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SNAP PROGRAMS
UPPER ATMOSPHERE EXPERIMENTAL
RE-ENTRY STUDY--PREFLIGHT ANALYSES

March 1962

The Martin Company
Nuclear Division

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FOREWORD

This topical report, prepared by the Martin Company, is submitted in compliance with requirements of Atomic Energy Commission Contract AT(30-3)-217, Task 7--Radiological Safety Studies for Space Power. The report has been approved by:

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SUMMARY

This report details the theory and hardware developed in support of experimental flight tests to prove and improve the safety of radioisotope-fueled auxiliary power systems for use in space. The principal objective of this study has been to verify and/or improve the analytical techniques used in predicting the heating effects of orbital re-entry on radioisotope-fueled generators and their components.

The following flight test programs were analyzed:

Phase I--Fuel Core Heating.

Phase II--Fuel Core Ablation.

Phase III--Generator Burnup.

Phase I testing is intended to evaluate and/or improve analytical procedures for predicting re-entry heating effects on generator fuel elements of four different aerodynamic shapes. Two shots are scheduled, with Atlas missiles as the test specimen carriers. The second shot is intended to measure the reproducibility of experimental results. Phase II testing is intended to evaluate and/or improve analytical procedures used in predicting the ablation rate of generator fuel elements. Again, two Atlas shots are scheduled to measure the reproducibility of test results. The Phase III program was restricted to design of a mockup generator test specimen. The specimen will not be tested in the present test series, because the Atlas missiles required have not been made available. The testing was planned to check and/or improve the analytical procedures used in predicting re-entry heating effects on a heterogeneous package.

One additional test package will accompany fuel capsule test specimens on each launch. It is a telemetry canister, the shell of which was designed, built and instrumented by the Martin Company to obtain the time-temperature history of re-entry.

Atlas vehicle performance was analyzed to determine the possible trajectories test articles might follow on re-entry. It was found necessary to consider re-entry flight path angles from -16.3 to -27 degrees. The equations of motion were then coupled with the equations of re-entry heating to predict the altitude, latitude and longitude of test specimen burnthrough. In order that this predicted burnup altitude could be checked experimentally, it was necessary to determine a satisfactory means of test data collection. Geophysics Corporation of America was subcontracted to recommend flare materials for

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encapsulation in the test specimens and to recommend photographic equipment for recording test results. Flare materials recommended were sodium, rubidium, potassium and cesium; photogrammetric equipment recommended included K-24 aerial cameras (filtered and unfiltered), spectrometers, photometers and cinecameras. It is planned to triangulate the re-entries from two aircraft and two land sites at Ascension Island. The predicted positions of test specimen burnthrough were used to establish aircraft viewing positions, courses and speeds, and to establish camera pointing angles in terms of azimuth and elevation. One of the aircraft, a C-54, will be operated by the 6555th Test Wing in support of the Atlantic Missile Range (AMR); it will carry Barnes Engineering Corporation cameramen and cameras. A second C-54, operated by the Air Force Special Weapons Center (AFSWC), will carry Air Force cameramen and equipment. One land site will be equipped and manned by AFSWC; the other, by AMR. Test data will be reduced by AFSWC and Air Force Missile Test Center (AFMTC), then analyzed and correlated with predicted results by the Martin Company. This analysis and comparison of results will be reported in an amendment to this report.

The contractual arrangement for prosecution of this program has been as follows:

- (1) The program is monitored and managed by the Auxiliary Power Branch of the U. S. Atomic Energy Commission.
- (2) The Air Force Special Weapons Center is designated Test Coordinator, with additional responsibility for portions of the test data collection and reduction.
- (3) The Air Force Ballistics Systems Division (BSD) will provide the Atlas missiles and prelaunch ground support and will arrange for distribution of missile performance data.
- (4) The Air Force Missile Test Center is responsible for missile launching and reduction of test data obtained by Atlantic Missile Range stations.
- (5) The Office of Aerospace Research (OAR) will coordinate test arrangements at Cape Canaveral.
- (6) The Astronautics Division of General Dynamics Corporation will supply to BSD the launch vehicles and scientific test pods to house test specimens.

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- (7) The Martin Company is responsible for designing tests, designing test specimens, manufacturing certain test articles, defining AMR support requirements, establishing accuracy tolerances for data and performing post-test analyses.

