280
11-30-66

NAA-SR-11934

COPY

MASTER

FINAL SNAPSHOT
PERFORMANCE REPORT

AEC Research and Development Report

RELEASED FOR ANNOUNCEMENT
IN NUCLEAR SCIENCE ABSTRACTS

ATOMICS INTERNATIONAL

A DIVISION OF NORTH AMERICAN AVIATION, INC.

DISCLAIMER

This report was prepared as an account of work sponsored by an agency of the United States Government. Neither the United States Government nor any agency Thereof, nor any of their employees, makes any warranty, express or implied, or assumes any legal liability or responsibility for the accuracy, completeness, or usefulness of any information, apparatus, product, or process disclosed, or represents that its use would not infringe privately owned rights. Reference herein to any specific commercial product, process, or service by trade name, trademark, manufacturer, or otherwise does not necessarily constitute or imply its endorsement, recommendation, or favoring by the United States Government or any agency thereof. The views and opinions of authors expressed herein do not necessarily state or reflect those of the United States Government or any agency thereof.

DISCLAIMER

Portions of this document may be illegible in electronic image products. Images are produced from the best available original document.

LEGAL NOTICE

This report was prepared as an account of Government sponsored work. Neither the United States, nor the Commission, nor any person acting on behalf of the Commission:

A. Makes any warranty or representation, express or implied, with respect to the accuracy, completeness, or usefulness of the information contained in this report, or that the use of any information, apparatus, method, or process disclosed in this report may not infringe privately owned rights; or

B. Assumes any liabilities with respect to the use of, or for damages resulting from the use of information, apparatus, method, or process disclosed in this report.

As used in the above, "person acting on behalf of the Commission" includes any employee or contractor of the Commission, or employee of such contractor, to the extent that such employee or contractor of the Commission, or employee of such contractor prepares, disseminates, or provides access to, any information pursuant to his employment or contract with the Commission, or his employment with such contractor.

Printed in USA. Price \$6.00. Available from the Clearinghouse for Federal
Scientific and Technical Information, National Bureau of Standards,
U. S. Department of Commerce, Springfield, Virginia

6.00 1.50

NAA-SR-11934

REACTOR TECHNOLOGY

289 PAGES

FINAL SNAPSHOT
PERFORMANCE REPORT

BY

G. E. BERG

RELEASED FOR ANNOUNCEMENT
IN NUCLEAR SCIENCE ABSTRACTS

LEGAL NOTICE

This report was prepared as an account of Government sponsored work. Neither the United States, nor the Commission, nor any person acting on behalf of the Commission:

A. Makes any warranty or representation, expressed or implied, with respect to the accuracy, completeness, or usefulness of the information contained in this report, or that the use of any information, apparatus, method, or process disclosed in this report may not infringe privately owned rights, or

B. Assumes any liabilities with respect to the use of, or for damages resulting from the use of any information, apparatus, method, or process disclosed in this report.

As used in the above, "person acting on behalf of the Commission" includes any employee or contractor of the Commission, or employee of such contractor, to the extent that such employee or contractor of the Commission, or employee of such contractor prepares, disseminates, or provides access to, any information pursuant to his employment or contract with the Commission, or his employment with such contractor.

ATOMICS INTERNATIONAL

A DIVISION OF NORTH AMERICAN AVIATION, INC.

CONTRACT: AT(11-1)-GEN-8
ISSUED: AUGUST 15, 1966

DISTRIBUTION

This report has been distributed according to the category "Reactor Technology" as given in the Standard Distribution for Unclassified Scientific and Technical Reports, TID-4500.

ACKNOWLEDGMENTS

SNAP 10A was designed and developed by Atomics International, a division of North American Aviation, Inc., under contract to the U. S. Atomic Energy Commission. Thermoelectric power converter module development and fabrication were subcontracted to the Radio Corporation of America. The Agena spacecraft was designed and built by Lockheed Missiles and Space Company who also provided vehicle integration services, under contract to the AEC and the U. S. Air Force. The USAF provided launch and orbital tracking facilities and support for SNAPSHOT.

Principal contributors to this report were R. B. Paulson, J. P. Hawley, P. M. Magee, G. H. Parker, E. A. Robb, A. K. Smith, and R. J. Thomson, Jr. Significant contributions were also made by J. R. Crosby, K. A. Davis, G. S. Drucker, R. A. Johnson, R. P. Johnston, S. R. Rocklin, and W. D. Whitaker.

CONTENTS

	Page
Abstract	14
I. Introduction	17
A. Scope of Report	17
B. SNAP 10A History and System Description	18
C. Development Experience	25
D. SNAPSHOT Objectives and Chronology	28
1. Preflight	28
2. Flight	29
II. Summary	33
III. System Performance Analyses	41
A. Launch	41
1. Structural Response	42
2. Individual Component Response	52
3. Conclusions and Recommendations	53
B. Prestartup Performance	54
1. Predictions and Prior Testing	55
2. SNAP 10A Orbital Orientation	56
3. Heat Shield	58
4. NaK Temperatures	58
5. Recommendations	61
C. Startup Period	62
1. Approach to Criticality	64
2. Initial Power Transient	64
3. Ramp to Full Power	66
4. Conclusions and Recommendations	66
D. Active Control Period	67
E. Passive Control Period	73
F. Shutdown	78
IV. Subsystem and Component Performance	87
A. Reactor Control System	87

CONTENTS

	Page
B. Reactor	92
1. Startup Performance	93
2. Active Control Period	99
3. Passive Control Period	104
4. Conclusions and Recommendations	107
C. Radiation Shield	109
1. Shield Performance	112
2. Neutron Fluxes	112
3. Gamma Fluxes	121
4. Payload Radiation Effects	121
D. Power Conversion	126
1. Performance	127
2. Isolation Resistance Degradation	135
3. Voltage Profile Analysis	146
E. Thermoelectric Pump	149
1. Low Temperature Operation	150
2. Startup	152
3. Full Power Performance	152
F. Expansion Compensator Performance	155
1. Launch	157
2. Orbital Prestartup	159
3. Startup Transient	159
4. Full Power	160
5. ECU Limit Switches	160
6. Conclusions and Recommendations	160
G. Heat Shield	161
1. Thermal Environment	163
2. Differences Between Space and Simulated Environment	171
3. Heat Shield Ejection	172
4. Recommendations	172
H. Instrumentation	174
1. Temperature Instruments	178
2. Electrical Instruments	180

CONTENTS

	Page
3. Position Indication Instruments	183
4. Radiation Monitoring Instruments	184
5. Other Instruments	185
6. Conclusions and Recommendations	186
I. Malfunction and Failure System	189
1. Malfunction Circuits	191
2. Failure Circuits	194
3. Low Voltage Signal	194
4. Failure Signal	194
5. Automatic Shutdown	195
6. Conclusions and Recommendations	195
J. Agena and Secondary Payloads	197
1. Electrical Subsystem	198
2. Telemetry Subsystem	210
3. Orbital Attitude Control System	214
4. Command and Tracking System	216
5. Thermal Control System	216
6. Secondary Payloads	216
7. Conclusions and Recommendations	220
K. Data System Accuracy	221
L. Thermal Control Coatings	228
1. Al-93 on Thermoelectric Converter Fins	231
2. Low Emittance Gold on Molybdenum Foil	235
3. Instrument Compartment Coatings	235
4. Conclusions and Recommendations	236
M. Thermal Performance and Sun-Shade Effects	237
1. Thermal Performance	238
2. Sun/Shade Effects	241
3. Conclusions and Recommendations	246
N. Flow Distribution	247
1. Preflight Analysis and Test Results	250
2. Flight Test Analysis and Results	252
3. Conclusions and Recommendations	254

CONTENTS

	Page
O. Space Effects.	254
1. Major Effects	255
2. Minor Effects	255
3. Negligible Effects.	255
4. Magnetic Torques.	256
V. Accomplishments and Recommendations.	259
A. Degree of Accomplishments of Flight Test Objectives.	259
1. Ground Handling and Launch Operations	259
2. Launch Survival	261
3. Remote Startup in Orbit by Ground Command	261
4. Unattended Operation After Initial Active Control Period	261
5. Production of 500 w for 1-yr in Space Environment	262
6. Demonstration of Simplicity and Reliability Potential of a Reactor-Thermoelectric Space Power Unit.	262
7. Measurement of Radiation Shielding Effectiveness in Space	262
8. Obtain In-Space Operating Data and Experience.	264
9. Nuclear Safety	264
10. Interface Engineering Verification.	265
B. Recommendations.	266
VI. Appendices	271
A. Converter Voltage Profile Analysis.	271
1. Variability of Thermoelectric Module Properties.	271
2. Average NaK Temperature Differences Among Tubes	272
3. Variations in Effective Heat Sink Temperature	274
4. Converter Electrical Malfunctions.	275
5. Flight Test Results.	276
B. Unexpected Expansion Compensator Deflection Under Zero-g Conditions	279
1. Gas Trapped in System	279
2. Alternate Explanations	281
C. Voltage Regulator.	283
VII. References	287

TABLES

	Page
1. SNAP 10A Weight Summary	22
2. Comparison of SNAP 10A Ground Test Systems Performance	27
3. SNAP 10A Qualification Systems Initial Performance	29
4. Final SNAPSHOT Orbital Parameters	30
5. SNAP 10A Performance in Space	34
6. Significant Rigid Body Accelerations	44
7. Low Frequency Vibration	47
8. SNAP 10A/Agna Structural Interface Lateral Shears and Bending Moments	49
9. Random Vibration Levels at Liftoff	52
10. Converter NaK Tube Outlet Temperature Variation During Prestartup	61
11. Mean NaK Temperature Variation During Prestartup	62
12. SNAP 10A System Performance During Active Control Period	69
13. Control Drum Steps During Active Control Period	71
14. SNAP 10A Performance During Passive Control Period for Sun Conditions	76
15. SNAP 10A Degradation Summary	78
16. Flight System Contact Point Material Combinations	91
17. Low Temperature Nuclear and Thermal Parameters for FS-4 Startup Transient Analysis	96
18. Comparison of FS-4 Startup Results with Predictions	100
19. Reactivity Losses During First 72 Hours of Active Control Period .	103
20. SNAP 10A Reactivity Losses During Passive Control Period (Normalized to 40 kw and 950°F Average Coolant Temperature)	106
21. Converter Isolation Resistance Extrapolations	144
22. Vehicle Power Margin	145
23. Differences Between Measured and Predicted Voltage Profiles . . .	148
24. Low Temperature Pump Performance	151
25. SNAP 10A FS-4 Expansion Compensator Deflection Summary	162
26. Inoperative Diagnostic Instruments	176
27. Agna Electrical System Performance	209
28. Estimated Accuracy of Some Important System Parameters	223

TABLES

		Page
29.	Relative Errors in SNAPSHOT Telemetry Data	226
30.	Optical Properties of the FS-4 System	230
31.	Converter Thermal Emittance History	233
32.	Converter Solar Absorptance History	234
33.	Detailed System Heat Balance (kw)	240
34.	Response of Several Important Parameters to Change from Sun to Shade Environment	243
35.	Comparison of Measured and Predicted Flow Distribution	254
36.	Best Estimate of the Reliability of SNAP 10A	263
37.	Percent Accomplishment of Engineering Mission Objectives	266
38.	Variation in Converter Double Leg Pair Internal Resistance	272
39.	Effect of Gas on Expansion Compensator Deflection	281
40.	Power Dissipated by Voltage Regulator	283
41.	Calculation of Maximum Voltage Regulator Electronics Temperature	285

FIGURES

		Page
	SNAPSHOT, 13:24 hr PST, April 3, 1965	15
1.	SNAP 10A Performance Characteristics	20
2.	SNAP 10A Flight Test System	21
3.	Spacecraft Configuration	23
4.	SNAP 10A/Agena/Atlas Vehicle Configuration	24
5.	Rigid Body Longitudinal Acceleration During Launch	44
6.	SNAP 10A/Agena/Atlas Third Lateral Elastic Mode of Vibration at Liftoff	45
7.	Comparison of Flight Test and Laboratory Acceleration Data	48
8.	Elastic Responses at Agena -- Second Ignition	50
9.	SNAP 10A/Agena First Longitudinal Elastic Mode of Vibration at Agena Engine Second Cutoff	51
10.	Random Vibration Response Level at SNAP 10A Upper Torque Box During Liftoff (Y-Y Axis)	53
11.	Initial SNAPSHOT Orbital Orientation, April 3, 1965	57
12.	Reactor Outlet Tube Temperature During Pre-Startup Period	59
13.	Converter Outlet Tube Temperature During Pre-Startup Period	60
14.	Comparison of Reactor Power Transients During SNAP 10A Startup Tests	65
15.	Comparison of Reactor Outlet Temperature Response During SNAP 10A Startup Tests	65
16.	Comparison of NaK Flow Response During SNAP 10A Startup Tests	67
17.	Reactor Temperature and Thermal Power History During Passive Control Period	75
18.	Possible High Voltage Shutdown Sequence	84
19.	SNAP 10A Converter Characteristics	84

FIGURES

		Page
20.	FS-3 Startup Transient Performance and Predictions	95
21.	Initial FS-4 Reactor Thermal Power Transient	98
22.	FS-4 Reactor Outlet Temperature During Initial Thermal Power Transient	99
23.	FS-4 Reactor Outlet Temperature During Active Control Period	101
24.	Total and Calculated Contributing Reactivity Losses During Active Control Period	102
25.	Reactor Average Temperature During Passive Control Period	104
26.	Predicted SNAP 10A FS-5 Average Temperature Drift	108
27.	Location of SNAPSHOT Radiation Detectors	110
28.	Fast Neutron Flux at SNAP 10A/Agena Interface	113
29.	Effect of Heat Shield Ejection on Normalized Radiation Levels at SNAP 10A/Agena Interface	114
30.	Comparison of Postflight and Preflight Estimates of Fast Neutron Flux at SNAP 10A/Agena Interface Due to TE Converter, Reflector and Control Drums	115
31.	Contributions to Fast Neutron Flux at SNAP 10A/Agena Interface	117
32.	Fast Neutron Flux in Agena Vehicle (26 in. off Centerline) . . .	118
33.	Fast Neutron Flux along SNAP 10A/Agena Vehicle	120
34.	Gamma Dose Rate at SNAP 10A/Agena Interface	120
35.	Gamma Dose Rate along Agena Vehicle (26 in. off Centerline) .	122
36.	Radiation Hardening Classification	123
37.	Equipment Location Options on SNAPSHOT Vehicles	124
38.	Converter Power Output During Startup and Early Part of Active Control	128
39.	FS-4 Converter Voltage, Current, and Resistance During Passive Control Period	129

FIGURES

		Page
40.	Comparison of FS-3 and FS-4 Power Output During Passive Control Period	130
41.	FS-3 Converter Degradation at Constant Average NaK Temperature	130
42.	FS-3 Power Adjustment to FS-4 Test Conditions	131
43.	Average TE Element ΔT	135
44.	Measurement of Converter Resistance to Ground in Flight System	136
45.	Measured FS-4 Leakage Current	136
46.	Comparison of Module Test Data with FS-4 Isolation Resistance Behavior	138
47.	Fit of Two Isolation Resistance Degradation Models to FS-4 Data	140
48.	Effect of Equivalent Slot Cover Diameter and Outgassing Rate on Converter Isolation Resistance Degradation Rate	142
49.	FS-4 Converter Isolation Resistance and Power Loss to Ground	143
50.	Effect of Variables on Converter Voltage Profiles (3.02 v Average Voltage for all Cases)	147
51.	Comparison of Measured and Predicted Voltage Profiles	148
52.	TE Pump Performance History	150
53.	NaK Pump Performance During Startup	153
54.	TE Pump Performance at Full Power	153
55.	Pump Degradation at Constant Reactor Outlet Temperature	154
56.	Relationship Between Flow and Terminal Voltage	156
57.	FS-4 ECU Position History	158
58.	FS-4 ECU Acceptance Test, Pressure-Deflection for Increasing Temperature	161
59.	FSM-1 Ground Test Simulation of Sun Shade Transient Orbit (Flowrate 6% of Design Value)	165

FIGURES

		Page
60.	Heat Shield TC Locations	166
61.	FS-4 Heat Shield Temperature Histories	167
62.	Typical Heat Shield Temperature Comparison with Prediction	169
63.	SNAPSHOT Vehicle and Heat Shield Orbital Planes for Orthogonal Ejection	173
64.	Heat Shields, Time Separation	174
65.	Malfunction and Failure Circuit	190
66.	Agna Unregulated Bus Voltage During Prestartup Test Phase	200
67.	NaK Pump Battery Performance	201
68.	System Voltages and Current During Startup	203
69.	Power Dissipated by Voltage Regulator	204
70.	Agna Voltage Regulator Electronics Temperature	207
71.	Link 2 Telemetry System	211
72.	Typical Telemetry Data Cycling Anomaly	214
73.	Extreme Bimodal Distribution Data Anomaly	215
74.	Example of a Link 2 Telemetry Channel Affected by Ion Engine Generated EMI on Orbit 18	217
75.	Calculated Space Heat Input to S10A as Function of SNAPSHOT Orbital Position	242
76.	Measured S10A Response to Changing Space Heat Input	244
77.	Flowpaths Through Thermoelectric Converter	248
78.	Top View of Thermoelectric Converter	248
79.	NaK Flow Distribution (% of Full Flow vs Leg No.)	249
80.	Converter Average NaK Temperature Distribution	252
81.	S10A Ground Handling and Launch Operations	260

ABSTRACT

On April 3, 1965 the world's first nuclear reactor space power system, SNAP 10A, was successfully launched from Vandenberg Air Force Base and placed in a circular, polar orbit. SNAP 10A is the first thermoelectric, direct conversion system for space application using a nuclear reactor as the heat source. The system achieved full power approximately 12 hr after launch and initially produced more than 600 watts of electrical power. After 43 days of successful operation, the reactor was shutdown as the result of a high voltage failure sequence in the electrical system of the Agena spacecraft. At the time of its shutdown, SNAP 10A was generating about 533 watts of electrical power.

The Preliminary SNAPSHOT-1 Performance Summary ⁽²⁾ described the flight test and provided a preliminary evaluation of the test results. The present report gives a detailed evaluation of both overall system and individual component performance during all phases of the flight test. In the evaluation special emphasis was placed on maximizing the amount of information obtained regarding the effect of the launch and space environments on the operation and performance of a nuclear thermoelectric system. In general the in-flight data agreed very closely with ground test results or extrapolation of those results to space.

SNAPSHOT, 13:24 hr PST, April 3, 1965

NAA-SR-11934

15

BLANK

I. INTRODUCTION

A. SCOPE OF REPORT

On April 3, 1965 the world's first nuclear reactor space power system was successfully launched from Vandenberg Air Force Base. Because of the highly successful and near nominal nature of the SNAP 10A/SNAPSHOT flight test prior to its unexpected shutdown, there were no major system problems or performance anomalies which require explanation in this final performance report. Hence, final analysis of the flight test prior to shutdown (the results of the extensive failure testing and analysis were reported in Reference 1) has taken two approaches, both intended to maximize the amount of information about the effect of the launch and space environments on the operation of a nuclear thermoelectric system:

- 1) Flight test data and results were compared to ground test and predicted space performance to determine whether the system or component behaved exactly as expected or whether there was any unexpected behavior. Unexpected behavior was then investigated with special emphasis on possible indications of incipient failure or effects which might have been caused by the space environment.

- 2) Effects on system performance and operation which could be caused by the differences between space and ground environments were postulated. The data were then analyzed to determine whether these effects did occur and whether they could be quantitatively evaluated.

To avoid unnecessary repetition of material, the Preliminary SNAPSHOT-1 Performance Report (Reference 2) is considered as a companion document for this final report. Hence, much of the descriptive material and data in Reference 2 is referenced, rather than repeated here. Where there are differences in conclusions or values, those in the final report should be considered to be correct. However, there are few significant changes in the numbers reported in Reference 2, and most of these are the result of further, and more detailed, analysis of the data for the final report rather than of errors in the original document.

The following paragraphs contain a brief history and physical description of SNAP 10A and a short chronology of SNAPSHOT test phases.

B. SNAP 10A HISTORY AND SYSTEM DESCRIPTION

The utilization of space will demand large quantities of reliable and long-lived electrical power. All spacecraft require electrical power for instrumentation, data processing, guidance, communication, and experiments. To date, electrical power has been supplied primarily by batteries, solar cells, and fuel cells. For significant power levels supplied for extended periods of time, power systems utilizing a nuclear reactor as a heat source offer advantages in weight, size, and reliability. Recognizing this, the Atomic Energy Commission (AEC) initiated the SNAP (Systems for Nuclear Auxiliary Power) program to develop compact, high-powered, and long-lived electric power units using nuclear heat sources.

The difficulty of supplying power for useful application in space, and the unique energy density advantages of nuclear fission, were recognized in the early 1950s. It was immediately recognized that significant power, greater than a few hundred watts in space, may require a nuclear heat source. System conceptual design studies identified the mutually dependent heat source and power conversion requirements and concept selection criteria. By 1955 these studies had resulted in the identification of a reactor concept for use over a broad power range. A U^{235} fueled and zirconium hydride moderated reactor with liquid metal coolant was chosen on the basis of minimum reactor weight, applicable operating temperature, and a reasonable extrapolation of reactor and materials state-of-the-art. These studies formed the basis for the establishment of the SNAP Program by the AEC in May 1956. Reactor concept verification by means of a critical assembly was achieved concurrently with the start of the space age in October 1957.

Interest in thermoelectric power conversion for space systems was initiated in 1958. The SNAP 10A program for the development of a reactor powered thermoelectric system was established in December 1960. The specific objective of the SNAP 10A program was to demonstrate 500 w of electrical power from a reactor thermoelectric system for 90 days in space. The program progressed through design, development, and qualification phases to a culmination in 1965-1966 with SNAPSHOT – the successful launch, remote startup, and subsequent

unattended operation of SNAP 10A in orbit for 43 days — and the ground testing of a duplicate nuclear qualification unit which operated continuously for 10,000 hr.

SNAP 10A employs direct thermoelectric conversion of heat to electricity. The reactor heat source is controlled by the inherent negative coefficients of reactivity and, as a result, requires no dynamic control or moving parts after startup in orbit. Heat generated in the reactor is transported to the power conversion system by means of a liquid metal coolant, NaK-78 (22% sodium - 78% potassium eutectic) pumped by a thermoelectric magnetic pump. The thermodynamic cycle and performance characteristics of the system are shown in Figure 1.

The complete SNAP 10A system is in the shape of a truncated cone with an overall height of 137 in. and a mounting base diameter of 50 in. This configuration was dictated by minimum weight shield requirements. The critical shielding problem was the elimination of neutron scattering around the shadow shield. As a result, the system configuration is conical, with a base diameter established by the vehicle payload, the upper diameter determined by the effective area of the reactor radiation source, and the length determined by the total radiator area requirement. Figure 2 illustrates the powerplant configuration, the location of components, and the flight unit, FS-4. The system weight breakdown is given in Table 1. A detailed description of the SNAP 10A design can be found in Reference 3.

The heat source of the SNAP 10A system is a compact nuclear reactor which employs a metal hydride fuel-moderator containing 4.8 kg of fully enriched U^{235} . The zirconium hydride moderator has a hydrogen density of about 6.5×10^{22} atoms/cc. This hydrogen density is approximately that of cold water and is responsible for the small size of the reactor (8-in. diameter by 12-in. long). The reactor is controlled by varying the neutron leakage through an external beryllium reflector assembly in which four rotatable control drums are located. Circulation of the liquid metal is accomplished by means of a dc faraday-type magnetic induction pump which derives its current from lead telluride thermoelectric elements operating between the hot NaK and a cold junction provided by a small pump radiator.

8-10-65

7561-0032 N

Figure 1. SNAP 10A Performance Characteristics

NAA-SR-11934
21

7-27-65

Figure 2. SNAP 10A Flight Test System

7561-02656A

TABLE 1
SNAP 10A WEIGHT SUMMARY
(lb)

Reactor	275
Shield	217
Converter	154
Pump	20
Expansion Compensator	28
Piping and NaK	45
Structure	83
Instrumentation and Compartment	106
Heat Shield	32
Total	960

The power conversion unit employs silicon-germanium in 2880 individual pellets electrically connected together in a series-parallel combination to produce over 500 w at a minimum of 28.5v. The individual pellets are in thermal contact with 40 parallel NaK tubes of the coolant circuit. A voltage is produced by the temperature difference which is imposed across the pellets by the flow of heat from the NaK tube, through the pellets, and radiated directly to space from individual radiator platelets coated with a high emissivity thermal control surface. Silicon-germanium was selected as the thermoelectrical material because of its thermal stability, the fact that stable, low-resistance electrical contacts can be produced, and because its mechanical properties are superior to competing materials, thereby permitting greater latitude in mechanical design.

Bellows-type expansion compensators are provided to allow for the volume changes of the liquid-metal coolant in the closed loop and to provide a system pressure high enough to prevent pump cavitation.

The attenuation of fast neutrons is accomplished by a lithium hydride radiation shield. The shield assembly is positioned to shadow-shield the instrument compartment and Agena vehicle and reduces the fast neutron flux in these regions by about five orders of magnitude.

The support structure is fabricated from 20-mil titanium formed to a corrugated cone. The structure transfers the reactor and shield loads to the Agena and directly supports the thermoelectric power system.

The instrument compartment houses the startup and control system, diagnostic instrumentation, and electrical connections which mate the SNAP 10A to the Agena vehicle. One hundred and twenty-one channels of diagnostic instrumentation provide detailed information on the system status and performance.

An ejectable heat shield covers the thermoelectric converter to prevent the liquid metal coolant from freezing prior to achievement of reactor power.

For the SNAPSHOT test, SNAP 10A was mated to the forward end of a modified Standard Agena space vehicle. Throughout the orbital operation SNAP 10A and the Agena operated as a single integrated spacecraft. (See Figure 3.) The Agena supplied such vital support functions as power distribution and conditioning, tracking and command capability, telemetry system, and attitude control and provided a test bed for a variety of secondary payloads. The spacecraft was placed in orbit by an Atlas/Agena launch vehicle. Figure 4 shows the launch vehicle configuration.

7561-02312A

Figure 3. Spacecraft Configuration
(on orbit)

Figure 4. SNAP 10A/Agena/Atlas Vehicle Configuration

In-flight operation of the SNAP 10A unit is initiated by a single ground command. This ground command releases locking pins holding the four control drums, applies power to the startup controller, and arms the heat shield eject circuit. Two of the control drums are spring loaded and immediately drive to the full-in position; the two remaining drums are automatically inserted by the control system at a rate of $1/2^\circ$ every 150 sec. Approximately 7 hr are required to reach criticality, during which time the reactor produces essentially no power and the heat shields protect the liquid metal coolant from freezing. Approximately 15 min after criticality, the reactor experiences a nominal power excursion which is terminated by the inherent negative temperature coefficient of the reactor. With the continued insertion of reactivity, the reactor gradually approaches full power. When the liquid-metal temperature reaches 275°F , the heat shield halves are released by pyrotechnics and pre-loaded springs eject them from the vehicle. The thermoelectric pump bootstraps flow immediately upon an increase in reactor temperature. When the system reaches full operating temperature, the ramp insertion of reactivity is terminated by the action of a temperature switch which senses reactor outlet NaK temperature. The temperature switch functions like a thermostat, causing an occasional reactivity insertion which maintains the reactor outlet temperature above the switch setpoint. After initial equilibrium, the control system is turned off and no further control actions are required. The reactor then behaves like a constant temperature source of power.

C. DEVELOPMENT EXPERIENCE

Development experience with the reactor preceded the inception of the SNAP 10A program. The first reactor of the zirconium-hydride (ZrH) type, SER, began operation in the Spring of 1959 and was operated until the scheduled conclusion of the test program in 1960. During that program a total of 225,000 kw-hr of energy were produced. A second reactor of the ZrH type, SDR, began operation in April of 1961 and the test program continued through December of 1962. Over 11,300 test hours were accumulated of which 6900 hours were at temperatures in excess of 900°F and 2100 hours were at 1200°F . The SDR reactor was the immediate predecessor of SNAP 10A. Further verification of the basic reactor capability is demonstrated by the recent operation of the S8ER. This reactor, while physically somewhat larger than the 10A unit, employs the

same basic technology. During its recently completed test program, S8ER accumulated a total of 10,320 operating hours at 400 kw or more, of which 8800 hours were at a coolant outlet temperature of 1300°F.

Development and test experience with the thermoelectric converter has been equally broad. During the course of the program the equivalent of over nine complete converters have been produced and tested. This development effort was directed principally toward maximizing the useful energy output of the converter by reducing extraneous losses, and improving overall converter reliability through detailed metallurgical improvements in the materials and stack. Some thermoelectric modules (modules being an assembly of 24 elements) have been on continuous life test for over 31,000 hours. The total module test experience exceeds 1 million module hours. The physical strength of an individual thermoelectric element stack assembly from NaK tube to aluminum radiator platelet is sufficient to support a tensile load of about 250 lb, which is a factor of 100 over the equivalent launch loads. A summary assessment of the development accomplishments indicates the SNAP 10A converter will survive the necessary environment and produce 500 w with a reliability in excess of 0.992.

The other principal components of the SNAP 10A system have been subjected to an equally rigorous development and qualification program. Thermoelectric pumps currently on test have accumulated over 20,000 hours endurance at full operating temperature in a simulated space environment. The system expansion compensators experienced over 10,000 hours of testing. High temperature instrumentation components have been put through a complete spectrum of tests to qualify them for the high temperature, hard vacuum, high radiation environment.

In addition to the extensive component engineering and development associated with the SNAP 10A program, primary reliance has been placed on extensive system tests. This system testing was basically of two types, (1) exploring the technical interface with the launch vehicle, and (2) performance evaluation, demonstration, and final qualification of the 10A design.

The SNAP 10A-SNAPSHOT system is wholly dependent upon the Agena launch vehicle for essential information feedback to earth during the flight. To ensure

combined system compatibility, prototype electrical SNAP 10A and Agena units were manufactured, mated, and subjected to extensive functional testing which continued over a period of approximately two years.

To explore performance of SNAP 10A, a total of eight prototype and qualification systems were built and tested; three of these were devoted to purely structural tests, three were prototype thermal vacuum performance tests, and two were final complete system qualification tests. The latter two systems consisted of a non-nuclear unit, FSM-4 (the reactor core was replaced by an electrical heater, but in all other respects the system was identical to the flight units), and a nuclear system, FS-3, identical to the flight units. These two systems were subjected to the complete spectrum of tests including initial performance record measurements, full qualification-level shock and vibration, transient startup tests, and long-term endurance testing in a simulated space environment. Test results from these two qualification systems, especially FS-3 which recently completed a completely successful continuous 10,000-hour endurance test, have been used extensively throughout this report to analyze flight system performance. It is interesting to note the improvement in performance between the prototype and qualification systems as shown in Table 2.

TABLE 2
COMPARISON OF SNAP 10A GROUND TEST SYSTEMS PERFORMANCE(*)

System	FSM-1	FS-1	FSM-4(†)
Date of Fabrication	May 1963	June 1963	June 1964
Type of System	Prototype	Prototype	Qualification
Average Converter NaK Temperature (°F)	920	920	920
Reactor Thermal Power (kw)	39.2	38.7	37.8
Electrical Power Output (w)	420	410	460
NaK Flowrate (gpm)	11.0	13.2	13.2
System Thermal Efficiency (%)	1.07	1.06	1.22
Overall Converter Efficiency (%)	1.20	1.19	1.37

(*)Thermal and electrical power are low because of the low average NaK temperature chosen for comparison.

(†)Performance of FSM-4 is typical of FS-3, FS-4 and FS-5 systems on the ground.

D. SNAPSHOT OBJECTIVES AND CHRONOLOGY

SNAPSHOT was the first flight test of a reactor space power system. The overall objectives of the SNAP 10A/SNAPSHOT program were to design, develop, and test a prototype reactor-thermoelectric space power system; to demonstrate that a reactor powerplant was capable of being handled, transported, checked out, and launched within acceptable safety criteria and without requirements for modifying normal launch procedures; to demonstrate that the powerplant could survive launch and be started up remotely in orbit; to demonstrate the simplicity and inherent reliability potential of a reactor-thermoelectric space power unit; to demonstrate that the radiation from the heat source could be adequately shielded and would not cause interference with operation of on-board equipment; to provide in-space operating data and experience as a basis for applications engineering and higher performance system design.

The SNAPSHOT flight test system, FS-4, was launched from Vandenberg Air Force Base (VAFB) on April 3, 1965. It operated successfully in orbit for 43 days at which point it was unexpectedly shutdown. The significant phases in the history of the first SNAP 10A system to be flight-tested are summarized below.

1. Preflight

- 1) Fabrication completed September 17, 1964
- 2) Acceptance-level shock and vibration tests successfully passed September 24, 1964
- 3) Fuel loading and dry critical testing completed October 26, 1964
- 4) NaK loading and thermal reference testing completed January 15, 1965
- 5) System shipped to VAFB February 18, 1965
- 6) System checked out and mated to Atlas-Agena on April 2, 1965. (Launch pad operations are discussed in Reference 4.)

During the thermal reference tests the FS-4 system was performance tested in a vacuum environment using electrical heaters to heat the NaK. System performance at full power was found to be substantially identical to that of the two ground test qualification systems as can be seen from Table 3. FS-4 thermal reference data were extrapolated to space, using suitable analytical techniques, to predict initial full power performance in flight. These predictions

TABLE 3
SNAP 10A QUALIFICATION SYSTEMS INITIAL PERFORMANCE

System	FSM-4	FS-3	FS-4
Average Converter NaK Temperature (°F)	960	960	960
Reactor Thermal Power (kw)	41.6	40.2	41
Electrical Power Output (w)	523	512	519
Open Circuit Voltage (v)	58.0	56.9	57.9
Internal Resistance (ohms)	1.63	1.60	1.61
NaK Flowrate (gpm)	14.1	14.2	14.1
System Thermal Efficiency (%)	1.26	1.27	1.27

are quoted throughout the report. By comparing the predictions to actual flight test data it was possible to determine that the system had not been damaged by launch loads and that the effects of the space environment on system performance had been properly taken into account in the SNAP 10A design.

2. Flight

1) The launch phase began at 13:24 PST on April 3, 1965 when the Atlas booster engines ignited. The phase ended 58.5 minutes later when the Agena engine second burn was completed. All data of structural interest, i. e. all system response to shock, vibration, and acceleration loads, were obtained during this period. The brief period prior to liftoff provided an opportunity to compare telemetry to land-line data, increasing the confidence in later flight data.

2) The prestartup phase covered the time period from orbital injection at 10 minutes after launch until the attainment of sensible heat on orbit 6, 10.3 hours after launch. This test phase was defined as that period when the the passive thermal response of the system to the changing solar heat inputs was of primary interest. Of particular importance during this period was the ability of the heat shield to maintain relatively constant NaK temperatures above a minimum of 50°F while the solar heat inputs changed significantly. (During this phase the reactor produced only an infinitesimal amount of power.)

3) The startup phase was initiated on orbit 2 when the reactor startup command was given 3.7 hours after launch. Confirmation of a long lived

orbit, a necessary safety precaution required prior to reactor startup, had been received on orbit 1. The final estimate of orbital parameters is given in Table 4.

TABLE 4
FINAL SNAPSHOT ORBITAL PARAMETERS

	Actual	Nominal
Apogee (n. mi)	722.27	700
Perigee (n. mi)	688.20	700
Inclination (degrees)	90.2°	90°
Eccentricity	0.0031	0.0
Period (min)	111.6	111
Orbital Lifetime (yr)	3800 to 20,000	3800

The startup period was ended by the initial reactor control temperature switch closure on orbit 7, 12.3 hours after launch. During this period of time the control drums were stepped in, the reactor went critical, heat shields were ejected, and the reactor reached full power. Of particular interest during this period of time were the performance of the reactor control system, the point at which the reactor went critical (indicative of the neutron source strength of space), and the response of the system to the nuclear startup transient. This was the most crucial phase for SNAP 10A since it involved by far the largest number of SNAP 10A mechanical and electrical operations.

4) The active control phase extended from the initial temperature switch closure on orbit 7 until the insertion of the last two control drum steps, by ground command, on orbit 82, 154 hours after launch. During the active control phase drum steps were inserted automatically by the reactor control system as required to maintain the reactor outlet temperature above the switch set point. The initial full power performance of the system and the short term reactivity losses were particularly noteworthy during this phase. The active control period was typically marked by unsteady state operation since the entire system responded to the temperature increase following each drum step insertion. Two unexpected events occurred in the Agena

during this time: (1) the ion engine caused severe interference with the telemetry and attitude control systems when operated on orbit 18 and (2) the flight tape recorder malfunctioned and was turned off on orbit 39.

5) The passive control phase was initiated on orbit 82 when the reactor control system was deactivated by ground command and continued until sometime after orbit 552 when the reactor was shut down. During passive control, the system was controlled solely by the inherent negative temperature coefficient of the reactor heat source. In this mode of operation, no electronic control devices or mechanical moving parts were required to function for the balance of the system life. Of particular interest during this phase were the rate of reactor temperature drift and the amount of normal degradation in such components as converter, pump, and emissivity coatings. The only flight anomaly involving SNAP 10A occurred during this time; converter isolation resistance degraded more rapidly than expected.

6) The shutdown phase began about 43 days after launch when contact with the spacecraft was lost, abruptly and without warning, between orbits 552 and 555. Telemetry data was next received on orbit 574 and indicated that the reactor had been shut down. Contact with the vehicle was maintained until the Agena batteries ran down on orbit 616, 47 days after launch. The major effort concerning this phase was devoted to locating the cause of the unexpected shutdown.

More detailed flight chronologies can be found in References 2 and 5.

BLANK

II. SUMMARY

Perhaps the most significant accomplishment of SNAPSHOT was the demonstrated absence of any major scientific or engineering unknown. The system operated almost exactly as intended and actual performances were extremely close to predicted performances. Table 5 gives evidence of this excellent agreement and shows typical system performance at the beginning and end of the flight test. This agreement provided important verification of the adequacy of the SNAP 10A design and testing and the analytical models used to predict its performance in space. SNAP 10A performed satisfactorily and met all design objectives except those associated with long-term performance, which could not be fulfilled because of premature reactor shutdown which resulted from the failure in the Agena electrical system. The reliability potential of reactor - thermoelectric space power systems was demonstrated by the nearly perfect performance of SNAP 10A on its first flight test.

While there were a number of informative deviations from expected behavior, which are discussed herein, the only significant deviation so far as SNAP 10A was concerned was the more rapid than expected decrease in converter resistance-to-ground. Conservative calculations showed that power loss-to-ground would not have caused mission failure; that is, at the end of one year net electrical power output would have been sufficient to meet vehicle power requirements.

The following paragraphs summarize the most important conclusions and findings from the system and component analysis sections of this report.

The launch vibration and acceleration load measurements verified that the SNAP 10A ground shock and vibration tests were more than adequate to assure structural reliability of the system. The support structure performed within the envelope of predicted flight-limit loads; the low frequency basic loads were near acceptance test levels as desired; and there was no system or component damage as a result of launch loads, with the exception of one thermocouple network which was temporarily short-circuited. Because of the exceptional length of the SNAPSHOT vehicle, the first and second bending modes of the booster system were excited to 50% higher levels than previously recorded on Atlas/Agena flights. However, for SNAP 10A these low frequency loads were not critical.

TABLE 5
SNAP 10A PERFORMANCE IN SPACE

	Initial Performance		Final Performance	
	Measured	Predicted	Measured	Predicted
Total Electrical Power Generated (kw-hr)	--	--	570	--
Total Thermal Power Produced (kw-hr)	--	--	44,000	--
Reactor Outlet NaK Temperature (°F)	1030-1062 [*]	--	1030	1034
Reactor Average NaK Temperature (°F)	966-997 [*]	--	962	964
Reactor Thermal Power (kw)	42.5-45.5 [*]	--	42.2	42.4
Flow (gpm)	14.9-15.1 [*]	--	13.3	13.1
Converter Isolation Resistance (ohms)	> 300 [†]	6000	34	2200
Electrical Power Output (w)	590-650 [*]	--	533	530
Converter Voltage (v)	30.3	29.8	30.25	29.75
Reactor Thermal Power at Average Reactor Temperature of 983°F (kw)	43.9	43.9	43.8	43.9
Flow at Reactor Outlet Temperature of 1040°F (gpm)	14.9	14.9	13.4	13.2
Electrical Power Output at Average Converter NaK Temperature of 980°F (w)	596	601	565	565
Overall System Thermal Efficiency (%)	1.36	1.37	1.26	1.26
Pump Degradation at Constant Temperature (%)	--	--	10.0	9.8
Converter Power Degradation at Constant Temperature (%)	--	--	4.2	5.0

*Approximate response to insertion of two control drum steps after control temperature switch reached its normal set point.

†Transducer could not measure resistances greater than 300 ohms.

The system thermal response during the prestartup phase conclusively verified the adequacy of the essentially passive thermal control techniques used on SNAP 10A to maintain the NaK temperature well above its freezing point. The heat shield experienced the predicted wide swings in temperature of nearly 300°F peak-to-peak in response to the changing solar heat inputs, but it held the maximum fluctuations in NaK temperature to 25°F which was even less than expected.

The automatic startup of the reactor power system in space was flawlessly accomplished. The response of the system was in excellent agreement with predictions based on analog computer studies and FS-3 ground test results. One area of uncertainty, the effective neutron source level in space, was bracketed more precisely. It is concluded that the startup behavior of SNAP reactor systems in space is well understood and no significant uncertainties remain.

SNAP 10A performance during the active control phase was very much as expected. The automatic control system inserted reactivity as required to maintain the core outlet temperature above 1030°F. The amount of reactivity insertion required was within 8% of that predicted, good agreement for these small absolute changes in reactivity. Both initial full power performance and the degradation observed during the six days of active control were within fractions of a percent of the predictions. Because of the continuous gradual temperature decrease observed on FS-3, it was decided to adjust the FS-4 temperature upward prior to terminating active control to ensure a gross electrical output of at least 500 w at the end of 90 days. This was done, demonstrating the ease with which temperature and power level adjustment could be accomplished from the ground.

System performance and degradation rates during the passive control period were as predicted. Premature shutdown of the system made it impossible to meet the 90-day goal and completely satisfy long-term endurance objectives. However, 43 days was long enough to demonstrate and evaluate component and system performance in space and to permit reasonable extrapolations of system performance to 90 days with the help of FS-3 data.

The most probable cause of the premature shutdown was high voltage on the Agena unregulated bus, the sequence beginning with a failure of the voltage

regulator and ending with the command decoder sending spurious commands to shut down the reactor. There is no evidence that a malfunction in the SNAP 10A system was the initiating cause of the shutdown.

Reactor startup transient performance and reactivity losses (both short term during active control and long term during passive control) were in good agreement with expected values based on the initial and long term performance of FS-3. The peculiar shutdown mode of the system provided invaluable data on the ability of the control system, including the reflector eject system, to operate properly after 43 days in space at a vacuum level unattainable in ground tests.

One of the principal objectives of SNAPSHOT was to determine the effectiveness of the radiation shield. These measurements cannot be duplicated on the ground because radiation scattering from test cell structures, or even air, bypasses a shadow shield, producing radiation levels which exceed those due to transmission through the shield by several orders-of-magnitude. The SNAPSHOT test confirmed that shield analysis and design for SNAP 10A is a fast neutron problem, and a specific gamma shield is unnecessary. The measured gamma dose rate at the reference dose plane (a 5-ft diameter circle located 17.5 ft below the reactor) was only 7×10^5 r/yr compared to a design objective of 1×10^7 r/yr. The measured fast neutron flux at the reference dose plane was 5×10^{12} nvt/yr, compared to a design objective of 1×10^{12} nvt/yr. Preflight estimates ranged from 1×10^{12} nvt/yr to 1×10^{13} nvt/yr, depending upon whether attenuation by Agena equipment was considered. The flight test confirmed the preflight analysis that fast neutron fluxes below the shield are controlled by scattering processes from partially-shielded reactor and system components rather than direct line-of-sight from the reactor core. A number of post-flight improvements and changes in the analytical model resulted in good agreement between measured and calculated levels, except in the area of the Agena aft rack where empirical equipment attenuation factors were required. There was no evidence of equipment degradation due to radiation effects and none was expected because of the low doses (for radiation hardened equipment) received by the time of shutdown.

The thermoelectric converter performance was essentially identical to module and system ground test data adjusted to space conditions with the exception of the previously mentioned decrease in resistance-to-ground. Preflight

calculations indicated that an 11 to 12% increase in electrical power output could be expected in space as compared to a ground test system operating at the same average NaK temperature. This was verified by flight test data. Net power output ranged from a transient high of 650 w shortly after reactivity insertions during the early part of active control, to a low of 523 w in the sun after 43 days when the power loss to ground was 10 w. The converter voltage profile showed no change during 43 days, indicating a healthy converter with no open or short circuits. The converter flow distribution was not significantly different from that predicted for space.

A gradual decrease in converter resistance-to-ground became apparent about 600 to 700 hours after reaching full power. While a similar phenomena caused by deposition of hydrocarbons on the insulator surfaces had been observed during ground tests, preflight calculations indicated that 1% power loss would occur only after 10,000 hours of operation, whereas flight test data showed 1% power loss after 835 hours. Subsequent to the flight test, more precise laboratory experiments have been performed which change the predicted time for 1% power loss to 1622 hours. This is relatively good agreement for a calculation of this sort and supports the conclusion that the mechanism of resistance-to-ground degradation is understood (although a significant amount of uncertainty in the magnitude remains). This agreement gives some confidence that the measures taken to avoid degradation on future systems will be effective. Calculations showed that net power output at one year would have been sufficient to supply vehicle power requirements.

The flow produced by the thermoelectric pump during the prestartup phase was more than three times greater than the minimum acceptable rate of 0.26 gpm. Information of value to the prestartup model was obtained on the effect of solar inputs to the pump radiators at low flows. During all phases of the test the pump provided the required flowrates with comfortable safety margins. Pump performance and degradation were nearly identical to that of similar pumps on ground test; there was no indication of malfunction, incipient failure, or flow instability.

The operation of both expansion compensators during all test phases satisfied all system design requirements. None of the data showed any indications of NaK leakage. Bellows deflection data indicated that at NaK loading conditions

there was probably 1 to 1-1/2 in.³ of gas trapped in the system. There was no evidence that this small amount of gas affected either the pump or the converter flow distribution.

The SNAP 10A instrumentation and SNAPSHOT data system provided satisfactory data of sufficient quantity and accuracy to obtain significant information from the flight test. Difficult problems such as the measurement of internal and isolation resistance of an energized thermoelectric converter and measurement of radiation dose rate distribution were solved using minimum weight and minimum power instrumentation. The much larger errors in telemetry data relative to data obtained from SNAP 10A ground tests caused rather substantial changes in the relative usefulness of different types of instruments and required substantial changes in the methods of data analysis. The flight test experience highlighted the unique, sometimes conflicting, requirements of real time operation compared to off-line analysis, and accurate analysis of normal performance compared to correct diagnosis of problems in the face of equipment failure. As a result of this experience a substantial number of improvements in instrumentation and the malfunction detection circuitry can be made to increase the accuracy of flight test information and the probability of obtaining at least the minimum amount of it required for unambiguous diagnosis.

The performance of the Agena vehicle was generally quite satisfactory and no major deviations or anomalies were encountered prior to system shutdown. Several minor deviations from normal or specified performance occurred primarily in the electrical and telemetry subsystems. The most important of these were the possible overstressing and overheating of the voltage regulator and the probable failure to meet the specified accuracy requirement of $\pm 3\%$ of full scale on as many as 30% of the telemetry channels. The severe electromagnetic interference generated by the ion engine was important not so much for the temporary upset it caused during orbit 18, but because of the resulting uncertainty which kept the SNAP 10A malfunction and failure system from being enabled and because of the reduced vehicle power demand which resulted in a greater load on the voltage regulator.

