4.5-yr at 1 kwe, and 3-yr at 1.5 kwe. Thus, the capability for achieving the 3- to 5-yr lifetime without any moving, mechanical, or electromechanical parts appears feasible for the range of power required for the advanced OMS. The advanced, high-temperature hydrogen barrier material required to achieve this performance has already been developed and proven in the SNAP 8 Experimental Reactor which just completed a successful endurance run at 450 kw with NaK temperatures of 1300°F.

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The inherently high reliability of the SNAP 10A type system would be maintained and improved in the advanced systems through a continuation of the vigorous quality control and test procedures already established for the flight test system and through numerous design simplifications and improvements. Thus, a reliability objective commensurate with that of the other major OMS subsystems is considered a realistic objective for the time period of interest.

Questions pertaining to nuclear safety should not affect the applicability of reactor powered SNAP systems for operational satellites. The inherent safety characteristics, design features, and handling procedures that have been developed for the SNAP reactors impose no special handling problems at the launch site. Radioactive fission products are not present until the system is started up in orbit (after a stable orbit has been achieved). The reactor is designed to burnup upon reentry to preclude criticality after hitting the earth; it can be shut down at any time by ground command and incorporates design features to shut it down automatically after the NaK temperature drops below a preset value. After the reactor has been shut down, the fission products would decay while the "dead" satellite continues to orbit the earth. For the high altitude specified for the OMS, the radioactivity would decay to a level lower than that of the uranium before the satellite reenters the earth's atmosphere.

The cost of the advanced, operational SNAP 10A systems (exclusive of development costs) would be from 1.5 to 2 million dollars per system for the power range applicable to the OMS. For semiproduction runs of identical units, the cost would be expected to drop to 1 to 1.5 million dollars. The intrinsic value of the enriched uranium fuel is only about \$60,000.

C. SNAP 10A POWERED OMS

The conceptual design evolved for a SNAP 10A powered OMS system is illustrated on the frontispiece. The general configurations shown consists of a 5-ft-diameter pillbox shaped structure which serves as a Van Allen shield and contains the highly redundant, densely packed electronic payload, an extendible boom to separate the payload from the nuclear power system, and a 1-kwe SNAP 10A power unit. The boom would be extended after the satellite has been placed in orbit to minimize the structural loads. Studies involving a tradeoff between shield and boom system weights indicate that a 40-ft separation distance is near optimum. With this arrangement, the gravity gradient torque should be sufficient to maintain the satellite vertical within 1 or 2 deg.

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The net electronic payload available in the SNAP 10A powered OMS was determined by combining the major subsystem weights with the launch capabilities of several potential launch systems for the specified altitude and orbit. The weight breakdown for systems using a 1-kwe SNAP 10A system (with the SNAP 2 reactor) are tabulated in Table 1 and the net electronic payloads are summarized in Figure 2 for a range of available power levels. It will be noted that sufficient payload exists using the standard Atlas/Agena to accomplish the basic meteorological mission and with a floxed Atlas/Agena a substantial margin exists to accommodate future power and payload growth.

TABLE 1
1-kw SNAP 10A POWERED OMS WEIGHT DISTRIBUTION
(lb)

Subsystem	Launch Vehicle		
	Atlas-Agena D	Floxed Atlas-Agena D	Atlas-Centaur
1. SNAP 10A power system	950	950	950
2. Gravity-gradient stabilization and control system	150	150	150
3. Payload structure, Van Allen shield, and electrical harness	265	335	455
4. Electronic payload	1335	2265	3845

Figure 2. SNAP 10A Powered Weather Satellite Electronic Payload vs Power

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D. CONCLUSIONS

The following conclusions may be drawn relative to the applicability of advanced, operational versions of the SNAP 10A system to advanced Operational Meteorological Satellites:

- 1) The weight and power characteristics of the nuclear power system are compatible with the requirements of the OMS payload and the capabilities of available launch vehicles. In the power range required for system growth and to obtain the level of redundancy necessary to achieve long-life and thus low system costs, the weight of the complete SNAP system is believed to be lower than that of solar cell-battery systems designed for a 3-yr life at the altitudes of interest.
- 2) The SNAP 10A system is also considered suitable for the OMS with respect to other important criteria such as reliability, safety, cost, availability, size, and development status. The system requires no preferential orientation space, would not deteriorate due to space radiation, and is insensitive to sun-shade transients.
- 3) The radiation levels due to the reactor heat source can be effectively reduced to levels which are insignificant compared to Van Allen radiation and acceptably low on an absolute basis, thus eliminating one of the usual disadvantages cited for nuclear power systems.
- 4) The optimum configuration for a SNAP 10A powered OMS results in an arrangement which is also ideal for a passive, gravity gradient stabilization system.

It is concluded, therefore, that advanced, operational versions of the SNAP 10A system would not only be applicable but would appear to be ideally suited for use in advanced Operational Meteorological Satellites.

III. ADVANCED OMS SYSTEM REQUIREMENTS AND CHARACTERISTICS

A. INTRODUCTION

Since the advanced OMS system requirements and characteristics are derived in part from the experience gained with TIROS and NIMBUS, a brief review of these programs is in order.

B. TIROS

A typical TIROS satellite is illustrated in Figure 3 and a summary of its major design features and operational characteristics is included in Table 2.

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Figure 3. TIROS

The basic feasibility and practical value of meteorological satellites have been firmly established by eight successful TIROS flights, the first of which was launched on April 1, 1960. Although TIROS I was intended to be experimental only, the TV pictures received the first day were of such quality that they were used in operational weather analyses. To date, more than three hundred thousand usable cloud cover pictures have been obtained similar to those shown in Figure 4. Numerous hurricanes and tropical storms have been spotted and

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TABLE 2
TIROS CHARACTERISTICS

Geometry	Pillbox
Orbital altitude	380 n. mi
Orbital inclination	48° equatorial
Stabilization	Spin stabilized
Earth coverage	10 to 25%
Design life	4 mo
Launch vehicle	Thor/Delta
Status	VIII launched 12-21-63
Experiments	TV cameras
	MRIR*
	APT System†
Camera resolution	500 lines/frame
TV resolution	1 mi
MRIR sensors resolution	30 mi
Maximum power	20 watts
Subsystems	Weight (lb)
Structure	104
Guidance and control	10
Telecommunications	15
Power supply	80
Sensors	72
Subsystem Total	281

*First TIROS' carried a 5-channel medium resolution infrared scanning radiometer (MRIR)

†TIROS VIII, first APT system, Automatic Picture Transmission system for weather satellites

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TIROS IV, 807/806, fr 18
1619 GMT, APRIL 5, 1962

TIROS IV, 880-direct, fr 26
2012 GMT, APRIL 10, 1962

TIROS III, 778/777, fr 16
1357 GMT, SEPT 4, 1961

TIROS III, 679/678, fr 12
1625 GMT, AUG 28, 1961

TIROS III, 793/792, fr 15
1506 GMT, SEPT 5, 1961

TIROS IV, 1005/1004, fr 8
1123 GMT, APRIL 19, 1962

a Open wave, (52°N , 37°W ,
1004 mb), April 5, 1962

b Wave cyclone beginning to
occlude, (46°N , 59°W , 996 mb),
April 10, 1962

c Cyclone 12-18 hours after
occlusion began, (48°N , 45°W ,
1006 mb), September 4, 1961

d Deep occluded cyclone near
maximum intensity, (30-36 hours
after occlusion began, 51°N , 24°W ,
978 mb), August 28, 1961

e Weakening occluded cyclone,
36-48 hours after occlusion
began, (50°N , 45°W , 1006 mb),
September 5, 1961

f Dissipating cyclone, more than
48 hours after occlusion began
(49°N , 4°W , 1009 mb),
April 19, 1962

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Figure 4. Stages in the Life Cycle of North Atlantic Cyclones

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tracked, ice reconnaissance and snow cover observation have been demonstrated, and measurements of the Earth's radiant heat losses have been obtained. In addition to these accomplishments, extensive ground data handling and distribution networks have been developed to the point where TIROS meteorological data are now made available to weather forecasters in the United States and abroad within 2 to 3 hr.

The first eight TIROS have been externally similar, but have differed somewhat in instrumentation, due to the experimental nature of the spacecraft. Typical payloads have consisted of two television cameras, infrared sensors, a magnetic tape recorder for storing the pictures, a command receiver, control programmer and means for transmitting data to ground stations. TIROS VIII carried an automatic picture transmission system (APT), from which a properly equipped ground station within line of sight of the satellite can receive the local weather picture directly.

Despite the outstanding success of the TIROS satellites, several basic shortcomings limit its operational usefulness. Since the drum-shaped spacecraft is spin-stabilized with the sensors pointing along the spin axis, the cameras are aimed at the earth during only about one-third of a single orbit, as indicated in Figure 5. The orbit and altitude are such that the polar regions are not covered at all and the cameras aboard a single satellite do not see certain areas of the earth for several weeks. Also, due to the limited size, weight, and power available, it has not been possible to provide for payload redundancy. The useful lifetime has thus ranged from 1.5 to 13 months, the average being approximately 9 months. The first of these deficiencies will be resolved in the configuration being developed for the interim operational system, details of which are not yet available.

C. NIMBUS

Plans for the NIMBUS weather satellite system were laid almost simultaneously with those for TIROS. Due to its greater complexity, sophistication, cost, and difficult technical problems, however, it has required a longer development period. Current plans call for several NIMBUS flights as a research and development satellite, but not as a part of an operational meteorological system.

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Figure 5. TIROS-Inclined Orbit Space Oriented

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Figure 6 illustrates the NIMBUS configuration; its major design features and characteristics are summarized in Table 3. Designed to operate in a near-polar orbit with the orbital plane in line with the sun, it would pass within viewing distance of every point on earth each day at approximately noon and midnight. Thus, the oriented solar paddles need rotate only about a single axis, 360° per orbit. A combination of gas-jets, inertial wheels and horizon sensors are used to keep the base of the spacecraft (which contains the TV cameras and infrared sensors) pointing toward the earth while sun-seekers and drive motors keep the solar paddles in a plane perpendicular to the sun's rays.

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Figure 6. The NIMBUS Weather Satellite System

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TABLE 3
NIMBUS CHARACTERISTICS

Geometry	Dumbbell
Orbital altitude	600 n. mi
Orbital inclination	80° polar, retrograde
Stabilization	active 3 axis, earth oriented
Earth coverage	100%
Design life	6 mo
Launch vehicle	Thor/Agena
Status	First flight scheduled for late 1964
Experiments	AVCS System* APT System† MRIR§ HRIR**
AVCS camera resolution	800 lines/frame
AVCS TV resolution	1/2 mi
IR sensors resolution, MRIR	30 mi
HRIR	5 mi
Maximum power	400 watts
Subsystems	Weight (lb)
Structure, thermal controls, electrical harness	172
Guidance and control	150
Telecommunications	81
Power supply	195
Sensors	152
Subsystem Total	750

*AVCS, Advanced Vidicon Camera System, 3 cameras to take 33 composite pictures during 68-min sunlight period.

†APT, Automatic Picture Transmission, continuous picture transmission of local cloud conditions.

§MRIR, Medium Resolution Infrared System, has been removed because of weight limitations.

**HRIR, High Resolution Infrared System, radiometer for night-time cloud cover.

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The design lifetime for the spacecraft and subsystems is 6 months. The original NIMBUS design called for a fully redundant electronic payload, but this objective had to be sacrificed when it later became apparent that the total spacecraft weight would exceed the Thor-Agena booster capability for operation at the desired orbital altitude. The nonredundant electronic payload weight and power requirements are summarized in Table 4. It is significant for this study to note that the NIMBUS payload weighs approximately 233 lb and consumes more than 1-watt/lb during interrogation.

TABLE 4
NIMBUS PAYLOAD CHARACTERISTICS

Equipment	Weight (lb)	Operating* Power (watt)
Advanced vidicon camera system	76	82
Automatic picture transmission system	26	48
High resolution infrared system	30	14
S-Band transmitter	20	145
Command/Clock	32	9
Telemetry/Beacon	49	8
Equipment Total	233	306†
		Power Demand (watt)
		Daytime Nighttime
Operation	206	170
Interrogation	400	342

*Guidance and Control system requires 78 watts

†The power supply can provide an average of 200 watts for the entire orbit

D. OMS OPERATIONAL REQUIREMENTS

The general requirements that have been set by the U. S. Weather Bureau for the advanced OMS system may be summarized as follows:

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- 1) Complete global coverage is required, four times a day at 6-hr intervals.
- 2) The primary instrument payload will initially be composed of broad and fine scale TV cameras, medium and high resolution infrared sensors. (The specifications for the resolution, zenith angle, uniformity, and other characteristics of these instruments have been set on the basis of obtaining useful meteorological data.)
- 3) The spacecraft should provide sufficient flexibility and growth capability to accommodate future experiments and payload variations in order to avoid early obsolescence.
- 4) Data is to be accumulated in orbit and transmitted to a single ground station located at Fairbanks, Alaska. (In addition, partial data readout of global data and direct readout of local data from the satellite shall be provided to regional forecasting offices.)
- 5) The total, annual program costs shall be minimized, consistent with meeting the technical objectives. A reasonable cost target is considered to be 25 to 40 million dollars/yr.

Based on these general criteria, a series of system design, cost, and reliability studies has been performed by the Weather Bureau and the Aerospace Corporation^{1,2} to determine the general characteristics of the system that could meet these objectives. The major features of the resulting OMS system which are pertinent to the present study are shown in Table 5.

It may be noted that the major improvements sought for the advanced OMS system (relative to NIMBUS) are the higher operating altitude, more frequent observation and increased satellite lifetime. The 1500-n. mi altitude is near-optimum for obtaining useful cloud cover pictures as shown in Figures 7 and 8 and permits data readout each orbit to a single existing ground station located at Fairbanks, Alaska^{3,4}. The requirement for global coverage, four times a day at 6-hr intervals can be met with a system of two satellites operating in polar orbits with their orbital planes displaced 90°.

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TABLE 5
OMS SYSTEM CHARACTERISTICS

Number of operating satellites in orbit	2 or more
Orbital parameters	Polar, circular displaced 90°
Orbital altitude	1500 n. mi
Launch vehicle	Atlas-Agena
Total spacecraft weight	2700 lb
Control and stabilization system	Gravity gradient
Electronic payload weight (nonredundant)	320 lb
Electronic payload power requirements (nonredundant)	320 watt
Useful life	
Minimum required	~1.5 yr
Design objective	3 to 5 yr
Required level of payload redundancy	4 to 5

The longer OMS lifetime is required to reduce the total system operating cost to the desired range (25 to 40 million dollars/yr). A useful lifetime of 3 to 5 yr has been set as a design objective. The results of the studies performed by Aerospace^{1,2} show that the upper limit on the desired system costs can be achieved only if the mean satellite lifetime can be extended to approximately 1.5 yr. The reliability analyses conducted indicate that this level of improvement is achievable by replicating the electronic payload four to five-fold and by employing simipassive stabilization and power supply systems. The increased cost of the larger booster systems required to place the heavier electronic payloads in a higher altitude orbit is more than compensated by the reduced launch rates necessary to maintain the system of two operational satellites. To reach the lower range of overall system costs would require an additional component improvement program similar to that undertaken for the Minuteman program. With this approach, an improvement in the satellite lifetime by a factor of five is considered possible and the total 10-yr program costs would approach an average of 23 million dollars/yr.