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I. INTRODUCTION

When designed for space use, a radioisotope-fueled thermoelectric generator can be described generally as a thin shell surrounding thermoelectric elements, their housing, their insulation and the encapsulated fuel. (See Fig. 1; insulation not shown.) Concern for human safety requires that the fuel be contained within the unpenetrated capsule under all conditions of generator manufacture, shipment, and handling at the launch site. Fuel capsules must remain intact when subjected to the heat released on a vehicle abort at the launch pad and must retain their integrity under impact stress loads if the vehicle aborts after liftoff. Generators now in orbit were tested to prove that these safety criteria had been met.

An additional safety criterion governs both vehicle abort just prior to orbital injection and re-entry from orbit: The generator, including the radionuclide fuel, must be reduced to micron-sized particulate by aerodynamic heating. Many analytical studies and laboratory tests have been conducted to ensure that generators and fuel capsules are designed to meet this requirement. However, the analytical techniques for defining the aerodynamic heating effects on a heterogeneous package of complex thermal design must be checked and/or improved by atmospheric re-entry testing. It is for this purpose that this study and test program were conceived.

Chapter II gives a description of these test programs and discusses briefly the following topics:

- (1) The test specimens.
- (2) The procedures followed in analyzing test specimen behavior.
- (3) Data acquisition and reduction.

Each of these subjects is treated in greater detail in the appendices.

Fig. 1. Low Power Radioisotope Thermoelectric Generator

II. DESCRIPTION OF TEST PROGRAM

The re-entry test program has two objectives:

- (1) To check and improve the methods used to predict heating effects on bodies aerodynamically similar to generator fuel capsules.
- (2) To check and improve the methods used to predict heating effects on bodies having the aerodynamic configuration and thermal characteristics of a heterogeneous generator package.

The test goals indicated in Item (2) will not be reached, because Atlas vehicles have not been provided for the program.

The precision with which parameters of re-entry are known (velocity, re-entry angle and atmospheric density) and the relative influence of test specimen re-entry orientation (stable, tumbling and/or spinning) are factors to be considered in test design. To check and improve the methods used to predict re-entry heating effects, the discrete influences of these factors must be separable in the test data collected. To meet Objective (1), two phases of testing are required. Phase I will utilize 32 test specimens of four different aerodynamic configurations. Eight specimens of identical shape will be used to check reproducibility of results within a specific configuration. Four distinct configurations will permit a check of the general applicability of predictive techniques. Phase II complements Phase I in that it will provide the data necessary to correlate Phase I test results with specific re-entry parameters, orientation or test specimen shape. Twenty-eight test specimens of identical external dimensions will be used--seven each, of four different wall thicknesses. Again, reproducibility of test results will be checked by testing several units of a single configuration. The use of four different wall thicknesses will distribute the burnthrough positions with respect to altitude, thus providing an indication of ablation rates experienced. Both the Phase I and Phase II tests will be run a second time to determine the reproducibility of test results.

One additional test article will be carried on each of the four firings planned: a telemetry canister of configuration and ballistic coefficient similar to that of other test specimens. The canister is instrumented to sense temperature at 26 points on its surface, throughout the re-entry heating phase; it will transmit this information to receiving stations.

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A. TEST PROGRAM--SEQUENCE OF EVENTS

The Phase I and Phase II test specimens and the telemetry canisters have been designed and built by the Martin Company. A portion of the internal electronic equipment of the telemetry canisters has been the responsibility of the Air Force Special Weapons Center (AFSWC). AFSWC has completed manufacturing the ejection mechanisms which will house the test specimens and telemetry canister. The ejection system, in turn, will be housed in a scientific passenger pod (SPP) provided by the Astronautics Division of General Dynamics Corporation. All test articles will be assembled in the SPP at Cape Canaveral. The safety procedures to be observed at the launch site in handling the test articles have been prepared and distributed by Martin. The pre-launch equipment tests will be conducted by Martin, General Dynamics, AFSWC and other government agencies.