There was no significant difference between the emissivity and absorptivity values of the SNAP 10A thermal control surfaces measured on the ground after the FS-4 thermal reference test and those inferred from flight data. There

was some evidence of a slight improvement (i. e. , a decrease) in solar absorptivity of the thermoelectric converter during the 43 days of operation.

SNAP 10A system heat losses were about 4 to 5% higher in space than on the ground as expected.

SNAP 10A was affected in a number of ways by the space environment, and in several cases the resulting change in system performance was measurable. In addition to those already mentioned (solar heat input during prestartup, neutron source strength, radiation levels, more uneven converter flow distribution, higher electrical and thermal power and flow) the system response at full power to changes in solar heat flux was suprisingly evident even though relatively small. For example, electrical power output increased by 20 to 25 w when the vehicle entered the earth's shadow. The magnetic torques resulting from the interaction of the fields produced by the pump magnet and converter current with the earth's magnetic field, were at most no greater than predicted; the exceptional stability of the SNAPSHOT vehicle did not permit oscillations large enough to measure. As expected, there was no evidence of micrometeorite damage, either to the NaK piping or thermal control surface, and the two micrometeorite detectors received no impacts large enough to detect (greater than 10 microns diameter).

The data received during the 43 days of successful spacecraft operation substantiated in all respects the technical soundness of SNAP 10A. The in-flight data closely agreed with ground test results or extrapolation of those results to space, thus demonstrating the repeatability with which the systems can be manufactured, and the small, well-understood effect of the space environment on their performance. In addition, the flight test and launch provided practical experience which can be applied to simplify and increase the reliability of future systems. The radiation environment from the reactor did not influence vehicle performance, having been properly accounted for in the design-development stage. Finally, success on the first launch of this reactor thermoelectric space power system attests in substantial measure to its simplicity and promise of high reliability.

BLANK

III. SYSTEM PERFORMANCE ANALYSES

A. LAUNCH

The flight test data of structural interest were acquired during the ascent phase of the SNAP 10A flight. It was during this brief period that the satellite encountered the critical environments of acceleration, vibration and acoustic noise that had set the bulk of its basic design criteria. The fact that SNAP 10A arrived in orbit without any apparent mechanical failure or functional difficulty confirmed its basic structural integrity and qualification for flight.

While the launch environment affected every component of the SNAP 10A system, the primary support structure received the heaviest loading during ascent. The initial responses of the SNAP 10A to the transient events of launch were the principal source of these loads. The responses were a function of the dynamic characteristics of the entire launch system. The SNAPSHOT launch configuration had the following characteristics that distinguished it from other Atlas/Agena programs.

- 1) The Agena forward auxiliary racks and nose cone for SNAP 10A defined the longest Agena flight configuration to date.

- 2) The SNAP 10A reactor and shield subassemblies were mounted at the top of a corrugated titanium shell structure. Thus, the heaviest components of SNAP 10A were cantilevered several feet from the interface by a relatively flexible support.

Flight testing is important from the standpoint of structural data because it allows accumulation of environmental data which can be used to compare the relative adequacy of design and ground test limits. For the SNAPSHOT flight test, structural loading data were obtained from each of the three principal axes at the SNAP 10A upper torque box, the SNAP/Agena interface, and the Agena forward rack. Evaluation of the launch environment experienced by the entire launch vehicle is presented in Reference 5.

The flight loads measurements verified that the SNAP 10A ground test specifications and test procedures were more than adequate to assure the structural reliability of the system. Based on a review of the data, the following conclusions can be made:

1) The SNAP 10A supporting structure performed within the envelope of predicted flight-limit loads and no critical conditions were exceeded; however, the low-frequency basic loads experienced by the SNAP 10A were near acceptance test levels.

2) At liftoff, the first and second bending modes of the booster system, 2.8 and 5.7 cps respectively, were excited to 50% higher levels than previously recorded on Atlas/Agena flights. This is attributed to the exceptional length of the vehicle. For the SNAP 10A however, these low frequency loads were not critical.

3) The high frequency vibration levels normally encountered at transonic speeds were relatively low due to the long streamlined nose cone and low winds aloft on the day of launch. This result was verified by the microphone in the Agena forward rack, which recorded 137 db overall sound pressure level at liftoff and 125 db overall during the transonic period. The design sound pressure level for SNAP 10A was 144 db overall.

4) The high-frequency vibrations, in terms of power spectral density plots and g (rms), were approximately two to four times less at the top of the SNAP 10A support shell than at the interface mounting ring. This effect was due to the SNAP 10A instrumentation being separated from the exterior nose cone structure, and the dynamic isolation above 200 cps afforded by the SNAP 10A sheet metal support structure.

5) The maximum rigid body longitudinal acceleration was 6.66 g's at booster cutoff. The lateral rigid body acceleration was near zero throughout ascent. Design levels were 7.5 g's and 1.0 g's, respectively.

6) All the SNAP 10A components successfully survived the launch environment. The expansion compensators and pump responded as predicted to the different acceleration loading. As expected, there was no detectable response of other components.

1. Structural Response

To compare the flight loads experience of SNAP 10A with its design and test criteria, the limited amount of acceleration data obtained during launch must be

evaluated in conjunction with dynamics analyses of the entire launch configuration. An evaluation of environmental data which produced the critical load conditions for SNAP 10A is discussed below.

a. Launch Environment

The launch period, as it affected the structure, was the time from liftoff to that of Agena second burn cutoff, a total elapsed time of 3507.54 sec. During this period, the structure was subjected to the dynamic environment consisting of rigid body accelerations, shock, sinusoidal and random vibrations, and acoustic noise. To measure these loads, accelerometers were located on the SNAP 10A and the Agena at the following locations: Three were mounted on the SNAP 10A upper torque box (SNAP 10A Station 97) and were oriented in three mutually orthogonal directions, corresponding to the vehicle's X, Y and Z axes. The X axis corresponded to the flight axis (longitudinal) of the vehicle. Two additional triaxially oriented sets were mounted on the Agena, one close to the SNAP 10A/Agena interface (Agena Station 189) and the other on the forward rack (Agena Station 255).

The launch environment as recorded by these instruments can be generally described as rigid body accelerations, with superimposed continuous low level high frequency random vibrations induced by the engine and aerodynamic effects. Punctuating this relatively steady condition were short periods of low frequency sinusoidal vibrations due to elastic response to vehicle staging and transonic passage.

b. Rigid Body Accelerations

A plot of the rigid body longitudinal acceleration recorded during ascent through SS-01B engine first operation is shown in Figure 5. This mission profile was as expected from previous flights of the Atlas/Agena. The significant rigid body longitudinal accelerations during ascent are listed in Table 6.

Rigid body accelerations along the lateral axes of the satellite were approximately zero throughout ascent. The rigid body acceleration levels used for design of SNAP 10A were 7.5 g's longitudinal, and 1.0 g lateral. Thus, the satellite was in no way critically loaded during this phase of flight.

6-14-66

7561-03186

Figure 5. Rigid Body Longitudinal Acceleration During Launch

TABLE 6
SIGNIFICANT RIGID BODY ACCELERATIONS

Events	Time From Liftoff (sec)	Acceleration (g)
Booster-Engine Cutoff	134.04	6.66
Sustainer Engine Cutoff	268.95	2.66
SS-01B Engine Second Shutdown	3507.54	2.90

c. Low Frequency Accelerations Due to Elastic Response

The low frequency vibration of the vehicle appeared to be caused mainly by normal mode responses to the transient events occurring during the flight.

The vibration data obtained from the flight consisted of acceleration levels at only three points along a rather complex elastic system. In terms of structural loads, acceleration data in itself are not sufficient to provide an understanding of the bending moments and stresses loading the system. Since it is impossible to monitor enough points along the structure to accurately define a

Figure 6. SNAP 10A/Agena/Atlas Third Lateral Elastic Mode of Vibration at Liftoff

dynamic mode shape, it becomes necessary to calculate the shape and to use the measured acceleration and the calculated mode shape to determine bending moments and shears.

The most critical parts of the flight occurred shortly after liftoff when the maximum vibration-induced lateral bending moment and shear occurred at the SNAP/Agena interface. Table 7 (from Reference 5) summarizes the conditions of peak low frequency vibration levels at the interface and the SNAP 10A upper torque box. It can be seen that the two most prominent modes excited were at 5.5 and 9 cps. Model analysis indicates that these frequencies corresponded to the second and third elastic body modes of the launch configuration.

SNAP 10A lateral loads were maximum at 9 cps in both Y-Y and Z-Z axes. The oscillograph record indicates these peak accelerations were in phase for approximately 2 sec at liftoff. Their vector sum then was 2.2 g. Normalizing the calculated mode shape to this measured value would produce an elastic curve as shown in Figure 6. This calculated third mode for the system at 8.4 cps compares closely with the measured 9 cps. Expanding the scale at the forward end would produce a curve such as that shown in Figure 7. It can be seen that the acceleration level was approximately 3 g and 3.3 g at the reactor center of gravity and top, respectively. A similar analysis revealed that the 5.5 cps second mode did not produce a high response of the SNAP 10A structure. This comparison is also illustrated in Figure 7.

The predicted maximum lateral bending load for SNAP 10A was based upon a similar modal analysis early in the design of SNAP 10A. The load prediction was also based upon previously obtained flight test vibration data, and an analytical dynamic model that reflected the specific structural characteristics of the SNAPSHOT vehicle. From these studies, the maximum lateral flight load based upon 99% probability data, was predicted to be 3.8 g's as measured at the reactor top. This vibration level was in turn established as a fundamental maximum load limit in qualification vibration testing of SNAP 10A systems, at frequencies below 200 cps. A fuller discussion of how this response restriction was implemented in the testing of SNAP 10A may be found in Reference 6. A response restriction was also observed during acceptance vibration testing of SNAP 10A systems. This vibration level was arbitrarily set at approximately 3.0 g.

TABLE 7

LOW-FREQUENCY VIBRATION

Event	Time	SNAP-AGENA Interface				SNAP 10A Upper Torque Box					
		Y Axis		Z Axis		X Axis		Y Axis		Z Axis	
		Frequency (cps)	Amplitude 0-to-Peak (g)	Frequency (cps)	Amplitude 0-to-Peak (g)	Frequency (cps)	Amplitude 0-to-Peak (g)	Frequency (cps)	Amplitude 0-to-Peak (g)	Frequency (cps)	Amplitude 0-to-Peak (g)
Liftoff	0	*	*	*	*	†	†	9.0	1.08	5.5 9.0	1.20 1.92
Booster Engine Staging	136.93	28.2	0.56	28.2	0.36	†	†	4.5	0.81	4.5	0.96
SS-01B Engine 1st Ignition 90% P _c	360.73	16.0	0.19	16.0	0.27	50.0	0.74	16.0	0.55	16.0	0.88
SS-01B Engine 2nd Ignition 90% P _c	3497.70	16.8	0.55	16.8	0.36	53.0	0.77	16.8	0.68	16.8	0.90
SS-01B Engine 2nd Cutoff	3507.54	53.0	0.55	16.0	0.27	53.0	0.90	40.0	0.50	16.0 42.0	0.66 0.33

*Low frequency obscured by high frequency

†Level too low to read

NAA-SR-11934
47

Figure 7. Comparison of Flight Test and Laboratory Acceleration Data

Figure 7 compares the vibration test mode shape at 11 cps, which most nearly matches the flight condition at 9 cps. (The frequencies do not coincide because of structural impedance differences between the shaker table and the Agena mounting ring.) However, a more meaningful comparison of loads can be made in terms of lateral shear and interface bending moments, rather than indirect quantities such as frequency and acceleration level measurements. Table 8 compares the basic flight loads with the SNAP 10A acceptance and qualification test maximum loads.

TABLE 8
SNAP 10A/AGENA STRUCTURAL INTERFACE
LATERAL SHEARS AND BENDING MOMENTS

Condition	Bending Moment (in. /lb)	Shear (lb)	Remarks
Flight	171,000	1,940	9 cps Elastic Mode
Acceptance	165,000	1,950	11 cps Vibration Test
Qualification	239,000	2,238	Static Ultimate

It can be seen that the flight loads experienced by the SNAP 10A structure were near the acceptance levels in vibration testing. The qualification ultimate static loads were in turn approximately 40% higher than the flight and acceptance level loads.

In the longitudinal direction, relatively strong vibrations were experienced by the SNAP 10A during the Agena firing sequences. For example, the instrument measuring X axis accelerations on the upper torque box (Station 97) registered 0.74 g at 50 cps during Agena first ignition, 0.77 g at 53.9 cps during Agena second ignition, and 0.9 g at 53 cps following the Agena engine second cutoff. These data are shown in Table 7. Figure 8, which is an oscillograph record of the flight instrumentation during Agena second ignition, illustrates the typical elastic response of the system to the transient events of launch.

During the above three widely separated occurrences the primary frequency excited was virtually the same and this frequency was very close to the first longitudinal mode measured during the FS-4 laboratory acceptance test,

6-14-66

7561-03188

Figure 8. Elastic Responses at Agena - Second Ignition

approximately 52 cps. The calculated first elastic body mode shape indicates that 3.6 g developed at the reactor top for the maximum 0.9 g response at Station 97 (see Figure 9).

The maximum acceleration response of the FS-4 acceptance vibration test was limited to 3.0 g at the reactor. The g level at the upper torque box was approximately 0.83 g during this mode. Thus the longitudinal flight loads, as with the lateral loads, were close to acceptance levels, but well below the qualification test requirements.

d. High Frequency Vibrations

The high frequency vibrations were for the most part random in nature and were generated from aerodynamic buffeting, acoustic noise, and engine vibrations.

The power spectral density plots at two time periods, liftoff (-0.13 to 18.87 sec) and at 44 to 46 sec after liftoff, indicate that the maximum energy input to SNAP 10A was at relatively high frequencies – between 500 and 700 cps in the X direction, and between 300 to 500 cps in the Y and Z directions.

Figure 9. SNAP 10A/Agena First Longitudinal Elastic Mode of Vibration at Agena Engine Second Cutoff

Table 9 lists the rms accelerations experienced by the vehicle over the entire frequency band of interest (cutoff above 1000 cps). The random g rms input at the interface were at generally low levels, and because the SNAP structure is a low frequency system, the vibration levels were considerably attenuated at the upper torque box.

TABLE 9
RANDOM VIBRATION LEVELS AT LIFTOFF
(g rms)

Axis	SNAP/Agena Interface	SNAP 10A Upper Torque Box
X	6.87	0.87
Y	5.17	1.47
Z	2.12	1.27

The maximum response occurred during liftoff in the Y-Y axis. The g rms level was 1.47 g, with the major portion of the energy between 300 and 500 cps (see Figure 10). Although the rms level was low, higher level peaks might have occurred for short periods of time in the random process, described by the power spectral density plot shown in Figure 10. Assuming a 3- sigma level would be the maximum peak acceleration that could be expected, there may have been a $3 \times 1.47 = 4.4$ g occurrence in this higher frequency load at liftoff. Since all SNAP 10A vibration testing was done by sinusoidal sweep procedures, no direct comparison can be made with the flight random vibration data except on a peak acceleration level basis. A review of SNAP 10A vibration test data reveals that, in the 250 to 500 cps bandwidth, several peak responses were measured and recorded at the SNAP 10A upper support structure. These peak responses ranged from 3.0 to 4.8 g during acceptance vibration tests, and 4.5 to 6.7 g during qualification vibration tests. Thus, in terms of peak response, the high frequency sinusoidal sweep testing of SNAP 10A adequately covered the maximum conditions of the actual flight test random vibration environment.

2. Individual Component Response

All SNAP 10A components responded in some way to the launch environment. Quantitative measurement of loads imposed were made for the structure only,

Figure 10. Random Vibration Response Level at
SNAP 10A Upper Torque Box During Liftoff
(Y-Y Axis)

discussed above. However, the diagnostic instrumentation of a few components indicated a response to the high accelerations caused by engine firings. The expansion compensators were deflected to their stops and, as a result, flowrate decreased to nearly zero during these periods. Both these results were expected and are discussed in more detail in Sections IV F and E, respectively. A network consisting of four thermocouples short-circuited during liftoff (see Section IV. H). This was the only indication of damage caused by launch loads. Response of other SNAP 10A components was not readily detectable from diagnostic instrumentation. However, because of the successful startup and operation of all components, it can be concluded that the components successfully survived the launch environment.

3. Conclusions and Recommendations

The flight test vibration data indicate that no critical design conditions were exceeded. The structural integrity of the entire SNAPSHOT vehicle was maintained during the critical ascent phase of flight. Detailed comparison of the flight test data with the design and test criteria of the SNAP 10A satellite has revealed that all loads encountered were near acceptance test levels, and well

within qualification levels for the system. The SNAP 10A structural design and test criteria were therefore near optimum for the launch environment. It is recommended that future flight tests of reactor powered satellites, similar in basic form to SNAP 10A, incorporate more accelerometer instrumentation. Since such payloads, with their heavy lumped reactor and shield masses, are configured differently than other space satellites, there are no directly comparable flight test data from which structural design criteria may be derived. In particular, the responses of the principal masses, reactor and shield, should be measured, as well as the vibration levels of the thermoelectric converters. A larger accumulation of such data would add confidence in the optimum design and development of SNAP systems.

B. PRESTARTUP PERFORMANCE

The principal SNAP 10A design consideration prior to the generation of nuclear heat was one of thermal control. Most satellites must be designed to operate in a variable thermal environment caused by changing heat inputs at different orbital positions. The specific requirements for a given satellite depend on its mission, type of orbit, and band of temperatures that can be tolerated. Passive control using emissivity coatings of various types has been successfully used for many satellites with near-earth orbits (Reference 7), is inherently reliable, and adds little additional weight to the satellite.

The prestartup temperature control of SNAP 10A was governed primarily by a design requirement to keep the NaK coolant above 50°F to prevent its plugging or freezing. The techniques used to accomplish this were passive and quasi-active: (1) an ejectable heat shield with emissivity coatings designed to effect a net heat input to the system, (2) the inherent heat capacity of the system, and (3) an induced low flow of the NaK to provide a more uniform NaK temperature distribution.

For the purposes of this report, the prestartup period was defined as the time interval from orbit injection to the attainment of sensible heat during orbit 6. Although this definition overlaps the startup period which began during orbit 2, there is no duplication because this section is concerned only with the passive thermal response of the SNAP unit to the heating inputs of the space environment. Prior to orbit 6, the reactor contributed no significant thermal power to the system.

A preliminary evaluation of the prestartup performance has been reported (Reference 2). A more detailed analysis of the orbital test data has since been made; the significant findings and verification of the preliminary conclusions are summarized as follows:

1) The prestartup thermal response of SNAP 10A in orbit conclusively verified the use of the above mentioned thermal control techniques for protection against liquid metal freezing. The NaK coolant was near or above 100°F throughout the prestartup period and was at no time in danger of freezing or plugging. The NaK was quite insensitive to the sun-shade transients of the orbit because of the following:

a. A high flow rate (1.0 gpm or 7% of full flow) was maintained; this flow was nearly twice the design requirement of 4% of full flow. The thermoelectric pump contributed 0.5 gpm superimposed on 0.5 gpm derived from the battery provided current.

b. The ejectable heat shield performed properly, thermally insulating the thermoelectric converter and preventing the NaK from suffering appreciable heat losses.

c. The dynamic thermal response of the SNAP unit was slow because of the large neutron shield and reflector heat capacities.

2) The heat shield was the only instrumented component that experienced large temperature changes during the sun-shade transients. Because these measurements were close to the predicted values, a strong confidence can be given to the analytical model and the simulated orbital environment used in ground system tests. The latter is particularly significant because the ground tests did not duplicate three orbital conditions related to the thermal response: the hard vacuum of space, gravity free environment, and the solar spectrum.

1. Predictions and Prior Testing

The thermal response of the SNAP unit to the environment in space had been analytically predicted⁽⁸⁾ by the use of a lumped parameter mathematical model. A digital computer program⁽⁹⁾ was devised from the model to compute the distribution of temperatures in the system as a function of satellite position in

orbit. To gain confidence in the validity of the computed predictions, orbital prestart conditions were simulated for two non-nuclear ground systems^(10, 11), and the tests conducted with these systems did provide a measure of confidence. However, there were several differences between the simulated and actual orbital environments:

1) The simulated solar heating during the ground system tests used infrared heaters with an energy spectrum ranging from 20,000 to 100,000 Å in wavelength. In contrast, 94% of the energy from the sun, the primary heat source in orbit, is at wavelengths below 20,000 Å.

2) The hardest vacuum levels attained during ground tests (2×10^{-5} torr) were several orders of magnitude "softer" than the 10^{-10} torr space vacuum at 700 nautical miles. As a result, molecular conduction and convection contributed to the heat rejection during ground tests.

3) Gravitational buoyant forces produced by NaK temperature differences acted on the flowing NaK during ground tests. Because the NaK pump develops only 0.01 psi head at the low prestartup flow rates, these forces were significant. They could be totally eliminated only in an orbital test.

Because of these rather substantial differences in environment, the actual prestartup performance of SNAP 10A in orbit was of particular interest. The good correlation between the orbital data, computed predictions, and ground system data was significant because it demonstrated that ground test results, combined with calculations accounting for known differences in environment, can be used to adequately predict performance in space.

2. SNAP 10A Orbital Orientation

The analytical model and the ground tests simulated two limiting cases of solar environment: (1) a noon launch in which the plane of the satellite's orbit would be parallel to the earth-sun line and (2) a twilight launch in which the orbital plane would be perpendicular to the earth-sun line. Because the actual launch was early afternoon (1324 PST), the plane of SNAPSHOT orbit was inclined 21° from the earth-sun line.

Figure 11 displays two views of the initial orbit and satellite orientation. With respect to solar heating, the orbit is classified as a sun/shade transient.

6-14-66

7561-03191

Figure 11. Initial SNAPSHOT Orbital Orientation,
April 3, 1965

Because of its inclination, however, the solar flux incident on the heat shield was not symmetrical (Figure 75, Section IV) as it would have been for a "noon" orbit. One portion of the heat shield was always shaded with minimum shade occurring as the satellite passed over the equator (Figure 76, Section IV).

3. Heat Shield

Heat shield performance is discussed in Section IV. G. The significant conclusion related to the heat shield is that it provided excellent thermal insulation for the radiator and converter.

4. NaK Temperatures

The analytical model predicted that NaK temperatures would experience periodic oscillations in a sun/shade orbit, and that the maximum amplitude would be $\sim 40^{\circ}\text{F}$ peak to peak at the outlet of a converter NaK tube for the anticipated flow of 4%. The average NaK temperature, however, was predicted to have an attenuated oscillatory response no greater than 10°F , peak to peak. The oscillatory behavior had been verified in the ground system tests. Detection of NaK temperature oscillations from the flight telemetry data was not straightforward because of the magnitudes of the random and bias errors in the data system. These errors are discussed quantitatively in Section IV. K. An assessment of the sizes of the random errors can be readily obtained from Figures 12 and 13, which display data from a thermocouple (TC) and resistance temperature device (RTD), respectively. These two displays are typical of the quality of data obtained from the two types of instruments. Data from RTDs generally contained smaller random errors than did the TC data. It is evident from Figure 13 that, while the NaK temperature at the converter outlet manifold varied with orbital position as predicted, the oscillation was only about 10°F peak to peak. The maximum observed variation in outlet temperature for a single tube was 25°F compared to the predicted 40°F .

The instruments that were designed to indicate NaK temperature variations most precisely were RTDs with a range from 0 to 300°F . Four of these RTDs were mounted on the converter outlet manifold, one in each quadrant; their locations were good for purposes of determining NaK temperature differences caused by sun/shade variations incident on the heat shield. The computed means and standard derivations from these measurements are given in Table 10 for pre-launch conditions and each real time pass.

6-24-66

Figure 12. Reactor Outlet Temperature During Pre-Startup Period

6-24-66

Figure 13. Converter Outlet Tube Temperature During Pre-Startup Period

TABLE 10
CONVERTER NaK TUBE OUTLET TEMPERATURE VARIATION
DURING PRESTARTUP

Instrument Number	Location on Outlet Manifold	Prelaunch		Rev. 1		Rev. 2		Rev. 4		Rev. 5	
		$\bar{x}$	s	$\bar{x}$	s	$\bar{x}$	s	$\bar{x}$	s	$\bar{x}$	s
		°F	°F	°F	°F	°F	°F	°F	°F	°F	°F
AI-51	Tube 26	94.8	±2.5	98.5	±2.9	102.1	±3.1	106.3	±2.2	107.8	±2.1
AI-53	Tube 16	89.2	±2.6	103.8	±3.7	107.6	±2.5	102.5	±2.7	104.6	±2.3
AI-55	Tube 46	82.7	±2.3	96.3	±3.8	100.8	±2.6	101.8	±2.9	103.9	±2.4
AI-57	Tube 36	92.3	±2.2	94.6	±3.3	98.4	±2.7	105.7	±2.6	107.0	±2.2

Although the variations from orbit to orbit were about the same size as variations from measurement-to-measurement, statistical analysis of variance revealed that the orbit-to-orbit variations were significant at the 1% level, while the measurement-to-measurement variations were not. Although the exact magnitude of orbit-to-orbit variations could not be determined because of limitations in the data, an estimate was made for an upper bound. The RTD data in Table 10 indicate that orbit-to-orbit variations were less than 5°F. For comparison, a best estimate of the average NaK temperature was made from all available instruments. These estimates are shown in Table 11; they indicate that the orbit-to-orbit variations were less than 10°F. It is concluded, therefore, that the NaK temperature was quite stable throughout the prestartup period. The lack of statistically significant differences between RTD measurements within a given orbit provided evidence that the NaK temperature was reasonably uniform in the four quadrants. Two factors were primary in producing the uniformity: (1) the relatively high flow rate (7% of full flow), and (2) the high effectiveness of the heat shield in insulating the NaK from variations in the sun/shade environment.

5. Recommendations

The recommendations are listed below.

- 1) The stable NaK temperature and high flow rate were enhanced by the launch pad preheating of the NaK to nearly 100°F. It is recommended that preheating be continued on future flights.

TABLE 11
MEAN NaK TEMPERATURE VARIATION DURING PRESTARTUP

Orbit Number	Best Estimate of Mean NaK Temperature (°F)	Satellite Environment During Pass
Prelaunch	97.0	
1	107.1	In sun, southbound over northern hemisphere
2	113.9	In sun, southbound over northern hemisphere
4	107.9	In shade, northbound over northern hemisphere
5	105.5	In shade, northbound over northern hemisphere

2) Assuming NaK preheating for future flights, it is recommended that the batteries which supplied current to the pump be eliminated. Although SNAPSHOT did not assure that a constant-sun/constant-shade orbit would permit a high flow, this uncertainty could be eliminated by repeating previous analyses using the pump low-flow performance information obtained on SNAPSHOT.

3) Additional consideration should be given to other means of implementing prestartup thermal control. The ejectable heat shield, although highly satisfactory for SNAPSHOT, may be surpassed in performance and reliability by other techniques that are not presently state-of-the-art.⁽¹²⁾

C. STARTUP PERIOD

Snap 10A was brought to criticality and full power automatically in orbit using a preprogrammed controller that signalled stepper motors to rotate reflector drums and insert reactivity. The purpose of this technique was to bring the system to full power safely and reliably without the need for continuous monitoring from tracking stations. The control system was designed to sense the NaK coolant temperature at the reactor outlet and to automatically halt the insertion of reactivity when the system reached its design operating temperature. The control system is described in Section IV. A.

The startup of Snap 10A in orbit conclusively demonstrated that a reactor power system could be brought to power safely, reliably and automatically after a single command from earth. After the command was sent from Hawaii Tracking Station during orbit 2, criticality was approached during a 6-hr interval. After attainment of criticality following the insertion of 148 steps of the fine control drums, continued insertion of reactivity resulted in a self-limiting power transient that reached 43 kw. The heat energy produced during the initial transient increased the temperature of the NaK coolant to approximately 340°F at the reactor outlet. As a result, the heat shield was ejected and the core's negative reactivity temperature coefficient limited the power increase and forced the power to decrease as expected. Subsequent stepping of drums placed the system on a power and temperature ramp until the temperature sensor switch closed when the NaK reached 1015°F during orbit 7. With the exception of an initial shifting of the control temperature sensor set point, the entire startup procedure proceeded as designed. System performance was in excellent agreement with analytical predictions. There were no unexpected differences between the FS-4 startup in space and the startup of the FS-3 nuclear ground test system. One area of uncertainty, the effective neutron source level in space, was bracketed more precisely. It was concluded that the startup behavior of Snap reactor systems is well known and that no significant uncertainties remain.

The automatic startup of SNAP 10A in orbit represented the culmination of extensive analyses and experimentation to determine the dynamic response of the system to preprogrammed reactivity insertions. In addition to analog computer simulations of the startup transients,⁽⁸⁾ three ground test systems had been programmed to represent the expected orbital startup. The first two of these were non-nuclear systems, FSM-1 and FSM-4, in which the transport of heat from reactor fuel to NaK coolant was simulated. This was done by programming electric heaters to produce the expected reactor outlet temperature transients. These non-nuclear startups provided good tests of the response of other components to the startup dynamics and verified the ability of the system to withstand the thermal stresses associated with the transients.⁽¹³⁾ The final ground test startup was the first nuclear system startup. With respect to the reactor, the FS-3 startup differed from the orbital startup in only two

significant ways: the use of ground-test-kits to control the reflector drums, and the use of an added neutron source near the core. The FS-3 startups verified⁽¹⁴⁾ in general the analytical simulations made of them. The expected dynamic response of SNAP 10A in orbit, therefore, had evolved during a 2-year interval. The analog computer simulations had been progressively improved to reconcile the experimental results and the earlier predictions. The final predictions for FS-4 orbital startup, therefore, were based on both computer simulations and an extensive body of experimental evidence.

A general description of the orbital startup of FS-4 has been published⁽²⁾ with a chronology of major events that occurred. Section IV. A, B of this report presents a quantitative analysis of the reactor and control system, the effective neutron source in space, and reactivity insertions during startup. The present section therefore will summarize the significant aspects of the overall SNAP 10A startup experience.

The dynamic behavior of SNAP 10A was determined primarily by the reactor, with the other components introducing second order effects. Reactivity insertion and the neutron source level of space influenced the attainment of criticality, after which the reactivity insertion rate, the temperature coefficient of reactivity and the heat capacity of the system governed the increase to full power. The effects of these factors were manifested in the time required to reach criticality, the magnitude of the peak power transient, and the shape and slope of the ramp increase to full power.

1. Approach to Criticality

During the subcritical phase of the startup, the control system inserted reactivity at 154 second intervals while the remainder of the SNAP 10A system responded to the changing solar heat inputs as already described in Section III. B. The reactor reached criticality when the control drums were within $0.6 \pm 0.7^\circ$ of the predicted position. Both the SNAPSHOT and FS-3 results demonstrated that the uncertainties pertaining to criticality are small, and that analysis can properly account for environmental differences.

2. Initial Power Transient

The magnitude of the initial power transient during the startup of a SNAP 10A reactor was determined by the effective source level and the negative

6-14-66

7561-03192

Figure 14. Comparison of Reactor Power Transients During SNAP 10A Startup Tests

6-14-66

7561-03193

Figure 15. Comparison of Reactor Outlet Temperature Response During SNAP 10A Startup Tests

temperature coefficient of reactivity. A qualitative comparison of the FS-4 orbital transient and the FS-3 ground transient is shown in Figure 14. The FS-4 initial power transient was much greater, as expected, because the low source level of FS-4 allowed more reactivity insertion between criticality and sensible heat than in FS-3. Consequently FS-4 was on a shorter reactor period than FS-3 when sensible heat occurred and the initial transient was greater. By measuring the peak power of FS-4 it was possible to measure the neutron source level of space. (See Section IV. B for details.)

3. Ramp to Full Power

Continued stepping of the drums after the initial power transient forced the reactor on a power ramp to which the system responded thermally, hydraulically, and electrically. The thermal response of the system is best described by the increase in reactor outlet temperature. A comparison of this response for FS-3 and FS-4 is shown in Figure 15. The differences between FS-3 and FS-4 were attributable to differences in drum stepping rate and to vault effects. These effects are discussed quantitatively in Section IV. B of this report.

The hydraulic response of FS-3 and FS-4 is shown in Figure 16 from which it can be seen that the NaK flow closely followed the reactor outlet temperature. This kind of behavior typifies the generally passive response to the reactor that the remainder of the SNAP 10A system exhibited. Similar responses can be seen in electrical power output (Figure 38, Section IV), expansion compensator position (Figure 57), and system voltages and currents (Figure 68).

Figures 38 and 53 (Section IV) compare the measured and predicted responses of electrical power and flow. The good agreement shown is typical of that obtained for all system parameters during startup and demonstrates the validity of the final analytical model.

4. Conclusions and Recommendations

The conclusions and recommendations are listed below.

- 1) Snapshot demonstrated that orbital startup could be achieved with a preprogrammed controller that required only a single ground command.

- 2) The inherent stability of SNAP 2/10A reactor has been thoroughly demonstrated; the rate of temperature and power increase, therefore, can be limited by thermal stress criteria rather than by reactor stability criteria.

Figure 16. Comparison of NaK Flow Response During SNAP 10A Startup Tests

3) The reproducibility of SNAP 10A startup performance and the excellent agreement between measured and predicted space performance demonstrated the certainty with which dynamic performance could be predicted. Future analyses can draw with confidence upon the body of experimental results and analytical models built up during the SNAP 10A program.

D. ACTIVE CONTROL

Immediately after initial full power was attained, reactivity losses occurred which were relatively short term. These short term losses were the result of hydrogen redistribution, xenon buildup, and the delayed heating of the beryllium reflector (Section IV. B). The effect of these reactivity losses, if not compensated for, was to lower system temperature and thermal and electrical power. Therefore, to maintain system power a period of active control was required to compensate for the short term reactivity losses. Reactivity was automatically inserted by stepping the control drums inward whenever the reactor outlet temperature fell below the set point of the controller temperature switches.

The performance of the system was generally as expected. Both initial full power performance and degradation rates were close to the predicted values.

Twenty drum steps were automatically inserted by the control system with decreasing frequency to maintain the reactor outlet temperature above its setpoint. Two drum steps were manually inserted by ground command on orbit 82, prior to turning off the controller, so as to produce 500 w at the end of 90 days.

The active control period proceeded as planned with the following exceptions:

1) Initial controller temperature switch closure occurred at a lower reactor outlet temperature than expected. (Section IV. A.)

2) Operation of the ion engine caused excessive electromagnetic interference, upsetting the attitude control and therefore vehicle stability, and making much of the telemetry data unintelligible. Further ion engine operation was deferred until the effects of this problem could be determined. (Section IV. J.)

3) The Link 2 tape recorder was not operated after orbit 39, since the tape quality appeared to be deteriorating with each orbit that it was used. (Section IV. J.)

4) Original plans had called for the malfunction and failure sensing system to be enabled by ground command at about orbit 55. However, at the time, the question of further ion engine operation had not been resolved. It was felt that the primary mission could be jeopardized or terminated if the sensing system were enabled during subsequent ion engine operation. Therefore the malfunction and failure sensing system was not enabled during this period. (Section IV. I.)

The initial and final system performance for the active control period are summarized in Table 12. The data presented have been normalized with respect to average converter temperature and effective space temperature. Normalization of this data was required to obtain a meaningful comparison of initial and final system performance because of the transient behavior of the system during this period and the variations of solar energy input with respect to the time of data acquisition (Section IV. M). The thermal, hydraulic, and electrical performance and degradation rates were all almost exactly as expected. Table 12 contains separate estimates for electrical power change and converter degradation. This was necessary because initial system power output is always

TABLE 12
SNAP 10A SYSTEM PERFORMANCE DURING ACTIVE CONTROL PERIOD

Variable	Initial Performance ⁽¹⁾		Final Performance ⁽²⁾	
	Measured ⁽³⁾	Expected ⁽⁴⁾	Measured ⁽³⁾	Expected ⁽⁴⁾
Reactor outlet temperature (°F)	1047	1045	1050	1050
Reactor inlet temperature (°F)	919	916	917	916
Reactor ave. temperature (°F)	983	983	983	983
Converter ave. temperature (°F)	980	980	980	980
Reactor thermal power (kw)	43.9 ⁽⁵⁾	43.9	43.6	43.9
Flow (gpm)	15.0	15.0	14.5	14.4
Electric power (w)	596	601	577	580
Pump degradation (%)	-	-	2	2.5
Electrical power change (%)	-	-	3.2	3.5
Converter degradation (%)	-	-	2.0	2.3

(1) Based on the average of orbits 10, 12, 13, 15

(2) Based on the average of orbits 70, 71, 72, 82

(3) Normalized to average converter temperature of 980°F and sun conditions. Load is not matched.

(4) FS-4 thermal reference test extrapolated to space and adjusted to same conditions as measured values.

(5) Includes 0.3 kw being absorbed by radiation shield at this time.

inflated compared to the equivalent steady-state operation. (Section IV. D.) As a result the initial power output change is always greater than the converter degradation.

The control drum steps during the active control period, and the times when the steps occurred, referenced to the time of initial switch closure, are listed in Table 13. A total of 22 drum steps were taken. This number includes the first two steps which were taken as a result of the temperature switch set point stabilization (Section IV. A). Also included are the final two steps manually inserted to produce the desired power at the end of 90 days. There is some uncertainty as to the exact number of steps because only 6 occurred during real time data acquisition. The Link 2 tape recorder was used to detect drum steps occurring between acquisitions. However, tape data were not always processed, and, in addition, the Link 2 tape recorder was not operated after orbit 39 because of deteriorating tape quality. In the absence of recorder data steps were detected by observing changes in system performance between two consecutive data acquisitions. Thus there is the possibility of more steps than reported in Table 13. However, they would have had to be single steps in order to be obscured by data scatter and there was no statistical evidence for them. The relationship between the number of steps and reactivity loss for the active control period is discussed in Section IV B.

The purpose of the power trimming operation at the end of the active control period was to ensure the production of 500 w electrical power at the end of 90 days of space operation. Based on analytical predictions of the most probable reactivity losses made early in the SNAP 10A program, 72 hours had been set as the nominal active control period. However, the FS-3 test demonstrated that the actual reactivity losses were closer to the maximum predicted values than to those which had been considered most probable. For SNAPSHOT it was decided to compensate for the increased downward temperature drift resulting from the higher reactivity losses by inserting additional control drum steps by (1) increasing the length of the active control period and (2) ground command. Since it was desirable to limit the active control period in order to demonstrate the self-regulating capability of the system, and the time between automatic steps was increasing (see Table 13; and Figure 23, Section IV) as the rate of reactivity loss decreased (see Figure 24, Section IV), it was decided to insert the last two steps by ground command.

TABLE 13
CONTROL DRUM STEPS DURING ACTIVE CONTROL PERIOD

Step Number	Accumulated Number of Steps During Active Control	Source of Step Detection	Estimated Time From Initial Switch Closure, (hr)
204, 205	1, 2	Orbit 8 T/R	0.30
206, 207	3, 4	Orbit 8 T/R	1.00
208, 209	5, 6	Orbit 8 R/T	1.84
210, 211	7, 8	Orbit 9 R/T	3.57
212, 213	9, 10	Data change between Orbit 10 and 11 R/T	6.1 - 7.5 ~ 7
214, 215	11, 12	Data change between Orbit 13 and 14 R/T	11.8 - 13.8 ~ 13
216	13	Orbit 17 T/R	18.53
217, 218	14, 15	Orbit 19 T/R	22.00
219	16	Orbit 25 T/R	33.54
220	17	Orbit 30 T/R	40.89
221, 222	18, 19	Orbit 36 T/R	53.89
223	20	Data change between Orbit 76 and 77 R/T	121-139
224, 225	21, 22	Orbit 82 R/T	139.6

Note: R/T indicates real time data acquisition
T/R indicates flight tape recorder playback

The procedure followed in deciding the number of steps required and the time for their insertion was as follows:

- 1) The average converter temperature required at the end of 90 days to produce 500 w electrical power was determined taking converter power degradation into account.
- 2) For the length of active control period being considered, the number of drum steps which would be taken automatically after reaching full power was determined. Based on FS-3 data and the reactivity to be inserted automatically, the temperature drift from the end of the active control period to 90 days was estimated.

3) This temperature drift, together with the average converter temperature required to produce 500 w at the end of 90 days, established the temperature required at the start of the passive control period.

4) From (1), the difference between the required temperature at the start of passive control and that prior to power trimming and (2), the temperature coefficient of reactivity, the reactivity insertion required was calculated. From the reactivity required and the differential drum worth the exact number of drum steps was then determined.

5) The transient thermal and electrical effects which would result from the required number of drum steps were investigated to determine if any SNAP or Agena components would be damaged. Of particular concern at the time were the possibility of pump damage (excessive degradation) due to high transient reactor outlet temperature, and the electrical system capability to dissipate the excess transient power.

Based on the procedure just outlined, it was calculated that two steps would be required after approximately 5-1/2 days of active control. The compromise was selected because it would produce the desired power at the earliest time possible without any danger of subjecting the system to undue thermal or electrical power transients. The power trimming was performed by overriding the control temperature switch by ground command until two steps were inserted. There were no operational difficulties, although precautions had been taken to ensure that no more than two steps would be inserted in case there were difficulties. Subsequent calculations (Sections III. E, IV. B, and IV. D) showed that the peak temperature was a safe 1062°F and the gross electrical power output would have been equal to 500 w at the end of 90 days, and that the power trimming operation was therefore successful.

The conclusions and recommendations resulting from active control analyses are listed below.

- 1) The performance of the system was generally as expected.
- 2) The method used to select the length of the active control period and the reactivity insertion to produce the desired power are satisfactory.

3) The tape recorder yielded valuable information, even during those periods when there had been no preflight plans to record data. Therefore it would be desirable to increase the reliability of this component so that it could be used at will throughout the test.

E. PASSIVE CONTROL PERIOD

The SNAP 10A reactor was designed so that after a few days of controlled operation the control system could be deactivated and the reactor would operate in a self regulating manner.

The period of time following controller deactivation is defined as the passive control period. During this period of endurance testing the thermal power and average coolant temperature very slowly decrease. These long term decreases are the net result of fuel burnup, fission product accumulation, hydrogen leakage and redistribution — which tend to reduce power and temperature — and samarium prepoison burnup, which tends to increase power and temperature.

The passive control period began during orbit 82 (21:24 PST, April 9) and extended until the last data acquisition before the unscheduled shutdown, orbit 552 (8:35 PST, May 16). The passive control period covered 874 hours, making a total of 1014 hours at full power.

The specific objectives during the passive control period were to demonstrate reactor and thermoelectric converter long term performance (500 w of electrical power at 90 days), to evaluate component performance in a hard vacuum, zero gravity, nuclear radiation environment, and to evaluate effectiveness of the radiation shield.

In general, test objectives were satisfied and system performance was as predicted. Premature shutdown of the system after 43 days at full power operation made it impossible to meet the 90 day goal and completely satisfy long term endurance objectives. However, 43 days was enough to evaluate the effectiveness of the radiation shield in the non-scattering space environment and to permit reasonable extrapolations of system performance to 90 days.

Nearly all components performed as predicted. Components which behaved as expected and performed identically to ground test included the reactor, reactor control system, and expansion compensator. Components expected to perform

differently in space than during ground test included the thermoelectric converter, thermal control surfaces, and pump. The performance of these components agreed with the analytical extrapolation of ground test results to space.

The major unexpected result was the decrease in converter resistance to ground (discussed in Section IV.D.). There was also some disagreement between preflight predictions and measured radiation levels. Since ground tests provided little information about the radiation shield effectiveness, predictions were based almost entirely on analytical methods and the initial disagreement was not surprising. The test data provided valuable information which was used to refine and correct the analytical models which then yielded results which were in good agreement with those measured. While the SNAP 10A instrumentation itself performed much as it did during ground testing, the additional error introduced by telemetry and processing did affect the relative performance and usefulness of some instruments.

Reactor thermal power and temperature histories during the passive control period are shown in Figure 17. The converter gross power output, NaK pump flowrate, and expansion compensator position are shown in Figures 40, 54, and 57 respectively. Measured and expected performance are presented. Table 14 summarizes expected, measured, and extrapolated system performance during the passive control period. The expected performance was based on a 6-day active control period starting with the same core average temperature as the measured value. The expected temperature drift and pump and converter degradation were based on the FS-3 test data adjusted to space conditions and FS-4 operating temperature. The expected performance shown is for the vehicle in the sun with an effective space temperature of 0°F. The measured data shown in Figures 17, 40, and 54 distinguish between data taken when the vehicle was in the sun from that taken when it was mostly in the shade to permit comparison with the expected sun performance and indicate the effects of sun and shade conditions. (These effects are discussed in detail in Section IV.M.)

The agreement between the expected and measured performance was excellent at the start of the passive control period. Measured and expected values agreed within 1% except for the converter voltage which was 1.7% higher. The higher voltage was attributable to a greater than predicted line voltage drop

6-14-66

7561-03195

Figure 17. Reactor Temperature and Thermal Power History
During Passive Control Period

TABLE 14
SNAP 10A PERFORMANCE DURING PASSIVE CONTROL
PERIOD FOR SUN CONDITIONS

	Beginning of Passive Control (5.7 Days)		Shutdown (43 Days)		90 Days	
	Expected	Measured	Expected	Measured	Expected	Extrapolation*
Reactor Thermal Power (kwt)	44.6	44.3	42.4	42.2	40.9	40.7
NaK Flowrate (gpm)	14.5	14.4	13.1	13.3	12.8	13.0
Electrical						
Gross Power (w)	593	590	530	533	500	500
Power Loss through Isolation Resistance (w)	0	0	0	10	0	35 ⁺⁰ ₋₁₂
Net Power (w)	593	590	530	523	500	465 ⁺¹² ₋₀
Gross Power Normalized to Converter Average Temperature of 980°F (w)	580	577	565	565	557	562
Converter Voltage (v) (Regulated)	29.8	30.3	29.75	30.25	29.7	30.2
Efficiencies (%)						
System Efficiency	1.33	1.33	1.26	1.26	1.22	1.23
Carnot Efficiency	23.5	23.5	23.1	23.3	22.7	23.0
Temperatures (°F)						
Core Inlet NaK	923	924	894	894	879	879
Core Outlet NaK	1057	1057	1034	1030	1018	1013
Core Average NaK	990	990	964	962	949	946
Converter Average NaK	987	987	961	959	946	943
Converter Radiator Average	647	647	634	630	628	623
Reflector Surface	600	595	590	584	580	575
Instrument Compartment Average	140 (max)	102	140 (max)	95	140 (max)	85

*Values measured at 43 days were extrapolated using FS-3 trends adjusted for higher operating temperature and space conditions.

between the converter and the voltage regulator. As discussed in the previous section of this report the average reactor NaK temperature of 990°F was required at this time to assure an electrical power output of 500 watts at 90 days.

After 600 to 700 hours at full power, a faster than expected decrease in converter isolation resistance (Section IV.D) became apparent. At the time of shutdown, the gross electrical power output was close to that expected, but the net power available to the load had been reduced by 10 watts lost to ground. It was estimated that after 90 days of operation the net power output would have been 7% lower than expected due to the decrease in isolation resistance. All other values at 43 and 90 days were in good agreement with those expected, with the exception of converter voltage which was higher than expected.