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Figure 7. Optimum Satellite Altitude

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Figure 8. Satellite Picture Worth

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E. OMS PAYLOAD CHARACTERISTICS

1. Component Description

The basic OMS payload consists of a combination of electronic, optical, and electrical-mechanical equipment which are required to perform the following functions:

- a) Data acquisition
- b) Data storage and readout
- c) Data transmission
- d) Control and command receiver.

A brief discussion of the equipment that has been developed (or is being developed) for use on the OMS follows.

a. Data Acquisition Equipment

The primary data acquisition equipment is expected to consist of TV camera systems for daytime cloud cover observation and infrared sensors for nighttime coverage and heat balance measurements.

(1) TV Systems

The characteristics of several types of TV cameras that have been used on TIROS and/or developed for NIMBUS are given in Table 6 along with those of an advanced system which would utilize fiber optics.

The exposure time indicated for each of the cameras is a reflection of the fact that discrete rather than continuous real time pictures are obtained. In TIROS pictures were taken either on command or automatically every 30 sec for specified 16-min intervals. In the AVCS system pictures are obtained from each of three cameras arranged like a fan every 108 sec, and with APT a picture is taken every 208 sec.

The use of a discrete picture system rather than a continuous one is justified on the basis that it is adequate to the task. The principal reason that discrete pictures are satisfactory is the finite period of time required for the satellite to move from one field of view to the next. This interval of time may be utilized to readout or transmit the picture information; the increased time permitting lower frequencies, narrower bandwidth and, hence, lower

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TABLE 6
CHARACTERISTICS OF SOME CAMERA SYSTEMS DEVELOPED
FOR METEOROLOGICAL SATELLITES

Satellite	TIROS [§]	NIMBUS [§]		OMS-Fiber Optic ^{**}
		AVCS QTV-2	APT QTV-3	
Vidicon (in.)	1/2	1	electro- static storage	1
Deflection and focus	electro- magnetic	electro- magnetic	electro- magnetic	—
Maximum resolution (TV lines)	350	800	700	800
Bandwidth (kc)	62.5	60	1.6	—
Exposure time (msec)	1.5	40	40	—
Weight (lb)	14*	25.3*,†,**	26.4*,**	30
Power (watt)	11.6	21 [†]	18	40
Volume (cu in.)	527	400	527	—

*Total weight of camera, electronics and controls.

†Per single camera. AVCS consists of three camera units or multiples thereof.

§Reference 5.

**Reference 1.

transmitter power for a given signal-to-noise ratio. An extreme case of this is the Automatic Picture Transmission (APT) system. This system is designed to permit direct transmission of picture information to numerous modestly equipped ground weather stations. In this case, however, power economy is a secondary consideration; the prime consideration is slowing the information transmission rate to the point where it is compatible with slightly modified, ground-based, conventional facsimile equipment. The result is an extremely narrow bandwidth as shown in Table 6.

A Fiber Optic camera system is suggested in Reference 1. In one arrangement of the concept, the visual image is focused on the face of a straight line fiber optic array. The image is then conveyed along the fibers of the bundle which at the other end is broken up into serial segments which are then arranged sequentially in straight lines on the face of the vidicon tube parallel to scan axis.

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Thus, the many scans which make up a single frame correspond to a single-line image. As each line image is scanned it is recorded. The rate of scanning is chosen on the basis of the satellite altitude and speed and optical considerations so that a complete picture is built up one line at a time. A typical line picture element in the broad-scale system (1500-n. mi altitude, 78.2° field of view) might be about 3100 n. mi broad and less than 10-mi wide. The line images could be recorded either on conventional tape recorders or on the new dielectric photosensitive storage film currently being developed.

(2) Infrared Systems

Two major infrared measuring systems have been developed for use in meteorological satellites: the High Resolution Infrared (HRIR) system and the Medium Resolution Infrared (MRIR) system. Both of these systems employ a rotating or oscillating mirror scanning technique to obtain a picture of earth and cloud infrared radiance. This information can be employed to estimate cloud cover on the dark side of the earth and to estimate the earth's energy budget.

The HRIR collects data in the range of 3 to 4 μ primarily to establish night cloud cover pictures. The MRIR measures radiance in five bands to provide cloud cover and energy budget data. These bands are:

- 6.5 to 7.0 μ : water vapor absorption
- 10 to 11 μ : atmospheric window
- 0.55 to 0.75 μ : visible reference and daytime cloud cover
- 7 to 30 μ : thermal radiation
- 0.2 to 4 μ : reflected solar radiation

The data from both the HRIR and MRIR are recorded on separate tape recorders for readout on command.

The weight and power consumption of the HRIR and MRIR systems including tape recorders is estimated at 70 lb and 10 watt for each unit.

b. Data Storage Equipment

The need for data storage arises from the limited range of the satellite data transmission equipment and, except for the APT system, the limited number of ground stations equipped to receive the specialized data transmissions. Accordingly, data must be accumulated over the bulk of the

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orbit and transmitted during the short interval that the satellite is within range of a capable ground receiving station. As previously indicated, the means for data storage for both camera and IR systems is magnetic tape. The weight and power consumption of these units is included in the values given for the corresponding camera or IR system.

c. Data Transmission Equipment

The data transmission systems employed in or proposed for meteorological satellites have consisted of a stored-data transmitter (frequency S-Band) and a low-power telemetry and tracking beacon used for monitoring the functioning of satellite components and for tracking of the satellite. Typical values of the weight and power consumption of these units are:

Stored-data transmitter	- 20 lb, 100 watt
Telemetry-beacon	- 50 lb, 10 watt

d. Control-Command Equipment

This equipment comprises the command receiver which accepts and decodes the ground commands and the control unit which converts the commands into operations. The control unit normally includes a crystal controlled oscillator clock which provides a time base for sequential operations. The weight and power consumption of a typical control-command unit for NIMBUS or OMS is 30 lb and 70 to 85 watts.

A summary of the "base line" (nonredundant) equipment complement, weight and power requirements estimated for the advanced OMS are shown in Table 7. It will be noted that the weight to power ratio is 1 lb/watt. The actual power and weight requirements necessary to achieve the required four to five-fold redundancy is discussed in the following section.

2. Redundant OMS Payload Characteristics

The studies reported in References 1 and 2 have shown that the mean useful lifetime of the relatively complex OMS electronic payload must be greater than approximately 1.5 yr in order to reduce the overall program costs to a reasonable range. With the current state-of-the-art of spacecraft electronics, it was further demonstrated that the only feasible way of achieving this goal is to replicate the baseline electronic payload four to five-fold. This implies that the total weight of the redundant payload would be four to five times that of

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TABLE 7
NONREDUNDANT OMS ELECTRONIC PAYLOAD -
BASELINE CONFIGURATION

Electronic Payload	Unit Weight (lb)	Unit Power (watt)
Broad-scale TV Line-scan subsystem	30	40
Fine-scale TV Line-scan subsystem	30	40
MRIR (Medium resolution infrared)	70	10
HRIR (High resolution infrared)	70	10
S-Band transmitter	20	100
Command/Clock	30	70
Telemetry/Beacon	50	10
Power Electronics	20	40
Total	320	320

the single nonredundant payload described in Table 7. The total electronic payload weight would thus be in the range of 1300 to 1600 lb, but the additional power required to achieve the necessary degree of redundancy was not investigated. In order to estimate the probable power requirements of the OMS, therefore, it is necessary to consider the circuit design techniques that might be employed and review what has been accomplished in this respect with other, similar systems.

To date, satellites have not made extensive use of redundancy. In general, the missions have been primarily experimental, and of relatively short lifetimes. However, the literature on redundancy in electronic applications is quite extensive and is growing rapidly.

There are two general categories of redundancy; active and standby. Active redundancy requires all components to be operative in conjunction with circuits or interconnections which permit failures of individual components without impairing the function of the system. Digital computers frequently make use of this technique and it is thus believed to be applicable to the Command-Clock and perhaps other electronic elements of the OMS payload.

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With standby redundancy, the spare components are inactive until a failure occurs, at which time they are switched into the circuit. This approach, therefore, requires the addition of sensor elements to detect a failure and switching circuits. Due to the nondigital nature of the TV cameras, the IR sensors, and the wear out mode of failure expected for the tape recorders in the OMS payload, the use of standby redundancy for these components would undoubtedly be preferred. A conceptual block diagram of a three-fold redundant TV camera system employing standby redundancy is shown in Figure 9 (Ref. 2).

In general, if active redundancy is used, electrical power requirements will increase in direct proportion to the payload weight. Thus, if four or five complete and separate OMS payloads were installed and operated simultaneously, the power requirement would be in the range of 1300 to 1600 watts (1 lb/watt).

With standby redundancy, an increase in power is required (relative to a nonredundant system) for the sensor elements and switching circuits. In order to prevent these added components from becoming the "Achilles Heel" of the system, they would undoubtedly incorporate active redundant elements at the module or piece part level.

Reference 6 presents the results of a detailed study of redundant spacecraft control systems. The results, which are summarized in Table 8 show that a three-fold redundant system (using one active and two standby systems) requires approximately twice the power of a single, nonredundant system. If this characteristic is typical of standby redundant systems, and if this mode of replication were used exclusively, then a weight to power ratio of approximately 1.5 would be expected for the redundant OMS payload. The required power would thus be in the range of 850 to 1070 watts.

In practice, it is expected that the four to five-fold increase in payload weight to achieve the desired level of reliability will be accomplished using only two or three discrete "sets" of components. Each set would then contain internally redundant circuits and elements of either the active or standby type. The optimum depth at which the redundancy is applied would undoubtedly vary with each component and subsystem, as would the choice between active and standby circuits.

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NOTE: (SE) INDICATES SENSING ELEMENT

Figure 9. Redundant Television Subsystem Block Diagram

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TABLE 8
EXAMPLE OF REDUNDANT SYSTEM POWER AND
WEIGHT REQUIREMENTS

Configuration	Reliability	Power (watt)	Weight (lb)
Nonredundant Reference System			
Complete system	0.108	50.0	80
System Level Redundancy			
Three active complete systems	0.202	157	245
One active, one standby system (switching by system)	0.20	97.5	210
One active, one standby system (switching by subsystem)	0.335	97.5	179
Axis Redundancy			
Three active systems	0.36	157	245
One active, one standby system	0.42	97.5	179
One active, two standby systems	0.556	97.5	214
One active, two standby systems (modified)	0.67	97.5	214
Hybrid configuration (active redundant amplifiers)	0.743	108.5	206
Hybrid configuration (standby redundant amplifiers)	0.753	97.5	206

The possible effects of the use of microelectronic circuits in portions of the OMS electronic payload should also be mentioned. (The nonredundant payload defined in Table 7 is based on the use of discrete component circuits.) The development of microcircuits is already an accomplished fact. Whereas the most sophisticated discrete component circuits can achieve packing densities of 10^5 equivalent parts/ft³, the new integrated microcircuits are achieving packing densities of 5×10^6 equivalent parts/ft³ for thin-film circuitry and 10^7 equivalent parts/ft³ for silicon-based microcircuits. Expressed another way, volume, and correspondingly weight, is being reduced by these new techniques by factors of up to 50 to 100. In practical equipment, volume or weight savings of these magnitudes are not realizable primarily due to the need for internal connections and for heat dissipation.

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A good example of the application of microelectronic circuits in an active redundant spacecraft control system is presented in Reference 7 and summarized in Table 9. It should be noted that the weight to power ratio of the complete, redundant system is less than 0.5 lb/watt. Also, the weight of the computer and Memory unit using microcircuits is only 40% of that of the same unit using discrete elements. The weight to power ratio of the redundant, microelectronic computer alone is only 0.23 lb/watt.

TABLE 9
GENERAL ELECTRIC COMPANY SPACE NAVIGATION SYSTEM[§]

Item	Nonredundant			Redundant*		
	Number	Weight (lb)	Power (watt)	Number	Weight (lb)	Power (watt)
Gyros and Gimbal Servos	3	36	38	6	72	75
Accelerometers	3	6	13	6	12	26
Structure	1	15	—	1	20	—
Power Supply	1	20	40	1	34	100
Computer and Memory	1	20(50) [†]	100	1	70	300
Inertial Unit Subtotal	—	97(127) [†]	191	—	208	501
Star Tracker	1	8	4	1	14	11
Horizon Scanner	1	4	4	1	11	11
Totals		109(139) [†]	199		233	523
Inertial Unit EMTBF (hr)**		5850(2480) [†]			150,000	
System EMTBF (hr)**		4870(2390) [†]			68,000	

*No redundancy in power supply.

†Paranthesized figures correspond to a system with a nonredundant computer using discrete elements.

§Uses microelectric circuits except as noted in Note[†]

**EMTBF defined for a mission of 10,000 hr.

In the OMS payload there are certain components that at present do not seem susceptible to dramatic weight or volume savings: these include the TV cameras, the IR systems, and the tape recorders. Nevertheless, it is estimated that at least 1/3 to 1/2 of the total electronic payload is susceptible to a

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significant reduction in weight and volume by use of integrated microcircuit techniques. If the weight of this portion of the payload were reduced by only 50%, a net reduction of approximately 15 to 25% would result. Since the power requirements for microcircuits are generally the same as those required for discrete element circuits, the weight to power ratio of the OMS electronic payload would decrease correspondingly. For a given available payload, therefore, the total power requirements would be expected to increase approximately 15 to 25%.

The weight savings made possible by microcircuits could either be used to further increase the redundancy or to accommodate additional experiments or operational equipment. As will be shown later, the reduced volume is particularly desirable for operation at the specified altitude, since the weight of the environmental shield which must surround the payload to reduce the Van Allen radiation to tolerable levels is appreciable. It is felt, therefore, that a strong incentive exists to incorporate microcircuit techniques in the OMS payload and that the use of such circuits will probably be standard practice for space electronic systems in the operational time period of interest for the OMS.

In summary, it is concluded that the achievement of a highly redundant OMS payload requires a substantial increase in electrical power, relative to that needed for a nonredundant payload. Considering that an appropriate combination of both active and standby redundancy techniques would probably be used in an actual system design, and that microelectric circuits will undoubtedly be applied to reduce the payload weight and volume, the weight-to-power ratio of the redundant OMS payload is expected to be in the range of 1 to 2 lb/watt, with the most probable value being around 1.5 lb/watt. The range of payload weight and power requirements that result for a four to five-fold redundant system are shown in Figure 10. A more detailed payload optimization study is required to narrow the range further.