Table 15 presents measured and expected system and component degradation at the time of shutdown and at 90 days of full power operation. The values for gross electrical power output include the initial transient effects (Section IV.D) and the decreases caused by both converter degradation and reactor temperature drift. The values for net electrical output include, in addition, the effect of power lost due to the decreasing converter resistance to ground. The normalized converter performance reflects only the actual converter degradation. The reactor temperature drift shown is that occurring only during the passive control period. The values for NaK flowrate include both pump degradation and the effect of reactor outlet temperature. The normalized pump performance reflects pump degradation only.

The system and component degradations during the test were as predicted, again with the exception of the degradation caused by the power lost to ground. The decreasing rate of degradation with time can be seen by comparing the amount of degradation which occurred during the first 43 days to the additional, smaller amount for the next 47 days.

It is concluded that the long term performance of SNAP 10A in orbit was shown to be generally the same as that of the nuclear ground test system extrapolated to space. The performance of the ground test system was, in turn, much the same as that predicted from analysis, and non-nuclear system and component ground test data. As a result, a strong measure of confidence can be placed on the reproducibility and predictability of SNAP 10A, and, to a lesser extent, advanced system performance in space.

TABLE 15
SNAP 10A DEGRADATION SUMMARY*

	SHUTDOWN 43 Days		90 Days	
	Expected	Measured	Expected	Extrapolated
Gross Electrical Power (%)	13.7	13.1	18.6	17.8
Net Electrical Power (%)	13.7	14.7	18.6	23.5
Actual Converter Gross Performance Normalized to Constant Temperature (%)	5.0	4.2	6.4	5.7
Actual Converter Net Performance Normalized to Constant Temperature (%)	5.0	5.9	6.4	11.6
Reactor Temperature Drift (°F)	26	28	41	44
NaK Flowrate (%)	12.0	11.3	14.0	13.0
NaK Pump Performance Normalized to Constant Temperature (%)	9.8	10.0	11.4	11.6

*Values shown include active control period degradation.

F. SHUTDOWN

On May 16, after 43 days of successful and troublefree operation, contact with the spacecraft was lost when it failed to respond to ground commands on orbit 555. The loss of contact was abrupt and entirely unexpected. The last acquisition of the vehicle on orbit 552 had been completely normal. Neither tracking nor telemetry signals were received (in spite of numerous attempts to contact and command the vehicle) from that time until 40 hours later when, on orbit 574, the vehicle was acquired with the Link 2 telemetry transmitter broadcasting at normal signal strength. Telemetry data revealed a vehicle status inconsistent with any logical mode of operation of the malfunction and failure detection system, failure of a number of independent electrical components, and a number of operations which could only be explained by ground commands that had not been sent. The SNAP 10A electrical power output was

zero; the spacecraft was operating from the failure battery supply; the command system was inoperative; the redundant 115 v ac power system was dead, resulting in the loss of control moment gyros attitude control system and about half the telemetry data, including all SNAP 10A temperature and flow data; the reactor reflectors had been ejected; the reactor controller was on and pulsing at its normal startup rate; and the vehicle was stable. Telemetry data were acquired on all possible orbits thereafter, but there was no change in vehicle status except for a gradual decrease in bus current and voltage as the failure batteries followed their expected discharge curve. The failure batteries were depleted and telemetry transmission ceased on May 21 during orbit 616. Further details of the shutdown chronology can be found in References 1 and 2.

Telemetry data gave no direct indication of the cause of system shutdown because of the loss of 40 hours of data between the last normal acquisition on orbit 552 and orbit 574 when transmission resumed, and because only half the normal amount of data was available on orbit 574. The failure analysis⁽¹⁾ performed jointly by LMSC and AI proceeded to an investigation of all conceivable occurrences which, alone or in concert, could account for the observed vehicle status on orbit 574. This investigation was followed by detailed analyses of all occurrences which showed any probability of leading to the shutdown. Among these were a detailed study of vehicle circuitry, wire routing, radiation and thermal environments, storage batteries and propellants, orbital behavior, ion engine operation, and low converter resistance to ground. These analyses showed that only a high or low voltage failure of the electrical system had a reasonable probability of occurring and causing the observed results. Detailed circuit analyses and equipment failure tests were then performed. The results of these tests upheld the hypothesized high voltage failure sequence, beginning with a voltage regulator failure and ending with a command decoder sending spurious commands to shut down the reactor and place the spacecraft in the observed configuration. Detailed results of this investigation are given in Reference 1.

The conclusions of the joint investigation by AI and LMSC into the cause of the shutdown are as follows:

- 1) The failure was unexpected, i. e. , impending trouble was not indicated by prefailure telemetry data.

2) The most plausible cause of shutdown of the nuclear power unit was spurious commanding which resulted from a high voltage failure of the command decoder. At no time preceding or subsequent to orbit 552 were reflector eject arm, reflector eject execute, or controller on commands ever sent to the vehicle.

3) The probable cause of the command and decoder failure was high system bus voltage due to:

a) A piece-part failure in the voltage regulator, or

b) Overstressing of the voltage regulator caused by reduced vehicle loads resulting from failure of other electrical equipment, or controller operation resulting in increased reactor power output.

4) The initial malfunction could not be determined.

5) Principal equipment damage noted was most probably due to high voltage.

6) All data anomalies can be credibly explained by a high-voltage condition. They cannot be accounted for by means of a low-voltage failure.

7) The failure batteries were probably connected to the bus by a spurious command, and not as the result of a low-voltage malfunction sequence. The normal low-voltage malfunction/failure sequence was neither initiated nor completed.

8) The thermal and radiation environment was not a significant factor in the shutdown.

9) The possibility of orbital collision, explosion, or other catastrophe was not supported by the data.

10) The flight anomalies noted (ion engine interference with telemetry and vehicle attitude, and the decline of thermoelectric converter resistance to ground) were not factors in the shutdown.

11) Residual propellants that were not dumped from Agena tanks did not appear to be involved in the shutdown sequence.

12) The possibility of vehicle status at orbit 574 being the result of short circuits anywhere in the vehicle is considered very remote because of the highly selective nature that such shorts must assume, as well as the absence of likely initiators of such short circuits.

In analyzing SNAPSHOT performance for this report, particular attention was paid to any anomalies which might have indicated impending component malfunction or failure with special emphasis on electrical equipment. There was no anomalous behavior which could be interpreted as either causing or foretelling vehicle shutdown. However, there was evidence, though not conclusive, that between passes when vehicle power consumption was low the voltage regulator was stressed to its design limit during as much as 20% of the flight and that during the same period of time the regulator electronics temperature exceeded its maximum limit of 160°F. There were also indications of loads near the maximum for good regulation on one phase of the 115 v ac inverter and the 28 v dc converter during the active control period. Both the anomalies are discussed in Section IV.J. of this report.

The following discussion summarizes the information presented in detail in Reference 1.

Since failure seemed almost certainly due to a high voltage or low voltage condition in the vehicle after orbit 552, resolution of this cause of failure required that the high and low voltage characteristics of appropriate vehicle equipment be determined. Testing at high, low, and transient voltages, generally under simulated orbital conditions, was conducted to determine the tolerance of several key vehicle equipment items to the abnormal voltages postulated. The devices tested were the following:

Agena Equipment

- 1) Doppler tracking and command; system, Type XI
- 2) Failure timer, Type IV
- 3) Emergency reset timer, Type VI
- 4) Voltage regulator
- 5) DC/AC 3-phase inverter, Type XI

- 6) DC/DC converter, Type XV
- 7) DC transfer relay
- 8) Control moment gyro heater

SNAP 10A Equipment

- 1) Low voltage sensors
- 2) Main relay box
- 3) SNAP 10A timer
- 4) Reactor controller
- 5) Converter impedance measurement device isolation transformer
- 6) Converter impedance measurement device
- 7) Differential current transductor
- 8) Reactor control temperature switch
- 9) Control drum position instruments
- 10) Expansion compensator position instruments

In general, an attempt was made to control the thermal environment of equipment under test and thereby simulate actual flight conditions. Tests on the voltage regulator, dc/ac inverter, dc/dc converter, dc transfer relay, and the CMG were performed in a vacuum chamber (10^{-5} torr); in most of these cases, temperature was sinusoidally cycled through a range of -25° to $+130^{\circ}\text{F}$ in a 2-hr period. Other equipment items were deemed less sensitive to thermal effects or were mounted in such a way as to compensate for the presence of air and convective cooling. All equipment was to be tested to failure.

No failures due to low or transient voltages occurred during the tests. Generally, the equipment ceased functioning as voltage decreased. The high voltage tests caused a number of components to fail and produced significant results that substantiated the high voltage postulated failure. Two of the most significant results obtained from the high voltage tests were, (1) the Doppler tracking and command system could produce spurious commands when input voltages equaled or exceeded 55 vdc, and (2) a critical failure could be induced in a relatively short time by overstressing the regulator by only 30%. Also, it

can be postulated that such a failure could occur under normal input/output loads if the thermal sensitivity or environment of the flight unit transistors were greater or more severe than the test unit. It can also be concluded that a failure of the flight regulator is an acceptable postulation. The regulator will fail when overstressed and can fail when not overstressed, but it is impossible to determine by ground testing how the unit failed or even that it failed.

Conclusions drawn from the tests and analyses were then applied to the various postulated failure sequences with the result that very few survived with any reasonable degree or probability. The key criteria that any explanation of vehicle failure had to meet to be considered possible were (1) account for the relay closures that resulted in the extraordinary payload manipulation observed (i. e. , reflectors ejected, controller stepping, etc.); (2) account for the failure of the vehicle equipments observed inoperative; and (3) permit survival of the equipment and circuits known to be functional after system shutdown. Further, the probability that any particular postulation was correct, even though it met these criteria, was considered to be inversely related to its restrictiveness.

The high voltage failure sequence was postulated to have been generated by failure of the voltage regulator, which in turn may have been caused by a number of different events. Every aspect of vehicle status on orbit 574 is reasonably explained by this sequence, although the results of the failure tests do not support the postulate completely, and some restrictions must be placed on the sequencing of events. The credibility of the high-voltage sequence is limited by the rather low likelihood of each of the possible causes of regulator failure, but the large number of such possible causes offsets the limitation. The postulated high voltage sequence is entirely probable. A possible sequence of events which could result in system shutdown and the vehicle status observed on orbit 574 is shown in Figure 18. This sequence assumes a total failure of the regulator, which resulted in an unregulated bus voltage sufficiently high to fail a large number of vehicle equipment items and ultimately generate the commands necessary to achieve the orbit 574 configuration. The SNAP 10A converter characteristics are shown in Figure 19. The normal operating point of the converter was 30 v; without the load provided by the regulator, the operating point moved up to 45 v. As other vehicle equipment failed, the operating point continued to shift upward.

6-14-66

7561-03196

Figure 18. Possible High Voltage Shutdown Sequence

6-14-66

7561-03197

Figure 19. SNAP 10A Converter Characteristics

The sequence of equipment failure is somewhat restricted, but a fair amount of flexibility remains. This postulated sequence satisfies the criteria established, since the results of the failure tests showed that the equipment could generally fail or survive as required and that appropriate commands could be generated. The postulated sequence is not firm, however, and requires certain favorable circumstances which cannot be completely verified by testing. The sequence presented here is considered by both LMSC and AI to be the most probable. A complete justification and a discussion of its limitations are given in Reference 1.

Any high voltage sequence requires the failure of the voltage regulator at a very early stage. While the sequence shown in Figure 18 assumes a total failure as the first step, it is quite possible that some other piece of equipment failed thus reducing the vehicle load, overstressing the regulator, and causing it to fail. As mentioned in the summary the initial malfunction could not be determined.

Many other possible causes of the vehicle shutdown were investigated with a variety of results: their occurrence was highly improbable, and they could not have resulted in the observed status of the vehicle on orbit 574, etc. Most of the rejected causes have been listed in the summary; all are described in Reference 1.

The following recommendations have been extracted from Reference 1; no results obtained during the preparation of the current report have altered them:

- 1) Since the conclusions point to a high voltage failure, all reasonable attempts should be made to prevent its reoccurrence. Extensive system and design analysis would be required to determine the best method or combination of methods to accomplish this goal. Appropriate methods recommended for evaluation include:

- a) Provide sufficient regulator capacity to tolerate the maximum converter output combined with a minimum vehicle load.

- b) Redesign critical equipment to provide greater tolerance to over-voltage either by limiting the input voltage or derating the components substantially.

c) Reduce the required operating range of the regulator by providing dummy loads to act in opposition with switched vehicle loads.

d) Provide a redundant regulator.

e) Provide a high voltage input to the malfunction detection system and a method of automatically removing or reducing the high voltage.

2) Since a credible initiator of the high voltage sequence is the inadvertent operation of the reactor controller as the result of selected wiring shorts, it is recommended that greater separation in plugs and harnesses be provided for command, power, and diagnostic functions.

3) In addition to the above recommendations, which are intended to prevent a high voltage sequence from starting and continuing, it is also suggested that certain design improvements be made to enhance the recovery of vehicle data after any type of failure. The design improvements are the rearrangement of diagnostic information on the telemetry multiplexers and the provision of several reactor temperature and flow monitors with high level outputs; then, in the event of the loss of a multiplexer or command power supply, a better combination of data will survive. For example, on SNAP-SHOT, loss of the AC power supply resulted in the loss of data from all reactor temperature and flow monitors; this data may have been useful to the failure diagnosis from the standpoints of actual data and proof of circuit integrity.

Reference 15 contains a detailed description and evaluation of each of the above recommendations.

Section IV.1 of the present report gives recommendations for the improvement of the SNAP 10A malfunction detection circuitry to improve the probability of detecting a malfunction and recording sufficient information to enable unambiguous determination of its cause. Reference 16 contains a number of recommendations resulting from the knowledge gained in analyzing the shutdown.

IV. SUBSYSTEM AND COMPONENT PERFORMANCE

A. REACTOR CONTROL SYSTEM

The functions of the reactor control system are to:

- 1) control the rate of reactivity insertion during the startup period,
- 2) establish reactor stability at full power, and
- 3) make the reactor subcritical at termination of the mission.

Complete control of the reactor is achieved by controlling the neutron leakage through the neutron reflector surrounding the core vessel by varying the reflector thickness. Rotation of the four semicylindrical drums which comprise a portion of the beryllium reflector has the effect of varying the effective thickness of the reflector, thus permitting startup and control of the reactor.

Principal components comprising the reactor control system are the controller, control drum actuators, thermal switch, and timer. At the time the start command is given, power is supplied to the control system, the control drums retainer pins are pulled, and the two coarse control drums are driven by springs to the full in position. The controller starts to provide sequenced power pulses to the control drum actuators to drive the two fine control drums in at a nominal rate of $1/2^\circ$ each 150 seconds. The controller continues to drive the control drums until the reactor outlet thermal switches close when the system reaches full power. The control system is maintained in the active condition while the reactor stabilizes. The control drums are actuated whenever the reactor outlet temperature drops below the thermal switch set point. At the end of the stabilization period, the control system is deactivated by ground command. The control system can be reactivated by ground command at any time during the mission.

Shutdown of the reactor can be initiated by a malfunction signal or by ground command. Either signal activates a timer. If the signal is not corrected in one minute, the reactor will be brought to a subcritical condition by the action of a second timer, one-hour later, by ejection of the reflector assembly. Detailed information describing the reactor control system components and operation can be found in Reference 17.

The following diagnostic instrumentation was provided to monitor the control system status: fine control drum position indicators, motor coil voltage dividers to provide verification of controller operation and drum movement, limit switches to provide full in and full out status information on both fine and coarse control drums, and limit switches to provide on and off status on each half of the reflector assembly and on the retaining band.

The control system operated flawlessly throughout the entire mission. An initial shift in control switch set point from about 1000 to 1026°F during the first two hours at operating temperature was observed. This was also observed on ground tests and has been attributed to trapped air bleeding from the temperature sensor thermal wells. The relatively poor heat transfer in these wells also slowed the temperature sensor-switch response so that two drum steps were normally inserted whenever the switch opened during the active control period. This, too, was observed with the FS-3 nuclear ground test system.

Two features of the SNAPSHOT mission provided unique information on the SNAP 10A control system. First, the power trimming operation described in Section III. E of this report demonstrated the ease with which ground control of the system could be accomplished. Second, the peculiar shutdown mode of the system provided invaluable data on the ability of the control system, including the reflector eject system, to operate after 43 days of operation in space at a vacuum level not possible in ground test.

After receipt of the start command, verification was obtained prior to the end of the orbit 2 pass that the coarse control drums had been inserted and that the fine control drums were moving. Review of the telemetry data showed that the average insertion rate for the drums was 1 step every 153.8 sec, which compares well with the specified rate of 150 ± 10 sec. The data also showed that no steps were missed and that the stepping period did not vary more than 1 sec from the average value.

Drum insertion continued until the temperature switch closed at an indicated temperature of 1015°F. With the known time lag in the temperature sensor, this would indicate that the switch was set at or below 1000°F. This low set point, at least 20°F below the expected nominal value, was cause for some concern during the time immediately following the initial closure on orbit 7. The

indicated temperature at the start of the next pass over the Hawaii Tracking Station, however, was 1026°F; within minutes after the acquisition, the temperature switch opened thus allowing insertion of two drum steps with a resulting increase in temperature to 1055°F. For the rest of the five-day active control period, the switch actuated each time the temperature dropped to an indicated temperature of $1026 \pm 2^\circ\text{F}$. This is illustrated by Figure 23 (Section IV. B) which shows the temperature at the reactor outlet; the temperature at the switch location was 4°F less.

The temperature switches had been set, checked, and the adjusting screw potted during acceptance testing. Following the potting operation, the set points were determined to be 1015°F on one switch and 1020°F on the other. Since the switches are in series, the switch with the higher set point controls the reactor. Following the potting operation, the temperature sensors were removed from their thermal wells to allow for a detailed inspection of the welds in that area. Based on acceptance test data, this introduced an uncertainty of $\pm 8^\circ\text{F}$ in the subsequent switch set point and also allowed air to be trapped in the thermal well upon insertion of the sensors.

The initial shift in set point from about 1000°F on orbit 7 to 1026°F on orbit 8 was attributable to the increase in heat-transfer resistance between the NaK and temperature sensor, caused by trapped air bleeding out of the thermal well. This had been witnessed on several other thermal vacuum tests and was demonstrated in a controlled experiment using a NaK loop. As the heat-transfer resistance increased, the NaK had to achieve a higher temperature for the sensor to reach its set point temperature.

Typically, when the outlet temperature fell to the set point during the active control period, two drum steps were inserted. This occurred eight times as opposed to four times when only single steps were taken. The double stepping was witnessed during FS-3 operation, also, and was attributed to the slow response of the temperature sensor to changes in NaK temperature. The increase in average sensor temperature during the 153.8 sec interval between steps was normally less than the slight ($\sim 0.5^\circ\text{F}$) difference between sensor opening and closing set points. Thus, a second step was taken before the switch could close, as a result of the first step. This was not a serious handicap in system operation, though in future systems the thermal coupling

between the sensor and NaK will be improved to eliminate this phenomenon and decrease the fluctuations in system output. The same improvement would eliminate the problem of trapping air in the thermal well.

At the time of shutdown, the control system had been inactive for 38 days, the controller had received a radiation dose of approximately 8×10^{12} nvt and 1×10^6 r, and the components mounted on the reflector assembly had received a dose of approximately 5×10^{17} nvt and 2×10^9 r.

The telemetry data on orbit 574 showed that the reflectors had ejected and that the control system was operating normally. (When the reflectors eject, the electrical wiring to the reflector assembly is not broken.) The pulsing period of the controller was still approximately 154 sec. Interpretation of the data from the motor coil telemetry channels showed that all motor coils were not only receiving the proper voltage pulses, but were drawing current, thus indicating that the circuits were not damaged. The limit switches indicated that the drums had moved to the full-in position. Since the reflectors were ejected and the outlet temperature was low, the control system continued to try to insert reactivity as long as electrical power was available.

These actions provided valuable information on self-welding of materials in the space vacuum at high operating temperatures, information which is limited because of the difficulty in truly simulating the space environment by ground testing. The fact that the reflectors ejected and the drums moved after the 43-day test in space, showed that no significant self-welding had occurred at critical contact points. Table 16 lists the various contact points (all of which were located on the reflector assembly), the bearing materials involved, and their operating temperatures.

The following items are recommended based on the SNAPSHOT flight test and the events leading to it:

- 1) The control temperature sensor should be redesigned so that it could be mounted in better thermal contact with the NaK. This would eliminate the initial set point shift problem due to trapped air, the large system output variations due to double-stepping of the drums and the calibration problems resulting from the long sensor time constant. The change would also allow calibration of the sensors in a test loop and would simplify the welds in the sensor area.

TABLE 16

FLIGHT SYSTEM CONTACT POINT MATERIAL COMBINATIONS

Contact Point	Materials	Operating Temperature (°F)
Drum Bearings	Kentanium (TiC) vs Molykoted Al_2O_3 on Ti	600
Control Drum Gears	Titanium vs Molykoted Stellite 6B	600
Drum Release Pin-Bracket Contact	Titanium vs Al_2O_3 on Ti	100
Actuator Bearings	P5N Graphite vs Al_2O_3 on Ti	600
Actuator Brake	Chrome-plated relay steel vs Al_2O_3 on Ti	500
Position Sensor Bearings	P5N Graphite vs Al_2O_3 on Ti	350
Limit Switch Contacts	P-3W Graphite vs Silver Alloy	350
Reflector Support Hinge (pivot pin)	Tungsten carbide on Ti vs Al_2O_3 on 316 SS	650
holddown eccentric pin	Chrome carbide on Ti vs Al_2O_3 on Ti	650
sides of hinge	Chrome carbide on 316 SS vs Al_2O_3 on Ti	650
Reflector Upper Positioning Bolts	Al_2O_3 on A286 steel vs Al_2O_3 on Ti	650
	Al_2O_3 on A286 steel vs Al_2O_3 on 316 SS	900
Ejection Spring-Reflector Contact	Graphite Foil (Grafoil) between Ti and 316 SS	650
Reflector Band Standoffs	Grafoil between Ti and 304 SS	500
Band End Fitting at TABRD	Grafoil between A286 steel surfaces	700
TABRD Stud - Band End Fitting Contact	P5N Graphite vs A286 steel	700

2) For thermoelectric power conversion systems the system could be controlled off output electrical current rather than outlet temperature. Test data show this is feasible. The control system could be simplified, ground checkout at ambient conditions made possible, NaK line penetrations eliminated, high temperature wire eliminated and the control system time response reduced.

3) Ground command functions could be given more serious consideration in system design and in establishing procedures, where simplification is desired.

4) The startup time should be decreased to minimize the prestartup temperature control requirements and increase reliability. Insertion of the coarse control elements should decrease reactor subcriticality enough that it can be made critical within 30 minutes rather than the 6 hours required on SNAP 10A.

5) The overall control system should be simplified by combining all temperature switch functions and the controller function into a single electronic unit.

6) Easily accessible test points should be provided for a complete control system checkout.

B. REACTOR

SNAP 10A uses a compact nuclear reactor as its heat source. The core consists of fully enriched uranium-zirconium alloy fuel elements clad in stainless steel. The fuel material is hydrided to also serve as moderator. The fuel elements are positioned and constrained in a stainless steel vessel by upper and lower grid plates and cooled by NaK-78. The core vessel is surrounded by a beryllium reflector which contains four rotating control drums. Detailed information about the reactor can be found in Reference 17.

After the first few days of operation (after the fission product xenon has reached equilibrium) the control system is deactivated. The reactor power level and temperature are then regulated by inherent nuclear effects for the remainder of the system operating life. A samarium-149 prepoison is added to the fuel to make the reactor self stabilizing over the long term. Burnout of the

prepoison compensates for reactivity losses due to U-235 burnup, fission product accumulation and hydrogen leakage. The reactivity loss due to redistribution of hydrogen within the fuel rods is not compensated for by prepoison burnout.

The reactor coolant inlet and outlet temperatures are measured with several thermocouples and resistance temperature sensors, and the coolant flowrate is measured at the NaK pump. The calorimetric power may be calculated from the above data. In addition, a neutron detector (calibrated to flight test data) provides an indication of reactor power.

The reactivity losses observed during FS-4 startup and operation were in good agreement with predictions based on analysis of FS-3 performance. Long term temperature drift was comparable to that observed on FS-3. No anomalies were observed.

The source level in space was calculated to be 10^{-11} kw unmultiplied, or 6.5×10^{-10} kw multiplied. The range of uncertainty was reduced from two orders of magnitude pre-flight to less than one post-flight.

Performance of the reactor during each of the principal test phases is discussed below.

1. Startup Performance

A total of 148 fine control drum steps were inserted to attain criticality. The fine control drums were at $56.8 \pm 0.7^\circ$ (0° is full-in, 135° is full-out) when the reactor was critical at an average coolant temperature of 112°F . The predicted position was $58 \pm 2^\circ$ at an average temperature of 80°F . This prediction is equivalent to 56.2° at an average coolant temperature of 112°F . The $0.6 \pm 0.7^\circ$ difference between the observed and actual critical drum position is an indication of the accuracy of the estimate of startup reactivity effects, including the vault effect. An allowance of 1.4° was included in the predicted critical drum position to account for reflection of neutrons from the test vault walls, which would not be experienced by FS-4 in space.

The reactor generated sensible heat (100 w) 17-1/2 minutes after reaching criticality; the peak of the initial reactor power transient was reached 3-1/2 min. later. Analysis of the FS-4 startup transient was performed with the

mathematical model used for all SNAP 10A startup transient studies. However, some modifications of the model were made prior to the final flight predictions on the basis of startup data from the FS-3 ground test reactor.

The major correction necessary in the thermal model of the reactor was the inclusion of the internal beryllium reflectors (which contribute significantly to the total reactor heat capacity). It was also necessary to make minor modifications in the thermal model to account for the time delays resulting from the location of the reactor inlet and outlet thermocouples at some distance from the inlet and exit plenum, respectively. The value of the overall heat transfer coefficient (UA) calculated for the low temperature range (0.70 - 0.85 kw/°F) was substantiated by FS-3 data and was employed in FS-4 analysis.

The changes made in the nuclear model were: (1) the nominal design 0.5° drum stepping angle was increased to the true value of 0.528°, (2) on the basis of FS-3 data, the absolute value of the negative fuel temperature coefficient at 300°F was increased to 116.3% of the S2DR experimental value and, (3) the nominal drum step interval of 150 sec was increased to the true value of 154 sec.

Before final FS-4 startup predictions were generated, an FS-3 simulation was made to check the accuracy of the revised model. The resulting FS-3 initial power transient is compared to the actual test data in Figure 20. The FS-3 prediction was based on the best estimate of the multiplied source strength (two drums in) for the ground test, 1.75×10^{-6} kw. The predicted peak power was 26.9 kw as compared to the observed peak power of 25.9 kw, a difference of 3.9%. Predicted outlet temperature matched the experimental data well until about 260 sec past sensible heat. Then, for a short time, the theoretical temperature increased more rapidly than did the data. The flow rate followed the reactor outlet temperature in an identical manner for both transients.

The various nuclear and thermal parameters used in the FS-4 startup transient analysis are summarized in Table 17. The most significant of these are (1) source strength, (2) fuel temperature coefficient and (3) overall heat transfer coefficient.

Figure 20. FS-3 Startup Transient Performance and Predictions

TABLE 17
LOW TEMPERATURE NUCLEAR AND THERMAL PARAMETERS FOR FS-4
STARTUP TRANSIENT ANALYSIS

Multiplied Source Strength (kw)	6.5×10^{-10}
Corresponding Unmultiplied Space Source (kw)	10^{-11}
Shutdown Margin (2 drums in) (%/°F)*	-52**
Fuel Temperature Coefficient (%/°F)	-0.0363** (at 300°F)
Upper Grid Plate Coefficient (%/°F)	-0.0146** (at 300°F)
Lower Grid Plate Coefficient (%/°F)	-0.0207** (at 300°F)
Overall Heat Transfer Coefficient (kw/°F)	0.76 (at 300°F)
Fuel Conductivity (Btu/hr ft °F)	13
Fuel Specific Heat (Btu/lb °F)	0.1038 (at 300°F)
Beryllium Internal Reflector Conductivity (Btu/hr ft °F)	83 (at 300°F)
Beryllium Internal Reflector Specific Heat (Btu/lb °F)	0.52 (at 300°F)
NaK Specific Heat (Btu/lb °F)	0.23 (at 300°F)

*Actual shutdown margin was -51.0%; however, this small difference in shutdown margin does not affect the calculated source strength significantly.

**Percentage of initial cold excess reactivity.

The latter two values (2) and (3) above, had been verified by ground testing, but there was a large band of uncertainty on the source strength in space which could only be reduced by flight test data. The multiplied source strength, or subcritical reactor power level, is equal to the unmultiplied space source times the multiplication factor, $1/(1-k_{eff})$. For the SNAP 10A reactor with two drums in, the multiplication factor is 63. Therefore, a 6.3×10^{-10} kw multiplied source would correspond to an unmultiplied space source of 10^{-11} kw. It should be noted that description of the unmultiplied source in terms of kw is merely a convenience. The space neutron flux cannot produce power until it interacts with (is multiplied by) the SNAP 10A reactor.

The multiplied (p,n) space source, due primarily to proton interactions with beryllium, was previously estimated to be between 6.3×10^{-11} to

6.3×10^{-9} kw. Although the (p, n) reaction is the most probable interaction at the lower energies (<10 Mev), the (p, 2n; p, 3n; p, xn), evaporation, spallation, fragmentation and fission reactions at energies greater than 10 Mev should also contribute to the total source. The evaporation and fission reactions are near collisions with nuclei which result in the excited nuclei emitting neutrons or fissioning. Spallation and fragmentation are collision reactions of very high energy protons with nuclei in which neutrons and protons are expelled from the nucleus or the nucleus is destroyed. Unfortunately, the cross sections and yields are not well known for these high energy interactions.

Interactions of energetic residual electrons from high altitude bomb tests provide further uncertainty in the calculation of the startup source in space. The anticipated reaction is electron slowing-down radiation (Bremstrahlung) resulting in (γ , n) reactions in beryllium. For the reactor to be influenced by this potential source contribution, it must be in one of the Van Allen electron belts during the startup sequence.

The minimum source that can be present in the SNAP 10A reactor is that due to spontaneous fission within the core. This source was calculated to be 6.3×10^{-12} kw multiplied, a factor of at least 10 times smaller than the minimum predicted (p, n) space source. However, all that could be said with confidence prior to the flight test was that the multiplied space source was greater than 6.3×10^{-12} kw.

The space source strength was calculated indirectly from flight test data by measuring the initial power peak following criticality whose magnitude was determined only by source strength and fuel temperature coefficient of reactivity.* By assuming the low-temperature fuel coefficient derived from analysis of FS-3 data was valid, the magnitude of the initial power peak could be used to "back-calculate" the space source strength. The excellent agreement between experimental and analytic power transients, based on a 6.5×10^{-10} kw multiplied source strength, is shown in Figure 21. (The experimental peak

*The heat transfer coefficient can be varied over a wide range with little effect on peak power; the fuel to coolant temperature difference changes greatly but the average fuel temperature remains relatively constant. The upper grid plate coefficient has little effect on the power peak as the increase in outlet temperature at the time of peak power is only $\sim 120^\circ\text{F}$.

Figure 21. Initial FS-4 Reactor Thermal Power Transient

power was 43.5 kw; the calculated peak power was 44.0 kw.) A similar but independent analysis, using analog rather than digital techniques, yielded a value of 3.5×10^{-10} for multiplied space source strength. Both these values lie at approximately the middle of the range predicted prior to the flight (6.3×10^{-11} to 6.3×10^{-9}). It is not unreasonable to say that the uncertainty in space source strength for the SNAPSHOT orbit has been reduced from two orders of magnitude to less than one.

The FS-4 reactor outlet temperature transient is shown in Figure 22. As in the case of the earlier FS-3 analysis, the theoretical values were in good agreement with the experimental data through the time of peak power, but exceeded the data during the latter

half of the outlet temperature rise. The reason for this discrepancy is not known. However, the theoretical and experimental curves converged as the power decreased. FS-4 and FS-3 results are compared directly in Figures 14, 15, and 16 for power, temperature, and flow, respectively.

FS-4 thermal stresses during startup were essentially identical to those experienced by FS-3 due to the similar rates-of-change of outlet temperature. Subsequent to the FS-3 startup, thermal stresses in the reactor core were recalculated using the measured initial power and temperature transients which were the most severe experienced by the reactor. The calculation revealed that there were no excessive thermal stresses.⁽¹⁸⁾

Fifty-five fine control drum steps were inserted from criticality to attain full power. The fine control drum position when full power was attained was 27.8° from full in. The reactivity inserted to attain full power was 66.5% of the initial cold excess reactivity. The preflight prediction was 69.8% for nominal design startup conditions. When the prediction was corrected to actual flight conditions, it became 67.1%, or 0.9% greater than the observed value.

Figure 22. FS-4 Reactor Outlet Temperature During Initial Thermal Power Transient

When full power was attained, the FS-4 control drums were further in than the FS-3 control drums (27.8° compared to 33° for FS-3) because the beryllium shim had been removed from the FS-4 reflector before launch to reduce the excess reactivity. FS-3 had two 1/8 in. thick shims on each of two reflector block pieces.

In summary, agreement between preflight predictions and observed performance during the startup phase of operation was excellent. The observed parameters are shown in Table 18 for comparison to predictions.

2. Active Control Period

A total of 22 fine control drum steps were inserted during the active control, 20 to compensate for short term reactivity losses and 2 for power trimming on orbit 82. The response of reactor outlet temperature to the drum steps is shown in Figure 23; the steps are itemized in Table 13.

The total reactivity inserted automatically was 15.5% .^{*} This included 1.1% used to overcome the low initial temperature switch closure, discussed in Section IV. A. The total active control period reactivity loss was therefore only 14.4% with about 13.3% occurring during the first 72 hours. The component reactivity losses are compared to predictions and FS-3 losses in Table 19.

The total active control period reactivity loss and the apparent contributing losses are shown in Figure 24. The predicted total FS-4 reactivity loss is shown for comparison. The good agreement between predicted and observed reactivity loss data tends to verify the assumption that reflector motion (which could not occur on FS-4) actually did occur during FS-3 reactor operation. The FS-4 reflector was pinned into position and essentially could not move relative to the reactor vessel. The FS-3 reflector attachment was modified (by removing

*Reactivity values shown are percents of initial cold excess reactivity of the FS-4 reactor.

TABLE 18
COMPARISON OF FS-4 STARTUP RESULTS WITH PREDICTIONS

Parameter	Nominal Prediction	Adjusted to Actual Temperature	FS-4 Observed or Indicated Results
Vault worth (%) [*]	1.67	1.67 ^{**}	2.38 ± 0.70
Multiplied space source power (kw)	6.3x10 ⁻¹¹ to 6.3x10 ⁻⁹	6.3x10 ⁻¹¹ to 6.3x10 ⁻⁹	6.5x10 ⁻¹⁰
Fine control drum position at critical (Degrees from full in position)	58	56	56.8
Fine control drum stopping rate (sec/step)	154	154 ^{**}	153.8
Peak power during startup transient (kw)	35 to 45	39 to 49	43.5
Temperature-power reactivity defect (%) [*]	69.8	67.1	66.5

^{*}Percent of initial cold excess reactivity

^{**}No adjustment necessary

ejection springs and holddown pins) to simplify system disassembly, and movement relative to the reactor vessel was possible. In fact, ejection switches on both FS-3 reflector halves activated during startup shortly after the reactor temperature began to rise. The approximate magnitude of this effect was +5.4 to 5.8%^{*} during startup, -4.1%^{*} during the 72-hr active control period, and about -1.3 to -1.7%^{*} during the first 100 hr of static control. Since the holddown pins were installed in FS-4, and the reflector eject switches did not actuate, it is assumed that no reflector motion occurred during FS-4 operation. It was important to know this since the reflector motion ascribed to FS-3 was subtracted in making predictions of FS-4 performance.

Figure 23. FS-4 Reactor Outlet Temperature During Active Control Period

Figure 24. Total and Calculated Contributing Reactivity Losses During Active Control

TABLE 19
REACTIVITY LOSSES DURING FIRST 72 HOURS OF
CONTROL PERIOD

Loss Mechanism	Predicted FS-4 Loss (%)*	FS-3 (%)*	FS-4 (%)*
Xenon	-6.31	- 6.79	- 7.12
Hydrogen Redistribution	-2.61	- 1.02	- ~1.15
Reflector heating	-5.56	- 4.98	- 4.98
Reflector motion (FS-3 only)	0.0	- ~4.1	0.0

*Reactivity values shown are percents of initial cold excess reactivity of the FS-4 reactor.

**FS-3 operated at 41 kw (average) during the active control period compared to 43 kw for FS-4. This difference explains the difference in xenon and hydrogen redistribution reactivity effects.

Analysis to date indicates that the reactivity loss due to xenon accumulation was 14% greater than predicted for both FS-3 and FS-4 and that the reactivity loss due to reflector heating was about 8.5% less than predicted. This accuracy is well within the accuracy of the predictions. The apparent 1.15% reactivity loss due to hydrogen redistribution during the first 72 hours of active control is only about 55% of the originally predicted loss which is shown in Table 19. However, this original prediction was based on a preliminary analysis of FS-3 performance, and a more recent analysis⁽¹⁸⁾ resulted in an estimated loss of 1.02%* for FS-3 and a new prediction of 1.05%* for FS-4. This new prediction is an excellent agreement with the apparent 1.15%* FS-4 loss due to hydrogen redistribution.

Reactivity losses due to fission product accumulation, U^{235} burnup, and hydrogen loss are essentially compensated for by Sm^{149} prepoison burnout during both the active and passive control periods. The net reactivity gain or loss

*Reactivity values shown are percents of initial cold excess reactivity of the FS-4 reactor.

Figure 25. Reactor Average Temperature During Passive Control Period

due to these effects during the active control period is entirely negligible (estimated to be approximately 0.017%,) compared to values shown in Table 19.

3. Passive Control Period

The observed FS-4 reactor average temperature drift during the passive control period is shown in Figure 25; FS-3 performance is also shown for comparison. An analytical model which includes each reactivity loss or gain mechanism as a separate effect was used to predict the FS-4 temperature drift. The only difference between the passive control models used for FS-3 and FS-4 was an additional 1.75% allowance for reflector slippage which was included in the model for FS-3 calculations, during the first 10 days of static control operation, and was deleted for FS-4 calculations. Agreement between the calculated temperature drift and the observed data is very good for FS-4, and also for FS-3 for operation through seven months. Reactivity values used in the temperature drift analyses are compared to calculated values in Table 20. The calculated values shown in the table are those calculated from basic nuclear data, although some of the values reflect improvements in nuclear models resulting from S2DR or other SNAP reactor tests. The parameters employed to

obtain a good fit of the FS-3 and FS-4 data reflect changes from the calculated values. These changes are all well within the uncertainty range of the calculated values. Only the prepoison burnout and hydrogen redistribution values were changed significantly to obtain the fit. The change in prepoison burnout from 2.9 to 3.7%/year resulted from a recent analysis⁽¹⁸⁾ which accounted for spatial burnout effects. The change in hydrogen redistribution resulted primarily from the need for a good fit to the observed FS-3 curves and cannot be completely substantiated by theory.

Both the magnitude and time constant of hydrogen redistribution had to be changed to obtain a good fit to the FS-3 and FS-4 experimental data. A time constant of approximately 1580 hr resulted in a good fit to the first seven months of FS-3 data, but it was necessary to vary this value later in the test to fit additional data. None of these values affected the FS-4 fit since they affected only the very long term drift. One thing that all these time constants had in common was that they were much longer than the preflight prediction of 90 hr (at 950°F) obtained with the HYTRAN code which bases its calculations on experimental values of hydrogen diffusion coefficients within the fuel elements. A postflight model which described variation in hydrogen redistribution time constant as a function of temperature was derived from S2DR test data.⁽¹⁹⁾ This model predicted a time constant between 550 and 1600 hours which is in reasonable agreement with FS-3 and FS-4. The reason for the large discrepancy between the experimental results and the analytical values predicted by HYTRAN is not known. Possible explanations are errors in the measured diffusion coefficients, oxide film buildup on the fuel rod (which would force axial hydrogen migration), or some other effect not accounted for in the HYTRAN model.

The magnitude of the hydrogen redistribution effect was calculated with the HYTRAN computer code as 7.5%/year. The HYTRAN analysis predicted that about 50% of this effect would occur during the static control period. As mentioned in the active control discussion, preliminary analysis of FS-3 performance indicated that 2.6% of hydrogen redistribution had occurred during active control. It was therefore expected that the remaining 4.9% redistribution would occur during the passive control period. However, a total of 6.1% was required to result in a good fit of the experimental data, with only 0.5% occurring during

TABLE 20

SNAP 10A REACTIVITY LOSSES DURING PASSIVE CONTROL
(NORMALIZED TO 40 KWT AND 950°F AVERAGE
COOLANT TEMPERATURE)

Reactivity Loss Mechanism	Calculated FS-3 Losses ^{**} (%/yr)			Losses ^{***} Resulting in "Good Fit" to Observed Data (%/yr)	
	Minimum	Nominal	Maximum	FS-3	FS-4
Primary Effects					
Hydrogen Loss	-1.0	-1.0	- 3.0	-1.0	-1.0
Hydrogen Redistribution	-2.8	-4.85	- 6.9	-5.6	-5.6
Prepoison Burnout	+4.9	+2.9	+ 0.9	+3.7	+3.7
Subtotal	+1.1	-2.95	- 9.0	-2.9	-2.9
Other Effects					
U ²³⁵ Burnup and Fission Product Accumulation	-4.05	-4.95	- 5.9	-5.0	-5.0
Reflector Motion (FS-3 only)	0	-1.4	- 2.7	-1.7	0
Radiation Damage	0	- .3	- 0.7	0	0
Thermal Effects	0	0	- 0.7	0	0
Subtotal	-4.05	-6.65	- 9.4	-6.7	-5.0
Total	-2.95	-9.6	-18.4	-9.6	-7.9

*Reactivity values shown are percents of the initial cold excess reactivity of FS-4.

**Calculated values are based on nuclear analysis of SNAP reactors.

***"Good Fit" losses are calculated values which have been varied (within their uncertainty range) as required to obtain a match to the observed temperature drift.

active control and 5.6% during the passive control period. This discrepancy was partially explained by the longer than HYTRAN-predicted time constant, but the change in magnitude has not been explained.

It can be seen that a substantial amount of uncertainty still exists concerning hydrogen redistribution. An extensive analysis of the problem is given in Reference 18. Two additional comments are in order. Additional long term FS-4 data would have been quite useful to verify the general trends observed on FS-3, but additional FS-4 operation probably would not have resolved the uncertainties in hydrogen redistribution because the telemetry errors were large (absolute accuracy of outlet temperature was $\pm 10^\circ\text{F}$) and the amount of uncertainty in absolute value of reactivity and temperature was small, e. g. the "large" discrepancy between the FS-3 fit described in Table 20 and the measured temperature at the end of one year was 9.5°F .

4. Conclusions and Recommendations

The conclusions and recommendations resulting from the analysis are described below.

The performance of FS-3 and FS-4 show that reliable power generation in space using nuclear reactors is a state-of-the-art technology. Improvement of reactor performance in the area of reducing the temperature drift appears desirable. Although there is some uncertainty in the reactivity loss model which was employed in FS-3 and -4 analyses, it was still possible to calculate the change in reactivity loss rate which would be required to minimize temperature drift and to predict the uncertainty in the calculated drift. The same reactivity loss model used to analyze FS-3 and -4 performance was used to determine the prepoison loading which would be required to assure 500 w of electrical power output throughout 1-year of operation. It was assumed that converter resistance to ground degradation could be made negligible so that a year-end average coolant temperature of 950°F would result in the desired electrical output. It was concluded that an initial average coolant temperature of 990°F, and an increase in Sm_2O_3 loading would provide the desired performance. The Sm_2O_3 loading would be increased from 8.6 mg/in. /element to 12.6 mg/in. /element. The reactivity penalty associated with the higher prepoison loading could be compensated for by replacing the three low hydrogen content fuel elements currently loaded in the SNAP 10A core with nominal 6.35 N_h fuel elements. Calculated performance of SNAP 10A reactors with increased Sm_2O_3 loadings is shown in Figure 26.

The amount of uncertainty remaining in the reactivity loss model is indicated by the large band of uncertainty (average $\pm 25^\circ\text{F}$) in Figure 26. Work to improve both the empirical fit and the theoretical models is continuing under the FS-3⁽¹⁸⁾ and reactor improvement programs.

6-14-66

7561-03203

Figure 26. Predicted SNAP 10A FS-5
Average Temperature Drift

C. RADIATION SHIELD

If no radiation shield had been provided the radiation level at the SNAP 10A/Agena reference dose plane, 17.5 ft below the reactor, would have been about 10^{16} nvt/yr due to fast neutrons, and 10^8 rad/yr due to gamma rays. Severe degradation of semiconductor devices can be expected at such levels. Consequently, after considering the radiation tolerance and hardening capability of such devices, it was specified that the maximum radiation level at the dose plane be 10^{12} nvt/yr and 10^7 rad/yr. The problem, therefore, was to design a shield which would attenuate the fast neutron flux by a factor of 10^4 and the gamma dose rate by a factor of 10. Lithium hydride, primarily a neutron shielding material, was chosen as the shield material since it provided the lightest shield, could be fabricated in the desired size, and would withstand the operating environment.

One of the principal objectives of the SNAPSHOT flight program was to determine the effectiveness of the radiation shield. A similar test cannot be duplicated on the ground because radiation scattering from test cell structures can bypass a shadow shield. The radiation level due to scattering can exceed that due to transmission through the shield by several orders of magnitude. Even in an open-air facility, the radiation level due to air scattering would greatly exceed the transmitted level.

In support of the shield effectiveness portion of the SNAPSHOT test, a total of 26 radiation detectors were installed on the SNAPSHOT vehicle. LMSC provided an instrument package consisting of a fast neutron detector, a thermal neutron detector, and a gamma ray detector at four axial locations in the Agena near the vehicle outside surface and at the SNAP/Agena interface on the vehicle centerline. AI provided eight solid-state fast neutron dosimeters in the SNAP instrument compartment to measure an expected radial and angular variation in fast neutron flux, a gamma ray detector, and a thermal neutron detector. A fast neutron detector was located at the reactor centerline. Figure 27 shows the locations of the radiation detectors. The postflight analysis used the readings from the LMSC detectors as reference because the LMSC fast neutron detectors were basically standard fast neutron fission chambers (neptunium) which indicated neutron flux directly while the AI devices were experimental solid-state dosimeters.

6-14-66

7561-03204

Figure 27. Location of SNAPSHOT Radiation Detectors

The SNAPSHOT test confirmed that shield analysis and design for SNAP 10A is a fast neutron problem and a specific gamma shield is unnecessary. The measured gamma dose rate at the reference dose plane was 7×10^5 r/yr compared to a design objective of 1×10^7 r/yr. Preflight estimates⁽²⁰⁾ ranged from 9×10^5 r/yr to 3×10^6 r/yr depending upon whether attenuation by Agena equipment was considered.

The measured fast neutron flux at the reference dose plane was 5×10^{12} nvt/yr compared to a design objective of 1×10^{12} nvt/yr. Preflight estimates⁽²⁰⁾ ranged from 1×10^{12} nvt/yr to 1×10^{13} nvt/yr depending upon whether attenuation by Agena equipment was considered.

The test data confirmed the estimates⁽²¹⁾ of radiation attenuation by equipment in the Agena vehicle except in the aft rack region. New factors were empirically derived for this area.

The maximum difference between the measured data and the preflight analysis was a factor of six underestimate of fast neutron flux at the center of the SNAP/ Agena interface. The post-flight analysis uncovered an error in estimating scattering from the thermoelectric converter. With this correction the analytical data were in complete agreement with the detector reading at this location.