3. Future Payload Variations

The availability of a system of operational satellites that can provide complete global coverage several times a day on a routine basis opens up many possibilities for useful functions in addition to those discussed in preceding sections. A payload and power margin is desired for the advanced OMS, therefore, both to accommodate the inevitable improvements and variations in the

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Figure 10. Weight and Power Requirements
for Redundant OMS Electronic Payload

equipment required to fulfill the basic meteorological mission and to allow additional experiments or operational equipment to increase the general utility of the system. Several examples of these variations are:

- a) Data Collection. By providing additional receiving, storage, and transmission capabilities, the OMS could pick up meteorological data from ships, floating buoys, and remote, unattended ground stations located anywhere on earth and transmit the information to the ground station network each orbit for rapid analysis and distribution. This "flying postoffice" function is expected to increase the effectiveness of world-wide meteorological forecasting appreciably.
- b) Ionospheric Mapping. Top-side sounding of the ionosphere (as demonstrated by the Alouette satellite) or microwave transmission measurements could be performed on a routine basis to improve the reliability of communication predictions. Two approaches to microwave transmission are possible; one would require a global network of ground-borne transmitters with the receiving, data

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storage, and retransmission equipment located in the satellite and the other would place the transmitter in the satellite and the receivers on the ground. The latter approach would require a satellite power capability of 1 to 2 kw.⁸ A detailed technical and economic study would be required to determine the optimum arrangement.

- c) Radiation Monitoring. Since the Van Allen radiation levels are known to vary in intensity, much like the weather, methods are now being investigated for monitoring both the spectrum and the flux of space radiation on a routine basis. By incorporating suitable detectors (and additional data storage and readout equipment) on the OMS, the electron and proton fluxes and spectra at a single altitude could be measured several times a day. This information would then be combined with that gathered by other satellites to obtain a complete three-dimensional map. The neutron background from the SNAP 10A system would not affect measurements with an ionization chamber and the gamma ionization level is about three orders of magnitude lower than that of the electrons and protons within the shadow shield angle.

The examples cited above are just a few of many possible and potentially useful payload variations that are being considered. However, both a payload and a power margin are required to accommodate any of them.

F. RADIATION SHIELDING REQUIREMENTS

1. Introduction

Operation of a weather satellite at any altitude between approximately 400 and 5000 n. mi requires that serious consideration be given to the environmental radiation encountered in the Van Allen Belt. It is an unfortunate manifestation of Murphy's Law that the altitude range which is most suitable for meteorological observation purposes (1000 to 2000 n. mi) corresponds to the region of maximum proton and electron radiation. Since these are charged particles, the dose rate can be reduced by surrounding the payload with a shield in the form of an aluminum shell of suitable thickness. The weight of this environmental shield can become appreciable, however, and it therefore

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becomes important to estimate the maximum radiation dose that the payload can withstand without degradation and to explore the possibilities of "hardening" the components; i. e., reducing their susceptibility to radiation damage.

The use of a reactor-powered SNAP system would at first seem to compound the problem of radiation damage to the OMS payload. The general approach taken in this study, however, has been to determine the approximate radiation tolerance level that the payload must withstand to obtain a reasonable environmental shield weight, and then provide sufficient reactor shielding to lower the neutron and gamma dose to a level where it would not prejudice the selection of a nuclear powered system.

A brief discussion of the radiation environment, environmental shielding, radiation damage and hardening possibilities, and conclusions regarding environmental shielding requirements follows.

2. Radiation Environment

A SNAP 10A powered OMS electronic payload will be subjected to four principal types of radiation; protons, electrons, neutrons and gamma rays. The intensity of all of these forms of radiation can be modified with suitable shielding. In order to establish reasonable reactor and environmental shielding criteria, however, it is necessary to obtain factors relating these four types of radiation in terms of equivalent damage to the most radiation sensitive electronic components. With proper care taken in the selection of other components and materials, it has been concluded¹ that the semiconductor devices will be the most critical items in the OMS payload. Thus, the radiation damage equivalence factors will be based on the behavior of this general class of device.

Relationships have been established for the displacement damage effect caused by electrons, protons, neutrons and Co^{60} gammas in semiconductors by determining the average radiation required to produce equal minority carrier lifetime changes of p on n and n on p silicon solar cells.⁹ These damage equivalents (for specific particle energies) are shown in Table 10. In addition to the displacement effect, recent data indicates that surface effects are important in gas-filled transistors. Since this phenomena is caused by ionization, the rad (=100 ergs/gm) is the appropriate damage equivalence factor.

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TABLE 10
DAMAGE EQUIVALENTS FOR SPECIFIC PARTICLE ENERGIES

Radiation	Radiation Damage Equivalent			
	Neutron/cm ²	Proton/cm ²	Electron/cm ²	Co ⁶⁰ γ (rad)
1 neutron/cm ² (fiss. spec.)	1	0.3	1.2×10^3	8×10^{-5}
1 proton/cm ² (30 Mev)	3	1	4×10^3	2.5×10^{-4}
1 electron/cm ² (1 Mev)	8×10^{-4}	2.5×10^{-4}	1	6×10^{-8}
1 rad Co ⁶⁰ γ	1.2×10^4	4×10^3	1.7×10^7	1

The Van Allen (and Starfish) radiation that would be encountered by the OMS in a polar, circular orbit was calculated and reported in Reference 1. The omnidirectional, incident flux of protons and electrons for a 5-yr operating period is shown in Figure 11 as a function of operating altitude. The differential energy spectra and penetration thickness of these particles in aluminum is shown in Figures 12, 13, and 14 for the particular altitude (1500 n. mi) selected for the OMS. The integrated 5-yr particle flux and ionization level that

TABLE 11
EFFECT OF SHIELD THICKNESS ON RADIATION LEVEL
(Based on incident flux at 1500 n. mi)

Shield Thickness in. of Al	Protons p. cm ⁻²	Electrons e. cm ⁻²	Ionization Level Rads (Air)
0.005	3.3×10^{13}	5.3×10^{15}	3×10^8
0.050	1.8×10^{12}	2.6×10^{15}	1×10^8
0.100	1.4×10^{12}	1.2×10^{15}	5×10^7
0.250	1.2×10^{12}	2.0×10^{14}	7×10^6
0.500	9.8×10^{11}	6.0×10^{12}	2×10^5
1.000	7.6×10^{11}	~0	3×10^4

would be incident on the electronic payload at 1500 n. mi is shown in Table 11 for aluminum thicknesses ranging up to 1-in. Finally, by combining the radiation dose equivalence factors discussed above (Table 10) with the data shown in Table 11, the 5-yr equivalent dose (expressed in terms of n/cm²) and the ionization dose are shown in Figures 15 and 16, respectively as a function of aluminum environmental shield thickness.

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Figure 11. Space Radiation

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Figure 12. Differential Proton Spectrum
for Circular, Polar Orbit at
1500-Nautical Mile Altitude

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Figure 13. Differential Electron
Spectrum for Circular, Polar
Orbit at 1500-Nautical Mile
Altitude

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Figure 14. Penetration Thickness in Aluminum vs Initial Kinetic Energy of Protons and Electrons

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Figure 15. 5-yr Equivalent Radiation Damage Dose vs Aluminum Thickness (Based on Displacement Damage to Semiconductors)

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Examination of Figure 15 shows that the proton dose drops off rapidly with the first 0.1 in. of aluminum, but decreases very little thereafter as the shield thickness is increased. Even with a 1-in. thick shield, the equivalent 5-yr proton dose would be approximately 2×10^{12} nvt. The displacement damage equivalent of the electrons is seen to be relatively insignificant compared to the proton damage for shield thickness greater than approximately 0.2 in. The high-energy electrons are the major contributor to the ionization dose shown on Figure 16, however, up to shield thicknesses of approximately 0.5 in. In order to reduce the 5-yr ionization dose to less than 10^7 rad, an aluminum shield approximately 0.250 in. thick would be required.

It should be mentioned, that recent data¹⁰ indicates that the high-energy fission electron flux is decaying more rapidly than was originally surmised, and would be several orders of magnitude lower than the values used for this analysis by 1970-71, barring further nuclear weapons tests in space. A 0.250-in.-thick shield would thus be highly overdesigned for a 5-yr ionization dose of 10^7 rad, but without quantitative flux and spectral data it is difficult to estimate exactly how much. If the fast proton data is correct, however, a shield thickness of at least 0.1 in. would be advisable.

3. Radiation Damage

a. General

Radiation damage to electronic components and systems has been the subject of extensive study for the past 10 to 15 yr in connection with the ANP, SNAP, NERVA, and Weapons Effects programs. The more recent encounters with space radiation problems have also stimulated a more general interest in this subject which heretofore was associated primarily with nuclear power or explosions. As a result, a vast amount of both theoretical and experimental data are becoming available which should permit the design of reliable radiation resistant electronic circuits.

Reference 1 summarized the current status of radiation effects on electronic components and materials in general and on the OMS payload in particular. The conclusions reached were that the natural, environmental radiation posed a difficult but tractable problem, that both environmental shielding and radiation "hardening" would be required, that the semiconductor devices would be the most sensitive items, but that care must also be taken in the

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Figure 16. 5-yr Proton and Electron Ionization Dose
vs Aluminum Thickness

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selection of other materials and components to ensure satisfactory, long-term operation. No specific recommendations for the tolerable radiation dose levels were ventured, but an allowance of 270 lb was made for environmental shielding.

b. Semiconductors

Since it has been indicated that the semiconductors will be the most critical radiation sensitive elements in the OMS payload, a brief discussion of the damage mechanisms involved and the probable range of radiation dose that might be expected without causing malfunction or failure is in order.

Radiation exposure of semiconductor components can cause crystal lattice (displacement) damage, ionization effects, and surface effects. Displacement damage has generally been considered to be the basic cause of radiation-induced degradation of the electrical characteristics of semiconductor devices, most commonly as a result of a reduction of minority-carrier lifetime and occasionally by changes in the electrical conductivity. Any radiation which is sufficiently energetic can cause, by either primary or secondary processes, lattice defects which alter the characteristics of the semiconductor. The most pronounced effect of displacement is reduction of current gain. The rate of decrease of current gain is proportional to K/f_{α} where f_{α} is the base transport frequency cutoff and K is a damage constant dependent on type and energy of the bombarding particle, nature of the semiconductor, and type and conductivity of the base material. As a consequence, low-frequency transistors, and silicon transistors degrade faster than comparable germanium transistors.

Little attention has been given to the physical nature and origin of radiation-induced changes in semiconductor surface properties and the associated changes in device characteristics; yet these changes have been frequently observed. The dominant effect is an increase in the reverse current of a p-n junction. However, some observed changes in transistor current gain are of apparent surface origin. The degree to which surface effects are observed is sensitive to types of surface-coating materials, such as silicon greases placed on the semiconductor, to contaminants such as water vapor within the device encapsulation, and to the nature (or absence) of the gas within the encapsulation. Current knowledge does not permit a detailed correlation between radiation effects of surface origin and the physical nature of a semiconductor surface. Reference 1 points out, however, that since this problem was discovered,

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encouraging results are being obtained with new types of transistors, and that by 1966, reliable devices may be expected to be available with damage thresholds of at least 5×10^7 rad (air).

Other than as related to surface effects discussed above, ionization per se should have little effect on the performance of semiconductors in the OMS. Ionization has its primary effect in connection with the intense ionizing fields associated with a nuclear weapons detonation.

Figure 17 shows the degradation of current gain with neutron exposure for typical silicon transistors over a wide range of base cutoff frequency, f_{α} .⁹ It may be seen that the degradation is slight for the high frequency devices ($f_{\alpha} > 100$ mc) up to approximately 10^{12} nvt and would be acceptable in certain types of circuits with equivalent radiation doses greater 10^{13} nvt.

Much radiation effects data is obtained by using a nuclear reactor as a radiation source. The result is radiation damage data that is based on simultaneous gamma and neutron exposure, with the neutron exposure usually predominant. In an attempt to separate the effects due to gammas and neutrons researchers at Atomics International recently "predosed" several groups of

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Figure 17. Degradation of Current Gain With Neutron Exposure

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2N744 transistors to various gamma exposure levels using the Atomics International's Co^{60} irradiator.¹¹ The samples were then irradiated in the SNAP shield test reactor (a low gamma content facility) to various integrated fast neutron doses. The results are shown in Figure 18. The discontinuities in the curves are due to scheduled experimental shutdowns. The results may be summarized by observing that the threshold for significant degradation (0.9 initial gain) for this particular transistor is about 10^6 r for gammas and about 10^{12} nvt (>0.1 Mev). An increase in the gamma dose rate by two orders of magnitude (to approximately 10^8 r) is roughly equivalent to a factor of 10 increase in neutron dose (to 10^{13} nvt).

c. Radiation Hardening

Several examples of the successful uses of semiconductors in "hardened" components may be cited. Reference 9 discusses the methods used to design a radiation resistant transistorized power inverter for the ARIEL I satellite that performed satisfactorily up to 2×10^{12} nvt. Figure 19 (from the same report) shows the performance achieved with a linear amplifier using high-frequency transistors; the unit performed within specification up to approximately 5×10^{13} nvt.

A number of radiation-hardened components or systems have also been developed at Atomics International under the SNAP Development Program. One example is the reactor temperature controller for the SNAP 10A system.

The initial stage of the radiation hardening task consisted of two parts: the selection of the type of circuitry to be used, and the selection of the radiation resistant components. For the temperature controller being considered, either a digital or an analog system could be employed. Since digital circuits can be designed for wide parameter variations, as would be encountered due to radiation degradation, the digital circuit approach was selected. The preliminary selection of components was based on the general principles discussed previously (in this case, specifically, the transistor selected had a high alpha cutoff frequency).

The second stage of the development was the testing of the selected components in the combined environments of reactor radiation, vacuum, and temperature. Transistors, diodes, capacitors, resistors, connectors, and insulation materials were tested. Curves of parameter variation versus dose

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Figure 18. Samples Irradiated to Various Fast Neutron Doses

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Figure 19. Performance Achieved With a Linear Amplifier Using High Frequency Transistors

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were obtained (e.g., transistor beta vs nvt). These curves in turn allowed conservative "worst case" design of the logic circuits required. Various circuits using these selected components were then also radiation tested to determine the adequacy of their design.

The final stage was the testing of complete controllers. Two were tested in a combined temperature-vacuum-radiation environment. The results showed successful performance for radiation doses in excess of 5×10^{13} nvt and 10^7 r, thus demonstrating that by careful component and circuit selection, component testing, and circuit testing, semiconductor electronics can be hardened for radiation environments.

Several other components have also been developed at Atomics International for this same radiation environment, including a temperature switch which utilizes a resistance temperature detector operating in a bridge circuit which controls a bistable magnetic amplifier; a demodulator which compares the phases of the outputs from a variable reluctance position sensor and converts to a dc ramp proportional to angle; and a programmer utilizing magnetic core incremental counting techniques to provide timed output contact closures.

The developmental approach for radiation hardening of all of these components has followed the basic plan outlined above.

4. Conclusions

Assuming that the environmental radiation data used is reasonably accurate, the high-energy proton flux encountered at the altitude specified for the OMS will require that the payload be "hardened" to a level equivalent to approximately 3×10^{12} n/cm² to achieve a 5-yr useful lifetime. Increasing the environmental shield thickness beyond approximately 0.1 in. is ineffective as a means of reducing the proton flux. If such a radiation hardening program is undertaken, then it should also be possible to extend the ionization dose tolerance level to approximately 10^7 rad. Attainment of the latter value depends in part on the degree of success realized by current efforts to solve the surface effects problems encountered with transistors. Considering the level of effort concentrating on this problem and the time-scale for the advanced OMS system, however, a tolerable dose in excess of 10^{12} nvt and 10^7 rad is thought to be a reasonable design objective. The environmental shield required to reduce the dose to these levels would be approximately 0.1 to 0.25 in. thick.

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IV. SNAP 10A SYSTEMS

A. INTRODUCTION

A description and a discussion of the development status of the SNAP 10A flight test system is included in Section IV-B to serve as a current base of reference. The advanced operational SNAP 10A systems which are later used to determine the OMS system performance are based on an extrapolation of the technology developed for the flight test system. In general, these extrapolations are consistent with the advanced SNAP development programs which are currently underway and an availability date of approximately 1968. The design characteristics and performance expected for these systems are discussed in Section IV-C and the safety and reliability aspects are then summarized in Sections IV-D and IV-E.

B. SNAP 10A FLIGHT TEST SYSTEM

1. Description

The SNAP 10A system is the first power plant designed for space employing a nuclear reactor heat source and thermoelectric power conversion equipment. The 10A system work was initiated in early 1961 to provide a space power plant for various potential Air Force requirements. The principal technical objectives established upon the initiation of the program, and still current, are:

- a) Produce a minimum of 500 watts, at 28.5 volts dc, over a period of 1 yr within the space environment.
- b) Provide shielding capable of attenuating radiation emanating from the reactor to approximately 5×10^{12} nvt of fast neutrons and 4×10^7 r of gammas, integrated over a period of 1 yr, at the NPU-Agena vehicle mating plane.
- c) Capable of remote startup in orbit by ground command.
- d) Capable of handling, transport, and launch within acceptable safety criteria and without incurring extensive modifications to normal launch base and checkout procedures.

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- e) After initial startup period, no need for dynamic control or mechanical moving parts throughout its life.
- f) Provide sufficient on-board instrumentation to fully monitor the system during operation in orbit.

Performance characteristics of SNAP 10A are summarized in Table 12 for the present flight configuration. A sketch of the complete unit and the thermodynamic operating cycle is shown in Figure 20. A photograph of a complete SNAP 10A unit is shown in Figure 21.

TABLE 12
SNAP 10A SYSTEM PERFORMANCE

Initial power (watts)	565
Reactor power (kwt)	~38
Overall length (in.)	137
Base diameter (in.)	~50
Total weight (lb)	970
Radiator area (ft ²)	62.5
Maximum coolant temperature (°F)	1010
Design life (yr)	1
Power conversion system	Thermoelectric, "n" and "p" SiGe
Material - Structure	Ti
Liquid metal system	316 and 405 SS
Radiation levels at vehicle mating plane	Neutrons, nvt/yr ~5 x 10 ¹² γ's, R/yr ~10 ⁷

SNAP 10A employs for the heat source a slightly epithermal neutron spectrum U²³⁵-ZrH fueled reactor. This reactor type is common to the SNAP 10A, 2, and 8 programs and has inherent power capabilities from a few kilowatts up into the megawatt range in advanced versions. For SNAP 10A the reactor will operate at approximately 38 kw with a liquid metal coolant outlet temperature of 1000°F.

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Figure 20. SNAP 10A System

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Figure 21. SNAP 10A System

A cutaway view of the reactor is shown in Figure 22. The entire active reactor core is densely packed within a 9-in. diameter by 15-in. long core vessel which in turn is surrounded by a 2-1/4-in.-thick beryllium reflector. Four semicylindrical control drums located in the beryllium reflector region are used to effect startup and short-term reactor temperature stabilization.

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Long-term system control is achieved through the use of burnable poisons and the inherently large negative temperature coefficient (inherent in this reactor type, being a physical characteristic of the fuel and geometry).

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Figure 22. SNAP 10A Reactor

The reactor is cooled with a circulating liquid metal coolant, NaK-78, a eutectic of sodium and potassium. The coolant transports heat to the thermoelectric power conversion system where some of it is converted directly into electrical energy and the balance is radiated to space as waste heat. The NaK is circulated in a closed loop between the reactor and power conversion system by means of a dc Faraday conduction-type pump. The hydraulic pumping power

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results from the cross vector product of a magnetic field, supplied by a permanent Alnico V magnet, and a high electrical current, supplied by a shorted thermocouple integral with the pump. The shorted thermocouple operates over a $300^{\circ} \Delta T$ established by the hot NaK and an integral waste heat rejection radiator, and provides approximately 600 amps at 30 mv. The pump provides 5 watts hydraulic power to the NaK, rejecting about 600 thermal watts to space.

The thermoelectric power conversion system, comprised of "n" and "p" doped SiGe material, is arranged in a pellet configuration thermally coupled to the hot liquid coolant, but electrically isolated from it. Heat flows from the liquid metal through the pellet stack, resulting thereby in a temperature gradient which in turn produces useful electrical power through the Seebeck effect. Waste heat is rejected directly to space from individual radiator platelets bonded to the element stack cold junction.

The mechanical configuration of the power conversion system is conical, deriving its shape from the supporting titanium structure underneath. There are 2880 individual power-producing element stacks in the converter, bonded to and heated by 40 parallel NaK tubes about the cone. The elements are electrically isolated from the hot coolant tubes by means of an Al_2O_3 disc incorporated in each element stack. Total radiator surface area is 62-1/2 sq ft. The power conversion unit produces a minimum of 500 watts at 28.5 volts dc. Electrically, the 2880 individual elements are connected in two parallel chains of 1440 elements each. The two chains are electrically paralleled every two elements. This paralleling substantially reduces the probability of a converter open circuit, for a given number of individual element failures, thereby increasing overall reliability.

The additional principal components which comprise the SNAP 10A system are a LiH radiation shield located immediately under the reactor; the titanium sheet metal support structure; a bellows type expansion compensator to accommodate NaK volume changes and to pressurize the system; and a startup and short-term control system. In addition to these basic system components, the flight test SNAP 10A units incorporate an extensive diagnostic instrumentation system in order to monitor flight behavior.

The overall vehicle power plant control system is diagrammatically illustrated in Figure 23. Upon verification of successful orbit attainment, the

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Figure 23. Flight Test On-Orbit Control Link

power plant start command is given, which through the vehicle command system actuates an "on" and "off" contact sensed by the power plant controller. Two independent actions are initiated:

- a) Squibs retaining the control drums are detonated and the two coarse control drums snap to the full-in position by spring action
- b) The controller inserts the two fine drums at a preset rate of $1/2^\circ$ rotation in 150 sec.

Fine control drum rotation continues until the correct outlet temperature is reached. The controller is left on for approximately 3 days during which time the initial reactivity transients die out and the outlet coolant temperature is stabilized at 1000°F . A separate ground command then removes power from the controller and completely deactivates the control system.

Flight test plans for SNAP 10A call for the launch to take place from Point Arguello, California, over the Pacific Missile Range. The target is a 700 n.mi circular orbit. A drawing of the on-orbit configuration is given in Figure 24. The SNAP 10A power plant is attached to the forward end of the Agena second stage vehicle. While in orbit the vehicle will be maintained in a constant nose up attitude maximizing antenna ground coverage. Throughout operation in orbit, the SNAP 10A power plant and the Agena vehicle remain attached. Initial boost of the total orbit payload is accomplished with an Atlas

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Figure 24. On-Orbit Configuration of SNAP 10A Power Source and Agena Vehicle

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first stage vehicle. During Atlas boost the SNAP 10A package is covered by an aerodynamic nose faring which is subsequently ejected shortly after first stage separation.

2. Development

Principal developmental effort has been concentrated on the thermo-electric converter, pump, expansion compensator, the reactor including fuel and related high-temperature control hardware, and the system instrumentation. In most cases the development span of this hardware encompassed from 1 to 3 yr; the principal exception is the reactor fuel which has been under continuous development since about 1956. A brief summary of the development experience on the significant items of the system is given in Table 13.

Early in the development cycle, work was directed toward establishing feasibility and of performance mapping the various units and their alternates so that the flight system design might proceed. The second phase of development consisted primarily of more detailed test prototype components in simulated space, thermal, and radiation environments. A major fraction of this total effort was devoted to the evaluation of materials and problems of fabricability, high-temperature hard vacuum stability, dissimilar metal interdiffusions, joining, and related areas. The third development phase, and by far the costliest and most time consuming, involved that work necessary to translate the development results into a uniform, reliable product. Considerable effort was expended here in establishing material and product reproducibility, testing techniques to reveal early undesirable process variables, and statistical testing techniques to ensure adequately low failure rates.

Of significance in Table 13 is the reactor experience to date. The SNAP Experimental Reactor (SER) was operated in September 1959 until the successful conclusion of the test program in November 1960. This reactor employed all of the materials features of the present day SNAP reactor unit. The SER operated at peak powers to 50 kw and output temperatures to 1200°F. Total energy output was 225,000 kwh. There was a total of over 6000 test hours of which 3300 were in excess of 900°F. This total energy output approaches that required by the SNAP 10A unit during its 1-yr design life.

The second generation SNAP unit, SNAP Development Reactor (SDR) was operated from April 1961 through December 1962. This unit likewise

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TABLE 13
SUMMARY OF DEVELOPMENT EXPERIENCE

	APPROX. NO. OF UNITS TESTED	TEST OBJECTIVES AND ENVIRON.	TEST EXPERIENCE UNIT HOURS	MAX. TEST DURATION (hr & OR % DESIGN LIFE
REACTOR	1	INTEGRAL DEMONSTR. OF REACTOR FUEL PERF. TO 50 kw, 1200° F FURNISH PRIMARY FLIGHT REACTOR DESIGN DATA	225,000 kwhr (6035 hr)	1000 hr @ 50 kw 1200°F
SDR	1	FINAL INTEGRAL REACTOR FUEL PERF. DEMONSTR	273,000 kwhr (11,300 hr)	1100 hr @ 50 kw 1200°F
DRM-1	1	VIBRATION SIMULATED THERMAL VACUUM	> 2200 hr	90 DAYS

STRUCTURE

THERMOELECTRIC PCS

PUMP

	APPROX. NO. OF UNITS TESTED	TEST OBJECTIVES AND ENVIRON.	TEST EXPERIENCE UNIT HOURS	MAX. TEST DURATION (hr & OR % DESIGN LIFE
	5	STRUCTURAL INTEGRITY DURING LAUNCH S/V ENVIRON. (MECHANICAL & ACCOUSTICAL INDUCED)	~20	QUALIF LEVEL & DURATION ≈15 TIMES REQUIRED LIFE
	~ 600 MODULES	COMPLETE MATERIALS PERF., S/V HARD VACUUM COMPATIBILITY INCLUDING ULTIMATE MARGINS OF SAFETY	500,000 MODULE hr	12,000 hr (≈1.4 % DESIGN LIFE)
	~ 25	COMPLETE MATERIAL & PERF., S/V TEMP etc.	30,000 PUMP hrs	6900 hr (≈80% DESIGN LIFE

EXPANSION COMPENSATOR

EJECTABLE HEAT SHIELD

INSTRUMENTATION

	APPROX. NO. OF UNITS TESTED	TEST OBJECTIVES AND ENVIRON.	TEST EXPERIENCE UNIT HOURS	MAX. TEST DURATION (hr & OR % DESIGN LIFE
	~ 12	COMPLETE MATERIAL & PERF. MAP, S/V, LIFE, etc	7600 COMPENS. hr	2200 (≈25% DESIGN LIFE)
	2	COMPLETE PERF. S/V & SIMULATED THERMAL VACUUM	—	>50 SEQUENTIAL SUCCESSFUL EJECTIONS
	VARIES BETWEEN 4 & ≈10 OF EACH TYPE	COMPLETE PERF. INCL. VACUUM, THERMAL RADIATION, EMI, S/V, & FUNCTIONAL SYSTEM	—	CONTROL SYSTEM ≈ 90 DAYS AND/OR 30 FULL CYCLES) (3 TO 10% DESIGN LIFE) DIAGNOSTIC SYSTEM ≈ 30 DAY (≈9% DESIGN LIFE)

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operated at power levels of over 50 kw and temperatures to 1200°F. Total energy production was 273,000 kwh of which 6900 hr were at temperatures in excess of 900°F. In neither of these reactor test programs was there a failure experienced on the reactor.

Similarly, test experience on other principal system components has been considerable. Over 250,000 thermoelectric module hours of performance data has been accumulated; over 30 pumps of various types have been built and tested with total experience of over 40,000 test hours; and four complete structures have been subjected to full qualification level vibration testing. This level of component experience is designed to reveal component weaknesses prior to the performance of major system tests.

Typical of the development cycle is the thermoelectric power conversion system. A module, 1/120 of the complete converter, is shown in Figure 25. Conceptually this converter operates as shown in Figure 26. Heat flows from the hot liquid metal through the thermoelectric stack to an integral heat rejection radiator. The electrical current flows alternately through "n" and "p" doped SiGe elements in series. To ensure that any open circuit failure in an element stack does not result in a catastrophic open circuit of the entire power conversion unit, the elements are hooked in parallel as shown. With this arrangement hardware developed to overall failure rates of 2 per 1000 or less (per individual stack) results in an overall power conversion system reliability approaching 1.

Over 1300 modules have been built (to date) representing about three distinct generations of hardware. A summary of the performance test data is contained in Table 14 and Figure 27.

This thermal-electrical performance has been consistent throughout the total production history. All testing to date confirms the approximately 4% loss in power for 10,000 hr operating time, which results primarily from a precipitation of arsenic dopant in the "n" material. There has never been any test data which would indicate a lifetime limit of the power conversion unit to 1 yr. The principal benefit resulting from the three generations of production manifests itself in reduced failure rates arising from increased stability

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Figure 25. SNAP 10A Thermoelectric Module

Figure 26. Conceptual Operating Mode for the SNAP 10A Thermoelectric Power Conversion System

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of the metallurgical joints. For reliability purposes, all testing indicates failure can simply be classified as an electrical open or nonopen of an individual thermoelectric element stack. With this definition, failure rate experience is shown chronologically in Figure 28. The latest flight type hardware produced for 10A, as established by testing on over 780 element stacks, has an overall failure rate in service of less than 1 part/1000. This corresponds to a complete converter reliability of greater than 0.996 at 90% confidence.