The SNAPSHOT test confirmed the preflight analysis that fast neutron fluxes below the shield are controlled by scattering processes from partially-shielded reactor and system components rather than by direct line-of-sight radiation from the reactor core.

The test data showed that fast neutron scattering in the reactor reflector had been overestimated by a factor of 1.7.

Two postflight investigations were conducted of methods of reducing the fast neutron flux in the Agena aft rack to 1×10^{11} nvt/yr. This permits the use of unhardened, off-the-shelf electronic equipment. The first method used an advanced shield which would be about 65% heavier for the SNAP 10A reactor, but the weight penalty could be lowered to about 10% with reductions in the reactor envelope. The other method used local shielding in the aft rack and required about 40 lb for each of the two equipment trays.

The results of the SNAPSHOT radiation shield behavior are discussed below.

1. Shield Performance

The design objective of the SNAP 10A radiation shield was to limit the radiation dose across a five foot diameter plane 17.5 ft from the reactor to 10^{12} nvt/yr from fast neutrons and 10^7 rad/yr from gamma rays. Although these were the reference conditions, the radiation analyses concentrated on the radiation level at the SNAP/Agena interface, about 8.5 ft closer to the reactor, for the reasons listed below.

1) The reference dose plane was not in an instrument rack area, whereas several radiation detectors were mounted in the interface region.

2) Since the space between the interface and the shield was vacant, uncertainty in estimated radiation level due to equipment attenuation was nonexistent.

3) The outer region of the interface was outside the basic shield shadow, which permitted an evaluation of radiation scattering around a shadow shield.

4) Control devices critical to the operation of the SNAP reactor were located at the interface.

5) Maximum dose rates in the Agena vehicle occurred at the SNAP interface.

2. Neutron Fluxes

Figure 28 compares the preflight and postflight analyses of the fast neutron flux at the SNAP 10A/Agena interface with the recorded data from LMSC detectors RA1-1 and RA1-2 and AI detectors 166 through 172. As mentioned earlier, the data from the LMSC detectors was taken as the reference. The maximum deviation between the measured data and the preflight analysis was a factor of six underestimate at the center of the interface. Since both preflight studies^(20, 21) showed the major contributor to the flux at this location is scattering off the thermoelectric converter, this component was reviewed in detail in the post flight analysis.

Scattering from the thermoelectric converter was proportional to the incident neutron flux and the scattering properties of the converter materials. It

Figure 28. Fast Neutron Flux at SNAP 10A/Agna Interface

was found that the incident flux had been underestimated because of a poor approximation in the analytical model used in the SCAR computer program. The SCAR program is a ray-tracing code used to compute the scattering from thin conical surfaces concentric with the reactor axis. Hand calculations and results from the RATRAP program, a general ray-tracing code, confirmed the error in the SCAR model. Additionally, the analytical model used for the converter underestimated its true mass. The combination of these two factors increased the predicted scattering off the converter from 4.6×10^{12} nvt/yr to 6.2×10^{13} nvt/yr. Since the reactor was operating at about 40 kwt rather than the 34 kwt used in the preflight analysis, the revised flux due to TE converter scattering alone would be 7.3×10^{13} nvt/yr. An analysis of scattering from the converter heat shield, to be discussed below, indicated that scattering in the reflector had been overestimated by a factor of 1.7. Since over 90% of the neutrons which scatter off the converter have previously scattered in the reflector, the TE converter scattering can be reduced by a factor of 1.7 to a flux of 4.3×10^{13} nvt/yr recorded by detector RA1-1.

6-14-66

7561-03206

Figure 29. Effect of Heat Shield Ejection on Normalized Radiation Levels at SNAP 10A/Agena Interface

Concurrent with the study of TE converter scattering, an investigation was made of an abrupt decrease in the reading from the fast neutron detector RA1-1, at the center of the SNAP/Agena interface, simultaneously with the ejection of the converter heat shield. This phenomenon was not noted at the fast neutron detector near the Agena skin nor at the gamma detectors at either location. Analysis showed that the radiation levels for these detectors were less affected by scattering from the heat shield. Since heat shield ejection occurred during a power transient, the detector data had to be normalized to a constant power to note any abnormality. Figure 29 shows the normalized data. Before heat shield ejection, detector RA1-1 indicated about 2.2×10^{12} n/cm²-yr-kw. After ejection, it indicated about 1.3×10^{12} n/cm²-yr-kw. The difference, 0.9×10^{12} n/cm²-yr-kw, was attributed to heat shield scattering. An analysis using the SCAR scattering code gave 1.5×10^{12} n/cm²-yr-kw, a factor of 1.7 higher. Since more than 90% of the neutrons incident on the converter heat shield had previously scattered in the reflector, this result empirically indicated that reflector scattering had been overestimated by a factor of 1.7.

Figure 30 compares the preflight and postflight analyses for the fast neutron flux at the SNAP/Agena interface due to scattering from the thermoelectric

6-14-66

7561-03207

Figure 30. Comparison of Postflight and Preflight Estimates of Fast Neutron Flux at SNAP 10A/Agena Interface Due to TE Converter, Reflector and Control Drums

converter and the reflector which are discussed above. Also compared is the estimate of scattering from the projecting corner of the control drums for an axis directly below the drums. The estimated control drum scattering was lowered in the postflight analysis because of the reduction in reflector scattering and a reduction in the quantity of beryllium taken for the projecting corner. In the preflight analysis, the drums were assumed to have two 1/8 in. shims attached and to be rotated 22° from full-in. The flight configuration had no shims and, at steady-state operation, the drums were about 16° from full-in. Consequently, the estimated control drum scattering was generally reduced by a factor of 3 to 5 but as much as a factor of 10 in some places. Figure 30 also indicates the reason for the installation of many AI fast neutron detectors in the SNAP instrument compartment. From the preflight analysis, scattering from the control drums—assumed to be the major contribution to the fast neutron flux in the instrument compartment—increased between two and three orders-of-magnitude in going from the inside to the outside of the compartment. Besides this radial variation in flux there was significant angular variation, which is not shown. The maximum radial variation of total flux recorded by the AI detectors was a factor of three between instrument nos. 168 and 166. The maximum angular variation was a factor of two between instruments 166 and 169. Instrument 166 was directly under a drum which was slightly out. The expected flux variation did not materialize because of the combination of (1) an increase in the scattering from the thermoelectric converter, which had essentially no radial nor angular variation, and (2) a decrease in the scattering from the control drums, which did have radial and angular variations.

Figure 31 summarizes the postflight analysis of the fast neutron flux at the SNAP 10A/Agena interface and shows the relative contributions from the major system components. The flux in the inner region of the plane was controlled by scattering off the thermoelectric converter. The flux in the outer region was controlled by scattering from the reflector. The postflight analysis fully agrees with LMSC detector RA1-1 at the vehicle centerline and is about a factor of two higher than detector RA1-2 at 26 in. off the centerline.

The preflight and postflight analyses of the fast neutron flux in the Agena vehicle are compared with the data from LMSC detectors RA1-2 through RA1-5 in Figure 32. Two preflight estimates are shown: (1) the study by AI,⁽²⁰⁾ which

6-14-66

7561-03208

Figure 31. Contributions to Fast Neutron Flux at SNAP 10A/Agna Interface

6-14-66

7561-03209

Figure 32. Fast Neutron Flux in Agena Vehicle (26 in. off Centerline)

covered the area between the interface and the reference dose plane without taking into account attenuation by Agena equipment, and (2) the study by General Dynamics for the USAF,⁽²¹⁾ which covered the entire Agena vehicle taking into account attenuation by equipment. With the analytical model that was developed for the postflight analysis of fast neutron flux in the interface region and the Agena attenuation factors developed by General Dynamics, the calculated flux in the Agena agreed very well with the measured flux except for the Agena aft rack region. A revised attenuation factor for the aft rack was empirically derived by fitting the postflight analysis results to the measured data. These attenuation factors are shown in Figure 32. The flux at the reference dose plane was 5×10^{12} nvt/yr which is a factor of five greater than the design objective of 10^{12} nvt/yr.

The shielding required to satisfy the design objective of 10^{12} nvt/yr at the reference plane was estimated. Figure 33 compares the flux values below an advanced shield and below the SNAPSHOT shield. In the advanced design, the shield covers the entire reactor envelope to eliminate scattering around the shield from reactor components. In addition, the thermoelectric converter is entirely within the shield shadow. This shield is approximately 22 in. thick as compared to about 25 in. for the SNAP 10A shield. The weight penalty over the SNAPSHOT shield is about 140 lb. However, recent studies have shown that a significant reduction in the reactor envelope can be achieved. With the smaller envelope the weight penalty over the SNAPSHOT shield is about 10%, or 230 lb compared to 217 lb. Since lithium hydride is not a very good gamma shield material, the gamma dose below the advanced shield designs should not vary significantly from SNAPSHOT values. Note also in Figure 33 that in the Agena aft rack region, 30-35 ft from the reactor, the fast neutron flux would be about 10^{11} nvt/yr. At this level radiation hardening of semiconductor equipment is not required (see Figure 36, this section).

A study was conducted using local shielding in the Agena aft rack to lower the flux in that region to 10^{11} nvt/yr to permit the use of unhardened electronic equipment. Levels of 7.2×10^{11} nvt/yr and 1.8×10^{11} nvt/yr were recorded at the forward and aft ends of the rack, respectively, during the SNAPSHOT test. It is estimated that about 40 lb of lithium hydride per tray in the aft rack would be required.

Figure 33. Fast Neutron Flux Along SNAP 10A/ Agena Vehicle

Figure 34. Gamma Dose Rate at SNAP 10A/ Agena Interface

3. Gamma Fluxes

The gamma dose rate at SNAP 10A/Agena interface is compared to the AI and General Dynamics preflight predictions in Figure 34. Since the preflight estimates were less than a factor of two higher than the measured values, the adequacy of the analytical models was confirmed by the SNAPSHOT test.

The preflight and postflight analyses of the gamma dose rate in the Agena vehicle are compared with the data from LMSC detectors RA3-2 through RA3-5 in Figure 35. Two preflight estimates are shown: (1) the AI study,⁽²⁰⁾ which covered the area between the interface and the reference dose plane without taking into account attenuation by Agena equipment, and (2) the study by General Dynamics for the USAF,⁽²¹⁾ which covered the entire Agena vehicle taking into account attenuation by equipment. With the preflight analytical model results normalized to the flight data in the SNAP 10A/Agena interface region and the Agena attenuation factors developed by General Dynamics, the dose rate in the Agena agreed very well with the data from the detectors, again except for the Agena aft rack region. A revised gamma attenuation factor for the aft rack was empirically derived using the same procedure as for fast neutrons. These attenuation factors are shown in Figure 35. The dose rate at the reference dose plane was 7×10^5 r/yr, a factor of 14 below the design objective of 10^7 r/yr. This confirmed the preflight studies which showed that specific gamma shielding was not required in this system.

4. Payload Radiation Effects

Payload radiation hardening—the development of payloads which will perform reliably in a nuclear radiation environment—may be categorized into four dose ranges according to the effort required and the functional limitations imposed. This classification is based primarily upon the performance of sensitive payload electronics and, in particular, upon the semiconductor circuits.

Below 10^{11} nvt or 10^5 rad, no consideration of reactor radiation damage is required for payloads which have been designed to withstand the natural space radiation environment. In this sense off-the-shelf components may be used below these levels. Up to 10^{12} nvt or 10^6 rad, the necessary hardening may be performed by expert inspection of design drawings or parts lists. No testing is required and adequate substitutes for all sensitive components are readily

6-14-66

7561-03212

Figure 35. Gamma Dose Rate Along Agena Vehicle (26 in. off Centerline)

Figure 36. Radiation Hardening Classification

available. In the range above these levels, the choice of useful devices and circuits becomes increasingly restricted. Above about 5×10^{13} nvt or 5×10^7 rad, few standard components may be used and a complete series of design and qualification tests is necessary. Figure 36 summarizes the preceding discussion and shows the maximum level at which typical components and circuits can be used. The SNAPSHOT payload tolerance level, indicated on this figure, refers to the upper test limit for reactor control and diagnostic instrumentation mounted in the SNAP 10A instrument compartment at the SNAP/Agena interface.

Based upon the foregoing analysis, the options for equipment location on a SNAPSHOT vehicle are shown in Figure 37. For comparison, a vehicle with an advanced shield, designed to reduce the large scatter dose observed in SNAPSHOT, is also shown. In each case, the arrows designate the farthest forward location permissible for the equipment type. For example, inspection hardened long term telemetry could be placed at the rear of the forward rack in the advanced shield vehicle, while it must be placed in the aft rack in SNAPSHOT.

A portion of the aft rack is available for off-the-shelf equipment in the advanced shield design. This can be enlarged to include one or more complete trays by the addition of sufficient local shielding to reduce the input dose to 10^{11} nvt.

6-14-66

7561-03214

Figure 37. Equipment Location Options on SNAPSHOT Vehicles

At the time of reactor shutdown both the fast neutron and gamma doses were well below the minimum levels to which the SNAP 10A and Agena radiation hardened equipment had been tested. Hence, no degradation or failure of hardened equipment due to radiation was expected, and, in fact, none was observed.

However, both Link 1 telemetry and the horizon sensors of the basic guidance and control system, which were unhardened, functioned without apparent degradation until the ascent batteries supplying them with power finally ran down on orbit 109. Because these equipments were both located at the front of the Agena where they were exposed to high radiation fluxes, an analysis was made to determine whether the equipment had shown an unexpected ability to resist radiation.

Link 1 telemetry was located in the forward auxiliary equipment rack just ahead of the forward section interface and had received a fast neutron dose of 3×10^{11} nvt at the time of battery run down. The horizon sensors were located on the outer perimeter of the forward section and received a dose of 8×10^{11} nvt.

The sensors, in contrast to the signal conditioning and control portions of the guidance system, would not be sensitive to these doses. Semiconductor analog circuits in the Link 1 telemetry, on the other hand, might be predicted to degrade. The symptom of this degradation would be a reduction of signal strength, particularly in the output stages, which is similar to the effect of reducing the power supply voltage.

Effectively, analog circuits which are designed to operate over a wide range of supply voltage would be less sensitive to radiation damage. The Link 1 telemetry, which demonstrated an ability to operate with only 82% of design voltage, is an example of such a circuit. Except for circuits with components which experience catastrophic failure under irradiation, such as unijunction transistors, design which incorporates a wide latitude for subcomponent parameter variation may be considered partially radiation hardened. Doses at which this approach becomes ineffective, and for which special design procedures are required, are 5×10^{12} nvt and above. Unfortunately, the dose received by the Link 1 telemetry at the time of battery rundown was not large enough to demonstrate that it had successfully used such an approach.

The conclusions and recommendations resulting from the analyses are discussed below.

1) The SNAP 10A radiation shield basically performed as expected. The flight test demonstrated that for low power, high dose tolerance applications analyses based on ray-tracing methods and techniques are adequate.

2) The SNAPSHOT test confirmed the results of analyses which indicated that fast neutron fluxes below the shield are controlled by scattering processes from partially-shielded reactor and system components rather than by direct line-of-sight radiation from the reactor core. Consequently, lithium hydride as a shielding material was at least as effective as predicted. In addition, for applications requiring lower dose rates, reactor and system components must be fully shielded to prevent scattering around the shield.

D. POWER CONVERSION

The SNAP 10A thermoelectric converter-radiator converts a portion of the heat supplied by the reactor directly to electricity. The converter was designed to produce at least 500 w continuous electric power throughout the one-year mission life.

The converter consists of N and P doped SiGe thermoelectric material thermally coupled but electrically isolated from the NaK heat transfer medium. These SiGe pellets are bonded to 40 parallel, NaK carrying, D-shaped stainless steel tubes arranged in a conical configuration about the structure (see Figure 77 Section IV.N.) Along any one tube or leg there are 72 SiGe thermocouples or a total of 2,880 individual power-producing elements. These pellets are electrically isolated from the hot NaK tube by thin alumina discs. The heat transport path is from the NaK tube through the insulator, through the SiGe pellet, to an individual aluminum radiator.

The electrical current flows through N and P SiGe pellets connected alternately at their hot junctions by a copper hot strap and at the cold junctions by the aluminum radiators. All 72 pellets on a tube are connected electrically in series. To increase reliability, the series string of couples (pairs of N and P pellets) on each NaK tube is connected electrically with the adjacent string at every couple. These adjacent series-parallel connected modules are then

connected in series and comprise the complete converter electrical network. Converter design and testing are described in detail in Reference 22. Converter performance was analyzed from the following data:

- 1) Converter load voltage.
- 2) Converter load current.
- 3) Converter internal resistance. (This also gave an indirect measurement of instantaneous open-circuit voltage.)
- 4) NaK and radiator temperatures.
- 5) Leakage current to ground.
- 6) Converter voltage profile.

The thermoelectric converter met the design requirements during the flight test. The flight test data were essentially identical to converter ground test and module qualification test data adjusted for flight test conditions with the exception of isolation resistance degradation. It was concluded that the degradation of isolation resistance was due to deposition on the alumina insulators of hydrocarbons from insulating materials used in the instrument compartment and Agena and to the high temperature wire inside the supporting structure, which tended to outgas at operating temperatures in the hard space vacuum. The leakage current at the time of shutdown was 0.5 amp representing a power loss to ground of approximately 10 w. The extrapolated gross power output at 90 days was 500 w. The extrapolated power loss to ground at 90 days was estimated to be 35^{+0}_{-12} w. At one year, the extrapolated gross power output was 460 w. The power loss to ground was estimated to be 54^{+128}_{-22} w, depending on the leakage model used for the extrapolation.

Converter performance, degradation, and voltage profile are discussed below.

1. Performance

Converter power output during startup and the early phase of active control is presented in Figure 38. With the exception of the precise control drum step timing, the converter behaved as predicted during startup and also during active control. Initial full power output was lower than predicted because the predicted power did not take into account the low initial control temperature

Figure 38. Converter Power Output During Startup and Early Part of Active Control

switch set point. Behavior during the remainder of the active control period was similar to that shown.

Expected and measured converter voltage, current, and internal resistance are presented in Figure 39. Neither voltage nor resistance was affected by sun/shade changes. Converter voltage was held nearly constant by the regulator, and resistance was insensitive to the small temperature changes. The converter voltage was 0.5 v higher than expected. This could have been due to a higher than expected voltage regulator set point and/or a larger than expected line resistance between the converter and regulator. Since telemetry data indicated that the regulator output voltage was 28.7 v, or 0.5 v lower than expected, the most probable explanation was the uncertainty in the line resistance. A line resistance of 0.08 ohms would have been required to account for the observed difference between converter and regulator output voltages compared to a predicted resistance of 0.03 ohms.

The expected current shown in Figure 39 was too high because the actual voltage was higher than predicted. Using the measured voltage instead, the expected current would have been lowered by 0.3 amps and would more nearly fit the sun data.

The internal resistance was initially slightly higher than expected. This was attributed to slight differences between the FS-3 and FS-4 converters.

6/10/66

7561-03223

Figure 39. FS-4 Converter Voltage, Current, and Resistance During Passive Control Period

Converter power output measured in space is compared to the expected power output based on FS-3 nuclear ground test data in Figure 40. Agreement between the expected power for sun conditions and FS-4 power output measured in the sun was good. In order to calculate the expected FS-4 power output in space, it was necessary to (1) correct FS-3 data to the expected FS-4 average NaK temperature, (2) extrapolate converter performance to space, and (3) correct for the fact that the FS-4 converter would not be operating at matched load. These calculations are described below.

6-10-66

7561-03224

Figure 40. Comparison of FS-3 and FS-4 Power Output During Passive Control Period

6-10-66

7561-03226

Figure 41. FS-3 Converter Degradation at Constant Average NaK Temperature

Converter power output is especially dependent on the average NaK temperature. Test data from FS-3 and other ground tests indicated that the converter power output variation with temperature at matched load followed the relationship

$$P_{ML} = g(t) T^{4.8}$$

where

P_{ML} = matched load converter power output

$g(t)$ = coefficient which is time dependent due to the time dependence of converter degradation (see Figure 41)

T = absolute average reactor temperature

This expression was used to adjust the FS-3 measured power output to the FS-4 expected reactor average temperature shown in Figure 25. The required adjustment is presented in Figure 42. It was time dependent because of the difference in the FS-3 and FS-4 temperature drifts. It was not necessary to evaluate the time dependent coefficient $g(t)$ since the measured FS-3 power output already included converter degradation.

Figure 42. FS-3 Power Adjustment to FS-4 Test Conditions

To extrapolate ground test results to space conditions, it was necessary to correct for changes in effective emittance and effective sink temperature as described below. The combined effect of these corrections was to increase matched load power in space by 11.3% for the same average converter temperature. The effect of these same factors on reactor thermal power and system heat losses is described in Section IV. M.

The effective emittance of the converter radiator is reduced on the ground because some of the heat rejected by the converter-radiator is reflected from the chamber wall back to the radiator. This was accounted for by defining an effective emittance of the radiator according to the following well-known expression:

$$\mathcal{F}_{R-C} = \frac{1}{\left(\frac{1}{\epsilon_R} - 1\right) + \frac{1}{F_{R-C}} + \frac{D_R}{D_C} \left(\frac{1}{\epsilon_C} - 1\right)}$$

Where

$\mathcal{F}_{R-C}$ = effective radiator emissivity from radiator to chamber wall

ϵ_R = 0.90 = actual radiator emissivity

D_R = 2.9 ft = average conical-shaped radiator diameter

D_C = 8 ft = average chamber diameter

ϵ_C = 0.85 = chamber wall emissivity

F_{R-C} = 0.99 = geometric view factor from radiator to chamber wall

Substituting these values, the effective FS-3 radiator emissivity was 0.85.

In the region of the SNAP-10A converter design point, near 20% Carnot efficiency, converter power output is directly proportional to the radiator heat rejection capability. Therefore, the adjustment to FS-3 power output to account for chamber wall emittance was given by,

$$F_{\epsilon} = \frac{\epsilon_R}{\mathcal{F}_{R-C}} = \frac{0.90}{0.85} = 1.06$$

This factor is presented in Figure 42 and represents a 6% increase in power output in the space environment.

The effective sink temperature for FS-3 was the average vacuum chamber wall temperature of 123°F. The effective sink temperature in space averaged 0°F in the sun (Section IV. M). The correction was calculated by the following expression which compared the heat rejection capability of the radiator in space to that in the FS-3 chamber,

$$F_{ST} = \frac{(T_R^4 - T_{SP}^4)}{(T_R^4 - T_{CH}^4)}$$

where

F_{ST} = power output adjustment for sink temperature

T_R = 650°F = 1110°R = typical average radiator temperature

T_{SP} = 0°F = 460°R = effective space temperature (sun)

T_{CH} = 123°F = 583°R = average FS-3 chamber wall temperature

This expression avoids a complete converter heat balance but is accurate only near 20% Carnot efficiency. Substituting in this expression,

$$F_{ST} = 1.05 \quad ,$$

indicating an additional 5% power output increase in space.

The FS-3 test was conducted with the converter at very nearly matched load conditions, and the adjustments that have been defined refer to matched load conditions. The following expression describes the adjustment in power output for operating at other than matched load. The specific case considered below is for the start of the passive control period.

$$F_{OD} = \frac{4V_L (V_{oc} - V_L)}{V_{oc}^2}$$

where

F_{OD} = ratio of actual power to matched load power

$V_L = 29.8 \text{ v}$ = expected converter output voltage in space

$V_{oc} = 68.6 \text{ v}$ = expected steady-state converter open-circuit voltage in space.

Substituting,

$$F_{OD} = 0.983$$

This corresponds to a power loss of 1.7% which decreased slightly with time as indicated in Figure 42 due to a decreasing steady-state open-circuit voltage.

The total adjustment factor for 29.8 v presented in Figure 42 combines all the individual factors discussed above and was applied to the measured FS-3 power to obtain the expected FS-4 power output shown in Figure 40.

Converter power output degradation in space was expected to be the same as that observed on FS-3, which in turn was similar to that observed during thermoelectric module testing and non-nuclear system tests. The good agreement between FS-4 results and FS-3 results extrapolated to space demonstrated that power degradation in space and on the ground were the same. Figure 41 shows FS-3 converter degradation at a constant average NaK temperature to eliminate the effect of reactor average temperature drift on converter power output.

As indicated, the initial power was based on the power output at 72 hours plus 1.5%. This approach was taken because of system temperature transients in the first 72 hours of operation at full power. To illustrate this, Figure 43 shows the average temperature drop across the thermoelectric elements normalized to a constant average NaK temperature for both FS-3 and FS-4. The data show an abnormally high ΔT during the first few days of testing which was not observed during module qualification testing. Since power output is proportional to the square of ΔT , FS-4 and FS-3 power outputs were abnormally high during the first few days. The high ΔT occurred because of the large heat capacity of the lithium hydride radiation shield and, in the case of FS-3, the

Figure 43. Average TE Element ΔT

large heat capacity of the test chamber. Approximately 72 hours were required to achieve both radiation shield and vacuum chamber temperature equilibrium. In addition, control drum steps during the active control period caused temporary increases in the normalized thermoelectric ΔT . Since the module qualification tests did not exhibit these temporary ΔT variations, initial power output was based on the relatively stable 72-hr data plus 1-1/2% which was the observed degradation in the first 72 hours of module qualification tests. The validity of this approach was verified on FS-3 where power output normalized to a constant ΔT showed 1-1/2% degradation in 72 hours.

The evaluation of FS-4 converter degradation was complicated by the uncertainties in the precise effect of the sun/shade environment and telemetry data errors. Since there was good agreement between the measured and the expected power output based on FS-3, Figure 41 is considered to be an accurate representation of FS-4 converter degradation.

2. Isolation Resistance Degradation

During the flight test, the converter leakage current was determined by measuring the difference between the converter output and return currents using a differential current transducer as shown in Figure 44. The telemetry data first indicated a continuous linear increase in leakage current beginning between 600 and 700 hours after reaching full power as shown in Figure 45. The calculated standard deviation about the zero reading is identical to the total random telemetry error standard deviation. This provides strong verification that the leakage did not begin to increase prior to 600 hours. The indicated zero value of 0.052 amps, equivalent to an isolation resistance of 300 ohms, is believed

6-10-66

7561-03228

Figure 44. Measurement of Converter Resistance to Ground in Flight System

6-10-66

7561-03229

Figure 45. Measured FS-4 Leakage Current

caused by a bias error rather than a constant leakage current and was subtracted from the experimental data after 614 hours to give a leakage current corrected for bias. All values given for power loss in this report, unless noted otherwise, were calculated from models based on these corrected data.

Possible explanations for the observed increase in leakage current fall into three general categories:

1) Faulty Current Transducer

Theoretical analysis of the current transducer circuitry indicated that degradation of certain components could result in a linear increase in the indicated leakage current. However, extensive ground testing of the transducer under simulated space environment conditions including radiation, temperature and vacuum resulted in the conclusion that this mode of failure was extremely unlikely.

2) Shorts in the Converter Wiring or Attached Instrumentation

The occurrence of a point short would most likely have resulted in a step change in leakage current rather than the observed linear increase. Further, such a short would have been expected to change the converter voltage profile, and the flight data did not indicate such a change.

3) Degradation of the Exposed Converter Insulators

The converter was electrically isolated from the NaK tube by means of an 0.020-in. thick alumina insulator at each of the 2880 thermoelectric elements. The converter isolation resistance was therefore dependent on the parallel resistance of the 2880 alumina insulators, all of which were exposed to the system environment. The observed behavior of the flight system leakage current was similar to that observed during ground system and module tests. During ground testing of FSM-4 and FS-4, the isolation resistance at operating temperature decreased significantly in a few hours (to less than 50 ohms on FSM-4). Measurements made on these systems established that the reduction in isolation resistance was due to a decrease in the resistance of the individual alumina insulators, which had changed from an initial average resistance of approximately 10^8 ohms to about 3×10^5 ohms during operation. As shown in Figure 46, the behavior observed in

Figure 46. Comparison of Module Test Data with FS-4 Isolation Resistance Behavior

flight was also consistent with observations made in module tests where the modules had been exposed to oxygen.

From the results of system ground tests and the fact that the resistance degradation curve for the flight system was similar to that obtained on module tests, it was concluded that the increase in leakage current during the flight was due to degradation of the individual alumina insulators.

The possible mechanisms by which the converter could have degraded are listed below:

- 1) Distributed deposits of metal oxides or whiskers on the insulators.
- 2) Unexpected effects resulting from the space environment, such as erosion or sputtering due to micrometeorites, solar emissions, Van Allen belt radiation, or high vacuum.
- 3) Distributed deposits of hydrocarbons.

Reference 22 discusses the experimental and theoretical evidence for each of these mechanisms based on both ground and flight test experience and concludes that

"...although it has been impossible to identify carbon on the alumina insulators by direct analytical methods, sufficient

experimental work has been done to allow the conclusion that the degradation was due to the deposition of outgassed hydrocarbons on the exposed surfaces of the insulators and that none of the other mechanisms are probable..."

The hypothesized mechanism for deposition of hydrocarbons is described as follows: At the operating temperature hydrocarbons pyrolyze to form a conducting but initially discontinuous carbonaceous film. As this process of deposition and pyrolysis continues, the film becomes thicker and more continuous resulting in a reduction of the insulator resistance. This mechanism is supported by experiments which show (1) degradation does not occur in the absence of hydrocarbons, (2) that the degradation can be produced by deliberately introducing hydrocarbon contaminants (see Figure 46), and (3) that carbon atoms can be identified on an alumina substrate heated in a hydrocarbon contaminated system.

An analytical model of the carbon film theory was developed to predict the resistance of the conductive deposits, or carbon film, as a function of time. The resulting equation for film resistance (R_F) is shown in Figure 47. Also shown is the electrical circuit equation for resistance to ground (R_G , which is the parallel resistance of R_F and R_I , the insulator resistance). This model is based on the theory of ultra-thin conducting films which are generally discontinuous and consist of a plane array of finite, but very small particles. The conduction process in such an "island" structure consists of charges moving from one particle to the next by tunneling. The constants in the equation represent the following parameters:

R_0 = Minimum film resistance occurring at infinite time,

G = Time at which 1% power loss occurs,

N_0 = Time at which film begins conducting.

Also shown in Figure 47 is a simpler model which assumes that the film does not conduct until it becomes continuous. After that point, the film thickness, hence conductance and leakage current, increase linearly. The constant A represents the rate of conductance increase.

The two curves shown in Figure 47 were taken from Reference 22 and the constants used are not those which give the best fit to the FS-4 data. However, they are close enough to qualitatively illustrate the important differences between the two models:

6-10-66

7561-03231

Figure 47. Fit of Two Isolation Resistance Degradation Models to FS-4 Data

1) Both models can be made to fit the data equally well since there were less than 400 hr of measurable leakage in flight. This is also shown in Figure 45. Hence, it was not possible to prove which model was more nearly correct using flight test data.

2) The carbon film theory predicts that isolation resistance approaches an asymptotic value. As a result, more than half the power loss which would take place in one year is predicted to occur prior to 90 days of operation. For SNAPSHOT this meant that even after one year the power loss to ground would have been small enough so that there would have been an excess of power supply over demand. For this reason the carbon film model, although justified by theory, is considered to be optimistic.

3) The linear conductance increase model does not level off. As a result, the power loss at the end of a year would have been large and after several years it would have been greater than the power output. Since this is an impossible situation, the model is considered to be pessimistic, and was used to estimate an upper limit for power loss to ground.

The probable source of the carbonaceous material deposited during the flight test was determined by analysis of results obtained from the ground system tests and specially designed component tests. Extensive analysis of deposits found in the vacuum chambers used for system ground tests revealed that many of the insulating materials used in the instrument compartment and the high temperature wire inside the SNAP 10A structure contained hydrocarbons which outgassed at operating temperatures. Prior to flight, the slots providing a path between the inside of the structure and the converter insulators were covered in an attempt to prevent the hydrocarbons from the instrument compartment and the high temperature wire from depositing on the insulators. It was estimated that the covers would leave an opening equivalent to no greater than an 0.06 in. diameter capillary tube and as a result the time required for a 1% power loss to ground would be 10,000 hr or more (Figure 48).

Subsequent tests have shown that the slot covers were ineffective. The experimental results indicate that the slot covers left an opening equivalent to an 0.125-in. diameter capillary tube. Assuming the outgassing rate predicted from tests performed prior to the flight (20 mgm/hr) the 1% power loss point would be predicted to be about 1622 hours (Figure 48), compared to 835 hours calculated from the flight data (Figure 49). Considering all the possible sources of error in the prediction, this is relatively good agreement and supports the conclusion that the source of contamination during the flight test was material inside the structure.

It is necessary to qualify the above conclusions because most of them are based on ground test results and it is conceivable that space environmental conditions are sufficiently different to invalidate them, though, there was no evidence to suggest this.

Using the measured flight test data shown in Figure 45, it was possible to evaluate the constants in the two converter leakage models and to extrapolate the calculated resistance and power loss at 43 days to 90 days and one year. A selection of the results is shown in Table 21. The postflight prediction discussed above and the experimental fit of Reference 22 (which was plotted in Figure 47) are shown for information only. The linear conductance model shown gives the pessimistic upper limit on power loss as discussed above and shown in Figure 49. The carbon film model fitted to the data corrected for bias

Figure 48. Effect of Equivalent Slot Cover Diameter and Outgassing Rate on Converter Isolation Resistance Degradation Rate

Figure 49. FS-4 Converter Isolation Resistance and Power Loss to Ground

gives the optimistic lower limit on power loss as discussed above and shown in Figure 49. The carbon film model which gives the largest estimates of power loss and still results in a fit to the data which is probable (but not the best probable fit) has been arbitrarily used to calculate the most likely values of power loss for this report.

Using the most likely and upper limit values of power loss to ground, vehicle power margins were calculated for a number of possible conditions at various times in Table 22. It can be seen that, in most cases, power margin was positive and there would have been no restriction on vehicle power consumption. Under conditions of upper limit power loss to ground some curtailment of power

TABLE 21
CONVERTER ISOLATION RESISTANCE EXTRAPOLATIONS

Model	Constants Used			Isolation Resistance (ohms)			Power Loss (w)		
				43 days	90 days	1 year	43 days	90 days	1 year
Least Squares Fit to Data	No	G							
As Measured ⁽¹⁾	618	814		30	9	2	10 ± 1	36 ± 2	183 ± 13
Corrected for Bias ⁽²⁾	618	835		34	9	2	9 ± 1	35 ± 2	182 ± 13
Linear Conductance Increase ⁽³⁾	No	G	A						
Best Fit (Corrected for Bias) ⁽²⁾	618	835	1.353×10^4	34	9	2	9	35	182
Carbon Film Theory	No	G	Ro						
Post-Flight Prediction (Ref. 22)	713	1622	10	2200	31	12	0.2	10	25
Experimental Fit (Ref 22)	397	945	7.35	34	13	8	9	24	37
Best Fit (Data as Measured) ⁽¹⁾	375	814	8.64	32	14	10	10	22	32
Best Fit (Corrected for Bias) ⁽²⁾	420	835	8.45	35	14	9	9	23	33
Fit Giving Largest Probable Loss ⁽²⁾⁽⁴⁾	520	855	5.10	29	9	6	11	35	54

NOTES: 1) Model shown fitted to experimental data in Figure 45.

2) Figure 45 shows the zero reading was biased by 0.052 amps. To correct for bias 0.052 amps was subtracted from the experimental data.

3) This model is exactly equivalent to a linear least squares fit of the data. Only the mathematical form of the equation is different.

4) Using the estimates for total telemetry random error, the experimental data were adjusted to their maximum probable limits (95% confidence) so that the resulting model gave the largest power loss to ground.

NAA-SR-11934
144

TABLE 22
VEHICLE POWER MARGINS
(W)

Time	Gross Power Output ⁽¹⁾	Vehicle Power Consumption ⁽²⁾			Power Loss to Ground ⁽⁴⁾		Power Margin ⁽⁵⁾			
		During Active Pass - Typical and Maximum	Between Passes		Likely	Upper Limit	During Pass		Between Passes	
			Typical	Maximum ⁽³⁾			Maximum	Minimum	Maximum	Minimum
43 Days	533	350	265	365	9	9	174	174	259	159
90 Days	500	350	265	365	35	35	115	115	200	100
1 year	460	350	265	365	54	182	56	-72 ⁽⁶⁾	141	-87 ⁽²⁾⁽³⁾

NOTES: 1) Power output in sun from Figure 40.

2) The typical values were substantially higher than the minimum possible since secondary payloads and Link 3 were operated continuously to reduce the load on the voltage regulator.

3) Assuming ion engine was operated, 100 w would be needed to recharge batteries for several orbits afterward. If necessary, consumption could be reduced by turning off other switchable loads while batteries were being recharged. Under worst conditions, ion engine would not be operated.

4) From Table 21.

5) Power Output minus Loss to Ground and Power Consumption.

6) Consumption could have been reduced by turning off switchable loads (see footnote 2). Negative power consumption could be supported by surge batteries if active passes were not too frequent and if power margin were positive between passes.

consumption would have been necessary. However, power output to the load would have been sufficient to support the minimum required vehicle power consumption under all conditions, and hence would not have resulted in mission failure.

3. Voltage Profile Analysis

The converter voltage profile was analyzed using data from ten voltage taps, each of which measured the output voltage of four converter legs. Details of the analysis and its results are given in Appendix A. Prior to analyzing the flight test data, the voltage profile in space was predicted, taking into account all known significant variables. Figure 50 shows the independent effect of each of the variables. Figure 51 shows the prediction made combining these variables, together with the measured profile. The measured values are the average of data from seven full shade passes before the leakage current became measurable. Measured and predicted values for the two ground test qualification systems are shown for comparison. It can be seen that considerably better agreement was obtained on the ground than in space. This is shown numerically in Table 23 together with estimates of the measurement errors.

The reason for the poor agreement between measured and predicted values in space can be seen when it is noted that the standard deviations of the difference and the measurement error are nearly equal. An investigation of other factors was made to ascertain that the differences between measured and predicted profiles were due to data error, rather than unexpected space effects or converter malfunctions. None of the factors investigated (Appendix A) could account for either the shape or the magnitude of the observed profile.

Because of the large amount of absolute error in the measured profile, it was not possible to verify either the predicted converter flow distribution or the decrease in resistance to ground. However, it was determined that there were no significant changes in the shape or magnitude of the profile between initial full power and shutdown. This indicated a healthy converter, free from open or short circuits or serious (~ 10 ohms) resistance to ground degradation. Had the flight test continued another 15 to 30 days, voltage profile data would have been able to provide verification of the decrease in resistance to ground and the assumption that the decrease was generally distributed throughout the converter (Appendix A).

Figure 50. Effect of Variables on Converter Voltage Profiles (3.02 v Average Voltage for all Cases)

The conclusions and recommendations resulting from the power conversion system performance analysis are presented below.

Converter performance was as expected and in excellent agreement with predictions, with the exception of the decrease in isolation resistance.

Because the flight test did not continue a long enough time to verify the applicability of the carbon film theory model, there remains a substantial uncertainty in the expected magnitude of power loss to ground at one year. It does not appear possible to verify the model by any means other than another flight test.

Figure 51. Comparison of Measured and Predicted Voltage Profiles

TABLE 23
DIFFERENCES BETWEEN MEASURED AND PREDICTED
VOLTAGE PROFILES

	Standard Deviation of Difference Between Measured and Predicted (v)	Standard Deviation of Measurement Error (v)
FSM-4 (Ground)	0.039	0.02
FS-3 (Ground)	0.036	0.02
FS-4 (Space)	0.128	0.120

Under the worst probable conditions, isolation resistance degradation would not have caused mission failure through an inability to supply the minimum required vehicle power consumption at one year.

Since the cause of isolation resistance degradation has been established by extensive ground testing, it is possible to recommend several methods for reducing degradation to acceptable or negligible levels:

- 1) Increase the insulator surface leakage path length to reduce the thickness and increase the length of the conducting contaminant layer.
- 2) Reduce the sources and amounts of hydrocarbons in both the SNAP unit and launch vehicle.
- 3) Protect the insulators by isolating them from the remaining sources of hydrocarbons.

E. THERMOELECTRIC PUMP

Flow during the prestartup period was required to prevent local cold spots which could result in NaK freezing. Flow after reactor startup was required to transfer heat from the reactor to the thermoelectric converter.

The pump was a liquid metal, direct current conduction pump powered by an integrally mounted thermoelectric generator. It was located at the reactor outlet. Included in the pump were an electromagnetic flowmeter, input and measuring terminals for the external battery-supplied current used for prestartup operation, terminals for measuring the generator and pump operating voltage, and two thermocouples and two resistance thermometers for measuring radiator fin base temperature. Performance is analyzed from data from: (1) flowmeter, (2) battery current (during prestartup only), (3) generator and pump terminal voltage, and (4) NaK and radiator temperatures.

Detailed information on the pump design, fabrication, testing, and analysis is contained in Reference 23.

During the flight test, the pump provided the required flowrates with comfortable safety margins, and actual performance was very close to that predicted. The pump behavior was similar to the behavior of substantially identical pumps on ground test, including the pump operating in the FS-3 system.

Figure 52. TE Pump Performance History

None of the data showed any indication of pump malfunction, incipient failure, or flow instability.

The operational history of the pump for the entire flight test is shown in Figure 52. The details relevant to each test phase are discussed below.

1. Low Temperature Operation

Table 24 summarizes the low temperature performance and the expected and minimum required flows. There was excellent agreement between the measured and expected flowrates for all low temperature test phases extending from the system thermal reference test on the ground to just before the generation of sensible heat in orbit. The measured values were all well above the minimum requirement.

The pumping rate on the launch pad prior to the application of battery current was 0.1 gpm. The system had been warmed by hot air and, as a result, the NaK temperature was 6°F higher than the radiator fin temperature. This small temperature difference across the thermoelectric generator produced sufficient power to deliver the 0.1 gpm. This small flow was not sufficient to keep the lower manifold NaK temperatures above the 75°F minimum required for launch when the Agena cooling air was turned on and the SNAP 10A heating air was

TABLE 24
LOW TEMPERATURE PUMP PERFORMANCE

Test Phase	Battery Current, (amps)	NaK Temp. (°F)	NaK to Fin (ΔT , °F)	Flowrate, (gpm)		
				Required ⁽¹⁾	Measured	Expected ⁽²⁾
Pump Acceptance Test	40	100	3	0.26	0.57	--
FS-4 Thermal Reference Test	40	82	2	0.26	0.51	0.50
Launch Pad	53.8	104	6	0.26	0.67	0.80
Orbits 1 and 2 (sun)	44.5	100	20	0.26	0.89	0.80
Orbits 4 and 5 (shade)	42.9	106	40	0.26	1.10	1.0

NOTES 1) Minimum required value as quoted in pump qualification test specification.

2) Pump acceptance test flowrate corrected to measured current, NaK temperature, and ΔT .

turned off. As a result, it was necessary to apply battery current to the pump several times during the countdown to collapse the ΔT across the system and increase the lower manifold temperatures. With current to the pump, lower manifold temperatures ranged from 85 to 95°F.

Within 30 sec after liftoff, and also after the start of Agena second burn, the flowrate decreased to zero although the normal battery current continued to flow through the pump. This decrease was due to displacement of the expansion compensators, during periods of high acceleration (Section IV. F), permitting the formation of a void in the piping above the pump outlet. The break in the NaK continuity resulted in a column of NaK at the pump outlet which was subjected to the high acceleration forces and thereby constituted a higher hydraulic load against which the pump had to work. This load greatly exceeded the pump capacity of about 0.01 psi during this phase of operation, so the flow stopped.

From flight test data taken in the sun and shade (Table 24) it was possible to determine that the dependence of flowrate on NaK to pump fin ΔT was 0.01 gpm/°F at low temperatures. Knowledge of this value is important to the prestartup analysis since it will allow accurate prediction of flow in orbits different from that of SNAPSHOT. Efforts to measure this value on pump and system ground tests prior to the flight were unsuccessful, and the contribution to flow of the increased ΔT in space was considered as an unknown safety factor in the prestartup analysis.

2. Startup

Predicted and measured flowrates during startup were in good agreement, as shown in Figure 53.

3. Full Power Performance

Measured and expected flowrates during both the active and passive control periods are shown in Figure 54. Again, agreement between measured and expected values is excellent, and the measured flow was well above the minimum required. The measured flowrate extrapolated to 90 days was 13.0 gpm, compared to an expected value of 12.8 gpm.

The pump flowrate in space was higher than on the ground for a given pump NaK temperature. There were two reasons for this:

Figure 53. NaK Pump Performance During Startup

Figure 54. TE Pump Performance at Full Power

6-10-66

7561-03239

Figure 55. Pump Degradation at Constant Reactor Outlet Temperature

1) An increase in pump heat rejection capability in space increased the ΔT across the thermoelectric elements. (See extrapolation of converter power and heat losses to space, Sections IV. D and IV. M.) From a heat balance around the pump, it was calculated that this would increase the flow by 2.1%.

2) The lack of buoyant forces opposing flow in space decreased the system pressure drop compared to the same flowrate on the ground. In Section IV. N, it is calculated that this would cause a 0.2 gpm or 1.4% increase in space.

Thus, the total adjustment for space conditions was a 3.5% increase which was included in the expected values in Figure 54.

To show true pump degradation without the effects of changing reactor outlet temperature, flight test data were normalized to 1040°F. These normalized data are compared to the expected degradation curve for ground test pumps in Figure 55. It can be seen that there was no difference between pump degradation in space and on the ground.

One technique which has proven useful in detecting defects, such as thermoelectric element bond separations occurring during pump life and qualification

tests is plotting terminal voltage as a function of flowrate. The terminal voltage is the voltage input to the conduction pump and is also the output voltage of the thermoelectric generator at the pump load. The theoretical relation, derived in Reference 23, is shown in Figure 56.

Changes from normal pump behavior during pump component tests are indicated by this technique as either step changes from the theoretical line or gradual deviations. The first instance represents a sudden change in the pump itself or the hydraulic system (e. g., flow instability caused by gas bubbles) and the second represents an unusual pump degradation.

On SNAP 10A system tests, it was found that normal data fell on a line parallel to and below the theoretical line. Deviations from this parallel line indicated non-normality of pump behavior. Flight test data are also shown in Figure 56, together with statistical 95% confidence limits parallel to the theoretical line. The confidence limits enclose nearly all the data, indicating that the observed data scatter can be adequately explained by random telemetry error. Also, there was no trend of the later data (lower flowrate and lower terminal voltage) to fall further to one side of confidence limit lines, indicating that there were no unusual long-term changes in the pump. It was also determined that there was no tendency of sun and shade data to deviate one way or another from the average. It is thus indicated that the theoretical model properly takes into account changes in thermal environment, a point which was not demonstrated by the pump tests which were operated at constant temperature.

In summary, it can be stated that the flight test results indicated no deficiencies in either pump design or performance. There was no evidence of malfunction or flow instability. Information of value to the prestartup model was obtained on the low temperature response of the pump to solar inputs. This information will be of most value in determining whether the pump batteries are unnecessary for all orbits.

F. EXPANSION COMPENSATOR PERFORMANCE

SNAP 10A uses two identical expansion compensators (ECU) to (1) accommodate the net volume expansion of the NaK that occurs due to the temperature increase between launch and full power operation, and (2) maintain a void-free NaK system under a sufficient pressure to ensure that neither boiling nor pump cavitation will occur. Each ECU is an all-welded metal bellows connected to a

6-10-66

7561-03240

Figure 56. Relationship Between Flow and Terminal Voltage

coolant return line near the bottom of the system via a static branch line of small diameter. For increased reliability, each ECU has a secondary NaK containment housing which permits continued system operation at full power in the event of a primary bellows failure. Under normal conditions, the net volume expansion is distributed equally between the two ECUs. Detailed information describing the ECU design and testing can be found in Reference 24.

Each flight ECU was instrumented with (1) a position indicator to measure the amount of bellows deflection, and (2) a microswitch which actuated to send a signal to the malfunction and failure detection system whenever the bellows was deflected less than 0.24 inches.

The operation of both expansion compensators during all phases of the flight test satisfied all the system design requirements. The deflection positions and relative deflection positions of the ECUs were normal prior to launch (including time zero), during launch acceleration loading, orbital prestartup, startup, and full power operation.

None of the data showed any indication of NaK leakage.

During the zero-g condition prior to the system startup temperature transient, the measured compensator deflection was 0.08 in. greater than the predicted 0.14 in. The most probable explanation for the additional deflection is the presence of 1 to 1-1/2 in.³ of gas, at NaK loading conditions, which was trapped when the system was sealed off after NaK loading. There was no evidence that this small amount of gas, ~0.15% of the total system volume, affected either the pump or the converter flow distribution.

The operational history of the expansion compensators for the entire flight is shown in Figure 57. Performance during each test phase is discussed below.

1. Launch

The deflection of the ECUs on the launch pad was normal. The indicated deflection was 0.14 in. per ECU at an average NaK temperature of 97°F, an increase of 0.04 in. and 20°F from the position and temperature at seal-off after NaK loading. According to calculations based on the NaK system volume expansion resulting from a 20°F increase in temperature, the ECU deflection should have increased by 0.03 in., which is in good agreement with the measured increase. The deflection of the ECUs during acceleration loadings was as expected. ECU acceptance test data show deflection positions of 0.40 in. for both ECUs at 33 psid pressure to simulate the pressure at launch. (The bellows are prevented from deflecting further than this by retaining pins which are removed by squib firing prior to startup.) Flight test data showing a deflection of 0.42 in. during Atlas burn and 0.40 in. during the less powerful Agena first burn are in good agreement with acceptance test data.

6-10-66

7561-03241

Figure 57. FS-4 FCU Position History

The deflection of the ECUs during the 90 sec interval between acceleration loadings following sustainer engine cutoff and during all succeeding zero gravity periods prior to startup, was 0.08 in. per ECU more than the expected 0.14 in. The most probable of the several possible explanations (Appendix B) is that some gas, probably argon, was trapped in the NaK system at the time of seal-off. (Experience has shown that it is impractical to make the system completely gas-free using the present NaK loading equipment.) By the time of sustainer engine cut-off, atmospheric pressure on the compensators was negligible, reducing the system pressure from 16.6 psia on the launch pad to 2.3 psia. Under these conditions, the trapped gas expanded, deflecting the compensators. Approximately 1 to 1-1/2 in.³ of trapped gas, measured at NaK loading conditions, would be required to produce the observed deflection. The fact that no deflection was noticed during FS-4 ground vacuum testing indicates that (1) most of the gas was trapped in the compensators themselves, thus causing a negligible deflection under the force of gravity and the ground test adapter springs and/or (2) fluctuations caused by the unstable 400 cycle voltage supply obscured the real change of from 1 to 3 mv. Other possible explanations for the observed deflections and the reasons for rejection of these alternatives are given in Appendix B.

2. Orbital Prestartup

During the 10 hours in orbit prior to the startup transient, the compensator position cycled slightly (~ 0.02 to ~0.03 in. peak-to-peak) in response to the sun-shade environment. This much deflection would have been caused by a cycling of 15 to 25°F in average NaK temperature, which is not unreasonable; but deflection could also have been due to instrument compartment temperature cycling. Because of normal tape recorder data noise, it could not be determined whether the ECU cycles were in phase with either of the temperature cycles. At the end of the prestartup period, the average NaK temperature was 10°F higher than at launch. This should have caused 0.012 in. increase in deflection which is about what was observed. However, the precision of the telemetry data (±0.03 in.) makes it unrealistic to measure changes of this size.

3. Startup Transient

As the temperature of the NaK increased during the transient, the deflection of the ECUs increased to compensate for the change in NaK volume. During

this deflection increase, one ECU lagged in deflection by about 0.05 in. A comparison of the acceptance test records of the ECUs (Figure 58) shows that a difference in deflection of from 0.05 to 0.075 in. was normal and occurred because the slightly higher prestress pressure and spring rate of one unit (No. 030) caused it to lag behind in deflection when exposed to the same system pressure.

4. Full Power

About one hour after the system reached full power, the difference in ECU deflection decreased until compensator No. 030 was actually deflected by a slightly greater amount than No. 029. This indicates that No. 030 relaxed more at high temperature than No. 029 which would be normal for the more highly stressed unit. (No. 030 also relaxed more during its acceptance test.) The effect did not occur earlier because the expansion compensator temperature changes at a much slower rate than the NaK temperature, a fact demonstrated by many ground tests.

Table 25 shows the measured and expected compensator positions at operating temperature. In no case was the difference between measured and expected values greater than the combined uncertainty of both values. It should be noted that there was no sign of increasingly negative differences with time which would have been indicative of NaK leaks in the system.

5. ECU Limit Switches

The operation of the ECU microswitches during system startup was in agreement with acceptance test data (Figures 57 and 58) showing deactuation to occur at a bellows deflection of ~ 0.24 in. The microswitches actuated after system shutdown, indicating that the bellows had closed as a result of the NaK temperature and volume decrease.

6. Conclusions and Recommendations

The flight test data indicated no deficiencies in the expansion compensator design. However, they did demonstrate that a small but measurable amount of gas could be trapped in the SNAP 10A system, probably as a result of entrainment and dissolution of the argon used as a cover gas in the NaK loading cart. Although it is improbable that all or most of the trapped gas would form into a

6-10-66

7561-03242

Figure 58. FS-4 ECU Acceptance Test, Pressure-Deflection for Increasing Temperature

single bubble, especially if most of it were trapped in the compensators, the bubble would amount to 6 in.³ at full power conditions—more than enough to temporarily fill the pump throat (1.2 in.³) or the end of an upper manifold half (1.1 in.³). It is therefore recommended that the NaK loading equipment and procedures be modified to further reduce the amount of trapped gas, and that future system designs minimize areas which could entrap gas such as the temperature switch wells, NaK filling line stubs, upper manifold ends, and ECUs.

G. HEAT SHIELD

The role of the SNAP 10A ejectable heat shield in providing thermal control of the system during the prestart phase has been qualitatively discussed in Section III. B. The present section will consider the heat shield in greater detail and will supplement Section III. B.

The SNAP 10A heat shield was designed to provide passive thermal control in either sun-shade transient or constant-sun orbits with an altitude range from 700 to 3000 nautical miles. The aluminum honeycomb construction material was coated on the outside surface with a pattern of stripes designed to effect a

TABLE 25
SNAP 10A FS-4 EXPANSION COMPENSATOR DEFLECTION SUMMARY

Test Phase	Elapsed Time (hr) ⁽¹⁾	Avg. NaK Temperature (°F)	Deflection (in.)		
			Measured ⁽²⁾	Expected ⁽³⁾	Difference
Thermal Reference (Ground)	5	972	1.41 ± 0.02	1.43 ± 0.02	-0.02
Start of Active Control (Orbit 10)	12	963	1.53 ± 0.03	1.52 ± 0.02	+0.01
Start of Passive Control	145	990	1.54 ± 0.02	1.57 ± 0.02	-0.03
Just Prior to Shutdown	1045	962	1.51 ± 0.02	1.52 ± 0.02	-0.01

- (1) The total time of system operation at reactor outlet temperatures above 900°F, corresponding to ECU temperatures above 650°F.
(2) Average of both compensators. 95% confidence limits are shown.
(3) Includes an allowance for 1.3 in.³ of trapped gas located at an average height of the upper manifold.

net heat input to the shield when in the sun. Alternate stripes of Alodine 1000 and aluminum silicone paint were used so that the average ratio of solar absorptivity to thermal emissivity (α_s/ϵ) was 1.82 ($\alpha_s = 0.33$ and $\epsilon = 0.18$). The inside surface of the shield had an emittance of 0.08 to minimize heat losses from the converter radiators whenever the shield might be acting as a heat sink to the converter, e. g., when the shield was shaded from solar energy by the earth. The shield was automatically ejected during the initial startup transient in response to a signal from a temperature switch located at the reactor outlet.

Based on the evaluation of the prestartup period (discussed in Section III. B), it is concluded that the ejectable heat shield well satisfied its intended purpose: it provided good thermal insulation for the NaK coolant. The thermal measurements made on the shield surface were always below the design limit of 375°F, demonstrating that the shield had not been subjected to temperatures that would have been detrimental to the bonding material used in its honeycomb structure. The repeatability of the temperature measurements on each orbit indicated that α_s/ϵ did not change significantly as a result of exposure to the space environment. These results verify ground tests⁽²⁵⁾ conducted in simulated space environments which indicated that the solar absorptance and thermal emittance would be stable for the duration of the prestartup period.

During the system startup transient the shield halves were ejected at approximately the correct temperature. Position switches indicated separation had occurred, and the shield halves, each with an orbital plane that differed from the SNAPSHOT vehicle, were later identified by radar and optical tracking.⁽¹⁾

1. Thermal Environment

Prestartup predictions had been made for the limiting cases of noon and twilight launches (Section III. B). Because the actual orbit was inclined 21° from the noon condition, the solar inputs to the heat shield differed from that limiting case. Since the heat shield thermal predictions had been made for average behavior in characteristic orbits, a general correlation can be made between the analytical predictions, data from ground tests, and orbital heat shield data.

The predictions for average heat shield temperatures for a sun-shade orbit had indicated periodic oscillations ranging from -40°F to 230°F with a near sinusoidal shape. A phase difference near one-half the orbital period would exist between opposite sides of the shield because of the changing orientation of the satellite with respect to the sun. These predictions had been used as the basis for performing ground system tests, i. e. , to program the on-off cycles of infra-red heaters that simulated the sun.

As discussed in Section III. B, the ground tests had been able to verify the general validity of the analytical predictions. The differences between vacuum chamber and actual orbital environment, however, precluded a ground test that duplicated the magnitudes of the heat shield thermal variations, as can be seen from Figure 59 which displays shield temperature measurements from a ground test simulation of a sun-shade transient orbit. The shield temperatures varied between only $210 \pm 10^{\circ}\text{F}$ and 120°F peak-to-peak but had the expected phase difference.

Experimental data on the thermal response of the SNAPSHOT heat shield was available from three thermocouples mounted on the outside surface and a fourth thermocouple inside the shield on the squib block. The locations of these instruments are shown in Figure 60.

The real time and flight recorder telemetry data from these instruments have been smoothed to minimize random errors and the smoothed data are presented in Figure 61. The expected oscillatory shape and the repeatability are clearly evident. The fact that the peaks were the same on each orbit was evidence that the thermal coating materials were not degrading in the space environment. The actual data from one of the heat shield thermocouples is shown in Figure 62 together with a prediction that had been adjusted to account for the 21° inclination of the orbital plane from the earth-sun line. A discussion of the data from each of the four instruments follows below.

a. AI-64

This instrument, located near the vehicle -Z axis, measured the largest amplitude changes of the four thermocouples. AI-64 measured maximum temperatures of approximately 270°F each orbit as the satellite passed over the south pole of the Earth and the -Z axis reached its most favorable orientation

6-10-66

7561-03243

Figure 59. FSM-1 Ground Test Simulation of Sun Shade Transient Orbit
(Flowrate 6% of Design Value)

6-10-66

7561-03244

Figure 60. Heat Shield TC Locations

with respect to the sun. As the satellite moved into the earth's shadow, the temperature dropped rapidly and continued dropping after part of the satellite had reentered the sunlight. This was because the -Z axis was a shaded portion of the satellite until the satellite approached the equator in the southbound direction. Just prior to reentering the sun, AI-64 measured its minimum temperatures, approximately -40°F . Upon exposure to sunlight, the temperature increased rapidly to the maximum once again; this cycle was repeated each orbit until the shield was ejected on Orbit 6.

The data from this instrument are considered to be the most significant heat shield data obtained. A verification of the selection of shield surface coatings can be made because the temperature extremes were near the predicted values

Figure 61. FS-4 Heat Shield Temperature Histories

BLANK

6-10-66

7561-03246

Figure 62. Typical Heat Shield Temperature Comparison with Prediction

for a sun-shade transient orbit. Also, the maximum measured temperature was well below the design limitation, 375°F.

b. AI-161

This instrument was located very near the -Y axis. It was shaded at all times from the sun because of the orbital inclination to the earth-sun line. Ascent data appeared normal, showing an increasing temperature above 100°F. At the beginning of Orbit 1, the data appeared to decrease to -50°F. All subsequent data were below -40°F. Data in this range were so near the lower limit imposed by the telemetry equipment that it is impossible to know whether AI-161 equilibrated near the predicted value of -40°F or whether, more probably, the measurements were limited by telemetry band width.

c. AI-162

Because the squib could be ignited by overheating, this thermocouple was installed on the heat shield squib block (Figure 60) to enable monitoring of the temperature of the squib during ascent and prestartup. The thermocouple also provided temperature data from a point on the inside of the shield which was helpful in analyzing heat shield performance.

The data from AI-162 displayed periodic oscillations between -10°F and 90°F in phase with the other shield measurements (Figure 61). The 90°F maximum provided evidence that the squib never approached temperatures that might have caused premature firing. The minimum temperatures measured by AI-162 are significant and this instrument provided evidence that the inside surface of the heat shield reached temperatures below 0°F. Therefore, the temperature gradient across the shield from outside to inside surface was small, 20 to 30°F under shade conditions. Although this would represent, in shade, a temperature difference of ~ 100°F between radiator and shield inside surface, it should be remembered that the radiant interchange under these conditions was smaller by a factor of two compared to a non-insulated radiator exposed to an effective space temperature of -167°F. Therefore, despite the small temperature gradient across the shield, it acted as an effective insulator for the converter/radiator.

d. AI-140

This thermocouple, located in the quadrant between the -Y and -Z axes, was in a less direct orientation with respect to the sun than was AI-64. The data indicated cycling between 50 and 180°F in phase with AI-64. However, data from this instrument are questionable because they had been erratic prior to launch. Data taken during the countdown indicated 52°F whereas the other three shield thermocouples measured ~96°F. Since the SNAP unit was nearly isothermal prior to launch, because of a forced air heat blanket enclosing the unit, the indication is that AI-140 was biased 45°F low prior to launch. The orbital data were probably attenuated by this same amount.

2. Differences Between Space and Simulated Environment

A comparison of the flight measurements with those made during ground system sun-shade simulated tests can be made only indirectly. The magnitudes are not directly comparable because the ground tests provided uniform infra-red heat alternately on each shield side but with no changes in the angles of incidence while the heaters were on.

Also, the softer vacuum of the space simulation vessel permitted convective and gaseous conduction to supplement radiative heat transfer during the ground tests. The response time of the flight measurements from minimum to maximum temperatures can be seen from Figure 61 to be 40 to 50 minutes, depending on the measurement. This verifies the 35 to 45 minutes "heater on" times that were used in the ground system tests.

An important distinction between the ground test solar simulation and SNAPSHOT is that the spectrum provided by the radiant heaters was entirely infra-red, 20,000 to 100,000 Å. Therefore, SNAPSHOT provided the first system test of the shield thermal coatings when exposed to the true solar spectrum. The fact that the amplitude peaks for the shield measurements were constant throughout the 10 hours from launch to heat shield is an indication that exposure to ultraviolet radiation caused no significant deterioration in the coating properties. This should be considered as important new experimental knowledge from the flight.

3. Heat Shield Ejection

The ejection of the heat shield halves during Orbit 6 was verified by the opening of the break-away position switches and step changes in the signal levels from three of the shield thermocouples. It occurred when the reactor outlet temperature reached $340 \pm 20^\circ\text{F}$, about 6 minutes after generation of sensible heat. The temperature was higher than the design set point of the temperature switch ($275 \pm 25^\circ\text{F}$) because of the relatively long time constant of the switch and the rapid initial NaK temperature transient ($1.4^\circ\text{F}/\text{sec}$). The subsequent orbital paths of the ejected shield halves were of concern because of the possibility of later collision with the SNAPSHOT vehicle. The parting plane of the shield halves was oriented so that the ejection vectors would be orthogonal to the orbital plane of the vehicle. The resulting orbital planes of the shield halves were determined by the vector sum of the vehicle motion and the shield vector, relative to the vehicle. Figure 63 displays the vector relations and the angle between the orbital planes of the vehicle and one-half of the heat shield. For an orthogonal ejection, as shown in Figure 63, preflight analyses indicated some probability of collision at half-orbit intervals when the orbital planes crossed.

Radar tracking identified one-half of the shield early in the mission, but the second half was not identified until after Orbit 500. Subsequent radar and optical coverage revealed a time phase relation between the vehicle and heat shield halves, as shown in Figure 64. The significance of the changing time separation between objects is not completely understood. It is evident, however, that there was a lack of symmetry in heat shield half ejection when the orbital paths of the halves are compared. This can be interpreted as evidence that the vehicle had somewhat rotated from its nominal orientation when ejection occurred and that the ejection was not orthogonal.

4. Recommendations

If SNAP 10A type heat shields are used in future nuclear space power systems, further experimental results should be obtained for shield performance in a constant shade environment. The one instrument on the heat shield that was always shaded did not provide conclusive evidence of the equilibrium temperature in shade. It is realistic to expect some missions to have constant shade condition for a segment of the heat shield. The minimum tolerable equilibrium

6-10-66

7561-03247

Figure 63. SNAPSHOT Vehicle and Heat Shield Orbital Planes for Orthogonal Ejection

temperature in shade will affect the choice of surface coatings. Therefore, the equilibrium temperature in shade should be determined more precisely for a known coating design, such as SNAP 10A.

For future systems, the method used to eject heat shields should be devised to preclude any possibility of later orbital collision. The SNAPSHOT vehicle contractor has recommended ejection parallel to the vehicle orbital plane, thus placing the shield halves in slightly higher and lower orbits than the vehicle.⁽¹⁵⁾ Before any technique is adopted, however, a detailed orbital analysis should be performed.

Figure 64. Heat Shields, Time Separation

H. INSTRUMENTATION

Sections IV. K and IV. J and Reference 2 describe the methods of remotely acquiring diagnostic data from the SNAPSHOT vehicle using a pulse-amplitude-modulated (PAM) telemetry system. The SNAP 10A system instrumentation contained transducers and limited signal conditioning equipment to develop suitable signals for additional signal conditioning and submultiplexing by the Agena telemetry system.

A total of 121 instruments were installed on the SNAP 10A system. The numbers, function, range, and type of transducers installed were determined by a study of the expected performance and malfunction modes of the system. Allowable volume, weight, and power for system instrumentation and the

number and accuracy of available telemetry channels imposed limitations on the selection of instrumentation for the initial flight. The relatively high temperature and high radiation environments also imposed restrictions on the primary payload instrumentation.

Insofar as the performance of the instruments is concerned, it was difficult to evaluate the absolute accuracy of the instruments from flight data because there were few known standards in flight to which instrument outputs could be compared. In addition, it was not possible to separate instrument errors from the larger errors introduced in signal conditioning, telemetry, transmitting, and ground processing. This problem of accuracy is discussed in Section IV. K. However, one indication of the quality of the instrumentation system was the relatively small number of instruments that were inoperative or erratic. A brief evaluation of the performance of the SNAP 10A instrumentation in terms of how well the number, type, multiplexer assignment, range, and accuracy of the instruments satisfied system requirements for analysis of performance, quick-look real time decisions, and reliability of obtaining necessary data in the event of system or component failure is contained in the following paragraphs.

The SNAP 10A instrumentation subsystem operated with a minimum of instrument failures throughout the entire flight. Quantitatively, 92.5% of the diagnostic instrumentation operated without any problem and 97.5% produced usable data. Of the total of 121 AI instruments, 112 operated without any problems and yielded meaningful data. The remaining 9 were erratic or inoperable for at least a portion of the flight, but only three provided no usable data. These instruments are itemized in Table 26 with a brief description of the observed problem and its probable cause. Most instruments had negligible errors when compared with the telemetry errors.

The temperature instrumentation performance was generally satisfactory, although the precision of the wide range thermocouples was severely degraded by telemetry noise. The resistance thermometers provided exceptionally noise-free performance because of their narrow temperature ranges. Although the absolute accuracy of installed resistance thermometers had been inferior to that of thermocouples on system ground tests, the accuracy of flight instrumentation was controlled by telemetry and signal conditioning. Hence narrow

TABLE 26
INOPERATIVE DIAGNOSTIC INSTRUMENTS

AI Instrument Number	Measurement	Problem
8, 9, 10, 11	Thermoelectric Converter Temperatures	An apparent short circuit that resulted in low readings occurred at launch somewhere on the differential or reference thermocouple circuit. The short apparently cleared during the pass on Orbit 18. Exact cause or location of short circuit cannot be determined from flight data. Usable data were provided after Orbit 18.
44	Reflector Temperature	Apparent poor thermal contact between TC and reflector block was indicated by low reading (~70° F lower than expected). Subsequent review of thermal reference test data also showed erratic changes in indicated temperature indicative of poor thermal contact between TC and block. Similar problems with this TC occurred on several ground system tests.
64, 140	Heat Shield Temperatures	Indicated temperature distribution was inconsistent with predictions. A review of circuit loop resistance data obtained during system checkout confirmed that TC connections were interchanged between these two instruments. Simple change in identification of data resolved this problem. AI-140 behaved erratically prior to launch. Launch pad data showed probable bias of -45°F. Negative bias indicated electrical circuit problem rather than poor thermal contact. Data were usable with 45°F correction.
137	Instrument Compartment Neutron Flux Detector	Instrument failed to develop an on-scale reading throughout flight and failed to clear a low flux monitoring signal. Instrument range selected for this single decade instrument based on pre-flight radiation flux predictions did not agree with actual flux experienced at detector location.
165	Solid State Dosimeter	Reversed polarity of instrument output resulting from undetected wiring error at installation.

range instruments, such as the resistance thermometers, generally proved to have superior accuracy. No drift in resistance thermometers was observed.

The converter output voltage instruments produced an output voltage signal that was a function not only of voltage but also of converter return current and the resistance of the interconnecting system. Inaccuracy in voltage measurements and difficulties in the evaluation of real time data resulted from this condition.

The converter voltage profile instruments did not have sufficient resolution to detect small changes in voltage distribution resulting from the converter resistance to ground degradation which was experienced by the flight system. They were, however, adequate to detect the effect of severe resistance to ground degradation and malfunctions such as open and short circuits and gross maldistribution of flow (e. g. plugging) had they occurred.

The lack of a suitable power transducer in the flight system resulted in errors in the determination of converter power output. The minimum error in converter power output calculated from converter output voltage and return current was, at best, ± 16 watts. Additional errors approaching 2% had to be corrected for at the time of shutdown because of the effect of converter leakage current on the converter current measurement used in the power calculation.

Satisfactory measurement of converter internal resistance was accomplished using a specially designed ac measurement technique on the combined SNAP 10A and Agena systems. The additional isolation between the converter and Agena circuits provided by the converter impedance isolation inductor was more than adequate to provide an acceptable measurement. The instrument's calibration signal proved useful in both real-time and off-line analyses of flight test data.

The differential current transducer was specially designed to measure converter resistance to ground and was the only instrument sensitive enough to detect the observed decrease in resistance. Premature system shutdown prevented verification of the data by other instruments.

Position indication circuits provided confirmation of drum motion and expansion compensator bellows motion. The resolution of the drum position instruments in the presence of system noise was not adequate to insure that positive verification of an individual control drum step could be obtained. Special calibration was required on these and all other ac instruments used in the SNAP 10A system because of interaction between the instrument output filters and the Agena telemetry multiplexers.

The performance of the radiation monitoring instruments was as expected from ground test experience and adequate except for two instruments. The instrument compartment thermal neutron detector failed to provide data because of a mismatch between the single decade range selected for the instrument and

the actual input realized in the flight system. The inadvertent reversed polarity connection of the output leads from one of the solid state integrating flux instruments was not detected during vehicle checkout because its normal output is zero prior to being irradiated.

The large number of instrument channels on the SNAP 10A system can be classified in a few instrument types installed to perform specific functions. The performance of the instruments is discussed in terms of these classifications rather than in terms of specific instrument applications.

1. Temperature Instruments

Chromel-constantan thermocouples and platinum resistance thermometers were utilized on the SNAP 10A system to measure system temperatures. Thermocouples were connected in wide range absolute, suppressed zero absolute, and differential configurations. An evaluation of the performance of these circuits indicated that the wide range thermocouple instruments were of limited usefulness because of the lack of resolution resulting from telemetry random error. For example, the 0 to 1200°F wide range instruments were subject to a $\pm 36^\circ\text{F}$ error with the nominal 3% full scale telemetry system accuracy. However, they did provide a gross indication of system temperatures and complete coverage of startup performance from sensible heat to full power operation. Suppressed zero thermocouple circuits (range of 800 to 1200°F) provided superior data compared to wide range thermocouple circuits; they reduced the degrees per mv multiplication factor applied to a given telemetry voltage error (e. g. 3% of 400°F is only $\pm 12^\circ\text{F}$ compared to $\pm 36^\circ\text{F}$ for the wide range) and they utilized the 20 mv submultiplexer which was subject to about 50% of the random noise experienced by the 50 mv submultiplexer utilized by the wide range thermocouples. The suppressed zero thermocouples, which had 800°F reference junctions, were much more susceptible to error caused by variations in the 400 mv excitation voltage than were the wide range thermocouples, which had 0 or 32°F reference junctions. For example, a deviation of only 1 mv from the nominal 400 mv would cause an error of 1.7°F in the suppressed zero instrument while a deviation of 10 mv would be required to cause an equal error in the wide range instruments. The observed deviation in the telemetered value of the excitation voltage (Section IV.J) was sufficient to have caused a possible error of $+4.8 \pm 5^\circ\text{F}$.

Differential thermocouple circuits operated as expected and those connected to the 20 mv submultiplexer were also superior to wide range thermocouple circuits in terms of the apparent accuracy of the reduced data. However, since the desired quantity in most cases was the absolute temperature of parameters such as radiator fin temperature, these differential thermocouple circuits were difficult to use to evaluate system performance, especially in real time.

In addition to problems resulting from the lack of resolution of certain of the thermocouple circuits, the low sampling rate (once per 16 seconds) of the 20 mv multiplexer utilized for many of the thermocouple circuits caused difficulty in detecting the effect of control drum steps and sun/shade transients on system performance from these instruments. The speed (once per 4 seconds) of the 50 mv submultiplexer was satisfactory.

Despite the shortcomings of many of the thermocouple circuits in the accurate resolution of small increments of temperature changes and in following transient temperatures, the thermocouples substantially contributed to the complete data picture of the SNAP 10A unit and provided a lightweight reliable backup to other temperature instruments and related measurements.

In contrast to the thermocouple circuits, where only the thermocouples were installed in the SNAP 10A and the reference junction networks were installed in the Agena, the resistance temperature detectors (RTDs) and associated bridge networks were all installed in SNAP 10A. The precision of the RTDs was considerably better than that of the thermocouples. For low temperature measurements, such as instrument compartment or prestartup fluid temperature, the resistance thermometers provided a much higher level signal than the thermocouples. For high temperatures the suppressed zero capability of the bridge networks gave low degrees per millivolt multiplying factors which resulted in the lowest telemetry errors of any temperature measurement (e. g. 3% of 200°F full range is only $\pm 6^\circ\text{F}$). Had the accuracy of the 50 mv submultiplexer been equal to that of the 20 mv submultiplexer, even better performance would have been obtained. The RTDs were most successful in detecting the effect of sun/shade transients during the prestartup period, measuring the response of the system to control drum steps at full power, and following the slow drift in NaK temperature during passive control.

The data from the high temperature resistance thermometers were not processed during most of passive control because of preflight concern for RTD drift at the SNAP 10A operating temperatures. Qualification test data on appropriately designed RTDs operating at about 1000°F indicate that the errors resulting from detector drift were small compared with the telemetry system error, and that resistance thermometers can be utilized to obtain improved resolution even at temperatures up to 1000°F. Flight data confirmed this and the high temperature RTDs showed no detectable drift after 1000 hours of operation. In addition, the absolute accuracy of the installed high temperature RTDs had been poor, especially on early ground system tests. However, changes in the detectors and installation methods gradually improved the installed accuracy, and there was no detectable difference in the absolute accuracy of RTDs and thermocouples in flight. (The absolute accuracy of the NaK tube thermocouples during ground testing of FS-3 and FSM-4, as well as FS-4, was excellent.)

Perhaps the single most outstanding deficiency of the entire instrumentation system was the inability to provide any temperature data after loss of vehicle 400 cps power on shutdown. This power was required to operate both the 20 mv and 50 mv submultiplexers, and all temperature signals were routed through these multiplexers. No high level temperature instruments which could operate independently of the vehicle 400 cps were provided. Consequently, when the 400 cycle power supply failed, a significant loss of data resulted.

2. Electrical Instruments

Voltage divider networks were utilized to measure the thermoelectric converter output voltage and the converter voltage profile. In the design of the signal conditioning circuits for the converter output voltage instruments, for input to the single-ended 5 volt multiplexer, extraneous voltages proportional to converter return current were inadvertently included in the multiplexer input signals. To obtain the true voltage output it was necessary to apply a correction factor based on indicated current flow. Since this correction factor was also a function of the effect of temperature on lead wire resistance and the resistance of an isolation choke, it was not a true constant for all flight conditions. Thus, some additional error was introduced into the converter output voltage measurements and, of course, into the calculated power output of the converter

which was based on these measurements. Neither did the need to correct the voltage measurement in terms of the measurement of current facilitate the real time evaluation of converter output voltage.

Converter voltage profile instruments had ranges capable of monitoring the maximum expected open circuit converter voltage as well as the normal voltage. Consequently, the normal signal on these instruments was only about 40% of full scale, making the nominal error of 3% equivalent to 7.5% of the measured signal. Therefore, the resolution of these channels was relatively poor, and it was difficult to determine the presence or absence of small variations in converter voltage profile. Also the long length of converter circuits monitored by each of these instruments made them inherently susceptible to noise, since the converter portion of the circuit could not be shielded. While this had been a problem during integrated ground tests of the SNAP/Agena, it was not noticeable in space compared to the telemetry error. These instruments were, however, adequate to provide indications of gross flow maldistribution resulting from a plugged tube or to provide indications of serious open circuit and short circuit conditions had these conditions existed. If converter resistance to ground had continued to decrease, the converter voltage profile instruments would have eventually provided a measure of the decrease (when the resistance reached 10 ohms or less, equivalent to a power loss of 30 w, which would have occurred after about 80 days of operation).

Shunts were utilized to monitor the pump startup battery current and the converter return current. Satisfactory data were obtained from both these instruments. However, it was necessary to subtract the leakage current from the measured converter return current when the leakage current was non-zero in order to obtain the true converter output current.

No power transducer was provided in the flight instrumentation. Converter output power was determined by calculation from corrected converter output voltage and corrected converter return current measurements. The resulting calculated power output was subject to a minimum telemetry error of ± 16 w at best. The use of a full range power transducer could reduce this error to between ± 4 and ± 11 watts depending on submultiplexer telemetry errors. In addition to reducing the error in power, the instrument would provide verification of the value of leakage current indicated by the differential current

transducer. If the leakage current is non-zero the power calculated by multiplying the true output voltage times the return current will be higher than the measured power, and the leakage current can be calculated from the difference between the numbers. Originally there was a second return current shunt on the system which had to be disconnected on FS-4 to allow read-out of the differential current transducer. If this were replaced by a shunt to measure actual output current, it would not only be a direct measurement of this important parameter, but would provide verification of both the power and leakage current measurements. Since degradation of converter resistance to ground is a possible failure mode, the measurement of output power and output current should be considered for future systems.

A differential current transducer was used to monitor the converter leakage current which is the difference between converter return and output currents (Figure 44). This difference was measured by utilizing the converter return and output current leads as single turn differential control windings on the transducer. By assuming that the converter generator voltage, isolation resistance, and internal resistance were uniformly distributed, the relationship shown in Figure 44 can be used to calculate resistance to ground. This instrument gave the only measurement of leakage current at high values of resistance to ground and as a result provided significant information concerning the effect of space operation on the thermoelectric converter isolation resistance. Since it was the only indicator of leakage current, extensive environmental qualification tests and analyses were performed in an effort to discover anything that might suggest that the indicated leakage was due to a malfunction or inadequacy in the instrument rather than an actual leakage current. Nothing was found to suggest that the instrument was reading improperly.

Thermoelectric converter internal resistance was determined by measurement of an ac voltage drop developed across the converter by a 400 cps constant current. A choke isolated the measurement current from the Agena circuits. A relay was used to connect the measurement circuit to the converter on ground command. When not connected to the converter the instrument measured the voltage drop across a 2-ohm calibration resistor. This instrument adequately measured the converter internal resistance, and its calibration resistor reading proved unexpectedly valuable. The calibration signal was used extensively

during real time operation as a quick indicator of data bias, especially tracking station bias. During off line analysis the calibration signal was an invaluable aid in determining the data system accuracy and source of errors since it made it possible to determine the effect of the entire telemetry system and data processing on a constant, reasonably well known and understood signal. For future tests it is recommended that a similar calibration signal be located on each submultiplexer which handles data whose values must be known accurately. Because it is important that the behavior of the signals be well understood, it is necessary to have these signals originate in SNAP 10A rather than the Agena. Analysis of the measurement circuit and experience, which developed late in the ground test program and could not be integrated into the flight vehicle design, indicates that improved internal resistance accuracy could be obtained if four wire measurements rather than two wire measurements were utilized. This change would eliminate the requirement for lead wire calibration during thermal reference testing and minimize the effect of instrument temperature on the instrument calibration.

3. Position Indication Instruments

Linear differential transformers and rotary differential transformers with demodulators were utilized to monitor the position of expansion compensator bellows and control drums, respectively. The performance of the expansion compensator position indicator was satisfactory and that of the control drum indicator was adequate to confirm control drum operation. Telemetry system random noise errors, however, were slightly larger than the signals resulting from one control drum step so that positive verification of control drum motion could not always be obtained. These instruments and all other ac instruments with rectified and filtered output required special calibration since their performance with a multiplexer load was different than with a resistive load. This difference was particularly noticeable in the calibration of the drum position instruments because of their relatively high output impedance and large filter network time constant. Transient changes in the instrument output resulted from the step changes in multiplexer input impedance as the channel was gated on. The 1/2-sec multiplexer sample period did not provide sufficient time for the instrument output to reach the value corresponding to the input impedance of the multiplexer. Consequently, special calibration factors based on test experience with the SNAP/Agena electrical mockup (FSEM-3) were required.

The control drum actuator motor coil voltages proved to be a particularly useful set of instruments in that they verified that the controller was delivering the proper voltage and current to each actuator after shutdown, even though the reflector had been ejected and was supported by high temperature wiring only.

4. Radiation Monitoring Instruments

Radiation levels in the SNAPSHOT vehicle were monitored by a total of 26 instruments. Eleven detectors were located on the SNAP 10A unit and 15 located in the Agena. The detector locations are shown in Figure 27. The Agena detectors, supplied by LMSC, were placed in groups of three (a fast neutron, a thermal neutron, and a gamma detector) throughout the vehicle. The construction, calibration, and performance of these instruments which measured flux, not dose, are described in Reference 26. Over half the total number of detectors monitored fast neutron radiation, which produces the largest portion of radiation damage in semi-conductors. Ten of the eleven SNAP 10A detectors were located in the instrument compartment; these consisted of 8 solid state fast neutron dosimeters developed by AI, 1 gamma ratemeter, and 1 thermal neutron ratemeter. The other detector was a fast neutron ratemeter mounted alongside the reactor.

The performance of the radiation detectors was adequate and as expected from ground test experience, with the exception of the problems with one solid state dosimeter and the instrument compartment thermal neutron detector noted in Table 26. Even though the loss of data from these two instruments caused little loss of information about flight radiation levels, the reasons for the loss of data are important because of their broader significance. The output leads of the dosimeter were reversed, an error which was not detected during system checkout because the instrument output prior to irradiation was zero. If the instrument had had a known non-zero output, the error would have been obvious. This rule applies generally: All instruments should have non-zero outputs under normal conditions; if the input is normally zero during checkout, a bias signal should be provided so that the output is non-zero. Not only is this rule effective in making reversed polarity and open circuits obvious, but use of a known bias allows more detailed trouble-shooting such as quick calibration checks. The fact that the Agena radiation detectors had such a bias made it possible to

determine that they were energized subsequently to the reactor shutdown, a key piece of evidence in concluding that a normal failure sequence had not occurred.

The mismatching of the selected range for the thermal neutron detector and the actual radiation level is evidence that a single decade is too narrow a range for a radiation detector since prediction of radiation levels at the exact location of the detector even to within an order of magnitude is difficult. This is especially true under the complicated conditions of scattering below the neutron shield, conditions which are impossible to simulate on the ground. Even the fast neutron detector which was located next to the reactor, where the difference between space and ground radiation scattering is much smaller, operated at the very top of its single decade range.

The first generation solid state fast neutron dosimeters proved to be low cost, minimum weight, low power, compact detectors capable of providing flux distribution information. Improvement in temperature compensation and reduction in gamma sensitivity can be provided in future units to obtain better performance. Reference 27 describes the design and development of the SNAP 10A solid state instruments and should be consulted for detailed information.

The fast neutron detector located next to the reactor proved to be the most precise indicator of reactor thermal power once the instrument was recalibrated by an in-flight comparison to measured calorimetric power. A preflight attempt was made to calibrate the detector directly in terms of reactor power, an attempt which again demonstrated the difficulty of predicting flux at an exact location. Using an accurate flux versus power correlation obtained from the FS-3 test with a detector at the same location and the FS-4 detector flux calibration, a flux versus power calibration was obtained for the FS-4 detector. Power indicated using this calibration in flight was 40% too high, and this was considered to be relatively good agreement considering the uncertainties involved.

5. Other Instruments

Additional instruments were provided to monitor pump flow, launch accelerations, position switches, and control system events. All of these instruments performed properly and no difficulty was experienced in utilizing the data they provided.

6. Conclusions and Recommendations

Satisfactory performance was obtained from the SNAPSHOT instrumentation subsystem with but few exceptions, although the exceptions have been emphasized in this report. No instrument failures occurred that in any way influenced the performance of the flight system in delivering power to the electrical load. The SNAP 10A instruments were basically compatible with the flight telemetry system and performance was generally in accordance with the specified requirements. As in any initial flight test, the instrumentation selected early in the program did not always provide the optimum range, sampling rate, or resolution to satisfy actual data requirements. A review of the performance of the instruments on SNAPSHOT indicated advantageous improvements in the instrumentation. Recommendations and conclusions resulting from this review are listed below.

1) Critical temperature measurements should be monitored by means of redundant high level output instruments to insure that all temperature data are not dependent on a single multiplexer or on vehicle 400 cps power.

2) The utilization of narrow range temperature instruments vs wide range temperature instruments should be carefully evaluated in any future high temperature space power system test with particular consideration given to the selection of appropriate sampling rates and instrument ranges to insure adequate resolution.

3) Resistance thermometers appear to be attractive for high temperature ($\sim 1000^{\circ}\text{F}$) measurements as well as for low temperature measurements on space nuclear power systems. The development of improved detectors and installation techniques should be continued.

4) The use of a tab thermocouple bolted to the reflector block to measure reflector temperature should be discontinued. Poor thermal contact between the reflector and the thermocouple has resulted in partial loss of reflector temperature data on several systems, including SNAPSHOT.

5) The use of ΔT connected thermocouples in cases where the absolute temperature, rather than the ΔT , is of interest should be carefully evaluated. The interconnection of several instruments decreases reliability, and the use

of a varying reference junction which is not accurately known complicates data analysis and decreases accuracy and precision.

6) The converter output voltage instrumentation on future flight systems should be revised to eliminate the extraneous voltage input to the telemetry system multiplexer that was present on the SNAPSHOT system. A wide range and a narrow range voltage measurement instrument should be provided.

7) Since the present design converter voltage profile instruments did not provide sufficient resolution to adequately resolve effects of sun/shade, flow distribution, and small changes in converter resistance to ground, the value of the instruments should be compared with the reliability disadvantages associated with maintaining them as part of the instrumentation system.

8) Converter output power calculations based on output voltage and converter return current are not adequate to monitor the performance of a thermoelectric converter with leakage current. It is recommended that future systems monitor converter output current with a dc current transducer and that a power transducer (Hall effect wattmeter) be utilized to measure true converter power output. Improved accuracy in the determination of converter output power should result if this change is adopted.

9) Future converter internal resistance instruments should utilize four wire measurement techniques to eliminate problems and errors associated with lead wire resistance.

10) At least one calibration signal, such as that in the converter internal resistance instrument, should be included on all future systems. On developmental systems, where accurate data is especially important, a similar calibration signal on each submultiplexer would be quite helpful in analyzing the test data.

11) Analog position indication circuits cannot provide adequate resolution of incremental control drum motion. It is recommended that a wide range drum position indication instrument be provided for each control drum to provide gross position indication and back-up indication to a step counter instrument. The step counter would provide precise position indication as long as proper controller and actuator operation occur.

12) The use of solid state fast neutron dosimeters is an attractive low cost, low weight method to provide information on vehicle flux distribution. Improved detectors with less gamma sensitivity and better temperature compensation can be developed without difficulty for this application. Check-out of the detectors, and of other instruments as well, would be enhanced if they were designed to develop a non-zero output at zero integrated dose (zero input to instrument) so that the presence and polarity of the output signal could be verified.

13) SNAPSHOT experience has shown the difficulty in attempting to use a single decade radiation detector. It is recommended that future detectors have a minimum of three decades, especially those instruments located below the neutron shield.

14) The SNAP 10A instrumentation system accomplished the task of providing satisfactory data in sufficient quantity and accuracy to obtain significant information from the flight. Difficult measurement problems such as the measurement of internal resistance and resistance to ground of an energized thermoelectric converter and measurement of nuclear radiation dose rate distribution were solved with a minimum weight, minimum power instrumentation system. Advanced high temperature space power systems should profit from the experience developed from this flight test program.

I. MALFUNCTION AND FAILURE SYSTEM

Malfunction detection circuits were provided to determine when critical plant parameters were outside of specified operating limits and to initiate automatic data recording of all Link 2 channels when these limits were exceeded. When the malfunction circuits were enabled by a real time ground command, the occurrence of any one of the following listed conditions would have developed a malfunction signal to initiate data recording.

- 1) Expansion compensators collapsed
- 2) Low reactor outlet temperature
- 3) High reactor outlet temperature
- 4) Low Agena bus voltage

A block diagram of the malfunction and failure circuits is shown in Figure 65. This functional diagram indicates the relationship between the components, real time commands, and telemetry monitoring channels.

Data recording was controlled by the Agena Failure Timer, Timer IV. A malfunction signal from SNAP 10A would have turned on the Link 2 telemetry system, with the exception of the transmitter, and initiated recording of all Link 2 (SNAP 10A and Agena) data by the Link 2 tape recorder. It also would initiate a 45-min timing sequence in Timer IV. (Once a malfunction signal initiated the 45-min timing sequence, it would continue until clockout of the timer or until the timer was reset. Reset of timer IV was accomplished by means of a real time command.) The presence of a malfunction signal from SNAP 10A at the Agena failure timer was monitored by a telemetry channel. With this arrangement a minimum of 45 minutes of data were recorded after first initiation of failure timer operation.

In addition to developing a malfunction signal, a low voltage condition would automatically disable the secondary payloads to reduce the system load and would initiate a timing sequence in the SNAP 10A electrical short-time delay timer (ESD).

If, after the malfunction and failure circuits were enabled, a low voltage condition developed that reduced the Agena bus voltage to below the set point of each of the redundant low voltage sensors, and if the voltage did not increase

Figure 65. Malfunction and Failure Circuit

above the reset point of each of these sensors within the one-minute timing period of the ESD timer, clockout of the timer would have occurred and a failure signal would develop. This signal latched in the SNAP 10A failure relay (K-5) and delivered a failure signal to the Agena. This failure signal picked up the Agena failure relay (Agena K-3) which developed a signal to close Agena transfer switch No. 1 to connect the Agena failure batteries to the unregulated 28 v bus, deenergized the Agena dosimeters, transferred the command system to Doppler Mode A, and prevented clockout of the emergency reset timer (ERT).

In addition, this failure signal reset the Agena timer IV, turned on Link 2 telemetry except for the transmitter, and initiated recording of an additional 45-min of data. It also initiated operation of a 24-hr failure timer in the Agena which turned on the Link 2 telemetry and switched the tape recorder to the readout mode 24 hours after receipt of a failure signal.

The SNAP 10A failure relay also developed a signal to initiate a 1-hr timing sequence in the SNAP electrical long-time delay timer (ELD). Clockout of the one hour timer energized the electrically actuated band release device (EABRD), which released the reflector retaining band and resulted in reflector ejection and irrevocable automatic shutdown of the reactor. Shutdown of the reactor could also be accomplished by means of a real-time command to energize the EABRD. A thermally actuated band release device (TABRD) was provided to insure end-of-life shutdown of the reactor in the event of failure of the electrical shutdown equipment or circuits.

The SNAPSHOT operation of the malfunction and failure system is summarized below.

1. Malfunction Circuits

Operation of the malfunction circuits was normal with the following exceptions:

- 1) Circuits were not enabled on orbit 55 as originally scheduled because of unresolved problems with ion engine operation.
- 2) Circuits were probably enabled by spurious Agena commands resulting from high voltage failure of the Agena command subsystem.

3) The low voltage malfunction detection circuit failed to develop a low voltage malfunction signal and failed to deliver a low voltage signal to the Agena. Probable cause of the failure was a reduction in trip point of one low voltage sensor (SW-5A) resulting from the effect of high bus voltage.

Operation of the expansion compensator position indication limit switches was monitored by auxiliary contacts on the associated follower relays. As described in Section IV. F, operation of these switches from launch through power operation was normal. A comparison of pre-shutdown with post-shutdown data indicates that both relays transferred to provide an expansion compensator bellows collapsed malfunction signal.

Operation of the low reactor outlet temperature sensor was monitored by auxiliary contacts on the associated follower relay. During startup this switch deactivated normally between 760 to 770°F. The specified set point was $780 \pm 40^\circ\text{F}$. A comparison of the pre-shutdown and post-shutdown data indicates that the follower relay transferred to provide a low reactor outlet temperature malfunction signal. Since failure of the 400 cycle voltage supply resulted in loss of all temperature instruments, no data were available to measure the performance of the switch during system cooldown.

Operation of the high reactor outlet temperature sensor was also monitored by auxiliary contacts on an associated follower relay. No data were available to indicate that the reactor outlet temperature was ever sufficient to actuate this switch or to indicate that the switch ever actuated. However, since the follower relay was non-latching there was no evidence to support the statement that a high temperature condition never existed.

Both low voltage sensors closed during the initial prestartup period as Agena battery voltage decreased. Because the sensors tripped on relatively instantaneous low voltage conditions compared to the period between voltage monitoring samples, and because the sensors had an allowable hysteresis of up to 2 v, it is impossible to evaluate the exact sensor trip point from the flight test data. However, both sensors did operate at or close to their original set points (Section IV. J).

A comparison of pre-shutdown and post-shutdown telemetry data indicates that SW-5B closed and remained closed. However, SW-5A remained open at a

low bus voltage of 19.2 v observed at the final vehicle acquisition. Extensive testing and analysis reported in Reference 1 indicate that SW-5A probably experienced a reduction in set point as the result of high voltage on the unregulated bus.