TABLE 14
SNAP 10A TE CONVERTER PERFORMANCE

Thermoelectric material	"n" and "p" SiGe
Figure of merit (Z) (effective)	$0.58 \times 10^{-2} \text{ C}^{-1}$
Average NaK temperature (°F)	936
Average radiator temperature (°F)	606
Materials efficiency (%)	9.4
Converter resistance (ohms)	1.6
Number of couples, total	1440
in series	720
Power output	565
Open circuit voltage	61
Load voltage	28.5
Current (amps)	20
Heat radiated (kw)	34
Overall efficiency (%)	1.65
Device efficiency (%)	~73
Element failure rate	$\leq 1 \text{ ppt}$
Expected total converter reliability	≥ 0.996 at 90% confidence

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MATERIAL
OPERATING TEMP °F
MATERIAL EFFICIENCY "Z" C⁻¹
STRUCTURAL CAPABILITY

SiGe
1000
 0.64×10^{-3}
> 300% ATLAS-
AGENA ENVIR

NUMBER OF MODULES BUILT > 1300
NUMBER OF ELEMENTS TESTED > 4500
MAX TEST DURATION-hrs > 14,000
TOTAL TEST EXPERIENCE,
ELEMENT-hrs ~ 6,500,000
DEMONSTRATED RELIABILITY 0.996 AT 90%

ELEMENTS/MOD = 24
MOD/LEG = 3
LEG/CONV = 40
TOTAL ELEMENTS = 2880

<u>CONVERTER PERFORMANCE</u>	<u>SPECIFICATION</u>	<u>EXP</u>
VOLTS/COUPLE mv	86.5	85.8
RADIATOR EMISSIVITY	0.85	0.89
ELEMENT $\Delta T^{\circ}F$	280	310
TOTAL CONVERTER POWER, watts	585	565
WEIGHT, lbs		146
LIFE, yrs	1	> 1

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Figure 27. SNAP 10A Thermoelectric Converter Summary

Figure 28. SNAP 10A Thermoelectric Element Failure Rate Experience

3. System Qualification

The SNAP 10A program is now in the final stages of flight qualification with principal testing effort directed toward integrated systems. This system effort is in turn divided into two principal areas:

- a) Exploring the technical interface with the launch vehicle, and
- b) Final performance demonstration and qualification of the power plant.

The former tests are directed almost solely to evaluating the electrical interactions between the power plant and launch vehicle, since the vehicle provides the power consumption and data transmission links. The power plant performance tests are scheduled for three flight prototypes, two of which will be nonnuclear in nature, and one of which will be a nuclear test. One of the nonnuclear tests has been successfully completed. The remaining nonnuclear test and nuclear test are in various intermediate points in the test program.

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The power-plant vehicle interface is being flight verified through combined testing of the power plant and the Agena. This testing is being conducted in two stages. Late in 1962 a developmental version of the SNAP 10A unit, employing all of the electrical circuits and instrumentation, but without liquid-metal components, was delivered to the vehicle contractor for performance testing with a similar electrical mockup of the vehicle. A photograph of these units in the test laboratory is shown in Figure 29. Testing was conducted over a time span of approximately 6 months, and technical details relating to power consumption, gross functional behavior, noise, other interferences, accuracy, etc., explored. While detailed technical problems were identified and solutions obtained, the satisfactory functional behavior of the vehicle and power plant combination was established.

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Figure 29. Combined SNAP 10A-Agena Electrical Compatibility Test

A second electrical mockup unit was delivered to the vehicle contractor in November of 1963. This unit employs flight qualified components and is similar in detail to the actual flight units. The Agena mockup was likewise updated to the flight configuration. The test program for these later units is

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directed toward a final verification of the complete performance behavior to be expected in orbit. In addition to exploring in considerable detail isolated electrical, control, power, pyrotechnic, and data transmission circuits, the test schedule calls for repetitive complete simulated in-orbit system testing and EMI mapping of these units.

In addition to the principal effort on electrical compatibility between the two systems, work has been performed to evaluate both the structural and the thermal interface. The major aspects of these latter tests have been successfully completed.

This thorough ground evaluation and demonstration of the interaction between the power plant and vehicle is expected to contribute significantly to the reliability of a successful flight including data return therefrom.

The first full SNAP 10A performance qualification unit (designated FSM-1) was completed and placed on performance test in May of 1963. This system was complete and virtually the same as the subsequent flight hardware with the following exceptions:

- a) The unit employed early generation flight components and thus differed slightly in engineering or manufacturing detail.
- b) The components had not themselves been fully flight qualified.
- c) The thermal performance test was nonnuclear in nature and thus the reactor fuel was replaced by electrical cartridge heaters.

The test sequence was designed to simulate as closely as possible the actual factory-to-flight delivery scheme proposed for the flight units. A pictorial comparison of these sequences is shown in Figure 30. The FSM-1 testing phase included shock and vibration to full qualification levels followed by simulated in-orbit prestartup, startup, and long term operation. Specific test objectives included:

- a) Final determination of structural interactions induced by simulated launch vibration loads
- b) A final engineering evaluation of significant design features prior to freezing the flight unit design; i.e., NaK flow distributions, system heat balance, efficiencies, temperature distributions, etc.

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FSM - I
GROUND PROTOTYPE
TEST SEQUENCE

ASSEMBLY
CHECKOUT

ACCEPTANCE LEVEL
SHOCK & VIBRATION

QUAL. LEVEL
SHOCK & VIBRATION

NaK LOADING

IN-ORBIT
PRE-STARTUP TEST

TRANSIENT
STARTUP TEST

FULL POWER
ENDURANCE TEST

REFERENCE FACTORY TO
FLIGHT SEQUENCE

ASSEMBLY
CHECKOUT

ACCEPTANCE LEVEL
SHOCK & VIBRATION

NUCLEAR
FUEL LOADING

NaK LOADING

ELECTRICAL HEAT
HOT ACCEPTANCE
TESTS

VEHICLE LAUNCH

IN ORBIT PRE-
STARTUP, STARTUP,
& POWER OPERATION

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Figure 30. Ground System Performance Test Program

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- c) Evaluation of the magnitude of the in-orbit prestartup NaK freezing probability
- d) Transient behavior of the unit during startup by means of a programmed analog simulation of the electrical core heaters
- e) Endurance testing to 90 days to verify stability of the overall unit and to confirm the applicability of previous component test data
- f) To experimentally measure the thermal interactions with the launch vehicle by means of a simulated forward section of the Agena.

A photograph of the test system is shown in Figure 31.

The shock and vibration testing was carried out in two phases, a preliminary 5-min sweep on each of three mutually perpendicular axes, followed by full qualification levels on each of three mutually perpendicular axes with 25-min sweep time. All testing was sinusoidal. Vibration input levels vary from 1/2-in. double amplitude at 5 cps to 7-1/2 g at 3000 cps. This mechanical testing phase indicated no unanticipated characteristics. Typical problems encountered were loosening of connectors which had been improperly lock wired, fraying of high-temperature wire terminations when improperly installed, motion of a few small brackets, etc. The full qualification levels were sustained without structural damage, electrical malfunctions or failures of the NaK system. As expected, the critical frequency for the structure was in the frequency bandwidth of 12 to 40 cps laterally, and 40 to 225 cps longitudinally. The most critical load condition of the structure occurred at the first fundamental lateral frequency of 15 cps. At this frequency the input was 0.45 g, a factor of 2 greater than specification. Figure 32 indicates typical response g levels at the first lateral natural frequency. Figure 33 indicates the response of some of the principal components over the entire test frequency range. In no case were component responses on the system greater than the individual test specifications.

Following shock and vibration testing the unit was subjected to a simulated space environment through the preorbit, startup, and operation phases. All testing was conducted inside of an 8-ft-diameter by 18-ft-high vessel, operated in the 1- μ pressure range. The vessel walls were water cooled.

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Figure 31. First SNAP 10A Nonnuclear
Flight Qualification System

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Figure 32. Typical Component Vibration Response Levels,
Lateral Mode

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Figure 33. Typical Component Frequency Bandwidth Response

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Within the vacuum tank were located radiant heaters to simulate solar energy input. A photograph of the test facility is shown in Figure 34. Initial testing was directed toward providing data relative to NaK temperatures in orbit prior to startup of the reactor. This is a critical problem since large heat losses from the system when in the shade portion of the orbit could result in freezing of the liquid-metal circuit. To prevent these heat losses from the radiator, the system incorporates a heat shield, having a high solar absorptivity to thermal emissivity ratio, which is ejected after the reactor produces sensible heat. The testing of the NPU in simulated constant sun-constant shade and sun-shade orbits confirmed the design of the shield. The shield provides a net positive heat input to the system over an orbit and effectively dampens major oscillation in the NaK system temperatures.

Transient testing was conducted to verify the temperature and flow characteristics of the unit during startup. The electrical heaters, used to simulate the reactor, were preprogrammed from analog computer studies and the unit was operated in an open loop fashion. The thermal responses of the system were well behaved and displayed no erratic characteristics. A comparison of the experimental core outlet temperatures with those predicted from the analytical model, as well as the flow characteristics, are shown in Figure 35.

Following transient testing the system was placed on endurance test in early October 1963. It was run unattended continuously for 90 days. At the completion of the test the system was shut down and a thorough post-mortem examination of the various components conducted. Typical initial performance parameters of the unit, as compared to original design estimates, is shown in Table 15.

The initial power output of the system was limited to 427 watts due to the ground test environment. The geometrical relationship of the system to the chamber walls, and the fact that the chamber walls were significantly warmer than the mean effective space temperature, reduced power output approximately 50 watts. These test results, plus known improvements in the flight converter, including use of higher emissivity coatings, improved all welded electrical converter connections, and improved thermoelectric elements, indicate the flight units will produce approximately 565 watts at design temperature and environment conditions. The performance of several of the important system parameters over the 90-day test, normalized to a constant average temperature and

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Figure 34. Nonnuclear Test Facility

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corrected to remove perturbations to the data, caused by test rig inadequacies and electrical connectors are shown in Figure 36.

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Figure 35. System Temperature Transient

TABLE 15

FSM-1 NONNUCLEAR SYSTEM TEST SUMMARY

	Design	Initial Test
NaK outlet temperature (°F)	990	990
NaK average temperature (°F)	935	922
NaK flow (gpm)	13.1	10.8
Heat to converter (kw)	32	35.2
Converter output power (watts)	455	427
Converter ΔT (°F)	279	278
Structure temperature (°F)	750	722
Instrument compartment temperature (°F)	140 max.	126
System pressure (psi)	10	9
Overall converter efficiency (%)	1.43	1.21

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Figure 36. Ground Test System Performance History

Other components of the FSM-1 behaved as expected based on individual component tests. Most technical difficulty during the test was experienced with various elements of the flight diagnostic instrumentation. While none of these instrumentation failures would have compromised the successful startup and operation of the unit in orbit, some would have interfered with complete data transmission from all channels to ground. Typical instrumentation problems included inadequate accuracy resulting from improper bonding and thermal shielding of the sensors, failure of one set of position transducers on the expansion compensator, nonactuation of some limit switches due to thermally induced bracketry shifts, and temperature drifting in a few of the high-temperature RTDs.

As an integral part of the endurance test, heat balance measurements were made to determine the amount of heat which would be conducted or radiated from the high-temperature reactor plant to the Agena vehicle. For this purpose the Agena mating ring and an Agena mockup gas calorimeter were incorporated into the test facility, the emissivity and physical properties of which simulated

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the Agena interface. At a mean effective Agena temperature of 75.1°F, approximately 70 watts total thermal energy is transferred across the interface. This confirmed almost exactly the original design calculations.

Upon successful attainment of all the principal test objectives, the unit was shut down for an extensive post-mortem evaluation on January 14, 1964. A photograph of the system at the conclusion of the test is shown in Figure 37. Examination of the unit indicated no visual degradation with the exception of slight discoloration of some titanium surfaces resulting from the residual oxygen in the tank and possibly very slight oxidation of the exposed copper materials.

Figure 37. Appearance of the
Nonnuclear Ground System
at Test Conclusion

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Final SNAP 10A system performance is to be evaluated by means of an additional nonnuclear ground test and a nuclear test. Both of these systems will be put through a representative factor-to-flight sequence, including shock and vibration, NaK loading, critical experiments as appropriate, and simulated

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in-orbit prestartup operation. These units will then be taken to full power in a simulated space environment with the intention of running for a year without shutdown.

The reference test sequence for the nuclear test system is the same as that shown previously (Figure 30) except for the addition of fuel loading and critical experiment prior to NaK loading. Testing through shock and vibration has been completed successfully as have the nuclear fuel loading and initial critical experiments. The photograph of the power plant in the critical facility ready to receive fuel is shown in Figure 38. This is the same facility in which fuel loading, critical experiments, and reactivity shimming for the flight units will be conducted. Following fuel loading, the unit is charged with liquid metal and the system sealed. After liquid-metal loading, system operation using electrical heat will be conducted to verify the hot-performance behavior in a simulated space environments. Following minor adjustments as might be dictated by the hot tests, the system will be removed from the facility and transported to a separate power test vault for nuclear transient and long-term operation.

Figure 38. Nuclear Ground Test System in Acceptance Test Facility Ready to Receive Nuclear Fuel

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Final flight qualification is to be performed on a nonnuclear system test basis. Like the other system tests, the complete spectrum of environment which the flight systems will eventually see will be sequentially imposed upon the unit. A photograph of the unit in the final assembly area, showing its status as of March, is shown in Figure 39.

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Figure 39. SNAP 10A Final Assembly Area

C. ADVANCED OPERATIONAL SYSTEMS

1. General Nature of Improvements

The major improvements that would be incorporated in SNAP 10A-type systems intended for operational use with the advanced OMS in the 1968-71 time period would be a reduction in the unshielded system weight and improved shielding. The weight reductions would be achieved by raising the maximum operating temperature from the current value of 1000 to approximately 1300°F, by eliminating most of the diagnostic instrumentation, and through miscellaneous, detailed mechanical and metallurgical improvements.

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The performance improvements achievable by means of higher-temperature operation can simply be illustrated by the relationship of temperature to thermoelectric conversion efficiency and radiator heat rejection capability. Neglecting secondary effects, the efficiency of a thermoelectric device is proportional to:

$$Z\Delta T,$$

and the heat rejection capability is proportional to fourth power of the radiator temperature (T_r^4). At a given system Carnot efficiency, these equations can be simply arranged to show that the electrical power per unit area is theoretically proportional to:

$$P_{elec} \approx ZT^5.$$

Actually, consideration of other secondary effects including slight Carnot changes will show that the actual performance gained is not quite this large, but more nearly approximated by:

$$P_{elec} \approx ZT^4.$$

This latter expression has been verified by operating SNAP 10A modules at elevated temperatures. The experimental data, as measured in power output per couple is plotted in Figure 40 for temperatures up to 1300°F. For a power system in which the heat source thermal power is relatively independent of weight or size, as typified by a nuclear reactor, the power per unit area vs temperature relationship is also approximately true for other yardsticks of performance; i.e., power per unit weight, dollar per watt electrical, etc. Thus, it is obvious that the direction of future development should be toward increased temperature capability and that this will result in superior nuclear-thermoelectric power plant performance.