Original plans had called for the malfunction and failure circuits to be enabled by ground command at about Orbit 55. However, because of the uncertain effect of additional ion engine operation on the system, an operational decision was made to defer activation of the malfunction and failure circuits until after ion engine operating problems were resolved. By Orbit 552, additional ion engine operation plans had not yet been resolved, and the circuits had not been activated. During this period of operation, contingency plans were prepared so that the system could be enabled by ground command at the first sign of serious degradation or incipient failure. Backup protection to insure reactor shutdown was provided by the TABRD.

After it became apparent that the vehicle was not operating normally on Orbit 555, the malfunction detection arm and execute commands were transmitted. However, no verification of any command was received subsequent to the telemetry off command of Orbit 552. Telemetry data obtained on Orbit 574 indicated that the malfunction and failure circuit enable relay K-12 had been picked up and latched closed and that the malfunction relay K-7 was also picked up (non-latching) to develop a malfunction signal. In view of the fact that two malfunction conditions (collapsed expansion compensator bellows and low reactor outlet temperature) existed at this time, it is presumed that the K-7 indication was valid and that a malfunction signal was indeed delivered to the Agena. Verification of receipt of a malfunction signal by the Agena vehicle is normally provided by a telemetry signal from the Agena failure timer IV. This signal, however, was not present. Reference 1 indicates that high voltage applied to the timer could result in removal of this signal and concludes that the malfunction signal was probably delivered to the Agena. Reference 1 also concludes that a spurious command signal, probably resulting from the effect of high voltage on the command system, resulted in enabling of the malfunction and failure circuits.

In summary, the operation of the malfunction circuits was normal with the following exceptions: enabling was by spurious Agena command rather than by

ground command, the low voltage malfunction circuit failed as the result of over-voltage induced changes in sensor set point, and confirmation of delivery of a malfunction signal to the Agena failure timer could not be obtained because of high voltage damage to the timer which removed the monitoring signal.

2. Failure Circuits

Because of the change in trip point of undervoltage sensor SW-5A, a failure signal was not developed by the failure circuits. Automatic shutdown of the reactor did not occur as the result of normal failure circuit operation, but probably occurred in response to a spurious reflector ejection signal developed by the Agena command system.

3. Low Voltage Signal

As previously described a low voltage condition also develops a low voltage signal (from K-17) to drop secondary payloads by operation of Agena relay K-1. No direct telemetry monitoring of relay K-17 status was provided in the system, so that it was necessary to depend on Agena secondary payload bus status monitoring provided by Link 3 telemetry. Inability to command Link 3 telemetry on made it impossible to determine if a low voltage signal was delivered to the Agena. However, failure of undervoltage sensor SW-5A to close probably resulted in no low voltage signal reaching the Agena.

4. Failure Signal

The conclusion that the SNAP 10A failure signal was never delivered to the Agena is based on the following observations:

- 1) Agena dosimeters remained on after failure
- 2) Absence of telemetry verification of failure signal at Agena failure timer.

Failure analysis and testing disclosed that high voltage would not destroy the Agena failure timer failure signal status monitor and that Agena dosimeters could remain energized only if the Agena failure relay K-3 did not receive a failure signal.

5. Automatic Shutdown

Automatic shutdown of SNAP 10A required that a failure signal be delivered to the ELD timer input. Absence of a failure signal indicated that automatic shutdown of the reactor was not initiated by SNAP 10A failure circuits. Extended malfunction and failure mode analysis and testing of shutdown components and circuits as reported in Reference 1 indicated that shutdown occurred as the result of a spurious reflector eject command. Telemetry verification of EABRD energization, band release, and reflector ejection indicated that electrical reflector ejection had occurred.

6. Conclusions and Recommendations

The fundamental objective of the malfunction detection circuitry is to detect system malfunctions and initiate the recording of all primary payload data in order to permit the unambiguous determination of the cause of system malfunction. The system failed to accomplish this objective. The flight test experience indicated the following deficiencies in the design of the system.

a. Lack of Operational Flexibility

A single enable command was provided for the independent diagnostic instrumentation function and automatic shutdown functions. Because of this design feature, the malfunction detection circuits could not be used to initiate automatic data recording in the event of a malfunction without also enabling the failure circuits. Consequently, when the fear of initiating premature system shutdown during subsequent ion engine operations prevented the enabling of the failure circuits, the malfunction detection circuits also remained inoperative.

Not only was a single enable command used, but, once commanded, it could not be disabled. This lack of operational flexibility with respect to the malfunction enable command resulted in excessive use of the Link 2 tape recorder to provide complete data coverage.

b. Insufficient Design Margins

The spurious commands generated by Agena command system, failure of the SNAP 10A undervoltage sensor, loss of failure timer input signal telemetry monitor, and other Agena equipment failures resulting from overvoltage on the Agena unregulated bus indicate a fundamental lack of design margin in vital control and instrumentation equipment with respect to input voltage. The inherent high voltage regulation characteristic of all thermoelectric converters essentially results in the open circuit voltage of the unregulated converter being approximately twice that of the normal full load voltage. Failure of a single voltage regulator can result in equipment being subjected to voltages approaching the open circuit converter voltage. The degree of approach, of course, is dependent on the system load at the time of failure.

c. Inadequate Instrumentation Redundancy

The only post failure information available on system temperature was provided by the low reactor outlet temperature switch. Closure of this switch indicated that the reactor outlet temperature was less than 750°F. All SNAP 10A temperature instruments utilized either the 20 or 50 mv sub-multiplexers which require 400 cycle excitation. Loss of the 400 cycle power supply resulted in loss of all temperature measurement data. In order to insure that adequate malfunction and failure data are obtained, additional redundancy in instrumentation designs and channel assignments are required. High level temperature instruments could be made to operate independently of the availability of 400 cps power.

d. Critical Set Point Parameters

The low voltage sensors utilized in the malfunction and failure circuits were required to trip on low voltage indicative of failure conditions and yet not trip at normal operating conditions. The sensor trip point was required to be set very close to the limit of the minimum equipment operating voltage level in order to meet this requirement. The effect of overvoltage on the sensors resulted in a shift in critical sensor trip point sufficient to result in failure of the circuit to perform the intended function. This experience indicates the difficulties in operating with circuits that require the maintenance of critical set points close to normal operating conditions.

The following recommendations have resulted from an examination of the performance of the malfunction and failure circuits:

- 1) Completely redesign the malfunction and failure circuits to provide increased operational flexibility by separation of diagnostic and automatic reactor shutdown functions.
- 2) Provide the ability to enable and disable the automatic shutdown function.
- 3) Provide increased redundancy in telemetry tape recorders, failure power supplies, and critical parameter instrumentation.
- 4) Provide backup protection against high voltage resulting from voltage regulator failure and provide increased design margin against overvoltage in vital satellite control and instrumentation equipment.
- 5) Reexamine critical malfunction and failure circuit parameters and eliminate the requirement for a set point close to normal operating conditions.

J. AGENA AND SECONDARY PAYLOADS

Throughout orbital operation, the SNAP 10A system and the Agena operated as a single integrated vehicle. The Agena supplied vital support functions for SNAP 10A and provided a test bed for a variety of secondary payloads. It was a standard Agena with considerable program-unique equipment added—required mainly for long-term orbital operation,—and especially designed to withstand the effects of nuclear radiation. The major orbital sub-systems are listed below.

- 1) The electrical system, which contained all vehicle power regulation, distribution, conditioning equipment, and batteries.
- 2) The telemetry system, used to transmit both SNAP 10A and Agena data to earth.
- 3) The attitude control system, which maintained the vehicle in a nose-up, stable position.
- 4) The tracking and command system, which enabled the vehicle to be tracked and commanded by the ground tracking stations.

5) The thermal control system, which relied on passive thermal control surfaces and electrical heaters to maintain Agena equipment within the desired temperature range.

The performance of the Agena vehicle during the SNAPSHOT orbital operations was generally quite satisfactory and no major deviations or anomalies were encountered prior to the system shutdown. Several minor deviations from normal or specified performance occurred primarily in the electrical and telemetry subsystems. The most important of these are listed below.

1) There is evidence that the voltage regulator between passes was stressed to near its maximum design limit of 380 w during as much as 20% of the flight test and that during the same time period the voltage regulator electronics temperature exceeded its maximum limit of 160°F, also between passes. However, the evidence is not conclusive.

2) It is probable that as many as 30% of the telemetry channels used by SNAP 10A did not meet the minimum accuracy requirement of $\pm 3\%$ of full scale.

The following paragraphs describe the Agena performance only to the extent that it affected the operation of SNAP 10A; no attempt is made to analyze Agena performance in general. Agena performance and design are described in detail in References 1, 5, 15, 28, and 29. The secondary payload experiments are briefly described at the end of this section.

1. Electrical Subsystem

The Agena electrical system contained all equipment necessary to regulate, condition, and distribute the power generated by SNAP 10A. All power used by SNAP 10A electronics and instrumentation was first regulated and conditioned in the Agena. The electrical system also contained batteries to supply power during the prestartup period, during power-demand surges, and in case of failure. The electrical system performed satisfactorily with the following exceptions:

1) The SNAP 10A low voltage sensors unexpectedly tripped during the pre-startup period, due to low battery voltage. Calculations, however, indicated that the behavior of both the switches and the batteries was normal.

2) The power dissipated by the voltage regulator between passes was not significantly less than the maximum design limit of 380 w during the time period between 70 and 270 hours after initial full power. Using 300 w as a criteria for regulator overstressing, it is possible that the regulator was overstressed up to a maximum of 200 hr, or 20% of the flight test. Calculations indicate that, during this same time period, the voltage regulator electronics box temperature may have exceeded its maximum temperature limit of 160°F by as much as 50°F while the vehicle was in the sun between passes.

3) All Agena bus voltages were lower than nominal though within specification limits.

4) There were indications of loads near the maximum for good regulation on two buses during the active control period.

The performance of the major electrical components is described below.

a. Batteries

The two type XV failure batteries were used during the prestartup phase to provide electrical power for the SNAP 10A and Agena. The battery voltage remained within the predicted performance envelope (Figure 66) and above the minimum specified voltage of 22.5 v (Reference 30). Near the end of the prestartup period, the SNAP 10A low voltage sensors unexpectedly tripped at what appeared to be a substantially higher voltage than the nominal range (Figure 66) which had been verified during the FS-4 system ground testing. However, calculations have shown that the relatively high internal impedance (0.1 to 0.3 ohms) of the batteries while operating in the peroxide region (where the voltage has not yet leveled off), coupled with the measured current drain of 10 to 13 amps, was sufficient to drop the bus voltage at the SNAP unit to the measured trip point of low voltage sensors. The sensors reset after the thermoelectric converter began to supply power to the unregulated bus. Since the converter was supplying almost half the load at the time the switches reset, the voltage drop between the unregulated bus and the switches was reduced. This accounts for the smaller difference between the unregulated bus and switch voltages at reset than at trip.

6-10-66

7561-03250

Figure 66. Agena Unregulated Bus Voltage During Prestartup Test Phase

The NaK pump batteries supplied auxiliary current to the SNAP 10A thermoelectric pump during the pre-startup phase to ensure that there would be a minimum NaK flow of at least 0.26 gpm at all times. The batteries, although located in the Agena, were provided by AI. Prior to the flight test, there was some concern that the batteries would not provide the required 40 amps to the pump because of the high resistance, in a vacuum, of the long length of wire necessary to reach from the Agena to the pump. Changes were made in both wire diameter and battery capacity, and 40 amps of current were supplied without problems in flight. Examination of the pump battery data in Figure 67 shows the following effects.

1) During prelaunch operation, circuit current was high, but within the predicted band. This occurred because the circuit wire resistance, which was the dominating resistance, was low due to the cooling effect of air.

2) During early orbital operation, the current stabilized at a lower value than prelaunch operation, but still within the predicted band. This was normal since the wire resistance increased due to operation at a higher temperature in vacuum.

Figure 67. NaK Pump Battery Performance

3) After sensible heat, the current fell to a value outside the predicted band. This occurred because the wire resistance increased as wire temperature increased with structure temperature, but was of no consequence since auxiliary pump current was no longer required.

4) Battery capacity was about 1300 amp-hr, well above the minimum 1000 amp-hr specified.

b. Voltage Regulator

The Agena regulator was a shunt type voltage regulator designed to consume a maximum of 380 w of excess SNAP 10A power while maintaining a constant output voltage of $29.2^{+0.0}_{-0.2}$ vdc. However, the regulator could dissipate up to 500 w before losing its regulating capability. The regulator was required because the SNAP 10A power output was nearly always substantially greater than the vehicle power demand, requiring the regulator to dissipate from 30 to 60% of the power output. (The only exceptions occurred during transient loadings caused by control drum stepping or load switching. If these momentary peaks exceeded the SNAP 10A output, they were supplied by a rechargeable 30

amp-hr surge battery.) The large excess of power supply over demand was primarily due to the restriction on ion engine operation which made it unnecessary to charge the ion engine batteries, a load of 100 w. If the voltage regulator had not been present, the converter would have adjusted to meet the lower power demand by providing less current at a higher voltage. Figure 19 shows this relation between converter output voltage and load. Regulation of the voltage to an upper limit was required because much of the electronic equipment in the Agena was susceptible to over-voltage damage.

The regulator remained off until the converter output voltage exceeded the regulator set point approximately 100 minutes after sensible heat as shown in Figure 68. The voltage regulator output voltage experienced an unexpected step increase at 100 minutes after sensible heat, and a similar step increase occurred in the Agena unregulated bus voltage. The cause of the step was investigated since it had been expected that the voltage would increase smoothly until it reached the regulation point. It was concluded that the step was caused by a reduction in vehicle loads, which allowed the line voltage to increase to the set point of the regulator. The observed step of 1-v required a load decrease of a minimum of 14 w. Since the vehicle was not within range of a tracking station at the time of the step, the reduction in load must be attributed to the switching off of either the CMG or transmitter heaters. Since the two transmitter heaters each required 27 w and the four CMG heaters each required 15 w, it is reasonable to assume that the observed voltage step occurred when one of the heaters turned off. The fact that the step occurred just before the converter power output was large enough to bring the voltage regulator on line was coincidental. Such steps could have occurred at any point between the time when the failure batteries were diode blocked from the unregulated bus, approximately 70 to 90 min after sensible heat, and the time when the regulator assumed a load.*

Following SNAP 10A full power operation and the assumption by the regulator of its power dissipating function, the regulator performed as expected without drifting and maintained a constant voltage output of 28.7 v for the duration of the

*As long as the batteries were connected to the bus, small load changes did not cause voltage changes because the batteries have a low internal resistance. Thus, increases or decreases in current demand caused only insignificant changes in battery output voltage.

Figure 68. System Voltages and Current During Startup

flight test. This measured value is 0.3 v less than the minimum value of the regulator set point tolerance of $29.2^{+0.0}_{-0.2}$ v, but is within the overall accuracy of the telemetry system. However, all regulated bus voltages were also slightly low, indicating a true low voltage.

Because overstressing the voltage regulator could have contributed to its postulated failure, calculations were made to determine the amount of power dissipated by the regulator at various times throughout the flight. Particular emphasis was placed on those periods of time immediately following control drum steps when the SNAP 10A power output increased temporarily by from 30 to 50 watts. The results of the calculations are summarized by Figure 69 and are shown in more detail in Table 40 of Appendix C. Even though none of the calculated values were greater than 380 w, it cannot be assumed that the regulator was never overstressed for the following reasons:

- 1) The accuracy of the power dissipation calculation is poor, primarily because of the inaccuracies in measuring and estimating vehicle power

6-10-66

7561-03253

Figure 69. Power Dissipated by Voltage Regulator

consumption. The estimated accuracy of the values given for power dissipated by the regulator between passes is ± 60 watts (95% confidence limits).

2) The values shown are averages over time periods varying from 1 to 20 orbits and do not take into account those times when higher than average power output happened to coincide with lower than average vehicle power consumption to produce higher than average regulator loads. An example of the kind of coincidence which could have occurred was all cyclic heaters turning off at the same time (a decrease of 114 w from all heaters on) while the vehicle was in the shade (the thermoelectric converter generated about 20 w more power in the shade than in the sun). This would have resulted in a 70 w increase over the average regulator loads plotted in Figure 69.

3) The maximum power load on the regulator (500 w) does not necessarily correspond to the maximum stress on its individual components whose failure could cause partial or complete regulator failure. For example, the maximum stress on some of the driver transistors occurs at about 25% and 75% of maximum load (about 125 and 375 w) and is at a minimum at about 0, 50%, and 100% of maximum load (0, 250 and 500 w). This is an important point since the transistors are among the components most susceptible to failure, and failure of one or two could cause complete regulator failure.

Because of these uncertainties in what the actual voltage regulator load was and what loads constituted overstressing of the regulator, a lower limit of 320 w (design limit of 380 w minus the uncertainty of calculated values of ± 60 w = 320 w) was placed on the maximum design capacity. Values greater than 320 w were considered to be not significantly different from the design limit, which was also the maximum stress point for some of the transistors, and hence, capable of possibly overstressing the regulator. Using this approach, it is possible to conclude that (1) the regulator could have been overstressed for as long as, but no more than, 200 hr, or about 20% of the total flight time, and (2) it is improbable that the regulator was ever loaded beyond its maximum regulating capacity of 500 w.

Only a small fraction of the high regulator loads occurred as a result of power output peaks resulting from control drum steps and none occurred during a pass. There were two reasons for this, (1) during the first half of the active

control phase, and during all passes, vehicle power consumption was high, and (2) the power output peaks were very short, the time between prestep and post-step steady state power levels being only 10 minutes. On the other hand, the high power output at the end of the active control phase and the beginning of passive control combined with decreased vehicle power consumption to give high regulator power loads for the long period of time between 70 and 270 hours when there were only three drum steps.

That the end of active control and the beginning of passive control was the time period when the regulator was most highly stressed is verified by the data in Figure 70 which shows that the voltage regulator electronics box temperature, where the transistors were located, was highest from 100 to 200 hours. It is also probable that the electronics temperature reached higher values between passes than those shown in Figure 70 which were all measured during passes. This is true because the power dissipated by the regulator between passes averaged 100 w, or 30%, more than during passes between 100 and 200 hr. Calculations (see Table 41, Appendix C) indicate that the maximum average temperature between passes would be $\sim 210^{\circ}\text{F}$ when the electronics were in the sun and $\sim 175^{\circ}\text{F}$ in the shade. Both of these temperatures are above the maximum limit of 160°F . The same calculations indicate that the electronics temperature in the sun exceeded 160°F whenever the regulator load exceeded 255 w. Unfortunately, no flight tape recorder data was processed for this temperature measurement so the results of the calculation could not be checked against actual flight test data. Hence, the accuracy of the calculated results is somewhat questionable.

c. Power Supplies

The Agena contained three types of power supplies which provided regulated and conditioned power to the SNAP 10A and Agena electronics throughout the flight test:

- 1) Two redundant converters each supplied regulated positive and negative 28.3 vdc power. The system was designed so that only one converter was active at a time, the other serving as a backup unit.

- 2) Two redundant inverters supplied three phase 115 vac power at 400 cps. Only one inverter was active at a time.

6-10-66

7561-03254

Figure 70. Agena Voltage Regulator Electronics Temperature

3) The Agena telemetry system contained a 400 mv power supply to provide bias voltages for the thermocouple reference junctions.

Table 27 summarizes the measured and specified power supply output voltages. The voltage regulator and unregulated bus voltages are shown for comparison.

Based on a statistical review of the flight test data, including that shown in Table 27, the following conclusions have been made concerning the Agena power supplies:

1) The power supplies operated satisfactorily with no significant drift over the entire flight test.

2) The output voltages were constant. With only one exception (see (4) below), the standard deviations of the bus voltages shown in Table 27 were less than the sub-multiplexer deviations caused by random error. This indicates that the measurement error was considerably larger than the bus voltage variations and that the actual bus voltage variations were even smaller than the values shown in Table 27.

3) All bus voltages, and the voltage regulator output and unregulated bus voltages as well, were lower than the nominal values although they were still within the specification limits required by Reference 30 (Table 27). Since all the values in Table 27, with the exception of the voltage regulator data, were obtained from the 5 v sub-multiplexer, a statistical test was made to determine whether all data from that multiplexer were biased by the same amount. The test indicated that there was no significant evidence that the entire sub-multiplexer was biased. Another test indicated that, by chance alone, the individual instruments would all be low biased less than one time out of ten. It was therefore assumed that the bus voltages were low and small corrections were made in the SNAP 10A instruments affected. Only in the case of the suppressed zero thermocouples were the corrections larger than the total random error of an instrument.

4) Both the positive regulated 28 vdc bus and phase bc of the inverter showed signs that they were loaded to near their maximum limits for good regulation during the active control period. This was evidenced by a step increase in both voltages when the load was reduced by turning off the

TABLE 27
AGENA ELECTRICAL SYSTEM PERFORMANCE

Converters	Test Phase ⁽¹⁾	Measured Values		Predicted or Specified ⁽³⁾
		Mean	Standard Deviation ⁽²⁾	
Positive Regulated (vdc)	a, b	27.90	0.064*	28.3±0.6
	c	28.12	0.034	28.3±0.6
Negative Regulated Bus (vdc)	all	-27.95	0.067	-28.3±0.6
<u>Inverter</u>				
Phase AC (vac)	b, c	113.2	0.24	115±2
Phase BC (vac)	b	113.5	0.25	115±2
	c	114.1	0.22	115±2
400 mv Power Supply (mv dc)	all	397.2	1.42	400±4 ⁽⁴⁾ 400±15 ⁽⁵⁾
Voltage Regulator (vdc)	b, c	28.68	0.15	29.2 ^{+0.0} -0.2
Unregulated Bus (vdc)	b	27.90	0.119*	28.6
	c	27.99	0.066*	28.6

- (1) The letters indicate the following test phases: (a) prestartup, (b) full power-active control, (c) full power-passive control.
- (2) This is the total random error deviation and does not include bias which was impossible to determine for these measurements. The deviation is shown to indicate the amount of bus voltage variation, but with the exception of those values marked with an asterisk (*) the deviation was equal to the sub-multiplexer deviation, which indicates insignificant bus voltage variation. The data from approximately 95% of the passes lie within two standard deviations of the mean.
- (3) The values for the voltage regulator and unregulated bus are predicted. All other are specified by Reference 30.
- (4) Required to give the specified accuracy of ±6.5°F for the suppressed zero thermocouples.
- (5) Required to give the specified accuracy of ±3°F for the 0 and 32°F thermocouple reference junctions.

SNAP 10A startup control system at the end of active control, as shown in Table 27 by the higher mean voltages during the passive control period. In addition, the positive regulated bus showed a significantly greater variation during active control, which is also indicative of poorer regulation. It is significant that the unregulated bus voltage reacted to reduced loads in the same way.

5) The postulated failure sequence (Section III. F) would be somewhat more probable if there were indications that transfer from one inverter to the other had occurred sometime prior to the start of the failure sequence. Such a transfer usually causes a small step change in all phases of the output and all instruments which are sensitive to 115 vac changes. There was no evidence of any such changes for either the inverters or converters.

2. Telemetry Subsystem

The three complete telemetry systems listed below were used.

1) Link 1 was a completely separate, high sampling rate system used to monitor launch and early orbit performance. It was powered by the ascent batteries, which were isolated from the main power bus, and operated satisfactorily until between Orbits 109 and 113 when the ascent batteries ran down. The only Link 1 data used in this report are accelerometer and acoustical data and a small number of Agena interface and structure temperatures.

2) Link 2 was used for monitoring Agena and SNAP 10A orbital performance. It supplied nearly all data used in this report and is the only link whose performance was analyzed for this report. It is described in more detail below.

3) Link 3 was used only to monitor secondary payload performance and was of little importance in preparing this report. It operated satisfactorily throughout the flight.

The Link 2 telemetry system was a PAM/FM/FM system. In a PAM system, a multiplexer samples each transducer in sequence, and the output of the multiplexer is a series of pulses. The amplitude of each pulse depends on the instantaneous amplitude of the data signal, hence the name pulse-amplitude

modulation (PAM). The output of the multiplexer goes to a voltage controlled oscillator where the amplitude of each pulse modulates a vhf carrier. This frequency modulated (FM) carrier is then transmitted to the ground tracking stations, hence the designation PAM/FM. The second FM designated in the acronym arises from the fact that flight tape recorder data can be transmitted to earth simultaneously with real time data by using a different carrier frequency. A more thorough discussion of PAM/FM telemetry can be found in References 31 and 32. A diagram of the Link 2 telemetry system is shown in Figure 71. SNAP 10A data were sent to one of three sub-multiplexers depending on the input signal level, or directly to the main multiplexer if a high sampling rate was desired. It should be noted that the telemetry system includes signal conditioners, thermocouple reference junctions, and magnetic amplifiers (part of the 20 mv and 50 mv sub-multiplexers) in addition to the data sampling equipment which is usually considered to be the entire data system. This means that all errors introduced by this equipment become part of the total telemetry system error. Such a definition is necessary from a practical standpoint as well, since their contributions to the total error were not calibrated out prior to flight and were indistinguishable during the flight. All SNAP 10A data did not pass through the same design or even the same type of equipment and hence were not subject to the same sources of error.

Figure 71. Link 2 Telemetry System

The Link 2 telemetry system performed satisfactorily with 100% of the Agena and SNAP 10A measurements being transmitted and recorded at the ground tracking stations. Approximately 99% of the instruments furnished usable data. There was no degradation of telemetry signal strength or accuracy during the flight. There were, however, a number of relatively minor deviations or unexpected results which are described in the following paragraphs.

a. Overall Accuracy

Possibly as many as one-third of the telemetry channels used by SNAP 10A did not meet the required accuracy of $\pm 3\%$ of full scale.⁽³⁰⁾ This conclusion is a significant change from previous opinions that the accuracy was excellent and the error might be as low as 1 to 2% of full scale. The reason for the previous conclusion is that the data were clean, i. e., the amount of random error, both within and between passes, was small. This gave the data the appearance of great accuracy. However, nearly all the telemetry channels tested showed indications of bias, and the overall accuracy is affected by both random error and bias. The results supporting the second conclusion are discussed in detail in Section IV. K.

b. Differences in Multiplexer Accuracy

The 5v and 20 mv sub-multiplexers were significantly more noise-free than the 50 mv sub and main multiplexers. The 5v sub-multiplexer also had a total random error less than one-half that of the 50 mv sub and main multiplexers. The random error of the 20 mv sub-multiplexer could not be determined accurately.

It is possible that the noise level was lower on the slower multiplexers because they sample each instrument for a longer period of time, and therefore make a better measurement of the true input. However, this would not explain the lower total random error since the 50 mv sub and main multiplexers sample each instrument so many times during a pass that, by calculating the average, random error from noise can be nearly eliminated. If the reason for the higher accuracy of the two slow multiplexers could be determined, it might lead to improvements in the rest of the telemetry system.

c. Link 2 Tape Recorder Degradation

Performance degradation of the Link 2 tape recorder was first observed on Orbit 36 when about 35 minutes of data, from a possible total of 135 minutes, were too noisy to be usable. On each succeeding orbit, the amount of unusable tape increased at a faster rate. By Orbit 39, the amount of usable tape decreased at the rate of 15 minutes per orbit and only 50% remained usable. The tape recorder was commanded off on Orbit 39 to conserve the remaining tape for use in case of impending failure or malfunction. No attempt was made to turn on the recorder again until after Orbit 552. Therefore, complete orbit data are available only prior to Orbit 39. The most probable cause of tape degradation was misalignment of the tape during the record mode rather than simple wearing-out of the tape or heads.^(5, 15) It is not known whether the extensive ground-testing and continuous in-flight use of the recorder contributed to the misalignment. The recorder operated for a total of 136 hours, of which 71 hours were in-flight. However, the Link 3 tape recorder, which was of identical design and was used intermittently throughout the flight, operated satisfactorily.

d. Data Distribution Anomalies

Most of the flight test data followed a normal error distribution. However, both data cycling and bimodal distributions were observed on occasion throughout the flight.

An example of data cycling is shown in Figure 72. The cycling occurred on random orbits and affected all channels to varying extents. The cycles on all channels were in phase. The example shown is typical in that, although the peak-to-peak variation was as high as 3% full scale, no apparent bias was introduced and the accuracy of the average value was not significantly different from a pass on which no cycling occurred. On the other hand, the cycling tended to obscure real trends in the data during a pass since the cycles were much larger than real data changes. (For example, the cycling in the example is equivalent to a converter inlet NaK temperature change of 70°F.) It is not known whether the source of the cycling was the Agena or the tracking stations. The fact that identical cycling occurred when duplicate tapes were processed at different times indicates that the cycling was not introduced by processing. Although data cycling was observed on Orbit 552, it occurred with equal intensity as early as Orbit 5.

Figure 72. Typical Telemetry Data Cycling Anomaly

An extreme example of the kind of bimodal distribution observed in some of the data on some orbits is shown in Figure 73. The amount of separation between the modes is about 3% full scale. Definite bimodal distributions were observed primarily on data from the main multiplexer, but not on all channels during the same pass. The bimodal distribution caused a small increase in error and, in less extreme cases than that shown, obscured trends. The cause of the anomaly is unknown.

3. Orbital Attitude Control System

The two separate attitude control systems used were the standard Agena gas-jet pneumatic system and a less powerful control moment gyro system. The pneumatic system was used to (1) reorient the vehicle from a nose-forward to a nose-up attitude during Orbit 1, and (2) to maintain the vehicle in a stable position against the influence of the comparatively large torques resulting from propellant venting during the first few orbits. The system was designed to maintain stability to within $\pm 5^\circ$ on each axis, but the actual performance

6-10-66

7561-03257

Figure 73. Extreme Bimodal Distribution Data Anomaly

exceeded this. Once the vehicle was stabilized in an upright position and startup was completed, the mode of operation was changed to high gain to rapidly deplete the remaining control gas, thereby precluding subsequent leakage and resulting attitude disturbances. This was especially important in the case of SNAPSHOT since the pneumatic control system had not been radiation hardened. On Orbit 54, when the control gas was depleted, the pneumatic control system was deenergized. During the high gain mode of operation, the control gas depletion rate was much lower than predicted, a fact which indicated lower-than-expected vehicle torques.

Upon deactivation of the pneumatic system, vehicle stability was maintained by (1) the gravity gradient effect which was enhanced by the shape and the weight distribution of the vehicle, and (2) the control moment gyros. The gyros dampened any attitude changes and the gravity gradient effect provided restoring torques. This system provided much smaller counter torques than the pneumatic system, but required no attitude sensors for its operation. The data indicate that the system held the vehicle stable. In view of the observed high level of attitude control performance, it has been recommended that there is no need for orbital attitude monitoring instruments on future SNAPSHOT flights.⁽¹⁵⁾

Vehicle attitude control is further discussed in Section IV. O.

4. Command and Tracking System

The 400 mc Doppler tracking and command system was used to track the vehicle and to send operational commands to it. SNAP 10A saw commands at the Agena interface only as a set of relay closures. The performance of the system was excellent and there were no times prior to shutdown when the vehicle was not acquired on schedule or the vehicle failed to receive and verify a command.

5. Thermal Control System

The Agena used a combination of exterior thermal control surfaces (coatings were used to give the desired absorptivity/emissivity ratios) and electrical heaters on a few vital components to maintain the desired temperature levels. Because of the high temperature at which SNAP 10A operated, there was some preflight concern that the heat transferred across the Agena interface would cause excessive temperature levels in the forward auxiliary equipment rack. Ground testing and calculations indicated that the temperatures in flight would not be excessive as long as the mating ring and diffusion barrier joining the SNAP 10A to the Agena were painted with a high emissivity coating. Interface temperatures measured during the flight test confirmed the calculations.

6. Secondary Payloads

In addition to the primary payload — SNAP 10A — the Agena carried a number of secondary payloads. There were a total of eight experiments, seven of which remained with the vehicle throughout the flight. All seven returned data, and six yielded useful results. Since the secondary payloads were the responsibility of the vehicle contractor, only brief descriptions of each experiment and the results obtained have been included in this report.

a. Ion Engine

The most important secondary payload was a cesium-fueled ion engine which was designed to produce approximately 0.002 to 0.003 lb of thrust along the vehicle's neutral axis, where it would produce no force to alter vehicle attitude. The system was intended to demonstrate cyclic operation of the engine in space for an extended period, to establish a correlation between laboratory testing and a true space environment, and to demonstrate compatibility of the engine with the nuclear power generating system and Agena.

The ion engine was operated for the first and only time on Orbit 18. Its operation caused severe electromagnetic interference throughout the vehicle which distorted and made unusable much of the Link 2 and all of the Link 3 telemetry data (Figure 74). The EMI also affected the attitude control system causing the vehicle to slew severely. The source of the EMI has been attributed to cycling off and on of the high voltage supply due to an overload condition caused by contamination and arcing at the high-voltage terminals of the cesium ion thruster; subsequent ground testing of a similar engine showed both conducted and radiated interference levels were much higher than permitted by standard practice.⁽²⁹⁾ Although analyses of Orbit 18 data and EMI susceptibility of SNAP 10A and Agena equipment indicated that no damage had occurred and none was likely to occur from future ion engine operation, vehicle shutdown

Figure 74. Example of a Link 2 Telemetry Channel Affected by Ion Engine Generated EMI on Orbit 18

occurred before any further operation was attempted. A detailed analysis of the ion engine operation is given in Reference 29.

The decision not to operate the ion engine meant that the ion engine batteries did not require charging, thus reducing vehicle power demand by a maximum of 100 w.

b. Stability of Surfaces Experiment

The experiment was intended to measure the degradation of several thermal control surfaces of interest to LMSC and AI. (This was the only secondary payload to which AI contributed.) Changes in solar absorptivity and emissivity after exposure to the space and nuclear environment were to be determined by comparing temperature changes of the specimens to those of reference surfaces. Unfortunately several of the temperature transducers failed and data from the others, which were differentially connected, were too close to zero to yield any useful results.

c. Environmental Research Experiments

These were scientific experiments supplied by Air Force Cambridge Research Laboratory (AFCRL) and designed to investigate the space environment at 700 nm. The experiments were kept energized almost continuously following Orbit 21 to absorb the power that would normally have been used to charge the ion engine batteries. Most of the data have not yet been reduced by AFCRL. They intend to publish detailed results in the technical journals upon completion of the analysis of the data and correlation with data from other sources. The experiments are described below.

(1) Micrometeorite Detectors. Two types of detectors were used. The grid-type consisted of a series of dropping resistors, each in series with a number of frangible resistors connected in parallel. Any micrometeorite of sufficient energy to fracture a grid would have decreased the voltage across the dropping resistor, thus counting impacts. This detector operated satisfactorily for the life of the flight. No grids were broken, indicating no impact of micrometeorites greater than 10 microns diameter.

The membrane detector had three cells pressurized to one atmosphere. Micrometeorite impacts would have punctured a cell membrane, and the rate at which gas leaked out would have indicated puncture area. Of the three outputs of the membrane type detector, two indicated open circuits during the entire active life. The remaining membrane type measurement and temperature measurement of the instrument operated normally during vehicle life. This instrument, also, recorded no micrometeorite impacts.

(2) Density Gauge. This device measured the density of the upper atmosphere by collection and ionization of atmospheric molecules and measurement of the resulting ion current. The instrument operated as designed for the life of the satellite. The disturbance due to the reactor radiation was less than expected, indicating that density measurements may easily be made from reactor powered vehicles. Diurnal variations were noted which agreed with the current theories of the structure of the exosphere.

(3) Electrical Structures. This experiment measured positive and negative charges by utilizing one probe to measure electrons and another to detect positive ions and protons. The instrument operated as designed for the life of the satellite. The electron and positive ion density variations were as expected for the altitude range of the vehicle. An unexpected result was the indication of a large negative charge of the vehicle. During the flight the charge varied from -9 to -30 v with respect to the ambient plasma flux.

(4) Erosion Gauges. The instrument consisted of a crystal-controlled oscillator connected to a scaler-type frequency meter. The crystal was exposed to the direction of satellite travel. Erosion was measured by change of frequency as the crystal became thinner. The instrument operated as designed for the life of the satellite. The data agreed with those previously obtained on other orbiting vehicles. The change in output indicated an upper erosion limit for the gold surfaces of 10 Å per year. The attempt to measure long-term effects did not succeed due to the limited active life of the vehicle.

(5) Impedance Probe. The probe measured antenna impedance changes as the satellite circled the earth. From these changes electron densities along the orbital path were calculated. Significant data were collected for the life of the satellite. The 5-megacycle portion of the experiment failed prior to the 26th orbit. Fortunately the flight data were in the three megacycle range and the

loss of the higher frequency did not markedly degrade the experiment results. The data were of good quality and agreed with those obtained from previous flights. The analysis of the overall electron density mapping will be very difficult due to the scattering of the collection of the data with respect to time and geoposition.

d. Ionospheric Beacon

The beacon was a self-contained satellite which was ejected as programmed during the first orbit. Signals from the beacon were to be used to determine the geodetic position of points on the earth.

7. Conclusions and Recommendations

The Agena flight test data which were examined for this report and the results reported by LMSC^(1, 5, 15, 29) all indicate that the Agena performed satisfactorily prior to the system shutdown. None of the minor discrepancies or anomalies observed up to Orbit 552 seriously affected either the performance of SNAP 10A or the analysis of its performance. A number of excellent recommendations for improvements in future SNAPSHOT Agena vehicles have been made in Reference 15 and will not be repeated here. The following suggestions are only advisory since they have not been coordinated with the Agena contractor. They are listed in descending order of importance.

- 1) There is evidence of possible voltage regulator overstressing, indicated by both power and temperature data. More conclusive evidence of excessive temperatures could be obtained by processing existing flight tape recorder data for the voltage regulator electronics temperature measurement, C182. These data have not been processed previously either on-line (with the exception of Orbit 19 data which were affected by ion engine noise) or off-line. It is also recommended that analyses and/or tests be performed prior to a second flight test to better define the conditions which overstress the individual regulator components.

- 2) Some of the bias errors on individual telemetry channels could be eliminated by more precise end-to-end calibration of each telemetry channel. A check of this type was performed on the integrated SNAP 10A electrical mockup-flight Agena during final acceptance testing at Sunnyvale, but the usefulness of the data for calibration purposes was mitigated by inaccuracies in the input voltages.

3) An analysis of the loads on the positive regulated 28-v bus and phase bc of the 400 cps bus could be made to determine whether they were really overloaded during the active control phase, and whether such overloading could have been detrimental to the power supplies.

4) An investigation of the low noise levels and higher accuracies of the 5 v and 20 mv sub-multiplexers, compared to the 50-mv sub- and main multiplexers, might lead to an explanation of the phenomena and to improvements in the accuracy of future telemetry systems.

K. DATA SYSTEM ACCURACY

The SNAP 10A instrumentation and the Agena telemetry system have been described in Sections IV.H and IV.J. Telemetry data from the satellite were acquired by one of three tracking stations, Hawaii (HULA), Vandenberg (COOK), and New Hampshire (BOSS). Each of the stations was equipped to process the data in real and near real time for operational control and quick-look analysis (on-line processing) and to record raw telemetry data for later processing by LMSC in Sunnyvale, Calif. (off-line processing). In either case the raw data were first recorded on magnetic tape, then demodulated, demultiplexed, and finally converted to the original voltage input to the submultiplexers. Although the same procedure was followed for both on-line and off-line processing, there were differences in the equipment used and the primary emphasis. On-line, it was most important to have results quickly and to present them in a form suitable for rapid interpretation. As a result accuracy was impaired and the form of the data was not suitable for extensive detailed performance analysis. Off-line, the emphasis was on accuracy and presentation of the results in a form compatible with further processing by digital computers. However, off-line results were not normally available until two weeks or more after the data were received from space. As a result, on-line data were used as the basis for much preliminary performance analysis. When off-line data became available, it was necessary to reconcile the differences between the two types of data. For this reason both on-line and off-line data accuracy are discussed, even though off-line data were used almost exclusively in the preparation of the report. Reference 2 contains some tabular and graphical comparisons of on-line and off-line data as well as a more thorough description of the SNAPSHOT data system.

Although the quality of SNAPSHOT test data was reasonably good for telemetry data, the data exhibited much more error than exhibited by S10A ground test data. The analysis of flight test data was thus greatly complicated since the data scatter from orbit to orbit was generally larger than the real changes in data levels at full power. A substantial effort was made, therefore, to determine the accuracy of the flight test data and the ways in which the data were inaccurate so that (1) variations due to inaccurate data could be differentiated from real system performance changes, and (2) the data could be treated to eliminate as many of the sources of error as possible. This effort was successful and resulted in quantitative estimates of SNAPSHOT data errors. The most significant results are summarized below.

1) Off-line data were both more accurate and somewhat more precise than on-line data. The differences in system performance parameters as indicated by on-line and off-line data were reconciled. The differences were caused by bias errors in the on-line results.

2) Both on-line and off-line data were clean, i. e. the amount of random error was relatively small. It is improbable that the off-line total random error exceeded $\pm 3\%$ of full scale on any channel, and on 75% or more of the channels, the random error was less than $\pm 2\%$ full scale. This low random error made it possible to determine trends fairly accurately. For example, with expanded scale instruments, it was possible to measure reactor outlet temperature changes as small as 3°F between passes. The effect of random error on some important system parameters is shown in Table 28, in the column labeled precision.

3) Each telemetry channel, hence each SNAP 10A instrument output, was subject to a permanent bias which ranged from zero to as much as 5% full scale. It is probable that 15 to 20% of the channels were biased by more than 3% full scale, but the biases of individual channels could not be determined in most cases. The absolute accuracy of most instruments, which was determined by bias and to a lesser extent random error, was significantly less than their precision, determined by random error only. Only when several instruments were used to measure the same parameter was there any assurance that the bias error was not large; e. g. , the probable range of

off-line bias error (excluding random error) in reactor outlet temperature was reduced from $\pm 3.4\%$ F.S. ($\pm 26^\circ\text{F}$) to $\pm 1.2\%$ F.S. ($\pm 9^\circ\text{F}$) by averaging the readings of eight instruments.

4) The overall accuracy of the telemetry data was substantially poorer than it first appeared. This was entirely due to previously undetected bias errors. The median accuracy of off-line data was approximately $\pm 2.4\%$ F.S., and it was probable that one-third of the channels did not meet the off-line accuracy requirements of $\pm 3\%$ F.S. The net effect of both bias and random error on some important system parameters is shown in Table 28 in the accuracy column.

TABLE 28

ESTIMATED ACCURACY OF SOME IMPORTANT SYSTEM PARAMETERS

	Typical Value	Precision ⁽¹⁾	Accuracy ⁽²⁾
Reactor Outlet Temperature ($^\circ\text{F}$)	1030	± 5	± 10
Change in Reactor Outlet Temperature ($^\circ\text{F}$) (Measured by average of 2 RTDs or suppressed zero TCs)	----	± 3	± 3
Converter Power Output (w)	550	± 16	---
Power Consumed by Payload (w)	350	± 34	---
Converter Voltage Output (v)	30	± 0.2	± 0.4
Voltage Regulator Output (v)	29	± 0.3	---
Flowrate (gpm)	14	± 0.10	---
Change in Control Drum Position (degrees)(Average of 2 fine range PnI)	0.528 or 1.056	± 0.6	± 0.6

- (1) The limits shown are 95% confidence limits on the average reading for a single pass considering only the total random error. The standard deviation for each instrument was assumed equal to the measured average standard deviation for that sub-multiplexer. Where a value was calculated using the output of two or more instruments, propagation of error techniques were used to calculate the standard deviation.
- (2) The limits shown are 95% confidence limits on the average reading for a single pass considering both bias and random error. Because the bias error of any particular instrument could not be computed statistically, the accuracy was calculated only for parameters measured by several instruments or for changes in parameters when the accuracy and precision were equal.

5) There was a definite bias in on-line data from different tracking stations which was due to equipment maladjustment at Hula; there was no station bias in off-line data. There was a difference in total random error and telemetry noise in data from different tracking stations, Hula having larger errors both on-line and off-line.

6) Both the total random error and telemetry noise error were significantly smaller for the two slowest submultiplexers. The reason for this is not known. It had been expected that the reverse would be true, and that the faster multiplexers would be more accurate because there were more samples during a pass to average out telemetry noise. However, since telemetry noise during a pass generally proved to be the least significant source of error, this effect was not important.

The errors in the telemetry data were examined using a mathematical variance model:

$$x = \mu + \beta + \rho + \frac{K\epsilon}{\sqrt{n}}$$

where

x = average instrument reading during a pass after telemetering and processing

μ = actual average instrument reading during a pass at the SNAP 10A-Agena interface. (Errors in the SNAP 10A instrumentation were ignored since they were small compared to signal conditioning, telemetry, and processing errors.)

β = error due to bias

ρ = error due to random effects between passes

ϵ = error due to random telemetry noise during a pass

K = statistical constant

n = number of samples averaged during a pass

To meet the SNAPSHOT accuracy requirement,⁽³⁰⁾ x had to lie within $\pm 3\%$ F.S. of μ ; (*) or, in terms of error rather than accuracy, the sum of the three types of error for any pass had to be less than $\pm 3\%$ F.S.

Some general statements can be made about the sources of each of the three types of errors. Each instrument (i. e. each sub-multiplexer channel) was subject to a unique and unknown bias which remained constant throughout the flight. These biases were very difficult to accurately estimate since the actual values of the readings at the SNAP 10A/Agena in space were unknown. However, a comparison of readings from instruments measuring the same parameters (including 25% of the total instruments) showed differences, assumed to be biases, ranging from 0 to 4.7% F.S. and averaging a little more than 1% F.S. These biases were probably mainly due to signal conditioning, impedance mismatching, and telemetry offsets, rather than S10A instrumentation or ground processing errors. The evidence for this conclusion was (1) the same SNAP 10A instruments were not biased on the ground, either on the FS-4 system or on other ground test systems where launch effects were simulated by qualification level shock and vibration testing; and (2) the individual biases were the same both on-line and off-line (after correction for known on-line biases affecting all channels) and they remained constant, on the average, regardless of the tracking station or the time the data were processed by the LMSC ground station.

In addition to the constant bias on each channel, on-line processing by the tracking stations introduced bias because of data truncation at all stations and out-of-adjustment equipment at Hula prior to 4/28/65. These biases were not constant but fluctuated from instrument to instrument and pass to pass around the average values shown in Table 29.

The error due to random effects between passes was the largest contributor to the total random errors shown in Table 29. Because these errors were random only when considered over a large number of passes, they caused some data from a single pass to be biased with respect to the adjacent

(*) The statistical meaning of the required accuracy limits was not given by Reference 30. The required accuracy limits of $\pm 3\%$ F.S. have been assumed to be 2σ or 95% confidence limits, i. e. only one out of twenty times will x be more than $\pm 3\%$ F.S. from μ .

TABLE 29

RELATIVE ERRORS IN SNAPSHOT TELEMETRY DATA⁽¹⁾

	Bias Error Compared to Boss Off-Line, % of Full Scale ⁽²⁾		Total Random Error per Pass Relative to Boss Off-Line ⁽³⁾		Telemetry Noise During a Pass Relative to Boss Off-Line ⁽³⁾	
	Off-Line	On-Line	Off-Line	On-Line	Off-Line	On-Line
Tracking Station						
Boss	0	$\left. \begin{array}{l} -0.7 \\ +1.2 \end{array} \right\}^{(4)}$	1.0	$\left. \begin{array}{l} 1.0 \\ 2 \end{array} \right\}$	$\left. \begin{array}{l} 1.0 \\ 1.4 \end{array} \right\}$	$\left. \begin{array}{l} 1.5 \\ 1.5 \end{array} \right\}$
Cook	0		1.0			
Hula	0		1.4			
Multiplexer						
Main	-	--	$\left. \begin{array}{l} 1.3 \\ - \\ 0.5 \end{array} \right\}$	-	1.0	-
50 mv Sub.	-	--		-	1.2	1.8
20 mv Sub.	-	--		-	0.7	$\left. \begin{array}{l} 0.9 \end{array} \right\}$
5 v Sub.	-	--		-	0.6	

NOTES: (1) The values shown are typical or average values only. For example, on a specific Boss pass, a given instrument (sub-multiplexer channel) may have read higher on-line than off-line even though, on most passes, most instruments read lower on-line than off-line, averaging -0.7% F.S. over many passes and instruments.