The improved shielding would be obtained through geometrical modifications of the shield and (for the present application) the use of an extendible boom system to increase the separation distance between the reactor and the payload.

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Using this approach, it is possible to obtain a net decrease in the total shielded power system specific weight (lb/watt) and reduce the payload radiation dose to a level which is small compared to that due to environmental (Van Allen) radiation.

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Figure 40. High Temperature Capability of SNAP 10A Thermoelectric Module

2. System Description

A preliminary design of a typical 1-kwe advanced operational system is shown on Figure 41. The major changes relative to the current SNAP 10A system are that the design shown uses the SNAP 2 reactor, an advanced thermoelectric converter design, and two electromagnetic pumps which obtain their current from separate thermoelectric modules. Other changes include an

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Figure 41. 1-kw SNAP 10A
for Weather Satellite

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improved titanium structure design and elimination of the instrument compartment, since the startup instrumentation and control equipment would be located in the OMS payload structure.

It should be mentioned that advanced reactor concepts are currently being studied which would have smaller shieldable envelopes than the SNAP 2 reactor in addition to other improvements. Details of these advanced designs are not yet sufficiently known to include in the present report, however, other than to indicate the approximate improvement in shield weight expected.

The SNAP 2 reactor is quite similar to the present SNAP 10A reactor, but uses more advanced components, materials, and design features to permit operation over a wider range of power level and operating temperatures.^{12,13} An exploded view of the SNAP 2 reactor, reflector, and shield assemblies is shown on Figure 42.

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Figure 42. SNAP 2 Reactor Assembly

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Modifications required to adapt the SNAP 2 reactor to the advanced operational thermoelectric systems considered for the OMS would include the addition of two NaK lines (one inlet and one outlet), modification of the hydrogen content and samarium loading in the fuel elements (depending on the temperature and power level) and the use of a more advanced hydrogen barrier material which has been developed and demonstrated in the SNAP 8 program.

A developmental SNAP 2 reactor system (S2DRM-1) is currently being assembled for a series of nonnuclear tests, using an electrically heated NaK loop operating in a high-vacuum test chamber. Photographs of this system in various stages of assembly are shown on Figure 43. The cast LiH shield (which represents a significant improvement over the cold-pressed SNAP 10A shield) is shown on Figure 44. Operation of this developmental system is expected to verify the capability of the mechanical and electrical components to operate satisfactorily at temperatures corresponding to approximately 120 kwf and with maximum NaK temperatures of 1300°F.

The advanced thermoelectric converter which is currently being developed uses silicon-germanium elements and aluminum radiator fins, similar to the SNAP 10A converter but is modified for higher operating temperatures and employs round NaK tubes. The complete converter consists of 36 legs connected in four parallel circuits. A copper saddle is placed on the tube at each element location and the stack is bonded to the saddle and tube. The configuration of the module assembly is shown in Figure 45. Hastelloy N is the current choice for the NaK tube material.

Two electromagnetic pumps would be used, one in each return line within the converter assembly. These pumps, consisting of a throat, permanent magnet, and low-resistance power leads, obtain their current from sections of the system converter. A sketch of a separate source pump and bus network is shown on Figure 46. Thermoelectric couples in those portions of the converter used for pumping are connected in parallel rather than in series to provide the necessary current.

3. Reactor Shielding

The discussion of Van Allen radiation and radiation damage presented in Section III shows that the OMS electronic payload would have to be hardened

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Figure 43. S2DRM-1

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Figure 44. Lithium Hydride Shield

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Figure 45. Reference Design of Module

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Figure 46. Separate Source Pump - Bus Network

to an equivalent dose of approximately 3.5×10^{12} nvt to survive the high energy proton environment encountered over a 5-yr period at the desired altitude (1500 n.mi). Assuming that this is done, then it is also considered reasonable to expect that the ionization dose tolerance of the payload can be extended to approximately 10^7 rad for the operational time period of interest.

In most shielding problems where weight is important and multiple radiation sources are involved, an arrangement which results in comparable dose rates from each source is usually near-optimum for a given radiation tolerance level. In the present case, where a fourfold increase in the thickness and weight of the aluminum environmental shield reduces the high-energy proton flux less than one-third, it is apparent that there is little opportunity for shield weight optimization with respect to displacement type radiation damage to semiconductors. Thus, to put reactor powered SNAP systems on a par with alternate power sources, the neutron flux level should be set sufficiently low that it would not materially change the scope of the radiation hardening task. Based on this

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criteria, a neutron dose rate of 10^{11} nvt per year was selected as a conservative design objective for this study. At this level, the equivalent displacement damage of the neutrons would be approximately one-seventh that of the high-energy protons.

Unlike the proton dose, the ionization dose due to high-energy electrons can be effectively reduced at the expense of environmental shield thickness and weight. The gamma ionization dose from the reactor can also be adjusted, using tungsten discs, if necessary, to augment the attenuation provided by the LiH neutron shield. Thus, it is theoretically possible to obtain an optimum (minimum weight) distribution of gamma and electron shield weight for a given total dose. Assuming that this optimum would occur with equal contributions from both sources, a tentative gamma dose limit of 10^6 R per year was selected for parametric shield calculations.

The thickness of the LiH neutron shield required to attenuate the neutron dose to 10^{11} nvt per year is shown on Figure 47 as a function of reactor thermal power and the distance separating the reactor from the payload. (A variation of LiH thickness of 6 in. changes the fast neutron dose rate by a factor of 10.) The tungsten shield thickness required to decrease the gamma dose to 10^6 R per year when the neutron shield is designed for 10^{11} nvt per year is shown in Figure 48 for the same parameters. It will be noted that no tungsten gamma shielding is required if the separation distance is greater than 20 to 30 ft, depending on the power level.

For a given dose rate and separation distance, the shield weight depends primarily on the size of the reactor envelope which must be shielded. The shielded envelope of the SNAP 2 reactor is a cylinder 18 in. in diameter and approximately 15 in. high. Although the actual reflector is only about 14 in. in diameter, the larger envelope diameter is required to shield neutrons scattered from the NaK pipes, drum bearings, and other hardware located outside of the reflector. In more advanced reactor concepts currently being investigated it is expected that most of these items can be eliminated or relocated and that the top of the reflector may be shaped to reduce the shielded envelope to a cylinder approximately 12.5 in. in diameter and 16.5 in. high. The total neutron shield weights required for these two cases is shown on Figures 49 and 50 as a function of separation distance and LiH thickness. By combining the data shown in

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Figure 47. Neutron Shield Thickness

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Figure 48. Gamma Shield Thickness
vs Reactor-Payload Separation
Distance

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Figure 49.
Fast Neutron Shield Weight as a
Function of Separation Distance
and Shield Length for
18.0 Diameter by 15.0 Reactor
(Total Shield Wt. = $W_{LiH} + W_{casing}$;
LiH Density = 0.02637 lb/in.³;
 $W_{casing} = 0.25 W_{LiH}$)

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Figure 50.
Fast Neutron Shield Weight as a
Function of Separation Distance
and Shield Length for
12.50 Diameter by 16.5 Reactor
(Total Shield Wt. = $W_{LiH} + W_{casing}$;
LiH Density = 0.02637 lb/in.³;
 $W_{casing} = 0.25 W_{LiH}$)

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Figures 47 through 50, the total neutron and gamma shield weights required to limit the yearly dose to 10^{11} nvt and 10^6 R is shown on Figure 51 as a function of reactor power and separation distance for both the SNAP 2 and the advanced reactor envelopes.

Figure 51. Total Shield Weight
Neutrons and Gammas

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Based on the above shielding data and conceptual design studies of various types of extendible booms, it has been determined that a minimum weight system is obtained with a reactor payload separation distance of about 40 ft (see Section V). The total shield weights that result for the SNAP 2 and the advanced reactor for this separation distance are shown on Figure 52 vs reactor power level. For these neutron shield weights the gamma dose rates are less than 10^6 R per year and, converting to rads, vary with the reactor power as shown on Figure 53.

4. System Performance

As a part of the advanced thermoelectric development program, extensive preliminary design and system optimization studies have been conducted to guide the development of the components and to define the overall system performance potential. Based on these studies and preliminary test data, the design and performance characteristics of systems with initial net power levels ranging from 300 to 3000 watts have been determined. The results are summarized on Table 16; the characteristics of the current SNAP 10A system are included for comparison.

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Figure 52. Total Shield Weight

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Figure 53. Gamma Ray Dose vs Reactor Power

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TABLE 16
SNAP 10A DESIGN AND PERFORMANCE CHARACTERISTICS

Design Characteristics	SNAP 10A Flight Test	Advanced Operational Systems				
Design lifetime (yr)	1			3-5		
Maximum NaK temperature (°F)	1000			1300		
Minimum NaK temperature (°F)	900			1100		
Payload diameter (ft)	5			5		
Separation distance (ft)	9.3			40		
Payload dose rate (nvt/yr) (rad/yr)	$\sim 5 \times 10^{12}$ 1.3×10^7	10^{11} 0.7×10^5	10^{11} 1.4×10^5	10^{11} 2.5×10^5	10^{11} 3.5×10^5	10^{11} 5.3×10^5
Minimum net efficiency (%)	1.5			2.5		
Minimum net power (watt)	500	250	500	1000	1500	2500
Initial net power (watt)	565	300	600	1200	1800	3000
Nominal reactor power [kw(th)]	38	10	20	40	60	100
Gross radiator area (ft ²)	62.5	15	30	60	90	150
Weight Summary (lb)						
Reactor reflector	275	295	295	295	295	295
Control and instrumentation	117	45	45	45	45	45
Power conversion system	213	82	123	201	287	490
Structure	83	15	25	60	120	230
Heat shield	33	9	17	29	41	63
Instr. comp't. and thermal barrier	32	5	7	11	15	20
Startup batteries	22	10	15	20	25	35
Subtotal-Unshielded Weight	775	461	527	661	828	1178
Shield (SNAP 2 Reactor)	218	250	270	290	300	320
Total Shielded Weight	993	711	797	951	1128	1498
Shield (Advanced Reactor)		140	150	165	170	180
Total Shielded Weight		601	677	826	998	1358

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The following items should be noted in reviewing the advanced performance data shown on Table 16:

- a) All of the advanced systems have been oversized by 20% to allow for thermoelectric element degradation and reactor temperature variations expected over a 3- to 5-yr period. Thus, the initial, net power is 20% greater than the minimum net power in each case.
- b) The thermoelectric converter/radiator area required for the advanced 500-watt system is less than half that required for the current 500-watt SNAP 10A system. This dramatic reduction in area results in a significant weight savings in the power conversion system, structure, heat shield, startup batteries, etc.
- c) The weight of the reactor-reflector assembly is based on the current SNAP 2 reactor design. With the more advanced reactor concepts now being investigated, a reduction in reactor weight may be possible.
- d) The weight of the cone-shaped structure is based on an improved skin and stringer design with a criteria that the first resonant vibration mode must be greater than 15 cycles/sec.
- e) The instrument compartment would not be required in the advanced operational systems for the present application, since the startup instrumentation would be located with the electronic payload. The weights indicated allow for thermal insulation to limit heat losses only.
- f) The startup battery weight is sufficient to provide NaK circulation for approximately 24 hr prior to startup in orbit. Since the actual time required for launch, orbit verification and startup is less than 12 hr, the weight allowed provides a safety margin of at least two.

The unshielded and shielded weights of these advanced operational systems is plotted as a function of the design, net power on Figure 54 and the approximate size of the various systems is shown in Figure 55. The shaded area on

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Figure 54 bounded by the shielded system weight curves for the SNAP 2 and advanced reactors defines the most probable range of the weight of advanced operational SNAP 10A systems which could be available for the OMS in the 1968 to 1970 time period.

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Figure 54. Advanced Operational SNAP 10A Systems Weight vs Net Power

POWER LEVEL (WATTS E)	RADIATOR AREA (ft ²)	A in	B (ft)	C (ft)	D (ft)	E (ft)	F (ft)	G (ft)	H (ft)	J (ft)	K(ft)	L(ft)
250	15	22.25	2.16					2.7				
500	30	24.0		3.9					3.20			
1000	60	26.5			7.0					4.0		
1500	90	27.5				9.16					6.6	
2500	150	28.5					12.3					4.68

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Figure 55. Power vs Size Comparison Chart

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The long term performance of the SNAP 2 reactor using passive control has also been investigated. Assuming that the automatic, active control system would be used for the first few days of orbital operation (as with the current SNAP 10A reactor), the long-term reactivity changes due to hydrogen loss, fuel burnup and fission product poisons would be compensated by the inherent negative temperature coefficient of the reactor and by gradual burnup of the samarium prepoison incorporated in the fuel elements. The optimum amount of samarium required to minimize the reactivity (and thus temperature) variation with time has been determined for an initial NaK outlet temperature of 1300°F and a range of power levels. The resulting NaK outlet temperature vs time is shown on Figure 56 for 25, 50, and 75 kw.

The variation of the converter electrical power output with maximum NaK temperature is shown in Figure 57. By combining the data shown in Figures 56 and 57, the variation in net electrical power level can be determined. Although the reactor power levels analyzed are about 25% higher than would be required for the 500-watt, 1-kwe, and 1.5-kwe systems, respectively, this degree of conservatism is assumed to compensate for uncertainties in hydrogen leakage, fission product poisons, and samarium loading variations.

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Figure 56. Reactor Outlet Temperature vs Operating Time

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Figure 57. Percent of Rated Power vs Maximum NaK Coolant Temperature

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The resulting electrical power variation for these three systems is shown on Figure 58. The curves shown include a degradation allowance for the thermoelectric modules of 5% the first year and 1.25% per year thereafter.

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Figure 58. Electrical Power Output vs Time

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It will be noted that the net electrical power of the 1.5-kwe system does not drop to 1.5 kwe for approximately 3 yr and thereafter continues to generate an appreciable percentage of the rated power. The net electrical output of the 1-kwe system does not drop below its rated power for more than 4 yr and the 500-watt system is still producing more than 500 watts after 5 yr. (The excess power would be shunted to wrap around heaters installed on the NaK lines.) Thus, it is concluded that passive control is feasible for the systems considered in this study for at least 3 yr, up to a power level at around 1.5 kwe and 4 to 5 yr for lower powered systems. To extend the range of passive control to higher powers or longer periods of time would require either some means for augmenting the temperature coefficient or operation at reduced temperatures.

D. SAFETY

1. General

Operational procedures for reactor-powered SNAP systems involve no nuclear operation (other than zero-power acceptance tests) until a satisfactory launch has been obtained. The principal radiological safety consideration during shipment and launch is therefore the prevention of radiation exposure from inadvertent reactor criticality. In general the solution to this problem has been approached in two ways:

- a) Provision of a combination of safeguard devices and operational procedures to prevent inadvertent criticality, and
- b) The maintenance of controlled access areas to reduce the possibility of personnel exposure if an accident should occur.

Typical procedures and safety devices which are employed to accomplish safety objectives are described below.

2. Acceptance Test to Launch Countdown

This phase is essentially the ground handling phase of the program. It begins with acceptance testing of the SNAP system at the contractor's facility, and includes transport to the launch range, operations in range facilities, transport to the launch pad and early countdown activities.

During this phase, safety is assured by positive prevention of inadvertent criticality. Primary assurance of subcriticality is obtained by the use

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of two specially designed safeguards, the Shipping Sleeve and the Void Filler Block Assembly, which maintain the reactor subcritical for all credible handling and transportation operations and accidents.

Acceptance testing of the SNAP system will be accomplished in Atomics International's nuclear facilities. Tests will be confined to "zero power" runs and with the anticipated time between the acceptance tests and shipment (on the order of 2 weeks), the dose rate from residual fission products will have decayed to a few mr/hr adjacent to the reactor.

Prior to shipment, the reflector control assembly will be removed from the reactor and a specially designed shipping sleeve installed around the core. The annular structure of the sleeve consists of a layer of void cells, surrounded by a layer of boron powder, a layer of energy absorbing structure and a stainless steel container. Figure 59 shows a SNAP 10A Shipping Sleeve. The sleeve consists of two halves each of which replaces one-half of the beryllium reflector. The sleeve is designed to withstand a 15-ft drop onto concrete without significant distortion, or without breaking loose from the core. Experiments have verified that this type of sleeve will maintain a bare SNAP core at least \$1 subcritical with total water flooding and submersion.

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Figure 59. SNAP 10A Reactor Shipping Sleeve

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The radiation shield, power unit and reflector control subassembly will be shipped in individual shipping containers. The power unit shipping container is a double walled vessel with the inner container shock mounted, Figure 60.

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Figure 60. Container - NPU, Storage, Shipping, Reuseable

Using the shipping sleeves and shipping container, the SNAP system can safely be shipped by truck, rail, ship or plane. When the systems reach the launch range they will be assembled into the flight ready configuration and final nonnuclear checkout and testing will be accomplished. When the reflector control subassembly is reinstalled on the core, the void filler blocks, assemblies bolted flush against the main Be reflector to minimize outside neutron reflection, will be in place. The blocks are constructed of stainless steel, partially filled with natural boron powder. An integral feature of the blocks is the portion which fits into the control drum voids, mechanically restraining the drums in the "out" position. With the void filler blocks in place, the reactor is maintained subcritical even with infinite water reflection. In addition to the void filler blocks, key operated locking arms are incorporated to prevent

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control drum rotation. Void filler blocks, control drums locking pins and control drum locking arms are shown in Figure 61. With these safety devices in place, an NPU can safely be handled, assembled to the satellite, shipped to the launch pad, and installed on the launch vehicle.

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Figure 61. SNAP 10A Reflector Safety Lockout Devices

3. Final Countdown Through Orbit Attainment

From final countdown through launch, the primary safety emphasis shifts from positive prevention of criticality to prevention of radiation exposure to personnel in the event that an accident should occur. Safety in this phase is achieved through a combined approach which utilizes personnel access and missile operational restrictions in addition to the inherent and design safeguards of the SNAP systems. Before final countdown, the reactor would be atop the missile with the void filler blocks and control drum lockout arms still attached. At the initiation of final countdown, all nonessential personnel will be cleared from the pad area. One of the final steps in the countdown will be the removal of the void filler blocks and control-drum lockout arms. After

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these devices are removed from the reactor, the gantry will be removed and the missile remotely fueled and launched.

With the removal of the external safety devices, the reactor is in the flight ready condition. In this condition there are two ways in which inadvertent criticality could occur: accidental insertion of the control drums, or accidental immersion of the reactor in a reflecting media such as water or missile fuel.

There are three separate safeguards to prevent inadvertent startup. First, a coded signal is required for startup. The probability that a false signal would accomplish reactor startup is very small (less than one in 10^{10}). Secondly, an electric interlock can be inserted to separate the missile ground link from the startup programmer so long as the missile nose cone shroud is in place. (The interlock could be based on some other appropriate mechanical situation if desired.) This interlock will protect against inadvertent startup signals and against spurious signals generated in the missile circuitry. Finally, the band holding the reflector control assembly halves to the core can be severed by ground command, thus providing a reactor scram. Rotation of the reactor control drums by external forces that might be encountered in handling accidents or aborts is prevented by the control drum lockout pins and by the reentry band (band holding reflectors in place). A control drum lockout pin is located in each control drum. These pins restrain the drums in their least reactive position and are of sufficient strength that external forces would tear the control drums from the reactor before the pins would break and allow control drum rotation. The pins are withdrawn by squibs at the time of reactor startup. The reentry band provides a backup safeguard to prevent forceful insertions of the control drums. Forces much smaller than those required to shear the lockout pins would break the reentry band, allowing the reflector assembly halves to rotate free of the core.

With the reactor in the flight-ready condition, criticality could result from accidental immersion of the reactor in a reflecting media. A number of safeguards can be used to assure personnel protection during these phases of operation when accidental immersion of the reactor could occur. Inherent safeguards in the SNAP design limit the magnitude and effect of a potential nuclear accident. The compact design and the hydrided fuel elements used in SNAP design lead to a relatively small limit on the integrated energy (and

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consequent fission product generations) which the core can generate during an excursion. An integrated energy release on the order of 70 Mw-sec is sufficient to increase the pressure of hydrogen in the fuel elements to the point where explosive disassembly of SNAP cores occur, thus terminating any excursion. In practice, low rates of reactivity insertion would probably lead to much smaller excursions. Furthermore, since the only credible method of obtaining accidental criticality is by external reflection, there will always be a significant attenuation of the prompt radiation associated with accidental criticality by this environment.

Figure 62 shows the direct doses that would be received downwind of a 70-Mw-sec excursion of a SNAP reactor. The prompt neutron and gamma doses shown include attenuation in the water which must be around the core to

Figure 62.
External Dose from a 70-Mw-sec
Excursion as a Function
of Downwind Distance

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cause criticality. Figure 63 shows the downwind thyroid dose from a 70-Mw-sec excursion. It can be seen that exposed persons more than 2000 or 3000 ft from the excursion would not be seriously hazarded. Normal missile safety requires greater exclusion distances than those. In the launch area, the only place an excursion could credibly occur is on the launch pad itself (in missile fuel or pad deluge water). The normal exclusion distance required for missile safety will protect personnel from any potential nuclear hazard in this area.

The primary emphasis on safety after launch is to assure that the location of the reactor is controlled such that an accidental criticality could occur only in controlled or remote, unpopulated areas (e.g., the ocean). All missiles are equipped with highly reliable and redundant destruct systems which can be activated by the Missile Flight Safety Officer. This destruct system, in combination with a redundant tracking system, assures that the missile and all fragments remain within previously defined impact limit lines. These impact limit lines can be defined in such a manner as to preclude accidental reactor criticality in an area where people would be unduly hazarded. It should be borne in mind that even in the case of downrange missile aborts, a reflecting media is required for reactor criticality. Dryland impacts would not result in criticality.

The evaluation of the actual potential hazard from downrange abort of a missile carrying a SNAP system requires a given launch site, missile and launch trajectory. One such analysis has already been accomplished for a southerly launch on the Pacific Missile Range. This analysis¹⁴ shows that no significant nuclear hazard could occur. It is not anticipated that different launch site and trajectories (within reason) would cause significant nuclear safety problems.

Should the orbit achieved be unsatisfactory for the objectives of the flight, the reactor could reenter the earth's atmosphere and impact at some point on the earth's surface. Since orbital reentry is an essentially random process, one of the primary nuclear safeguards, locational control, would be lost. However, a compensating factor is present, the reentry environment. The high aerodynamic forces and thermal fluxes inherent in orbital reentry can be utilized to render the reactor safe. Since the reactor is launched in the shutdown conditions the only problem is to assure that criticality does not

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Figure 63. Thyroid Dose from a 70-Mw-Excursion as a Function of Downwind Distance

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occur at or subsequent to reentry. (The fission product inventory from acceptance testing and the radioactivity from the uranium fuel itself are not sufficient to cause a significant hazard.) The first safeguard against reentry criticality is the reentry band shown in Figure 22. This band holds the SNAP reflector-control subassembly in place against spring tension. This band can be severed by ground command allowing the reflectors to be ejected from the core. If a ground signal is not sent or fails, it has been demonstrated by actual flight test that reentry heating will sever the band early in the reentry phase, and expose the core to aerodynamic heating.

The SNAP reactor core vessels are constructed of thin stainless steel. Once exposed to aerodynamic heating they will rapidly melt, scattering the fuel elements in the atmosphere. In addition to the inherent susceptibility of the core vessels to aerodynamic heating, design feature such as extended lips are included to enhance or speed melting. Once the geometry of the core has been destroyed there is no possibility of subsequent criticality and, therefore, no nuclear hazard. Additionally, if the fuel elements are exposed at sufficiently high altitude it is possible that the elements themselves will melt and be dispersed in micro sized particles.

4. Post-Orbital Operation

At the end of the useful life of the SNAP system, automatic shutdown of the reactor is provided through the employment of failure detection devices (e.g., low voltage and/or low NaK temperature). The outputs of these sensors will effect the severance of the reflector retention band allowing the reflector to eject from the core vessel. The subsequent radioactive decay of the fission products during the remainder of the orbital lifetime will result in a rapidly diminishing inventory, eventually falling below that radioactivity inherently associated with the uranium fuel. Failure to achieve reactor shutdown at the end of useful life could result, theoretically, in continued operation for some time, but at an ever decreasing power level as the uranium is used up and hydrogen leakage from the fuel continues. In this case, the fission product inventory also decreases during the orbital lifetime, but at a considerably slower rate than with complete shutdown. Upon reentry, core disassembly and dispersion of the fuel elements by atmospheric heating would minimize any potential post-reentry hazard. Previous analyses of these potential

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hazards have indicated that the probability of injury to the world's populace due to possible residual radioactivity is very small, even if complete reentry burnup does not occur.

E. RELIABILITY

Since the attainment of a long satellite lifetime is basic to the economic operation of the advanced OMS system, it is imperative that the source of electrical power have a high reliability. This is the main reason why the SNAP 10A-type system with passive thermoelectric power conversion equipment was selected for this study.

For most applications it would be impractical to use two or more complete, separate SNAP systems in an effort to achieve high reliability through redundancy at the system level. Thus, the approach that has been pursued since the inception of the SNAP program has been to attain high reliability through inherent design simplicity, by development of hardware adequate to the task, by establishing and enforcing rigorous quality control procedures, and through extensive qualification testing of the developed product at the component, subsystem, and system levels under conditions equal to or more severe than those expected in actual operation. In addition, redundancy techniques are incorporated in those portions of the system which may be subject to statistical failure, such as the thermoelectric converter.

An indication of the scope of the SNAP 10A development, production, and qualification program may be obtained from the discussion included in Section IV-B. In most cases, experience has shown that the reliability of the first few components produced outside of a laboratory environment is relatively low, but increases rapidly as more and more units are manufactured, tested, debugged and ultimately qualified for flight systems. The thermoelectric converter offers one of the more striking examples of this reliability growth pattern, as shown on Figure 64.

Current estimates for the reliability of the first SNAP 10A flight system are approximately 0.8 for launch through startup and 1 yr operation. A significant portion of the 0.2 failure probability is associated with the launch through startup phase for this first system. The components and functions involved for this 3 day period are not subject to wearout and do not depend on the system

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operating temperature or power level. Therefore, from the experience that will be obtained with the highly instrumented flight test system and with subsequent design improvements and simplifications that could be made during the next 3 to 5 yr, it is felt that the failure probability for this initial phase of operation can be reduced to 2 or 3% in the advanced, operational systems which would be available in the late sixties.

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Figure 64. Reliability Growth of SNAP 10A Converter

The reliability of a SNAP 10A-type system after it has been started up in space depends mainly on the fuel-moderator elements in the reactor core, on the pumps which circulate the NaK, and on the thermoelectric elements which convert the heat to electricity. The basic technology required to assure satisfactory long-term operation of the fuel-moderator elements at the power levels and temperatures considered in this study has already been established in the SNAP reactor development program. For example, the SNAP 8 Experiment Reactor (S8ER) has just completed a successful 60-day endurance run at 450 kw (thermal) with maximum NaK temperatures of 1300°F. This is equivalent to more than 3.5 yr of operation at the reactor power level needed for a

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500-watt SNAP 10A system with respect to total kwh per unit core volume. Equally important is the fact that the advanced hydrogen barrier material which has been developed for SNAP 8 has been tested at temperatures substantially higher than 1300°F without noticeable degradation.

Thus, the mechanisms affecting fuel performance can be predicted with reasonable accuracy and accounted for in the design of the reactor by adjusting such parameters as the initial hydrogen density, fuel-rod diameter, burnable poison loading and reflector thickness. The production and quality control techniques which have been developed and enforced for the SNAP 10A program are such that the probability of getting a "bad" element in a core is much less than 1%. Also, it is important to realize that even if a major flaw somehow developed in the hydrogen barrier of a fuel element, the results would not be catastrophic. For example, if a clad area as large as 1 cm^2 were completely exposed, the increased hydrogen leakage from an average element would only reduce the reactivity of the reactor approximately 7¢, the NaK temperature 20°F and the electrical power 4%. Even if such a gross malfunction to one out of every 37 elements in the core were considered credible and if a gradual loss of power of this magnitude can be considered a failure, then the reliability of the core would still be greater than 97%.

The single EM NaK pump with an integral thermoelectric current source used on the current SNAP 10A system would be replaced by two EM pumps with separate current sources in the advanced, operational systems. The current source consists of a section of the main thermoelectric converter wired in parallel to obtain the high current and low voltage required. The pump consists of a section of the return NaK pipe formed into a rectangular throat, permanent magnets mounted adjacent to the throat section, and copper bus bars brazed on the sides of the throat so that the current can be applied at right angles to the magnetic field. Due to the inherent simplicity of this design, the pumping section (magnet and throat) has a reliability greater than 0.99. The minimum number of separate thermoelectric elements connected in parallel is approximately 15 per pump for a 1-kwe system. Even if the thermoelectric failure rate were as high as 10 per thousand it may be seen that the total reliability for this pump arrangement can be made to approach unity.