(2) A negative bias indicates that the data were lower than Boss off-line.

(3) The values shown were calculated as follows:

$$\text{Error relative to Boss off-line} = \frac{\text{calculated standard deviation}}{\text{calculated standard deviation for Boss off-line}}$$

The typical total random error standard deviation for Boss off-line was 0.65% F.S. and included the net contribution from noise. The typical noise standard deviation for Boss off-line was 0.75% F.S.

(4) Prior to April 28, 1965.

passes. Although the amount of random error and bias were unique for each instrument, the random error varied much less than the bias from instrument to instrument and was strongly influenced by the tracking station which recorded and/or processed the data and by the sub-multiplexer to which the instrument was connected. The data in Table 29 show the influence of tracking station and sub-multiplexer.

The telemetry noise during a pass had relatively little effect on the accuracy of the average instrument reading during the pass since so many data points were used to calculate the average. Only for the two slowest sub-multiplexers, 20mv sub and 5v sub, was noise a significant contributor to error; only for the 5-v sub was noise the larger contributor. The main effect of telemetry noise was to obscure trends and changes. This was especially true on tape recorded data — where the noise was from two to four times worse than for real time data — on full range instruments, and on slow sampling rate multiplexers. As with the total random error, the amount of noise was unique for each instrument but was more strongly influenced by the tracking station and sub-multiplexer (Table 29).

Histograms compiled from the flight data showed that both the total random error and the telemetry noise error were normally distributed.

Table 29 shows the effect of method of processing (on-line versus off-line), tracking station, and sub-multiplexer on the relative size of the three types of telemetry error which were measurable on SNAPSHOT. The results shown support the conclusions made earlier concerning the effects of each of these three sources on the different types of errors.

The conclusions and recommendations resulting from the data system accuracy analyses are summarized in the following paragraphs.

As expected, telemetry errors in flight proved to be much larger than data errors on SNAP 10A ground tests. In spite of this it was possible to analyze flight test results in detail by recognizing the sources of error and either using the data least susceptible to error or analytically correcting for the errors. Prior to the flight it was expected that telemetry noise would be the primary source of error in the flight test data, and it was not anticipated that bias errors would be so large or so frequent. As a result the low noise

levels observed in flight gave rise to the initial and misleading conclusion that the data system accuracy was better than expected. The need for improvement in accuracy is demonstrated by some of the data in Table 28. For example, the amount of error in control drum position was larger than the change resulting from a control drum step, and the amount of error in the change in reactor outlet temperature was the same as the total change during a week of operation at the time of shutdown.

As a result of the work done to find the sources and types of error in the flight test data, a number of improvements and changes in SNAP 10A instrumentation were suggested in Section IV. H of this report. In addition, improvements can be made in the on-line data analysis procedures and the off-line data reduction and analysis computer programs to increase both their accuracy and efficiency. The most important of these improvements would use the average data from all instruments reading the same parameter for a large number of passes to estimate the in-flight bias of each instrument. Only those instruments least susceptible to random error (and corrected for bias) would then be used to measure changes in the parameter during the test.

It was also found that it was necessary to perform statistical calculations using actual flight test data in order to determine the most accurate and efficient methods of calculating various parameters. This suggests that it would be useful to have two separate computer programs, one to supply the required data and perform the statistical calculations and another, more flexible program, which could be changed to incorporate the findings of the first program.

L. THERMAL CONTROL COATINGS

All component surfaces of the SNAP 10A system contained a coating or finish either applied or natural to the substrate. The design, application, and testing of the coatings are described in References 25 and 33. Of particular interest in this effort were evaluations of the (1) Al93 coating on the thermoelectric converter and pump fins; (2) gold on molybdenum foil utilized on liquid metal piping; and (3) the instrument compartment coatings.

The use of Al93 coating on the pump and converter fins was predicated on the fact that a white coating which exhibits a high thermal emittance and a low solar absorptance maximizes the net heat rejection from the radiation surface. The

gold on molybdenum foil was utilized as radiative insulation on the high temperature liquid metal components. In order to keep the instrumentation cool, it was necessary to reject heat emanating from instrumentation and leaking from the thermoelectric converter. High emissivity, low solar absorptivity coatings were used on both the inside and the outside of the instrument compartment. Information utilized for prediction of coating performance was obtained from optical measurements performed on the FS-4 system prior to flight.

Analysis of data indicated the following: (1) The thermal emittance of the Al93 coating in space was not significantly different from the value of 0.92 measured on the ground. This value varied by no more than $\pm 2\%$ throughout the 43 day SNAPSHOT mission. (2) The solar absorptance of the Al93 coating during the early portion of the test was 0.46, compared to the value of 0.50 measured on FS-4 following ground thermal tests. Test data indicated a 6% reduction to a value of 0.43 during the duration of the test. The behavior with time in space may be explained by a self-cleaning of the carbonaceous residue present at lift-off or by the sensitivity of the relationship used to calculate solar absorptance to data scatter.

Analysis of FS-3 data proved that the low emittance property of the gold on molybdenum foil remained stable throughout 5000 hours of system operation. Correlation of FS-3 and FS-4 data indicated that the coating was also stable in space.

The instrument compartment was maintained well within its design temperature limits, and there was no evidence that coating properties changed during the SNAPSHOT flight.

Prior to the flight test, the thermal emittance and solar absorptance properties of the FS-4 system were measured twice: (1) prior to NaK loading of the system, and (2) following the thermal reference test (Table 30). The evaluation of coating performance during flight used these data as initial reference conditions.

TABLE 30
OPTICAL PROPERTIES OF THE FS-4 SYSTEM

Component and Coating ⁽¹⁾	Thermal Emittance		Solar Absorptance	
	FS-4 pre NaK Loading	FS-4 After Thermal Test	FS-4 pre NaK Loading	FS-4 After Thermal Test
T/E pump radiator fin (Al93)	0.94	0.95	0.40	0.41
T/E converter radiator fin (Al93)	0.94	0.92	0.39	0.50
Control drum - outer surface (beryllium)	0.09	0.13	(2)	(2)
Reflector-outer surface (Cr ₂ O ₃)	0.88	0.86	(2)	(2)
Support shell (titanium)	0.11	0.17	(2)	(2)
Radiation shield container (stainless steel)	0.24	0.34	(2)	(2)
Instrument compartment interior (PT404A)	0.90	0.85	(2)	(2)

1. Measurements of the core vessel and the instrument compartment exterior could not be performed due to inaccessibility.
2. Solar absorptance could not be measured because of inaccessibility.

1. AI93 on Thermoelectric Converter Fins

For a spacecraft coating operating at elevated temperatures ($\sim 650^\circ\text{F}$), the error or uncertainty associated with determining an absolute emittance value is a strong function of the error or uncertainty in temperature measurement since $d\epsilon/\epsilon$ is a function of $(4 dt/T)$. For example, at 650°F a 1°F error or uncertainty in temperature affects the emittance by 0.36%. The minimum possible error in absolute radiator fin temperature in flight could be obtained by averaging the readings of the four inlet fin thermocouples and applying an analytical correction to obtain the average radiator temperature. The absolute error was $\pm 13^\circ\text{F}$ or an absolute emissivity error of $\pm 5\%$. The average radiator temperature uncertainty was further compounded by the fact that it was obtained from temperature readings based on only four fins sampled from the 2880 fins making up the radiator surface.

In brief, any attempt to ascertain an absolute emittance of the AI93 coating from the FS-4 flight data resulted in a large associated error or uncertainty. Thus the good agreement between measured and predicted heat losses in space, reported in Section IV.M, indicated only that the emissivity in space was not significantly different from that measured on the ground, i.e. $\epsilon = 0.92 \pm 0.05$.

A previous heat transfer analysis⁽³⁴⁾ performed on the FSM-1 system test yielded an excellent approach for determining the AI93 coating stability. During this test, it was determined that the total converter heat losses were primarily made up of three losses. Eighty-six % was transferred to space by the radiator platelets, 11% was transferred through the voids between individual radiator platelets and, leakage losses through the ends of the truncated conical support structure accounted for the remaining 3%.

Reference 34 showed that the 11% heat loss was independent of any emittance changes occurring on surfaces underneath the platelets, e.g., the NaK tubing or titanium structure. This heat loss was a function only of structure temperature, and analysis of FS-3 data throughout 5500 hours of operation showed this temperature to be a constant when normalized to the NaK converter inlet temperature. The 3% leakage loss was also considered a constant since it ultimately was rejected from stable, low temperature surfaces such as the PT404 coating on the instrument compartment. The following relationship can therefore be made,

$$\begin{array}{rclcl} \text{Heat Delivered to} & = & \text{Heat Rejected} & + & \text{Heat Rejected} & + & \text{End Losses} \\ \text{Converter Section} & & \text{from Radiator} & & \text{Through Radiator} & & \\ & & \text{Platelets} & & \text{Platelet Voids} & & \end{array}$$

With the last two terms being essentially constant,

$$WC_P \Delta T = \epsilon A \sigma (T_r^4 - T_s^4) + K$$

where,

W = converter flowrate

C_P = NaK specific heat

A = effective radiator area

T_r = average radiator temperature

T_s = equivalent space temperature

ΔT = converter NaK temperature difference

ϵ = radiator emissivity

K = heat loss independent of emittance

Rearranging

$$\epsilon = \left[\frac{W \Delta T - K}{A \sigma (T_r^4 - T_s^4)} \right]$$

for equal environmental conditions.

Table 31 shows the parameters used to analyze the emittance stability. The measurements taken during the six orbits shown were all made in as near full shade condition as possible so as to negate the effects of solar input on the heat balance.

TABLE 31

CONVERTER THERMAL EMITTANCE HISTORY

Orbit No.	Time (hr)	Average Fin Temperature (°F)	NaK Temperature Drop, Converter Inlet to Outlet (°F)	NaK Flow Rate (gpm)	Emittance (Assumed as 0.92 on Orbit 18)
18	20	647.5	119.0	15.01	0.92
19	22	643.1	121.8	15.06	0.95
212	388	650.9	129.2	14.02	0.92
237	428	647.9	129.2	13.95	0.93
430	786	640.5	131.6	13.51	0.93
494	935	640.3	132.2	13.48	0.93

The table shows that the thermal emittance of the AI93 coating system remained constant throughout the 43 day flight. (The minimum detectable change in emissivity was limited to ± 0.02 by telemetry error.)

The solar absorptance was determined by comparing test data while the vehicle was in the sun with that while in the shade. Equations describing the heat balance on the radiator fins were written for both sun and shade conditions. By subtracting one equation from the other, the following relationship was obtained

$$\alpha_s = \frac{\epsilon A \sigma}{SA} \left[T_{\text{sun}}^4 - T_{\text{shade}}^4 \right] + \left[\frac{Q_{\text{shade}} - Q_{\text{sun}}}{SA} \right]$$

where,

Q_{sun} = reactor thermal power delivered to converter fin in sun

Q_{shade} = reactor thermal power delivered to converter fin in shade

S = net heat flux incident on the radiator surface due to direct and earth reflected solar radiation

A = effective radiator fin area

α_s = solar absorptivity

ϵ = emissivity of radiator surface

T_{sun} = radiator temperature in sun

T_{shade} = radiator temperature in shade

Two approaches using this equation were considered. One method compared test data from one orbit while in the shade with data from another orbit while in the sun. The problem with this approach was that data scatter from one orbit to another caused large variations in the calculated solar absorptance. The preferred method, the results of which are shown in Table 32, considered orbits in which the vehicle was in the shade at the time of vehicle acquisition, entered the sun shortly thereafter, and remained in the sun until vehicle fade.

TABLE 32
CONVERTER SOLAR ABSORPTANCE HISTORY

Orbit No.	Time (hr)	Fin Temperature in Sun	Fin Temperature in Shade	Solar Absorptance
17	17	654	633	0.45
30	42.9	651	631	0.45
45	70.8	654	637	0.47
146	258.6	654	639	0.46
276	500.3	653	637	0.44
403	736.5	641	623	0.41
494	905.6	634	618	0.44

A fit of the data indicated an initial value of about 0.46 compared to the measured value following the thermal reference test of 0.5; or about 8% below the prediction. The best estimate at the time of shutdown was 0.43, indicating an apparent decrease in solar absorptance with time.

Definite conclusions about the initial 8% difference and the apparent 6% decrease with time could not be made. A possible explanation is that a carbonaceous residue was present on the converter radiator fins upon completion of the ground thermal test. (The absorptance did increase following thermal testing. See Table 30.) Previous testing has shown that a residue of this type will slowly evaporate when subjected to a high temperature in a clean environment such as space. Another explanation is that the sensitivity of the above relationship to errors in measurement of fin temperature and thermal power and calculation of solar input caused the observed change.

2. Low Emittance Gold on Molybdenum Foil

No temperature measurement of the low emittance gold on molybdenum foil was made during the FS-4 flight and its stability could only be inferred. Since the converter efficiency and power output decreased normally during the flight, it was concluded that no major emittance degradation of the foil occurred. Information from the FS-3 system test was analyzed to support this conclusion. It was shown in Reference 34 that temperature changes in the converter structure could be caused only by changes in the amount of thermal energy being reflected, absorbed, and re-emitted by the structure. The source of most of this thermal energy was radiation losses from the NaK tubing, 75% of which was shielded with gold-plated Mo foil. Thus, a relation existed between the emittance stability of the gold-Mo foil and the temperature stability of the support structure. Analysis of FS-3 data showed that throughout the first 5000 hours of system operation the support structure temperature, normalized to the initial converter NaK inlet temperature of 1033°F, varied at most by 4°F from the nominal structure operating temperature of 750°F. Relatively small foil emittance increases would have resulted in structure temperature increases well above these limits. Such a temperature increase would have caused a sharp increase in shunt heat losses from the structure and these would have been indicated in flight by unusual decreases in electrical power output and converter efficiency.

3. Instrument Compartment Coatings

As previously discussed, the temperature measurement error made absolute emittance and emittance stability determinations difficult. This was particularly true for coating surfaces operating at low temperature. Instrument compartment temperature changes due to a change in compartment coating emittance would have been masked by the measurement uncertainty.

However, acceptable coating performance was indicated since the design operating temperature was 120°F, and the flight data showed that the mean compartment temperature operated in a range of 95-110°F and decreased normally as reactor outlet temperature decreased.

4. Conclusions and Recommendations

Thermal emittance and solar absorptance could not be accurately measured in space. However, it was possible to determine that they were not significantly different from the values measured on the ground and that there was no significant degradation with time. Thermal control surface degradation measurement could be improved on future flights by more accurate radiator temperature data collected over entire orbits.

M. THERMAL PERFORMANCE AND SUN/SHADE EFFECTS

Because of the greater radiator heat rejection capability in space, SNAP 10A system heat losses were higher in space than they were on system ground tests conducted in a vacuum vessel. Since the increase in heat radiated from the thermoelectric converter in space was the cause of increased electrical power output and efficiency in space, determination of system heat losses during ground tests and extrapolation of these losses to space was the subject of a major analytical effort.⁽³⁵⁾ The sources of the individual heat losses were also considered in detail so that they might be reduced on advanced systems.

The SNAPSHOT vehicle was launched into a sun/shade transient orbit in which it periodically passed through the earth's shadow. As a result of this orbit, the solar flux incident on SNAP 10A, and hence the effective sink temperature, changed continuously. These sun/shade effects were of major concern during the pre-startup phase discussed in Section III. B. However, they also affected the system during full power operation. Many system parameters responded to changes in solar inputs. Because one of the suggested advantages of nuclear space power systems is that they require no orientation with respect to the sun and are relatively insensitive to changes in solar inputs, the actual full power response of SNAP 10A to sun/shade transients was of interest. In addition, the scatter introduced in a number of parameters by sun/shade effects was large enough to obscure long-term changes such as thermoelectric converter degradation. Hence it was necessary to quantitatively evaluate the sun/shade effects to eliminate this scatter.

The number of flight temperature measurements was not sufficient to allow calculation of all the individual heat losses measured on the ground. Only the total system heat loss (i. e. reactor thermal power) and a few of the major loss terms could be calculated in flight. These measurements, however, were sufficient to verify the extrapolated space heat losses. Reactor thermal power in space was calculated calorimetrically from flow and reactor temperature rise measurements. After an in-flight calibration based on the calorimetrically determined power, the neutron leakage flux instrument was used to measure reactor thermal power directly. It was less subject to random error than the calorimetric power and was also more accurate during transients caused by reactivity insertions.

Preflight calculations indicated that total system heat losses would be 5% higher in space than on the ground for an effective space temperature of 0°F. This was confirmed by flight measurements. The initial values of thermal emittance and solar absorptance in space were not significantly different than indicated by the final ground measurements, a fact confirmed by the good agreement of measured and predicted heat losses and sun/shade effects.

The effects of the changing solar heat inputs were surprisingly evident once the data had been treated to eliminate extraneous noise. Radiator, converter tube outlet NaK, reflector surface and instrument compartment temperatures, converter leg voltages and electrical power all changed rapidly as soon as the system came out of the earth's shadow into the sun. Changes in reactor thermal power and average temperature and flow were smaller, but still measurable. The changes in both electrical and thermal power from sun to full shade were predictable using simple relationships based on the methods used to extrapolate electrical power output to space. Methods were developed to eliminate data scatter due to sun/shade effects by using flight test results. It was determined that an effective space temperature of 0°F corresponded to the average, rather than the maximum, space heat input when the system was in the sun.

1. Thermal Performance

System heat losses increase in space because the controlling resistance to heat transfer is radiation from the system surfaces to the vacuum chamber walls or to space. In space, this resistance is reduced by the three factors listed below.

1) The effective sink temperature of space is lower than the temperature of the vacuum wall chamber. The average effective space temperature in the sun is 0°F. In comparison the FS-3 vacuum chamber wall temperature was 123°F. All system heat losses increased because of this factor, losses from the lower temperature surfaces being increased by greater percentages than those from high temperature surfaces. However, the absolute amount of the increase was small for all but the converter radiator.

2) No heat was reflected back to the system in space. In contrast to this, the vacuum chamber wall absorptivity was only 0.85, and its diameter (8 ft) was of the same order of magnitude as the average converter diameter

(2.9 ft). As a result, a significant amount of heat radiated from the system was reflected by the chamber wall back to the converter radiators. The other system surfaces were too small to reabsorb a significant amount of reflected heat.

3) Wiring trays with low absorptivity surfaces located in the vacuum chamber slightly obstructed the view that some system surfaces, principally the converter radiator, had of the chamber wall. There were no such obstructions in space.

Detailed system heat balances for both ground and space are given in Table 33. Also shown are those losses which could be measured in space and the expected ratio of space to ground values for the more important efficiencies, losses and powers. The losses were calculated using the methods of Reference 35. The original prediction of reactor thermal power in space which was made in Reference 36 (based on FSM-1) was only 0.3 kw different from that shown in Table 33 when adjusted to the same average temperature.

Several of the results shown in Table 33 deserve mention. First there is excellent agreement between the measured and expected values in space. The agreement is so good that part of it must be attributed to chance. Next it is evident that both the overall system and converter efficiencies increased in space. The reason for this can be understood by examining the individual heat losses. While nearly all the losses increased, only the heat rejected by the converter radiator increased by a substantial amount; it accounted for 95% of the increase in reactor thermal power. Since nearly 90% of this additional heat was conducted through the thermoelectric pellets, it can be said that $0.95 \times 0.90 = 85\%$ of the additional heat loss in space was directly useful in producing electrical power compared to only 75% usefulness of the total ground test heat loss. Finally, it can be seen that the reactor functioned as a constant temperature heat source, producing as much thermal power as system heat losses demanded. It would have been possible to as much as double the power output of the reactor by increasing the heat rejection capability of the system. Because so much extra heat generation capacity was available on SNAP 10A, it was not necessary to expend a great deal of effort to reduce all the individual losses, although there was one exception. Care was taken to reduce those losses which shunted heat from other parts of the system to either the converter or pump radiators

TABLE 33
DETAILED SYSTEM HEAT BALANCE (kw)

	Measured on Ground		Extrapolated to Space		Measured in Space	Expected Ratio of Space to Ground Value
Average Reactor NaK Temperature (°F)		970		970	970	
Average Sink Temperature (°F)		123		0	0	
Overall System Efficiency (%)		1.27		1.35	1.35	1.063
Overall Converter Efficiency (%)		1.35		1.43	1.43	1.067
Reactor Thermal Power		40.90		42.95	42.96	1.051
Gains -Reflector Internal Heating	0.48		0.50			
Losses -Reflector	0.83		0.87			
Reactor Top	0.29		0.30			
Reactor Bottom	0.24		0.25			
Total	0.88		0.92			1.045
Heat Gain by NaK In-Core		40.02		42.03		
Gains -Neutron Shield Internal Heating	0.17		0.18			
Losses -Pump	1.03		1.07			
Supply Lines	0.15		0.15			
Reactor Support Legs	0.59		0.60			
Neutron Shield	(0.84)		(0.88)			
Total	1.60		1.64			1.025
Heat to Converter		38.43		40.39	40.3	
Losses -Upper Torque Box	0.81		0.84			
Structure to Neutron Shield	0.13		0.13			
Structure to Sink	1.43		1.47			
NaK Legs to Sink	0.33		0.34			
Return Lines to Structure	(0.14)		(0.14)			
Lower Torque Box	0.80		0.85			
Instrument Compartment	0.50		0.50			
Low Pressure Convection	0.15		0.00			
Electrical Power Produced	0.518		0.577		0.578	1.113
Total	4.66		4.71			1.011
Heat Rejected by Converter Radiator						
From Heat Balance		33.77		35.68		1.057
From Measured Radiator Temperature		(35.0)				
Shunt Losses to Radiator	3.17		3.4			1.07
Conducted through Thermoelectric Pellets		30.60		32.28		1.055

NOTES 1) Measured ground data is from FS-3 test at beginning of passive control period. Electrical power output corrected for 1-1/2% degradation in first 72 hr of operation. No degradation of thermal control surfaces.

2) Terms shown in parentheses are for information only and are not included in the heat balance. Terms may not balance exactly because of rounding. Data are accurate to one decimal place only, second decimal place is given only to show difference between losses on ground and space.

since these shunt losses directly reduced the useful heat rejection capability of the radiators and this capability was already the limiting factor on power production.

Figure 17 showed the measured and predicted reactor thermal power as a function of time during the passive control period. The predicted values were obtained by adjusting the measured FS-3 thermal power to the predicted SNAP-SHOT average temperature and then extrapolating to space conditions as in Table 33. Since the predicted average temperature was slightly higher than that measured, the predicted thermal power was also high, by about 0.08 kw/°F. The shade data points are all higher than the prediction which was made for sun conditions. This is discussed below under sun/shade effects.

2. Sun/Shade Effects

The space heat input to the SNAP 10A converter radiator for the actual SNAPSHOT orbit and measured FS-4 solar absorptivity is shown in Figure 75 as calculated by the methods of Reference 37. It can be seen that the input varied not only as a function of latitude but also as a function of the date, although this second effect was not large during the active life of SNAPSHOT. The change with latitude was due to the changing vehicle orientation with respect to the sun as the vehicle circled the earth and was repeated every 111.6 minutes. The changes with date were more complicated and were due to two factors listed below.

- 1) The plane of the satellite's orbit was fixed in inertial space by the date and time of launch. As the earth followed its orbital path around the sun, the angle between the plane of the satellite's orbit and the plane through the earth's poles joining the centers of the earth and sun decreased at the rate of $360^\circ/365 \text{ days} = 0.98^\circ/\text{day}$. This angle was initially 21° (Figure 11), decreased to 0° by April 25 (equivalent to a noon launch orbit), and had further decreased to -21° by the time of shutdown on May 16. On July 25, the angle would have been 90° and the orbit would have been a constant sun/constant shade orbit with solar inputs at a maximum and unchanging with latitude.

- 2) The earth's celestial axis is tilted 23.5° with respect to the sun's ecliptic axis (Figure 11). (The plane in which the earth revolves around the

Figure 75. Calculated Space Heat Input to S10A as Function of SNAPSHOT Orbital Position

sun every 365 days is known as the ecliptic plane.) As a result, the sun is at the zenith, normal to the earth's surface, at the time of the equinoxes, — March 21 and September 23—far north on June 21, and far south on December 22. This affected the latitude at which the satellite entered and left the shade. For example, on March 21, the satellite would have entered and left the earth's shadow at equal latitudes south and north of the equator, respectively. However, as the sun moved closer to the solstice on June 21, the satellite entered the shade sooner, i. e., at greater south latitudes, and left earlier. This effect is evident in Figure 75, though somewhat obscured by the changing angle between the orbital plane and the earth-sun plane.

The heat adsorbed by the SNAP 10A system in space came from three sources: direct solar radiation, solar radiation reflected from the earth, and thermal radiation from the earth. The first two sources were most important since they were the larger and varied with orbital position. Their relative contribution varied according to latitude. Direct solar radiation was at a maximum near the poles, accounting for nearly all the solar input at that point, and reached a minimum near the equator, where it was nearly equaled by the contribution of reflected solar radiation. Thermal earth radiation was not

quite negligible for SNAPSHOT because of the high thermal absorptivity of the radiator surfaces, their large surface area, and the relatively low orbit. Because the earth's temperature varies little with time, the thermal earth input was constant.

The space heat inputs affected electrical and thermal power for the same reason as heat reflected from the vacuum chamber walls did; they reduced the ability of the radiator to reject useful heat. The effect of solar inputs was reduced by covering the radiators with a tailored coating which combined a relatively low solar absorptivity (0.5) with a high thermal emissivity (0.9). This coating and its performance are described in Section IV. L. It was possible to accurately predict the effect of the changing space inputs on reactor thermal power and electrical power using the methods developed to account for the effect of the change in sink temperature from ground to space. The results for a change in effective space temperature from 24°F (maximum solar input) to -167°F (full shade) are shown in Table 34. The calculated and measured results are in good agreement. Pump flowrate exhibits the same dependency on NaK to fin ΔT as during prestartup (0.01 gpm/°F).

TABLE 34
RESPONSE OF SEVERAL IMPORTANT
PARAMETERS TO CHANGE FROM SUN
TO SHADE ENVIRONMENT

	Change from Sun to Shade	
	Measured	Expected
Electrical Power (w)	+23	+20
Reactor Thermal Power (kw)	+0.55	+0.6
Flow (gpm)	+0.1	--
Pump NaK to Fin ΔT (°F)	+9	--

Figure 76 shows the typical response of a number of system parameters to the changing space heat input. The data shown were measured during Orbit 17 when reactivity was still changing at a fairly rapid rate; the dashed lines show the expected trend of each parameter due to reactivity changes. Each data

Figure 76. Measured S10A Response to Changing Space Heat Input

point represents the average value of data taken over a 54° interval of latitude and is plotted in the center of that interval. The sketches at the top of the figure show that the first and last data points were full shade values, the second point was in transition from full shade to sun, and the remaining four points were sun values.

The radiator fin temperature data in Figure 76 illustrate how the portion of the system exposed to the sun changes with latitude. Referring to the sketch at the top of the figure, it can be seen that the +Z axis fin thermocouples are the first to increase in temperature since they are the first ones exposed to the sun. The thermocouples on the other axes react in a similar fashion. One interesting result was the tendency of the thermocouples on the shaded side of the system to register lower temperatures in the sun than when the entire system was in the shade. This phenomenon is believed to be due to the reduced electrical current produced in the sun, which results in a smaller Peltier effect. It is also interesting to note that the outlet fins which are colder increase more in temperature in the sun than the inlet fins. This is reasonable since the solar heat flux to the inlet and outlet fins is the same, and the cold fin temperature must increase by more than the hot fin to reject the same incremental amount of heat.

In the introduction to this section, it was noted that sun/shade effects caused significant increases in data scatter. Using the information discussed in this section, it was possible to correct for these effects in many cases. The typical tracking station pass windows shown in Figure 75 illustrate the cause of the scatter. On any pass when the vehicle was southbound, the data obtained from all three ground tracking stations were for the vehicle exposed to about the same amount of solar heat input. Hence with the exception of those temperatures which were very dependent on latitude (e. g. , radiator fin temperatures), southbound data were relatively consistent regardless of the tracking station or date. However, the situation with northbound passes, which were commonly assumed to be full shade passes, was quite different. At the beginning of the flight test all Hula and most Cook passes were full shade, but the vehicle entered the sunlight during the middle of the Boss passes. As a result, electrical and thermal power were higher at Hula than Boss, giving the impression that there was an off-line station bias similar to that observed in on-line data. The

resulting difference in converter power could be calculated by determining the average latitude of each station's northbound pass window from Figure 75 (22° for Hula and 45° for Boss) and then finding the power output for that latitude from Figure 76 (620 w for Hula and 605 w for Boss). A slightly more sophisticated approach was used when it was necessary to correct flight test data for sun/shade effects. As the flight test continued there was first a slight increase in the amount of full shade data from northbound passes, until April 25, and then a decrease. At the time of shutdown, even most Hula northbound passes were not full shade. The difference in performance between northbound and southbound passes can be seen in a number of figures in this report, including Figures 17, 40, and 70.

Additional scatter was caused by instrument compartment temperature fluctuations (10°F peak-to-peak). Most SNAP 10A instruments were not sufficiently temperature sensitive to be adversely affected by fluctuations of this magnitude. However, a temperature effect was noted and corrected for on the calibration signal of the converter internal resistance measurement. Fluctuations with temperature were also noted on the solid state neutron dosimeters, but they were not large compared to the normal errors in flux measurements. Fluctuations were also noted in the Agena bus voltages, but is not known whether these were real or the result of instrument temperature sensitivity.

3. Conclusions and Recommendations

The good agreement between calculated and measured system heat losses in space demonstrated that the effect of the space environment was taken into account properly and that the thermal emittance and solar absorptance in space were not significantly different than the values measured on the ground. Useful knowledge of the sources of system heat losses resulted from the detailed evaluation of individual system heat losses which was necessary for space extrapolation. This knowledge can be, and has been, used to reduce heat consumption and increase efficiency of future systems.

Sun/shade effects at full power were small and predictable. They caused no significant electrical power loss or oscillation from the standpoint of spacecraft power demand or regulation. However, the ease with which the effects could be observed in the flight test data was rather surprising and contributed significantly

to the effort required to analyze flight test data. The effort required for the analysis of any future flight test can be substantially reduced by utilizing the results and experience of SNAPSHOT. The design of advanced systems should continue to take into account sun/shade effects at full power. This should include not only the effects on power production, but also the effects on instrument accuracy. The optimum approach would be to continue to reduce instrument sensitivity to the ambient temperature. In any future flight test operation, more attention should be paid to sun/shade effects when selecting (1) the orbit and tracking station on which to acquire data, or (2) the off-line data to be processed and analyzed.

N. FLOW DISTRIBUTION

The NaK flow path through the thermoelectric converter is shown in Figures 77 and 78. Two supply lines bring hot NaK from the reactor to the upper manifold which feeds the 40 D-shaped tubes. The NaK is collected in the lower manifold and returned to the reactor through two return lines. The flow through the 40 D tubes is not identical because resistance to flow between the supply and return lines is not equal for each of the 40 possible paths. Instead, the flow distribution is symmetrical about the return line axis with the flow in individual tubes varying by up to $\pm 28\%$ from the average. Figure 79 shows a typical flow distribution characteristic of all SNAP 10A ground test systems operating at full power conditions. Since the largest resistance to heat transfer across the thermoelectric pellet is radiation from the fin to space rather than conduction across the NaK film, differences in flow from tube to tube do not affect the total amount of heat transferred in direct proportion but primarily affect the tube NaK inlet to outlet ΔT (that is, higher flow tubes have smaller ΔT s). The average NaK temperature and temperature drop across the thermoelectric elements are affected, causing the voltage and power output to vary from tube to tube. Because the original SNAP 10A design electrical power output was based on the assumption of equal flow in each tube, the effect of the unequal flow distribution was calculated after it was observed on the first SNAP 10A thermal-hydraulic ground test system (PSM-3). These calculations showed that the observed flow distribution would cause a power loss of less than 3 w.⁽³⁸⁾

6-10-66

7561-03261

Figure 77. Flowpaths Through Thermoelectric Converter

6-10-66

7561-03262

Figure 78. Top View of Thermoelectric Converter

Figure 79. NaK Flow Distribution
(% of Full Flow vs Leg No.)

There was concern that the flow distribution in space might be more unequal than on the ground since the buoyancy forces which oppose the unequal flow distribution on the ground would not be present in space. The original calculations⁽³⁸⁾ indicated that the additional variation would be only $\pm 0.05\%$ of full flow per tube, and the effect on power would be negligible.

Although no problems were anticipated, operation of fluid systems in zero gravity is a relatively new problem in hydraulics. Therefore, the flight test data were reviewed in detail to verify the calculations by (1) determining that there were no unexpected effects, and (2) comparing the measured to the predicted effects. The results obtained are as follows:

1) There was no evidence of either converter tube plugging or flow instability, which would have been indicated by gross changes in the flow distribution.

2) The predicted effect of the space environment was to exaggerate the unequal flow and temperature distributions observed on the ground by the small amount of $\pm 0.2\%$ full flow and 4°F maximum. This was slightly larger than the original prediction of Reference 38. The flight data were in excellent agreement with the prediction, although they were by no means conclusive because of their large uncertainty. There was, however, a definite increase in the inequality of the distribution in space compared to ground tests.

3) The predicted increase in flow of 0.2 gpm, due to elimination of gravitational buoyant forces in space, and the predicted electrical power output decrease of 3 w, due to the change in flow distribution, could not be verified. However, the measured values of flow and power in space were very close to the predicted values which took these factors into account.

1. Preflight Analysis and Test Results

A majority of the SNAP 10A ground test systems were instrumented to measure flow distribution indirectly. Thermocouples were attached to each converter tube approximately two thirds down the tube. The measured temperature drop between the inlet manifold and the thermocouple was then used to calculate the relative flow in each tube using the method of equal heat flux per tube described in Reference 38*. An example of the results obtained using this method is shown in Figure 79. While the results shown are for the non-nuclear qualification system (FSM-4), the shape of the distribution is identical to that observed on all other SNAP 10A systems. However, because of thermocouple error, the absolute values shown were susceptible to a maximum error of $\pm 0.2\%$ full flow.

*The assumption of equal heat flux per tube is not quite correct since the high flow, high temperature tubes actually lose more heat than the average and the low flow tubes less. However, the much more involved method of making a heat balance around each tube gives nearly identical results. In addition, the equal heat flux method slightly exaggerates the shape of the flow distribution, making it more conservative for the present purpose.

The major concern from an unequal flow distribution is that converter voltage and power output will suffer. Since voltage and power are dependent on average NaK temperature, the average temperature for each tube was calculated from the same data used in Figure 79. The results are shown as a solid line in Figure 80. Although individual tube temperatures varied as much as $\pm 15^\circ\text{F}$ from the overall average of 941°F , the power loss as compared to the case where all temperatures are equal was only 3 w.^{(38)†}

As long as the system was on the ground, exposed to the earth's gravitational field, the unequal flow distribution was modified by buoyant forces. The reason for this was that the positive NaK head in each tube was opposed by the negative NaK head in the return lines. Because the temperature of the NaK in the return lines was about 50°F cooler than the average converter tube NaK temperature, the NaK density, and hence the head, in the return lines was greater and the net effect was opposition to flow. Since the high flow tubes had the highest average temperatures, the opposition to flow from buoyancy was greatest in the high flow tubes. Conversely, the buoyant forces were least in the low flow tubes. In space the buoyant forces disappeared, the total flowrate was calculated to increase by 0.2 gpm, for a flow of 14 to 15 gpm, and the flow distribution became more unequal. The flow distribution in space was calculated using the method described in Reference 38; the results of the calculation are shown as a dashed line in Figure 79. It can be seen that the changes in the distribution were small, with the largest changes of 0.1 to 0.2% of full flow occurring in the tubes with the most extreme flows. The effect of the changes in flow distribution on the average NaK temperature of each tube was also calculated; the results are shown as a dashed line in Figure 80. Again the changes were small, the largest being only 2 to 3°F . The data shown also include a correction of a maximum of 1°F required by the larger heat losses and flow in space compared to ground tests. Although the effect was too small to be visible in Figure 80 there was a slight decrease in the effective average NaK temperature which would cause a maximum power loss in space of less than 3 w compared to the ground.

†A power decrease occurs because the relation between power and average NaK temperature is $P_1 = P_0 (T_1/T_0)^{4.8}$. Although $\sum T_i/40 = T$ (where T_i is the average NaK temperature of tube i and T is the overall average), the ratio $\sum T_i^{4.8}/T^{4.8}$ is less than one. Since $\sum P_i = P (\sum T_i^{4.8}/T^{4.8})$, $\sum P_i < P$.

Figure 80. Converter Average NaK Temperature Distribution

The results of the preflight analysis can be summarized as follows:

- 1) The predicted effect of unequal converter flow and average temperature distribution on electrical power output was negligible, both on the ground and in space.
- 2) The predicted effect of the zero gravity environment of space on the flow distribution was to make it more unequal although the change might have been too small to measure.
- 3) The total NaK flowrate in space was expected to be approximately 0.2 gpm higher in space because of the absence of an opposing head due to buoyant forces.

2. Flight Test Analysis and Results

The flight system was not instrumented to measure converter flow distribution. However, differences between ground and space could be inferred from changes in two types of measurements, (1) converter inlet to outlet NaK temperature drop measured on four tubes and (2) converter voltage profile as measured by 10 voltage taps, each of which measured the output of a series-parallel combination of four thermoelectric legs. Figure 78 shows the location of these instruments.

Based on pre-flight analysis (Figure 80) the ΔT s measured in space would be $\sim 2^\circ\text{F}$ less than on the ground for high flow tubes numbers 17 and 37 and $\sim 3^\circ\text{F}$ more for low flow tubes numbers 27 and 47. Unfortunately, the accuracy of the flight data was such that it was unrealistic to attempt to measure such small absolute changes. For example, the repeatability of the ΔT measurements from pass to pass was only $\pm 3^\circ\text{F}$ due to random error, and the typical absolute error due to bias was 5°F . In addition such effects as sun-shade transients, control drum steps, and increases in flow and reactor thermal power in space compared to ground testing caused significant changes in NaK ΔT . Therefore an approach was developed which compared the average ΔT of the two high flow tubes to that of the low flow tubes. The method of calculation made the results independent of all changes in average system ΔT over the range of interest—eliminating the effects of drum steps, thermal power level, and total flow—and reduced the effects of both bias and random error by averaging data from four instruments and seven passes and by increasing the sensitivity of the test. To eliminate the relatively large sun-shade effects, only data taken when the entire vehicle was in the shade were analyzed. The results of the calculations are summarized in Table 35.

The agreement between the predicted and measured ratio of flow in space was excellent, although the confidence limits on the measured value were too large to be sure that true value in space was as close to the prediction as it appears from the data. However, it is possible to state that (1) the ratio, and therefore flow distribution, in space was significantly different from that on the ground and 2) the measured ratio, and therefore flow distribution, was not significantly different from that predicted.

Because of the large uncertainty in the measured ratio of ΔT in high and low flow tubes, it is possible that the ratio was as high as 1.52 and that the flow distribution was significantly more unequal than predicted. To determine whether this was probable, the converter voltage distribution was examined in detail as described in Section IV. D. Unfortunately the scatter due to telemetry error was too large to draw any conclusions concerning flow distribution, although it was concluded that the observed shape of the voltage profile was not caused by an exaggerated flow distribution.

TABLE 35
COMPARISON OF MEASURED AND PREDICTED
FLOW DISTRIBUTIONS

Source of Data	Ratio of Average ΔT in High and Low Flow Tubes ⁽¹⁾
Ground Tests	
FSM-4	1.39 ± 0.024
FS-3	1.36 ± 0.024
Average	1.37 ± 0.017
Flight Test	
Measured Average	1.45 ± 0.04 to 0.07 ⁽²⁾
Predicted	1.45 ± 0.036

- (1) The limits shown are estimates of the absolute accuracy of the ratios. They are 95% confidence limits.
- (2) The range of limits shown reflects the uncertainty involved in estimating the average bias of only a few instruments. The smaller limit used the lowest reasonable estimate of bias.

3. Conclusions and Recommendations

The effect of the space environment on converter flow distribution was apparently close to that predicted, although the data were not conclusive because of the necessity of measuring the distribution indirectly using instruments with large telemetry errors. If there is any need to measure the flow distribution of future systems, the instrumentation or its accuracy should be changed. Improving the accuracy of the voltage profile monitors would be the most reasonable change since this would not only give better information about flow distribution but, more importantly, it would aid detection and measurement of converter resistance to ground degradation or converter electrical malfunctions.

O. SPACE EFFECTS

Since most of the effects of the space environment on SNAP 10A could be only partially simulated on the ground, much reliance was placed on analytical models which predicted SNAP 10A behavior in space. SNAPSHOT provided the opportunity to verify the models and the predicted satisfactory performance of

SNAP 10A in space. The various space effects have been grouped together, as shown below, according to whether they had major, minor, or negligible effects on SNAP 10A. Since most of the effects have already been discussed in previous sections, they are only listed here. Unless stated to the contrary, the observed effect was within the range predicted. No effect had any detectable detrimental effect on system operation.

1. Major Effects

- 1) Varying solar heat inputs during prestartup. See Sections III. B and IV. G.
- 2) Vehicle radiation environment. See Section IV. C.
- 3) Effective neutron source level in space. See Section IV. B.
- 4) Combined launch environment of shock, acceleration, and acoustic loads. See Section III. A.
- 5) Increased electrical power output, reactor thermal power, and flow due to better heat rejection in space. See Sections IV. D, M, and E respectively.

2. Minor Effects

- 1) Varying solar heat inputs during full power operation. See Section IV. M.
- 2) Effect of zero-gravity on converter flow distribution and power output. See Section IV. N.
- 3) Effect of solar radiation on solar absorptance of converter radiator. Tests in a simulated space ultra-violet environment resulted in a 10-15% degradation (i. e. increase) in solar absorptance compared to an 8% decrease calculated from flight test data. See Section IV. L.
- 4) Interaction of SNAP 10A magnetic moments with earth's magnetic field. See discussion below.

3. Negligible Effects

- 1) Effect of solar radiation on thermal emittance of converter radiator. See Section IV. L.

2) Self-welding due to hard vacuum. The effect of space vacuum was unknown. Exposure for 43 days had no effect. See Section IV. A.

3) Micrometeorite damage. Micrometeorites could have resulted in erosion of thermal control coatings or NaK tube puncture. Neither one was expected, especially after only 43 days, and neither was observed. The non-puncture probability of the NaK system was calculated to be 0.97 for one year using the most recent information available, May, 1966, on micrometeorite flux and penetration. For 43 days of operation, the non-puncture probability was increased to 0.996. Neither of the small secondary payload micrometeorite detectors recorded an impact. (See Section IV. J.) The secondary payload erosion detector indicated an upper erosion limit for gold surfaces of 4×10^{-5} mil/yr, or only 5×10^{-6} mil in 43 days. (See Section IV. J.) The layer of gold on the gold-Mo insulating foil was the thinnest (0.01 mil thick) of the thermal control coatings. Even this was several orders of magnitude greater than the maximum indicated erosion at one year.

4. Magnetic Torques

A design requirement of SNAP 10A was to impose no special attitude control requirements on the vehicle in orbit. Since the flow of electrical current through the thermoelectric converter and a permanent magnet in the pump resulted in magnetic dipole moments, which, interacting with the earth's magnetic field, produced rather large magnetic torques, a detailed analysis was performed before the SNAPSHOT flight to determine the magnitude of these torques.⁽³⁹⁾ This study predicted that the most significant torque from these sources would be about the vehicle pitch (y) axis, and that it would be nearly sinusoidal with a period equal to that of the orbital period. The predicted peak amplitude was approximately 0.5 oz-in. which should have induced oscillations about the y axis of no more than 9° in an unstabilized vehicle.

The predicted torques were obtained from computerized numerical calculations that determined the cross product of the SNAP 10A magnetic dipole moments with the appropriate components of the earth's field. One of the significant features of the model was the representation of the earth's magnetic field by spherical harmonics, that is, a series expansion of latitude, longitude and altitude into sinusoidal terms and Legendre polynomials. The numerical

coefficients of the series were based on experimental measurements of the earth's field and provided a precise representation of the field.⁽⁴⁰⁾

The SNAPSHOT attitude control systems are described in Section IV. J. During the first 54 orbits when attitude was controlled by the inertial reference platform with associated horizon sensors and attitude control gas, the rate of gas usage was substantially lower than had been anticipated. From this, it follows that the magnetic torques, which were calculated to be the largest source of attitude disturbing torques on the vehicle, were not large enough to produce significant vehicle oscillations.

After deactivation of the pneumatic system attitude control was provided by the less powerful control moment gyros and the inherent gravity-gradient effect (See Section IV. J). Several instruments measured the CMG performance; the data quality, however, was assessed to be marginal by the vehicle contractor⁽¹⁵⁾ because of imperfect signal conditioning. Thus the error in these data was larger than the nominal $\pm 2^\circ$ and they provided only a qualitative estimate of low torques. It was obvious, however, that the oscillations were less than 9° .

After shutdown, attitude control was provided by gravity gradient only. Under these conditions, the opportunity was greatest for torques induced by the pump magnet (the largest contributor) to disturb the vehicle attitude stability. Unfortunately, the opportunity for detecting attitude stability was not good after failure. Radar signature analysis techniques⁽⁴¹⁾ were applied in an effort to determine oscillations or tumbling of the vehicle. No tumbling or oscillations were detected within the nominal acquisition time, 35 minutes, indicating that the gravity gradient effect was sufficient to provide vehicle stability after shutdown.

It was concluded that SNAP 10A induced magnetic torques did not impose detectable motions on the SNAPSHOT vehicle and that the gravity-gradient effect alone was able to dominate induced magnetic torques. The magnitudes of the torques, although not determinable, were concluded to be within the limits predicted by Reference 39.

Attention will undoubtedly be given to magnetic torques when future systems are being designed. There may be concern about torques imposing attitude problems, especially for systems that employ several NaK pumps. Therefore,

it is recommended that judicious choices of pump location and orientation be made to provide counter-torquing if needed to null potential sources of torque. If applications for orbits as high as a few earth radii are considered in the future, the applicability of the model used to predict SNAPSHOT torques should be reconsidered. Because new knowledge is being gained about distortions in the earth's magnetic field by the solar "wind",⁽⁴²⁾ the classical model for describing the field may become obsolete and an updated model may be needed.

V. ACCOMPLISHMENTS AND RECOMMENDATIONS

A. DEGREE OF ACCOMPLISHMENTS OF FLIGHT TEST OBJECTIVES

The overall objective of the SNAPSHOT flight test was to demonstrate the feasibility of a nuclear reactor-thermoelectric space power system. This general program objective can be broken down into a number of specific objectives which were listed in Section II. The degree of accomplishment of each of these objectives, as well as several other significant objectives, is given below.

1. Ground Handling and Launch Operations

The ground handling and launch support activities for the SNAPSHOT flight test introduced no new equipment or special procedural requirements. The SNAP 10A system, with its reactor fueled and its liquid metal system filled and sealed, was shipped from Santa Susana, on the outskirts of Los Angeles, to Vandenberg Air Force Base by normal commercial carrier, over the most direct route. No restrictions were imposed on either time of shipment or on routing.

The receipt, checkout and mating procedures at the launch site were similar to those employed for other payloads and were accomplished in a normal launch preparation cycle. The nuclear power unit was first thoroughly checked for proper electrical and mechanical functioning. It was then mated with the Agena in the gantry and was functionally exercised through the Agena to check proper interfacing and subsystem operation. In this total checkout sequence, access to the reactor system and to the rest of the spacecraft was direct and unrestricted. Figure 81 shows the SNAP 10A unit during various phases of the shipping and launch site operations. No unique operations were introduced by the presence of the nuclear power system, and no special handling tools or unusual ground support equipment were required to support the launch.

The time required on the launch pad for mating, functional checkout, and complete booster payload preparation for launch was similar to that experienced with other payloads. The launch was conducted with a normal countdown procedure and with no unusual delays or difficulties. In all respects, the SNAPSHOT activities at Vandenberg were described by Lockheed and Air Force launch personnel as routine and smooth. The launch and prelaunch operations from SNAP 10A acceptance testing through launch are described in Reference 4.

Figure 81. S10A Ground Handling and Launch Operations

2. Launch Survival

SNAP 10A survived the launch environment of simultaneous accelerations, vibration, and acoustic noise without any apparent mechanical failure or functional difficulty other than the temporary short-circuiting of one thermocouple network. Since these environments could only be simulated in the laboratory, the structural loads observed during launch, and the system response to them, were quite significant. The launch loads were close to those used for system acceptance testing and well below those used for system and component qualification testing.

3. Remote Startup in Orbit by Ground Command

The startup control system performed flawlessly and the reactor and power system responded just as predicted. The reactor went critical, reached full power in Orbit 7, and was automatically held there by the control system for nearly six days, all in response to the startup enable and execute commands sent on Orbit 2.

4. Unattended Operation After Initial Active Control Period

After the initial 6-day control period required to compensate for short term reactivity losses, the reactor control system was deenergized. Thereafter, the reactor and power system required no dynamic control or moving parts, relying on the inherent stability of the reactor and thermoelectric materials for passive control, closely following the trends observed on the nuclear ground test qualification system, FS-3. FS-3 further demonstrated unattended operation when it operated for more than the 1-year design life. Following this, the FS-3 reactor temperature and power were increased to the initial operating conditions, and less than 1 day of active control was required before the system was again placed on passive control for a final month of operation. The total operating time of FS-3 was 10,000 hours, which set a record for continuous operation of a nuclear reactor.

5. Production of 500 w for 1-year in Space Environment

SNAPSHOT did not meet this goal because of the premature shutdown. However, extrapolations of flight data show that the power output would have been 503 w at the end of 90 days and 460 w after one year, had there been no power loss due to converter resistance to ground degradation. Taking into account this loss, power output would have been 468 ± 12 , -0 w at 90 days and 406 ± 22 , -128 w after one year.

There were two reasons the extrapolated system performance fell below 500 watts at one year on this first SNAP 10A flight system. The first of these was the unexpected decrease in resistance to ground and the second was the greater than originally predicted decrease in average temperature which had already been foretold by the FS-3 test which began some two months before launch.

The solution to these two problems is sufficiently clear that the already completed FS-5 system could be modified to eliminate the unexpected temperature drift and to reduce the resistance-to-ground degradation to such an extent that there would be a reasonable probability of producing 500 w at one year. On any new system, resistance-to-ground degradation could definitely be reduced to the required level, thus assuring 500 w. It is interesting to note that, while the unexpected temperature drift was detected initially by ground testing and verified by the flight test, the resistance-to-ground degradation problem could be detected only by a flight test because of outside sources of contamination present during ground testing and the inability to approach the hard space vacuum on a system ground test. One additional comment is appropriate. On SNAPSHOT there was a power excess. However, if it had ever been necessary to increase the power output, this could easily have been accomplished by inserting the required reactivity by ground command as was done on Orbit 82.

6. Demonstration of Simplicity and Reliability Potential of a Reactor-Thermoelectric Space Power Unit

The reliability potential was demonstrated by the nearly perfect performance of SNAP 10A on its first launch and flight test. The uneventful ground operation of FS-3 for 10,000 hours, 1200 hours more than its one-year design life and 9900 hours of which was unattended passive control, provided further evidence of reliability and simplicity. Additional demonstrations of reliability were provided by FS-3 during short periods of active control for special tests after one year and after 10,000 hours, when the system responded to manual changes in reactivity just as predicted with no operational problems. Table 36 shows the present best estimate of SNAP 10A reliability.

7. Measurement of Radiation Shielding Effectiveness in Space

The effect of nuclear radiation on the SNAPSHOT vehicle was counteracted by using a neutron shadow shield to reduce radiation levels in the instrument compartment and Agena and by placing the most sensitive Agena electronics in

TABLE 36
BEST ESTIMATE OF THE RELIABILITY OF SNAP 10A*

System and Major Component	Launch Survival and Orbiting	Plus Startup and Stability	Plus Endurance	
			90 day (500 w)	1 Yr (~500 w)
Reactor and Shielding				
Fuel (core)	0.9996	0.9991	0.998	0.995
Reflector Assembly (partial)	0.9998	0.9993	0.999	0.998
Radiation Shield	0.9997	0.9996	0.999	0.998
Conversion and Distribution				
TE Converter System	0.9997	0.9994	0.9987	0.995
Shunts, Taps, Wire	0.9998	0.9993	0.9990	0.997
Heat Transfer System				
Pump and Battery	0.9993	0.998	0.997	0.994
Expansion Compensators (2)	0.9988	0.995	0.985	0.974
NaK Piping (include meteoroid)	0.9978	0.994	0.985	0.965
Structural System				
Ejectable Heat Shield	0.9991	0.996	0.996	0.996
Primary Structure and Compartment	0.9984	0.997	0.997	0.997
Control System				
Controller	0.9989	0.997	0.994	0.990
Drive Motors (2)	0.9993	0.997	0.995	0.993
Temperature Switches (4)	0.9990	0.997	0.996	0.996
Squibs and 7 Pin-pullers	0.9991	0.992	0.992	0.992
Relay Boxes	0.9988	0.994	0.992	0.990
Wire and Connectors	0.9981	0.995	0.992	0.984
Diagnostic System (as required)	0.9990	0.996	0.994	0.989
Shutdown System (as required)				
Delay Timers	0.9998	0.9996	0.998	0.994
EABRD and TABRD	0.9995	0.9990	0.998	0.996
Low Voltage Switches (2)	0.9998	0.9993	0.997	0.995
Total reliability inherent in NPU design:	†(0.984)	0.944	0.905	0.842

* Reliability of SNAP 10A flight system assuming success of Atlas/Agena.
(Reliability = 1 minus probability of causing mission failure.) Data are from Reference 43.

† The probability is less than 0.005 that launch-induced failures of the NPU will preclude an initiation of startup. Residual launch damage has probability of 0.012 of contributing to premature mission termination.

areas of lowest flux. One of the most valuable flight measurements, since radiation levels could not be simulated on ground tests because of scattering from equipment and air, was the radiation data. The flight data made it possible to construct fairly detailed flux distributions for the Agena and to determine the sources of the scattered neutron radiation. These measurements allowed a number of improvements in the computer programs used to predict radiation scattering, changes which will give greater confidence in future shield designs. They also enabled determination that the Agena equipment would have survived the one-year radiation doses without serious degradation. However, the substantial differences between the preflight predictions and measured radiation levels show the need for radiation measurements on future flights with different vehicle or shield configurations.

8. Obtain In-Space Operating Data and Experience

As a result of the flight test data and experience, a firm basis has been provided for improvements in SNAP 10A, design of higher power systems, and applications engineering. Suggestions for improvements resulting directly from the flight test experience are given in Section V.B. Improvements based on the range of SNAP 10A experience are described in Reference 16.

9. Nuclear Safety

The SNAP 10A flight provided a significant first with regard to development of safety plans and approval for launching of a nuclear space reactor. One of the earliest steps in the SNAPSHOT planning was preparation of a flight safeguards report⁽⁴⁴⁾ which defined and evaluated all possible nuclear hazards associated with the flight test, from the time of prelaunch testing to reentry. This evaluation was supplemented by an extensive aerospace safety analytical and test program.

The devices and procedures which evolved were intended to reduce to very low levels the possibility of nuclear exposure to program participants and to the general population. They were not only effective at accomplishing this, as shown by the accident-free record of the program, but required no significant changes in normal launch pad procedures. The automatic shutdown of the reactor in orbit assured that, when the reactor finally reenters the atmosphere from its long-life orbit, there will be no safety hazard. Despite all preflight fears, the

U.S. accomplishment with SNAPSHOT resulted in no known public outcry of safety from any quarter, foreign or domestic. This factor, which has at times been considered a major drawback to the acceptance of nuclear power, can now be considered a "ghost".

10. Interface Engineering Verification

Early in the program the decision was made not to attempt to verify the compatibility of the SNAP 10A and Agena by means of a full scale qualification test in a simulated space environment; instead it was decided to qualify them separately, verifying the structural, thermal, and electrical interface compatibility by a series of tests and functional mock-ups of the SNAP/Agena structural⁽⁴⁵⁾ and electrical systems.⁽⁴⁶⁾ The validity of this approach was demonstrated by the trouble-free mating of the flight models of the SNAP 10A and Agena for the first time on the launch pad, as well as by the success of the integrated vehicle in orbit.

The major SNAPSHOT engineering objectives and their percent of accomplishment are listed in Table 37. Based on this preflight evaluation of objectives, 43 days of test operation resulted in accomplishment of approximately 80% of the engineering mission objectives. Such important objectives as remote start-up of the nuclear reactor, initial stabilization of the reactor, and initial performance characteristics of the thermoelectric power converter in space were 100% attained. The major engineering objectives that were not completely attained were those associated with long-term performance of the various components. However, in even these areas much information was gained, as can be seen by the percentage accomplishment figures.

When the 100% accomplishment of the nuclear safety goals is considered, the SNAPSHOT flight test resulted in the accomplishment of 85% of the flight test program objectives.

TABLE 37

PERCENT ACCOMPLISHMENT OF ENGINEERING MISSION OBJECTIVES

Mission of SNAP 10A was to Demonstrate in Space Environment the Following Major Objectives	Weighted % of Total Mission	1 Day	5 Days	30 Days	43 Days
1. Remote startup of nuclear reactor	10	100			
2. Stabilization of nuclear reactor	10	25	90	100	
3. Long-term performance of nuclear reactor	20		10	40	80
4. Radiation shield effectiveness	10	20	80	100	
5. Initial performance of thermo-electric power converter	5	100			
6. Long-term performance of power conversion system	25			30	60
7. System and instrument integrity following launch	10	60	75	90	95
8. Component performance with exposure to actual space environment and nuclear radiation	10	20	30	35	45
Percent of Mission Completion		28	43	55	80

B. RECOMMENDATIONS

Although the flight testing of SNAP 10A was most remarkable for its excellent, as-predicted performance, it also clarified needed and possible improvements to SNAP 10A and future reactor or thermoelectric space power systems. The changes required to assure the production of 500 w after one year are the most obvious and are listed below.

1) The reactor temperature drift experienced by SNAPSHOT and its companion ground test, FS-3, can be minimized and adjusted by increasing the prepoison loading of the fuel elements and by a minor change in composition of 3 of the 37 fuel elements.

2) Resistance-to-ground degradation can be reduced by minimizing the sources and amounts of outgassing hydrocarbons in both the SNAP unit and

Agena, by protecting the converter insulators by isolating them from the remaining sources of hydrocarbons, and by increasing the insulator surface leakage path to reduce the thickness of the conductive contaminating layer and to increase its electrical resistance.

The other recommendations are intended to simplify and improve system operation and reliability and to improve the quality and amount of information available from the spacecraft, especially under conditions of malfunction or failure. The recommendations listed here are those prompted by SNAP 10A flight test experience; most have already been mentioned and described in more detail in previous sections of this report. A more extensive discussion of recommended improvements resulting from a review of the experience gained from the entire SNAP 10A program and the SNAP 10A optimization study is given in Reference 16. The results of a postflight design review of the SNAP-SHOT Agena directed toward product improvement are given in Reference 15.

1) Future tests of reactor powered satellites should incorporate more accelerometer instrumentation. Because these payloads, with their heavy lumped reactor and shield masses, are configured differently from other satellites, there is no body of directly comparable flight test data from which structural design criteria may be derived.

2) Launch pad preheating of the SNAP 10A unit was a significant contributor to the stable NaK temperature and high flow during prestartup. This procedure should be continued on future flights. With preheating, the requirement for batteries to supply current to the pump during prestartup can be eliminated.

3) The inherent stability of SNAP reactors has been thoroughly demonstrated; the rate of temperature and power increase during startup should therefore be limited by thermal stress criteria rather than reactor stability criteria. There is no requirement for the long time delay between the beginning of startup and criticality or between the initial power transient and full power.

4) The control system temperature sensor should be redesigned so that it can be mounted in better thermal contact with the NaK, or it should be replaced by a current sensor. Either change would simplify calibration

procedures, decrease control system response time, and simplify or eliminate NaK tube penetrations. The time required for reactor startup should be reduced; insertion of the coarse control elements should decrease subcriticality enough that the reactor can be made critical within 30 minutes rather than the 6 hours required on SNAPSHOT. The use of ground commands in controlling the system can be given more serious consideration where simplification is desired.

5) Payload radiation doses in the Agena aft rack can be reduced to 1×10^{11} nvt/yr which will allow the use of unhardened, off-the-shelf electronic equipment. On FS-5 this could be accomplished by local shielding. On more advanced systems an improved shield (similar to that used on SNAP 2 and SNAP 8) together with a tapered reflector, could be used to reduce neutron scattering.

6) By minimizing areas where gas could be entrapped and establishing a maximum amount of trapped gas which could be tolerated, advanced system designs could resolve the difficulty of removing all gas during NaK loading. One of the factors which should be considered in designing facilities used for NaK loading is elimination of gas from the SNAP system.

7) Recommendations for instrumentation improvement and changes are given in Section IV. H. Implementation of the suggestions would increase the accuracy of data transmitted from space, its believability, and reliability under both normal and malfunction or failure conditions. Instrumentation for any future system should take into account the differing requirements of real-time and off-line analysis, and normal and anomolous operation.

8) The malfunction and failure circuits should be redesigned to provide (1) increased operational flexibility by separation of diagnostic and automatic reactor shutdown functions; (2) ability to enable and disable the automatic shutdown functions; and (3) high voltage detection. Redundancy of telemetry tape recorders, failure power supplies, and critical parameter instrumentation should be increased. Backup protection against high voltage, resulting from voltage regulator failure, and increased design margin against over-voltage in vital control and instrumentation equipment should be provided throughout the entire spacecraft.

9) Tests and/or analyses of the voltage regulator should be performed prior to a second flight test to better define the conditions which overstress individual regulator components. An attempt should be made to eliminate the bias errors on individual telemetry channels by precise end-to-end calibrations. An investigation of the causes of the lower noise levels and higher accuracies of the 5 and 20 mv sub-multiplexers compared to the 50 mv sub- and main multiplexers could lead to improvements in the accuracy of future telemetry systems. The tape recorder provided valuable information for engineering analysis and explained several apparent performance anomalies during SNAPSHOT; providing its reliability can be improved, the recorder should be used routinely on future flights as well as for malfunction-failure situations.

10) The type and causes of telemetry errors observed on SNAPSHOT should be taken into account when selecting instrumentation for future flights.

11) An experiment to measure the change in thermal emittance and solar absorptance of SNAP 10A thermal control surfaces in space (similar to that which failed on SNAPSHOT) should be included on a future flight test. Alternatively, more accurate system radiator temperature measurements taken over complete orbits could serve the same purpose on surfaces at operating temperatures, though probably not as accurately.

12) Sun/shade effects at full power were small, but not negligible. The effect of sun/shade on both power output and instrumentation performance should be considered in the design of future systems to assure that they will continue to be small.

13) The effect of magnetic torques on spacecraft attitude control should be considered early in the design of future systems so that the pumps can be located and oriented to provide counter-torquing if required.

14) SNAPSHOT provided a wealth of information which not only indicated a number of ways in which future reactor or thermoelectric space power systems could be improved, but also demonstrated the adequacy of the SNAP 10A design and test program. Thus SNAP 10A and SNAPSHOT provided a firm, proven basis for the design, fabrication and test of more advanced systems.

BLANK

VI. APPENDICES

A. CONVERTER VOLTAGE PROFILE ANALYSIS

The converter voltage profile was analyzed in an effort to verify the conclusion of Section IV. N that the converter flow distribution in space was in good agreement with the predicted distribution, and more important, to determine whether the converter was operating as expected and whether the decrease in resistance to ground indicated by leakage current could be verified independently. The voltage profile was monitored in flight by 10 voltage taps, each of which measured the output of four series-parallel connected thermoelectric legs as shown in Figure 78.

The converter voltage profile was affected by a number of factors both internal and external to the converter. Those which have a significant effect have been listed below together with a short explanation of the reason for the effect. Figure 50 shows the effect of these factors on the FS-4 predicted voltage profile.

1. Variability of Thermoelectric Module Properties

The smallest portion of the converter monitored during system test consisted of four legs, each of which contained three modules. Each module contained from 20 to 28 thermoelectric pellets. It would seem reasonable that the use of so many pellets, all of which were manufactured and acceptance-tested to a common specification, would tend to eliminate differences between modules. The SNAP 10A converter modules had nearly identical open circuit voltages, but exhibited variations in internal resistance which persisted in the finished converter. (See Table 38.) Since the only measurements available were made at room temperature it was necessary to assume that the relationship between electrical properties and temperature and time was the same for all double leg pairs. This assumption was verified during long term testing of FS-3.

TABLE 38
VARIATION IN CONVERTER DOUBLE
LEG PAIR INTERNAL RESISTANCE

System	Internal Resistance at 75°F		
	Mean (mohm)	Standard Deviation	
		(mohm)	(%)
FS-4	91	1.1	1.2
FS-3	88	0.9	1.1
FSM-4	89	1.0	1.1

Variation in internal resistance affected the voltage profile because it resulted in different internal voltage drops among double leg pairs in accordance with the circuit equation,

$$E_i = E_{oc} - IR_i$$

where E_{oc} = open circuit voltage per double leg pair (equal for all modules).

I = total converter current (flows through all double leg pairs)

R_i = internal resistance per double leg pair (varies from leg to leg)

E_i = measured closed circuit voltage at voltage tap

The calculated effect at full power of the measured variation in FS-4 internal resistance is shown in Figure 50.

2. Average NaK Temperature Differences Among Tubes

Since the voltage output of thermoelectric elements is strongly dependent on temperature and the average NaK temperature varies widely from tube to tube, the NaK temperature variation was the most important source of voltage variation. Its effect can be calculated as follows.

$$P_i = \overline{P} \left(\frac{\overline{T}_i}{\overline{T}} \right)^n \quad \dots (1)$$

where

P_i and $\overline{T}_i$ = the power output and average temperature of a double leg pair,

$\overline{P}$ and $\overline{T}$ = the average converter power per double leg pair and average converter NaK temperature,

n = a theoretical constant = 5.15 on the ground and 4.9 in space.*

But,

$$P_i = E_i I \text{ and } \overline{P} = \overline{E} I \quad \dots (2)$$

where,

E_i and $\overline{E}$ = the closed circuit voltages of the desired double leg pair and an average pair, respectively, and

I = total converter current.

Substituting,

$$E_i I = \overline{E} I \left(\frac{\overline{T}_i}{\overline{T}} \right)^n \quad \dots (3)$$

or

$$E_i = \overline{E} \left(\frac{\overline{T}_i}{\overline{T}} \right)^n \quad \dots (4)$$

which is the desired relationship.

*The exact value of n used in this application is not important. Varying the value of n from 4 to 6 has no significant effect on the results. A value of $n = 5$ was used in the calculations. A value of 4.8 was measured on FS-3.

The calculated effect of temperature distribution on FS-4 voltage profile is shown in Figure 50, for both ground and space based on the temperature distributions shown in Figure 80.

3. Variations in Effective Heat Sink Temperature

The effective sink temperature — an artificial temperature which includes the effects of emissivity and view factors — need not be considered if it affects the whole system equally. Use of the measured average voltage and individual NaK temperatures in Equation 4 above takes effective sink temperature into account by variation of the exponent n . However, when the effective sink temperature varies circumferentially around the converter (e.g., when the converter sees both sun and shade simultaneously in space or when the view of the vacuum vessel wall is obstructed on the ground) the normal relationship between average NaK temperature and thermoelectric ΔT , the actual driving force for voltage, does not hold for individual legs. In these cases a heat balance must be made around each double leg pair using the local effective sink temperature to obtain the converter radiator temperature and thermoelectric ΔT , after which the voltage can be computed. Unfortunately the calculations are involved and lack the precision required for accurate prediction of voltage profiles, primarily because of the difficulty of obtaining accurate geometric factors and temperatures. Hence an effort was made to predict voltage distribution only for conditions where sink temperatures were constant, such as full shade periods in space.

Examples of the effect of variations in sink temperature are given in Figures 50 and 51. Figure 50 compares a typical FS-4 voltage profile measured in full shade to a profile measured in the sun 1/2-hr later. Those legs exposed to the sun decreased in voltage while the voltage of the remaining legs, which were shaded, was forced to increase because the total voltage was held constant by the voltage regulator. Figure 51 shows the depression of FSM-4 and FS-3 voltages below predictions at those points where the converter view of the vacuum vessel wall was blocked by wiring trays covered by low absorptivity aluminum foil.

4. Converter Electrical Malfunctions

Decreases in converter isolation resistance and converter open or short circuits can cause changes in the voltage profile. The change caused by a distributed decrease in isolation resistance can be calculated.⁽⁴⁷⁾ The results are shown in Figure 50 for two values of leakage current, (1) the measured current just before shutdown and (2) the extrapolated current after 90 days. The voltage increases in the legs with the highest potential above electrical ground because the total converter output voltage was held constant by the voltage regulator. Without the regulator there would be a decrease in all legs with largest decrease in the legs closest in potential to electrical ground.

Random short and open circuits can result in either high or low voltages affecting varying numbers of legs depending on their locations and magnitudes. Reference 48 discusses methods of examining unexpected changes in the voltage profile together with other converter data to determine whether the changes could have been caused by one of several different classes of shorts or opens.

To determine the accuracy of the methods to be used to predict the voltage profile, voltage profiles were calculated for both the FS-3 and FSM-4 ground test qualification systems. The measured profiles were then compared to the predictions which took into account both temperature and internal resistance variations. The results are plotted in Figures 51. It can be seen that the agreement between measured and predicted values was quite good except for those legs which had higher than normal effective sink temperatures because wiring trays or cold traps with low thermal absorptivities partially blocked the "view" these legs had of the vacuum vessel wall. The amount of deviation between measured and predicted values for these legs was directly proportional to the extent to which the effective sink temperature had been increased.

The FS-3 voltage profiles were also examined to determine the effect of time, radiation, and changes in operating conditions. The shape of the voltage profile after 9300 hr of full power operation in a radiation environment more severe than that experienced by FS-4 in space was identical to the initial profile. This demonstrated that the ten sets of thermoelectric elements experienced similar changes in internal resistance and open circuit voltage as the elements degraded and the temperature changed. It also showed that the voltage profile of a healthy

converter was not affected by changes in average NaK temperature or thermoelectric degradation, so that changes in the flight voltage profile could be interpreted as indications of converter problems rather than as normal responses to time and temperature changes.

The response of the FSM-4 voltage profile to low isolation resistance was also studied. Previous work had shown that a number of analytical techniques for comparing leg voltages would permit a qualitative determination of isolation resistance only if it became low enough, about 10 ohms, to cause noticeable power loss. Since FS-4 isolation resistance was still as high as 30 ohms at shutdown, these methods could not be used. However, the shape of the FSM-4 profile was changed in a characteristic way which was discernible at isolation resistances as high as 50 ohms although the magnitude changes remained fairly small until the resistance dropped to about 10 ohms. Although the expected deviation from the normal profile was small, it can be seen from Figure 50 that the change is negative at one end of the converter and positive at the other, a fact which allows real changes to be differentiated from random error while the isolation resistance is still fairly high. While the voltage profile allows detection of decreased resistance to ground earlier than allowed by the analytical method, it does not provide any better estimate of its absolute value.

5. Flight Test Results

The measured and predicted voltage profiles for FS-4 in space are shown in Figure 51. The prediction procedure was identical to that used on the ground with the addition of the small effect caused by change in flow distribution in space (Figure 50). The measured values are the average of data from seven full shade passes. It can be seen that the agreement between measured and predicted values is poorer than for the ground test systems. This is shown numerically in Table 23, which also shows the error involved in measuring the voltage profile. The measurement error on the ground was significantly smaller statistically than the difference between the measured and predicted values; this suggests that there may have been some secondary factor which was not considered in the predictions. The measurement error in space, in addition to being about six times larger than the ground error, was approximately the same size as the difference between measured and predicted values, this implying that the observed discrepancy was due to data error. The fact that the measured shade profile was fairly repeatable from orbit to orbit, in spite of the large total

data error is reasonable since only a small portion (2%) of the measurement error for the average profile shown was random error while 98% was due to bias. Because there is always a good chance for error in assigning numerical values of bias to specific instruments using statistical inference, an investigation was made to determine whether there was any other factor which could cause the observed profile. Effort was concentrated on differences between the space and ground environments and converter malfunctions.

Since the largest source of variation in the voltage profile was the flow and temperature distribution, it was logical to suspect that the distribution in space might be more extreme than predicted. Even though the measured range of double leg pair voltages was twice as great as predicted, it can be seen that the measured profile was not an exaggeration of the predicted profile. Since all the known and suspected differences between ground and space — e. g. lower effective sink temperature with attendant higher heat losses and flow, zero gravity, and regulated voltage which caused a mismatched load — and any reasonable modification of the prediction method (e. g. assuming non-linear temperature distribution along the length of the NaK tubes, treating each leg separately to calculate voltage rather than assuming a double leg pair operates at the average temperature of the four tubes) only exaggerate the existing ground profile or have no effect, it was concluded that the flow and temperature distributions were not different than predicted. This view was reinforced by the analysis of NaK tube ΔT s which indicated measured and predicted values were equal (Section IV. N).

At the time of shutdown low isolation resistance should have resulted in the change shown in Figure 50. Neither the shape nor the magnitude of the change is like that measured, however. Partial short or open circuits could have resulted in a profile with the shape observed, but shorts or opens severe enough to cause the observed deviations would have resulted in a measurable decrease in power below the predicted value and discrepancies between the readings of the total voltage instruments and the sum of the voltage tap readings. Neither of these was observed. In addition, any shorts to ground would have caused a measurable leakage current as soon as the system reached full power. It was concluded that the observed profile was not the result of converter malfunction.

The only remaining explanation was a local change in effective space temperature which could have been caused by one of the heat shield halves remaining close to the vehicle after ejection. This would have produced several effects: (1) Local depression in the shade voltage profile at the location of the legs affected and slight increases elsewhere; (2) shading by the shield of some legs from the sun's rays, preventing a decrease in voltage in those legs when the system came into the sun; and (3) change in the legs affected in both sun and shade as the SNAPSHOT orbit changed with respect to the orbits of the heat shield and sun. None of these effects was observed. The observed profile exhibited large scatter both higher and lower than the prediction. The voltage of each double leg pair was observed to react as soon as that part of the system was exposed to sunlight. The shade profile showed no significant change with time during the flight. In addition positive heat shield ejection was indicated by position indicator and thermocouple readings, radar tracking, and nuclear radiation measurements.

Since none of these alternative explanations was reasonable, the observed voltage profile was attributed to data scatter.

An attempt was made to verify the degradation in converter resistance to ground indicated by the leakage current increase. To eliminate error due to sun/shade effects it was necessary to use full shade data only, a fact which limited the usable data to a small portion of Orbit 548. (The previous shade data had been taken 100 hours earlier when the resistance was still fairly high.) Because only one set of data was available the random telemetry error was too large to detect any change. (The telemetry bias error was eliminated by comparison with the profile in Figure 51 which was measured while the isolation resistance was still high.) In order to be 90% sure of detecting the change in the voltage profile with only a 20% chance of mistakingly saying there was a change, it would have been necessary to use data from seven shade passes after 42 days of full power operation. The change would have been large enough for detection 90% of the time with only a 10% chance for error with data from only one shade pass after 70 days when the resistance to ground would have been about 16 ohms. At the end of 90 days the change could have been detected using data from one shade pass 95% of the time with only a 2% chance of making a mistake if the isolation resistance had been 8 ohms.

B. UNEXPECTED EXPANSION COMPENSATOR DEFLECTION UNDER ZERO-G CONDITIONS

During zero-g condition prior to the system startup transient, the measured expansion compensator deflection was 0.08 in. greater than the expected 0.14 in. (See Section IV. F and Figure 57.) A number of possible reasons for this behavior were considered, but the only satisfactory explanation was that it was caused by the presence of a small quantity of gas trapped in the system. Supporting calculations for the trapped gas theory are given below, together with the reasons for rejecting alternative explanations.

1. Gas Trapped in System

During NaK loading of SNAP 10A systems, precautions were taken to avoid trapping gas. However, experience has shown that it is impractical to expect a completely gas-free system. If we assume that the entire 0.08 in. per ECU additional deflection was due to trapped gas we can calculate the volume of gas trapped as follows

	<u>Launch Pad</u>	<u>Zero-g</u>
Pressure at top of system, psia	12.7	2.3
Pressure at bottom of system, psia	16.6	2.3

The change in average NaK temperature between the launch pad and sustainer engine cut-off was negligible, therefore, from the perfect gas law,

$$P_L V_L = P_E V_E$$

or

$$P_L V_L = P_E (V_L + \Delta V)$$

where,

$$\Delta V = 0.085 \text{ in. deflection per ECU} \left(\frac{40.7 \text{ in.}^3}{\text{in. deflection per ECU}} \right) \times 2 \text{ ECU}$$
$$\Delta V = 6.92 \text{ in.}^3$$

Substituting numerical values and assuming the gas is located at the bottom of the system,

$$16.6 (V_L) = 2.3 (V_L + 6.92)$$

$$V_L = \frac{2.3 (6.92)}{14.3} = 1.11 \text{ in.}^3$$

$$V_E = 1.11 + 6.92 = 8.03 \text{ in.}^3$$

If the gas is located at the top of the system where the NaK head is lower we obtain, $V_L = 1.54 \text{ in.}^3$ and $V_E = 8.46 \text{ in.}^3$ by the same calculation. Thus the volume of the gas bubble at launch pad conditions was between 1.1 and 1.5 in.^3 depending on its location.

It was next necessary to calculate the volume of the gas at the conditions prevailing during NaK loading to determine whether it was reasonable to trap that amount of gas in the system.

	<u>Launch Pad</u>	<u>NaK Loading</u>
Pressure at top of system (psia)	12.7	11.4
Pressure at bottom of system (psia)	16.6	15.3
Temperature (°F)	97	77

$$V_N = \frac{(V_L)(P_L)(T_N)}{(P_N)(T_L)} = \frac{(1.11)(16.6)(537)}{(15.3)(557)} = 1.16 \text{ in.}^3$$

Similarly for gas at the top of the system $V_N = 1.65 \text{ in.}^3$. Therefore, the amount of gas trapped ranged from 1.16 to 1.65 in.^3 at NaK loading conditions. Considering the volumes of the places where gas could most likely be trapped (thermoswitch wells, upper manifold ends, ECUs), and the amount of gas observed in the two non-nuclear ground test systems after long-term testing, a reasonable estimate is 1 to 2 in.^3 of trapped gas.

Since the assumption of trapped gas was reasonable, the effect of the gas on the original predictions of bellows deflection (which took into account only the

change in NaK volume with temperature) at several different test conditions was determined by calculating the change in bubble size due to the changes in system temperature and pressure. The original predictions and those modified for the presence of gas are tabulated in Table 39 together with the measured values.

TABLE 39
EFFECT OF GAS ON EXPANSION COMPENSATOR DEFLECTION

Test Condition	Bellows Deflection (in. /ECU)				
	Measured	Predicted			
		Original (No Gas)	All Gas in ECU	All Gas in Upper Manifold	All Gas in Thermoswitch
NaK Loading	0.10 ⁽¹⁾	0.10	0.10	0.10	0.10
Vacuum vessel, system cold	--	0.10	0.13	0.17	0.23
Vacuum vessel, system hot	1.41	1.45	1.46	1.43	1.37
Launch pad	0.14	0.13	0.13	0.13	0.13
In orbit, system cold	0.22 ⁽²⁾	0.14	0.22	0.22	0.22
In orbit, system hot	1.53	1.54	1.52	1.52	1.52

(1) Reference position. All predictions are based on this value.

(2) Estimates of gas volume were calculated using this value. Hence all cold values equal 0.22 in.

The assumed location of the gas bubble has a significant effect on the prediction in a vacuum when the system is under the force of gravity. Since the gas was probably trapped as small bubbles in a number of places it was rather arbitrarily assumed to be in the upper manifold, which gives an average of the probable entrapment locations. In every case, the assumption of gas in the upper manifold gives equal or better agreement with the measured value than the original prediction with no gas.

2. Alternate Explanations

a. Telemetry Errors

It is totally improbable that either telemetry bias or noise caused the 0.08 in. discrepancy; the agreement of both telemetry channels with the predicted values was excellent both on the launch pad and when acceleration forces were applied.

These data were acquired by the same tracking station and were obtained both immediately before and after the discrepant data.

b. Position Indicator Errors

For the same reasons as telemetry bias is unlikely, calibration errors or changes in the position transducers are unlikely explanations. There was no change in instrument supply voltage to account for the change in output. Hysteresis in the instrument output has been observed at times during component acceptance testing, but it was never greater than 0.01 to 0.02 in.

c. Bellows Racketing and Hysteresis

During component testing of the ECUs installed on FS-4, a slight racketing or hanging up was observed at times as the bellows collapsed at zero deflection. However, this effect was random and did not occur when the bellows were extended more than 0.01 to 0.02 in. It is therefore highly improbable that both bellows would ratchet three successive times at the same deflection so far from collapsed (0.22 in.) The bellows returned to the same position following sustainer engine cut-off and the end of Agena first and second burns.

Mechanical hysteresis of the bellows has been observed on many component tests. That is, the deflection of the bellows is greater on decreasing pressure than on increasing pressure, the difference amounting to as much as 0.04 in. However, hysteresis cannot appear in the same way when the ECU is installed in the SNAP 10A system since it would require the formation of voids in the NaK under zero gravity. The bellows must return to the same deflection and the hysteresis appears as a slightly lower pressure on decreasing deflection.

C. VOLTAGE REGULATOR

TABLE 40
POWER DISSIPATED BY VOLTAGE REGULATOR⁽¹⁾

Control Drum ⁽²⁾ Step No.	Orbit No. ⁽³⁾	SNAP 10A ⁽⁴⁾ Power Output (w)	Power Used ⁽⁵⁾ by Payload (w)	Power Dissipated ⁽⁶⁾ by Voltage Regu- lator (w)	Regulator ⁽⁷⁾ Power Margin (w)
0 Peak	7	590±25	380±50	210±60	170±60
1,2 Peak	7-8	600	340	260	120
3,4 Peak	7-8	610	340	270	110
5,6 Before	7-8	560	340	220	160
Peak	8	650	380	270	110
After	8	610	380	230	150
7,8 Before	8-9	590	340	250	130
Peak	9	640	380	260	120
After	9	620	380	240	140
9,10 Estimated Peak	10-11	630	270	360	20
Est. After	10-11	600	270	330	50
11,12 Est. Peak	13-14	630	320	310	70
Est. After	13-14	600	320	280	100
13 Before	16-17	590	320	270	110
Peak	16-17	630	320	310	70
After	17	590	370	220	160
14,15 Est. Peak	18-19	650	280	370	10
Est. After	19	620	410	210	170
16 Before	24-25	580	340	240	140
Peak	24-25	630	340	290	90
After	24-25	580	340	250	140
17 Est. Peak	29-30	620	310	310	70
Est. After	29-30	590	310	280	100
18,19 Before	35-36	580	340	240	140
Peak	35-36	630	340	290	90
After	35-36	600	350	250	130

TABLE 40
POWER DISSIPATED BY VOLTAGE REGULATOR⁽¹⁾
(CONTINUED)

Control Drum ⁽²⁾ Step No.	Orbit No. ⁽³⁾	SNAP 10A ⁽⁴⁾ Power Output (w)	Power Used ⁽⁵⁾ by Payloads (w)	Power Dissipated ⁽⁶⁾ by Voltage Regu- lator (w)	Regulator ⁽⁷⁾ Power Margin (w)
20 Est. Peak	76-77	620 ± 25	240 ± 50	380 ± 60	0 ± 60
Est. After	76-77	590	240	350	30
21,22 Before	82	580	380	200	180
Peak	82	620	380	240	140
Est. After	82-83	585	265	320	60
Beginning of Passive Control (between passes)		605 shade	265	340	40
		585 sun	265	320	60
Beginning of Passive Control (during a pass)		605 shade	350	255	125
		585 sun	350	235	145
End of Passive Control (between passes)		540 shade	265	275	100
		520 sun	265	255	100
End of Passive Control (during a pass)		540 shade	350	190	190
		520 sun	350	170	210

- (1) The data are intended to show the maximum power loads which the voltage regulator had to dissipate. The peak loads are not typical but were chosen to show the effects of the SNAP 10A power output peaks corresponding to control drum steps. The before, after, and passive control data are indicative of average loads.
- (2) The number shown is the count of control drum steps occurring after full power (Table 13). The before data were taken five minutes before the power peak and the after data were taken five minutes after the peak. Before data were not affected by the step, and after data were taken after the primary effects of the step had died away. When the peak was not observed in off-line data, it was estimated using average values.
- (3) A single number indicates that the data were taken during a pass in real time. Two numbers indicate the data were taken between passes, either from the tape recorder or estimated.
- (4) The precision of the SNAP 10A power output data and estimates shown is ±25 w (~95% confidence limits). The amount of peak power was dependent on both the number of drum steps and whether the system was in the sun.
- (5) The precision of the payload power data and estimates shown is ±50 w (approximate 95% confidence limits).
- (6) The power dissipated by the voltage regulator was calculated by subtracting the payload power from the SNAP 10A power output. While this was not quite correct, the error involved was less than the error in the data. The precision of the data shown is ±60 w.
- (7) The power margin was calculated by subtracting the power dissipated by the regulator from the maximum specification requirements of 380 w.⁽¹⁾ However, the regulator was designed so that it would continue to regulate up to a load of 500 w.

TABLE 41

CALCULATION OF MAXIMUM VOLTAGE REGULATOR
ELECTRONICS TEMPERATURE

Time Period (hr)	Electronics Temperature During Passes (°F) ⁽¹⁾		Power Dissipated by Regulator During Passes (w) ⁽²⁾
	Sun	Shade	
0 to 60	110	75	190
100 to 200	140	105	225
650 to 1040	110	77	185

(1) From Figure 70.

(2) From Figure 69.

The calculation of change in electronics temperature with change in power is shown below, for 0 to 60 hours compared to 100 to 200 hours, sun:

$$\frac{\Delta T}{\Delta P} = \frac{140-110}{225-190} = \frac{30^{\circ}\text{F}}{35 \text{ watts}} = +0.85^{\circ}\text{F/w}$$

for 0 to 60 hours compared to 100 to 200 hours, shade:

$$\frac{\Delta T}{\Delta P} = \frac{105-75}{225-190} = \frac{30^{\circ}\text{F}}{35 \text{ watts}} = +0.85^{\circ}\text{F/w}$$

Similarly comparing 650-1040 hours to 100-200 hours, gives +0.75°F/w in the sun and +0.70°F/w in the shade. All four values are very close considering the accuracy of the data used.

The calculation of electronics temperature during the period of highest power consumption between passes is described as follows:

The average regulator power dissipation between passes from 100 to 200 hr was 325 w (see Figure 69). At the same time, dissipation during passes averaged 225 w.

Using the lowest, conservative, calculated value for $\Delta T/\Delta P$, the increase in temperature between passes is

$$(325-225) (+0.7^{\circ}\text{F/w}) = 70^{\circ}\text{F}$$

Temperature during pass +70°F = temperature between passes

∴ Maximum average temperatures between passes are

$$140 + 70 = 210^{\circ}\text{F in sun}$$

$$105 + 70 = 175^{\circ}\text{F in shade}$$

The voltage regulator loads at which electronics temperature exceeds maximum limit of 160°F are:

a) In sun, $T = 160 - 140 = 20^{\circ}\text{F}$

$$P = \frac{20^{\circ}\text{F}}{0.7^{\circ}\text{F/watt}} \approx 30 \text{ w}$$

∴ Maximum regulator load in sun = 225 + 30 = 255 w

b) In shade $T = 160 - 105 = 55^{\circ}\text{F}$

$$P = \frac{55^{\circ}\text{F}}{0.7^{\circ}\text{F/watt}} \approx 80 \text{ w}$$

∴ Maximum regulator load in shade = 305 w

VII. REFERENCES

1. J. F. Pohlman, "SNAPSHOT Failure Analysis Report," LMSC-A764759, NAA-SR-MEMO-11635 (U) (September 30, 1965)
2. R. M. Galantine, et al., "Preliminary SNAPSHOT-1 Performance Summary," NAA-SR-11430 (U) (August 1, 1965)
3. L. L. Bixson, "SNAP 10A Power Unit," NAA-SR-MEMO-7886 (CRD) (June 30, 1964)
4. B. F. Ureda, "SNAPSHOT Launch Operation," IEEE Transaction on Nuclear Science, Vol. NS-13, No. 1 (February 1966)
5. Staff, "SNAPSHOT Program System Test Evaluation and Performance Analysis," LMSC-A745746 (CDI) (June 16, 1965)
6. H. L. Henley and W. H. Dauterman, "Structural Test on the Final SNAP 10A Prototype System (PSM-1A)" (U), NAA-SR-9820
7. R. Kidwell, "Thermal Control of Spacecraft," Space/Aeronautics, Vol 44, pp 55-61 (July 1965)
8. T. Boyle and G. Drucker, "SNAP 10A Pre-Startup and Startup Analysis," NAA-SR-9720 (CRD) June 30, 1964)
9. R. Blaine and R. Berland, "500-THGC," NAA-SR-TDR-9664 (U) (March 27, 1964)
10. T. Boyle and E. Ross, "FSM-1 Pre-Startup Orbital Test Results," NAA-SR-TDR-9709 (U) (July 23, 1964)
11. C. E. De Cook, "FSM-4 Orbital Operations Prior to Startup," NAA-SR-11439 (U) (June 14, 1965)
12. P. Jalichandra, "Pre-Startup Coolant Temperature Control," NAA-SR-TDR-9953 (February 27, 1964)
13. G. H. Parker, "SNAP 10A FSM-1 Startup Test Results," NAA-SR-MEMO-9808 (CRD) (February 1964)
14. S. Miner, et al., "Preliminary Test Results, SNAP 10A FS-3," NAA-SR-11206 (CRD) (September 1, 1965)
15. Staff, "SNAPSHOT Postflight Design Review," LMSC-A769348 (October 15, 1965)
16. P. Soske, "SNAP 10A System Optimization," NAA-SR-11935 (U) (June 1966)
17. W. W. Davis and J. Susnir, "SNAP 10A Reactor Design Summary, Rev. 1," NAA-SR-MEMO-8679, Rev. 1 (CRD) (October 26, 1964)

18. J. Hawley and R. A. Johnson, "S10FS-3 Reactor Performance," NAA-SR-11397 (CRD) (July 1966)
19. D. Staub, "Progress Report, SNAP 10A, July-September," NAA-SR-11693 (CRD) (December 29, 1965)
20. J. A. Belcher, W. A. Flynn, and R. J. Thomson, "SNAP 10A Radiation Shield Analysis," NAA-SR-9647 (CRD) (October 15, 1964)
21. T. R. Strayhorn and N. L. Wilson, "Evaluation of Radiation Environment and Shield Design Parameters for the SNAP 10A/Agena-D and SNAP 2/Agena-D Vehicles - Phase B," WL-TDR-64-32 (June 1964)
22. S. R. Rocklin, et al., "Design and Testing of SNAP 10A Thermoelectric Power Conversion System, NAA-SR-11955 (CRD) (June 1966)
23. K. A. Davis, "SNAP 10A Thermoelectric Pump," NAA-SR-11879 (U) (March 1966)
24. S. Sudar, et al., "SNAP 10A Expansion Compensator," NAA-SR-9778 (U) (October 1, 1965)
25. J. Crosby, "The Development and Qualification of Thermal Control Coatings for SNAP Systems," NAA-SR-9908 (U) (September 30, 1965)
26. W. G. Kruse, "Radiation Dosimeters Developed for SNAP 10A Flight Test," IEEE Transactions on Nuclear Science, Vol NS-13, No. 1 (February 1966)
27. A. J. Chesavage, "Solid State Neutron Detectors - SNAP 10A," NAA-SR-11762 (U) (to be issued)
28. Staff, "SNAPSHOT Technical Review," LMSC-A701018 (August 13, 1964)
29. Staff, "Ion Engine Flight Test Analysis, SNAPSHOT," LMSC-A748484 (May 6, 1965)
30. D. J. Sobo, "SNAP 10A Interface Requirements and Agreements," NR7561-10 (March 15, 1965)
31. Staff, "Satellite Control Facilities Capabilities Manual," 6594th Aerospace Test Wing, Air Force System Command
32. H. L. Stiltz, Aerospace Telemetry, Prentice-Hall, New Jersey (1961)
33. J. R. Crosby and M. A. Perlow, "SNAP 10A Thermal Control Coatings," AIAA Thermophysical Properties Specialist Conference, Paper No. 65-652 (September 13, 1965)
34. R. P. Johnston, "10FSM-1 NNPU Thermoelectric Converter Heat Loss Analysis and Performance Summary," NAA-SR-10993 (U) (February 5, 1965)

35. L. L. Bixon, "SNAP 10A Flight System Prototype (FSM-1) Performance," NAA-SR-10735 (U) (May 15, 1965)
36. D. Sobo, "SNAP 10A Quarterly Progress Report, July through September 1964" (CRD), NAA-SR-10493
37. M. G. Coombs and R. A. Stone, "Space Heat Rejection," AI-9068 (October 1963)
38. W. Martin and J. Van Osdol, "Thermal Testing of the SNAP 10A Prototype System (PSM-3)" (U), NAA-SR-9380 (July 15, 1964)
39. T. Boyle and R. Galantine, "Magnetic Moments and Vehicle Torques in the SNAP 10A" (U), NAA-SR-11207 (September 1965)
40. S. Chapman and T. Bartels, Geomagnetism, Oxford Press (1940)
41. C. Brindley, "Target Recognition," Space/Aeronautics, Vol 43 (June 1965)
42. C. Cahill, "The Magnetosphere," Scientific American, Vol 212 (March 1965)
43. R. M. Ohlenkamp, et al., "SNAP 10A Component Qualification Program Final Report" (U), NAA-SR-11914 (May 6, 1966)
44. R. S. Hart and W. T. Harper, et al., "Final SNAPSHOT Safeguards Report," NAA-SR-10022, LMSC B-107701 (CRD) (March 20, 1965)
45. W. Dauterman, "PSM-1B/Agena Combined Static Test," NAA-SR-11130 (U) (March 10, 1965)
46. J. F. Pohlman, "Ground Testing of the SNAP 10A/Agena Space System," IEEE Trans. on Nuclear Science, Vol NS-13, No. 1 (February 1966)
47. A. K. Smith, "Performance of a Thermoelectric Converter with Leakage Current," NAA-SR-11417 (U) (June 7, 1965)
48. C. J. Ehlers, "Interpretation of Telemetry Data-Converter Electrical Failures (U), NAA-SR-10866 (December 31, 1964)