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The electrical portion of the thermoelectric converter is a series-parallel arrangement of thermoelectric elements. The number of elements which must be placed in series to obtain the required output voltage and the power level determine the number of parallel current paths. In addition, connections can be made to parallel every couple (two elements in series, one "P" and one "N" type) with each couple in the adjacent parallel paths, as shown on the sketch below:

For a given thermoelectric element failure rate, the reliability increases rapidly with the number of parallel current paths. Thus, for a given required voltage output, large systems with high voltage outputs per thermoelectric element would be more reliable than smaller systems with low voltage per element. For example, the 500-watt SNAP 10A flight-test system will be connected as in the above sketch, has two parallel paths to supply the power at 28.5 volts, and operates at a maximum temperature of 1000°F. The advanced, operational 1-kwe system described in Section IV-C would contain approximately the same number of thermoelectric elements as the 500-watt flight-test system, but would operate at a maximum temperature of 1300°F. The higher operating temperature of the advanced system results in a voltage output per element approximately twice as high as that obtained at 1000°F. For the same number of thermoelectric elements and final output voltage, therefore, the number of parallel paths can be doubled. A comparison of the reliability of the complete thermoelectric converter for these two systems is shown on Figure 65 as a function of the element failure rate. (The number of thermoelectric element failures expected per thousand elements for the flight test modules for launch, 90-day operation and 1-yr operation is 0.7, 1.5, and 4.5, respectively.) It may be seen that the total number of thermoelectric

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Figure 65. Thermoelectric Converter Reliability vs Number of Element Failures per Thousand

element failures for the 1-kwe advanced systems could be approximately 10 times greater than that of the flight test system for the same reliability. One of the objectives of the advanced thermoelectric development program currently being pursued is to produce improved thermoelectric elements capable of 1300°F operation with failure rates as low as those which have been obtained with the flight test system. Depending on the degree of success actually realized in this program, and the power level selected for the advanced OMS system, a converter reliability greater than 95% for 3- to 5-yr operation should be achievable.

It should be mentioned that an advanced thermoelectric element is also being investigated which would permit an additional doubling of the number of parallel paths without significant increase in the converter weight. By cutting the cylindrical thermoelectric stack in half lengthwise and combining an N-half with a P-half, the number of separate thermoelectric element couples and parallel paths can be effectively doubled. If this approach can be successfully developed, the reliability curve for the 1-kwe advanced system shown in

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Figure 65 would apply to an advanced 500-watt system and the number of thermoelectric element failures allowable for the 1-kwe system would be limited only by power degradation considerations.

In summary, an overall reliability of approximately 0.9 for 3 yr operation is considered to be a reasonable and realistic objective for the advanced operational SNAP 10A systems intended for use in the late 60's and early 70's. This goal is thought to be commensurate with that of other OMS subsystems and better than that achievable with most alternate power generating systems.

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V. SNAP 10A POWERED OMS CONCEPTUAL DESIGN

A. INTRODUCTION

A conceptual design has been evolved for a SNAP 10A powered Operational Meteorological Satellite for the purpose of assessing the performance of such a spacecraft over a range of design variables. The essential features of this concept and its parametric performance characteristics are described in this section.

B. DESCRIPTION

1. General

An illustration of the general configuration which is considered to be the most suitable for the advanced OMS using a SNAP 10A power system is shown on the frontispiece and on Figure 66. This configuration was chosen after considering several alternate possibilities, including that of using the Agena as the basic spacecraft structure as well as the final stage boost vehicle. This approach was discarded because of the difficulty of locating the payload in a position where the sensors would be earth-oriented and still obtain a configuration that would lend itself to passive (gravity gradient) stabilization.

Perhaps the most unique aspect of the arrangement selected is the long, extendible boom separating the SNAP system from the payload compartment. This feature permits a major reduction in the payload radiation levels due to neutrons and gamma rays emanating from the reactor, for a given shield weight. It also provides a restoring torque (due to the gravity gradient effect) which has been calculated to be sufficient to maintain the spacecraft oriented normal to the earth's surface within the accuracy required for meteorological observations.

The densely packed, highly redundant electronic payload is contained in the pillbox shaped structure which also serves as a Van Allen radiation shield and passive thermal control surface. The meteorological sensors would be installed at the base of the cylinder which would be oriented towards the earth.

Thus, the major subsystems for the advanced OMS are the electronic payload, the nuclear power system, the payload structure and environmental shield, and the gravity gradient stabilization system. The electronic payload and the SNAP 10A power system have been described in detail in Sections III and IV of this report. The environmental shield thickness requirements have also

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Figure 66. 1000-Watt SNAP 10A Powered Weather Satellite

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been discussed for various ionization dose rates in Section III. A brief discussion of the gravity-gradient stabilization system and the factors affecting the weight of the payload structure and environmental shield follow:

2. Gravity Gradient Stabilization System

Two basic OMS requirements suggest the selection of a stabilization or attitude control system based on the gravity gradient principle: earth orientation of the primary meteorological sensors and a long useful satellite lifetime.

The principle of a gravity gradient system is that an orbiting, elongated body will tend to align itself so that its major axis is pointing towards the center of the earth. If the satellite is designed in the form of two masses separated by a boom (i. e., a dumbbell configuration), then the mass on the end of the dumbbell closest to the earth will receive a greater gravitational pull because of its relative position. This principle has already been demonstrated by Satellite 22A and development of gravity gradient system components is currently being pursued for communications satellites.

A discussion of both the theory and the practical aspects of applying the gravity gradient concept to the OMS, achieving proper damping, yaw control, and the approximate magnitude of the disturbing and restoring torques is presented in Reference 1. Figure 67 shows the alignment accuracy estimated about the pitch and roll axes as a function of boom length for the expected range of disturbing torques. The case shown is for a 2700-lb OMS with an 800-lb SNAP 10A power system located on the "top" end of the boom. (The restoring torque and accuracy would be slightly higher with an equal mass distribution.) The attitude error is directly proportional to the magnitude of the disturbing torques and decreases in proportion to the extension distance squared.

The magnitude of disturbing torques due to phenomena such as the earth's asymmetry, orbit eccentricity, solar pressure, etc., at the operating altitude of interest are estimated to be in the range of 10^{-5} to 10^{-4} ft-lb. The maximum disturbance due to the interaction of the earth's magnetic field with the pump magnets, control components and electrical current loops in a 1 kwe SNAP 10A system has been calculated to be approximately 4×10^{-4} ft-lb if no compensation is provided, other than to orient the pump magnets in opposite directions. By using small compensating magnets and/or retracing the current loops around the top and bottom of the converter, it is expected that these effects can be

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Figure 67. Satellite Attitude Error

effectively reduced to approximately 10^{-4} ft-lb. Assuming that the disturbances due to payload components are such that the total disturbing torque is less than 10^{-3} ft-lb, an attitude error of 1 or 2° appears feasible with a boom length of 30 to 50 ft. This error is considered acceptable for the OMS.

In addition to the basic requirement for a spacecraft configuration consisting of two masses separated by a rigid boom, means must be provided for damping librations and stabilizing the system about the yaw axis. Typical damping systems being developed which employ spring, magnetic, or fluid dampers and additional rods arranged along the pitch and roll axis are described in Reference 1. The weight of these systems ranges from 45 to 60 lb.

The sequence for deploying the spacecraft in orbit would be as follows:

- a) The extendible boom and auxiliary rods and dampers would be retracted during launch, with the SNAP 10A power system fastened securely to the top of the payload compartment.

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- b) Upon achieving the designated orbit altitude and direction, the final stage of the boost vehicle (i. e., Agena) would be oriented in a vertical direction and the satellite would then be separated. (Some short-term means of reorienting the satellite in the proper attitude for "capture" after separation may be required.)
- c) The boom, dampers, and rods would be extended, after which the satellite is expected to oscillate for several orbits. Startup of the SNAP 10A power supply would be initiated after the boom is extended to avoid exposing the payload to excessive radiation.

With the above sequence, the loads imposed on the extendible boom are extremely small, and the boom system can therefore be relatively light. Table 17 summarizes the design features and weight estimates for a telescoping boom concept based on the use of either titanium or Loc-Alloy, a lightweight beryllium-aluminum alloy with high thermal conductivity, strength, and modulus of elasticity. A sketch of the boom portion of this system is shown on Figure 68. Although the design shown is based on the use of electric motors, cables, and pulleys, several alternate methods could be used to deploy the boom for approximately the same weight. The total deflection for this design (based on a minimum wall thickness of 0.090 in.) would be less than 1 in. for the miniscule loads expected.

The weight estimated for the complete gravity gradient stabilization system including the primary boom, sensors, dampers, and electrical cable is shown on Figure 69 as a function of separation distance. By combining these weights with those obtained for the reactor shielding in Section IV (Figure 52), an optimum separation distance of approximately 40 to 50 ft results, as shown on Figure 70. At 40 ft the total weight of the gravity gradient system is estimated to be approximately 150 lb.

3. Payload Structure and Environmental Shield Weight

The weight of the environmental shield, naturally, depends on the material used, the required thickness, and the shield area. Although graphite has a slightly higher theoretical weight efficiency, aluminum was selected for the present study so that it can serve a dual role as the primary payload structure and shield. The payload configuration is chosen to be a right, circular cylinder 5 ft in diameter, consistent with the choice of the Atlas-Agena launch vehicle.

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TABLE 17
EXTENDIBLE BOOM SYSTEM

Item	Weight (lb)	
	Loc-Alloy	Titanium
Actuator, support, cables	9	11
Boom (5 segments)	32	70
Rollers, locks, pulleys, etc.	7	14
Batteries and hardware	12	12
Total weight	60	107
Design Characteristics (Loc-Alloy)		
Boom		
Length compressed	8 ft	
Length extended	38 ft	
Segments	5	
Base diameter	6 in.	
Wall thickness	0.09 in.	
Deployment speed	4 fpm	
Action type	Ball on groove	
Actuator		
Speed	10 rpm	
Voltage	28 v	
Current	1.5 amp hr	
Battery type	Silver-Zinc (0.5 amp)	

The only remaining variables are thus the tolerable dose rate (which establishes the thickness) and the volume occupied by the electronic payload equipment.

The density of current spacecraft electronic components (including the TV camera systems used on NIMBUS) is usually in the range of 80 to 120 lb/ft³. However, when these components are installed in a typical system, the average packing density is often reduced to the range of 15 to 30 lb/ft³ to accommodate irregular shapes, harness runs, etc., and in some cases, because the space is available. With payload volume (and shield weight) at a premium, however, a packing density of approximately 50 lb/ft³ is considered to be a reasonable objective, particularly considering the advances being made in the development of

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Figure 68. Extension Boom-Gravity Gradient Orientation-SNAP Powered

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Figure 69. Gravity-Gradient Stability and Control System Weight

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Figure 70. Optimum Reactor-Payload Separation Distance

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compact microelectronic circuits. Even with current systems, this would still provide an average void fraction of approximately 40 to 60% for harness runs and to account for the irregular shapes of the components.

Figure 71 shows how the shield weight varies with packing density and aluminum thickness for a 1500-lb electronic payload. The actual aluminum shield thickness required is believed to be in the range of 0.1 to 0.25 in. (see Section III). Figure 72 shows the environmental shield weight as a function of electric payload weight for these two thickness limits, based on a packing density of 50 lb/ft³. In addition to the thick, outer shell, a relatively light internal support structure would be needed. However, since the 0.25-in.-thick shell is believed to be highly conservative (i. e., does not account for the decay of fission electrons to the 1970 time period), the higher weight will be used with the assumption that it represents a reasonable estimate for the combined structure and shield weight. Making an additional allowance for the electrical harness the total weight estimated for the payload environmental shield, structure, and electrical harness is shown on Figure 73.

C. PERFORMANCE

The performance characteristics of a range of SNAP 10A powered OMS systems are determined by superimposing the parametric subsystem performance data on the total payload capabilities of various launch systems which are assumed to be available in the time period of interest. By this process, the net electronic payload which can be launched to the desired orbit and altitude can be determined for several boosters and compared with that required to satisfy the mission requirements. Since the weight of the nuclear power system depends on the power level, which is in turn a function of the net electronic payload, the available electrical power is retained as variable.

The following booster and upper stage vehicle combinations were considered in this study:

- 1) Thrust-Augmented-Thor (TAT)/Agena D
- 2) Atlas/Agena D
- 3) Floxed Atlas/Agena D
- 4) Atlas/Centaur

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Figure 71. Environmental Radiation Shield Weight
vs Packing Density

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Figure 72. Environmental Shield Weight vs Electronic Payload

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Figure 73. Environmental Shield, Internal Structure, and Electrical Harness Weight Estimate

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The total spacecraft weight which these launch systems can place in a circular polar orbit is shown in Figure 74 as a function of orbit altitude and on Table 18 for the desired altitude of 1500 n.mi. It will be noted that the TAT-Agena is incapable of boosting even the basic redundant electronic payload required for the OMS to the desired altitude range. Thus, the choice of boosters narrows down to the Atlas or a floxed Atlas combined with the Agena or Centaur final stages.

The distribution of the total allowable spacecraft weight between the four major subsystems in a SNAP 10A powered OMS launched by the Atlas/Agena combination is shown on Figure 75 as a function of the available electrical power. Similar weight distributions for satellites launched into the required orbit by the floxed Atlas/Agena and Atlas/Centaur launch systems are shown on Figures 76 and 77. It will be noted that approximately one-half of the total satellite weight is available for electronic payload using the "standard" Atlas/Agena at a power level of 1 kwe. With the floxed Atlas/Agena and the Atlas/Centaur launch systems the percentage of the total spacecraft weight available for electronic payload increases to about 60 and 70%, respectively, at the same power level.

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Figure 74. Launch Systems Capability (Polar Orbit)

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TABLE 18

LAUNCH VEHICLE PAYLOAD CAPABILITY

Circular Polar Orbit 1500-n.mi Altitude

Launch System	Total Spacecraft Weight (lb)
TAT/Agena D	700
Atlas/Agena D	2700
Floxed Atlas/Agena D	3660
Atlas/Centaur	5400

The results shown in Figures 75 through 77 include the shielded weights of the advanced, operational SNAP 10A systems which were based on the use of the SNAP 2 reactor. Similar satellite weight distributions have been calculated based on the lower SNAP 10A power system weights that could be obtained by using a more advanced reactor concept (see Section IV, Figure 53). The affects of the variation in power system weight are indicated by the width of the shaded bands in Figure 78, which shows the net electronic payload available for all three booster systems as a function of electrical power level.

Finally, by combining the weight and power requirements for the redundant OMS payload (Figure 10, Section III) with the capability shown in Figure 78, the overall system performance shown on Figure 79 is obtained.

Inspection of Figure 79 reveals the following important results:

- 1) Using the "standard" Atlas/Agena launch vehicle to place the SNAP 10A powered OMS in the specified orbit and altitude, electronic payloads weighing from 4 to 4-1/2 times that of a single, nonredundant payload and consuming as much as 1300 watts of power appear possible.
- 2) Using the floxed Atlas/Agena or the Atlas/Centaur launch systems, a substantial payload and power margin would be available to accommodate additional experiments, operational equipment or further redundancy.

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Figure 75. SNAP 10A Powered Weather Satellite Weight Distribution vs Power

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Figure 76. SNAP 10A Powered Weather Satellite Weight Distribution vs Power

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Figure 77. SNAP 10A Powered Weather Satellite Weight Distribution vs Power

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Figure 78. SNAP 10A Powered Weather Satellite Electronic Payload vs Power

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Figure 79. SNAP 10A Powered Weather Satellite System Performance

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