The SPP will be attached to the side of an Atlas launch vehicle (Fig. 2). It contains a programmer which sequences events occurring prior to re-entry. The SPP will be separated from the Atlas vehicle, at a velocity of approximately 5.5 fps, approximately 9 seconds after the Atlas vernier engines cut off (launch + ~300 seconds). It will be ejected down and to the right at an angle of approximately 45 degrees with the nose cone trajectory. At 3.75 minutes before the SPP is scheduled to reach 400,000 feet in the re-entry portion of its trajectory, the SPP programmer will initiate ejection of the test specimens from the pod. Ejection will occur at approximately 4 fps. All specimens of a specific configuration will be ejected simultaneously; ejections of different configurations will occur at 25-second intervals. The telemetry canister will be ejected 2 seconds after the final test specimen. The use of 25-second ejection intervals will provide a separation of test specimens compatible with the resolution achievable with photographic recording equipment.

Alkali metals are encapsulated in the test specimens. On burn-through of a test specimen, the alkali will be exposed, giving off a brilliant flash of light. A different metal is encapsulated in each of the four families of test specimens so that filtered cameras and spectrometers can distinguish among configurations or wall thicknesses by recording light of a specific wavelength. Burnthrough of the test specimens has been predicted to occur between 209,000 and 135,000 feet. The specific altitude depends on the test specimen in question and the trajectory achieved. This portion of the re-entry trajectory will be triangulated with camera equipment located on two C-54 aircraft and land sites on Ascension Island. The aircraft will be vectored to the prescribed viewing position by radar at Ascension Island. Signals from the telemetry canister will be monitored and recorded by a receiving station on the island.

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Fig. 2. Pods on Atlas Missile

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Photographic data obtained by Air Force personnel and telemetry data recorded at Ascension will be reduced by AFSWC. Test specimen velocity, re-entry angle and burnup altitude, and the time-temperature history of the telemetry canister will be provided for post-test analysis to the Martin Company. Nose cone trajectory data and weather to 100,000 feet in the re-entry test area will be provided by the AFMTC personnel operating on the AMR.

B. TEST ARTICLES

Test articles are described in the following paragraphs and are analyzed in detail in Appendices A through F.

1. Test Specimens--Phases I and II

Phase I test specimens comprise 8 spheres and 24 cylinders. The spheres are sodium-filled. The cylinders are divided into three groups of eight, each group having a different length-to-diameter ratio and each containing a different alkali metal (cesium, rubidium or potassium). Figure 3a shows the sodium-filled spheres; Fig. 3b, the cesium-filled cylinders. All Phase II specimens are cylinders--seven each, of four different wall thicknesses. Table 1 gives the physical characteristics of all test specimens.

The walls of both Phase I and Phase II specimens are lead-silver alloy (Pb + 0.1% Ag). The alkali metals in their liquid form corrode this alloy. To preclude corrosive attack (and subsequent hazard to personnel should the wall be penetrated), these metals, excepting sodium (which is in the solid state except on re-entry), are encapsulated in polypropylene containers prior to insertion into the test specimen. A summary of the laboratory tests conducted to prove that the polypropylene liners are adequate appears in Table 2. Two sets of Phase I test specimens and three sets of Phase II test specimens were manufactured.

TABLE 2
Polypropylene Corrosion Tests

Alkali Metal	Initial (lb)	Final (lb)	Weight	
			(lb)	Loss (%)
K	3.0546	3.0546	0	0
Rb	3.0005	2.9909	0.0096	0.32
Cs	2.8033	2.7972	0.0061	0.22
Cs	3.2964	3.2850	0.0114	0.35
Cs	3.6359	3.6280	0.0079	0.23

Length of test--55 days
Temperature--93° C

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TABLE 1
Physical Characteristics of Test Specimens*

<u>Phase No.</u>	<u>Group No.</u>	<u>No. per Flight</u>	<u>Dimensions (diameter x length)</u>	<u>Wall Thickness</u>	<u>Total Weight (per specimen)</u>	<u>Aerodynamic Ballistic Coefficient (W/C_DA)</u>	<u>Alkali</u>	<u>Alkali Weight</u>	<u>Total Weight, 8 Capsules</u>
I	1	8	Sphere, 3 D	3/8	3.51	77.5	Na	0.163	28.08
	2	8	Cylinder, 2 D x 5 L	3/8	4.60	86.0	Cs	0.153	36.80
	3	8	Cylinder, 2-1/2 D x 5 L	3/8	6.37	77.7	Rb	0.303	50.96
	4	8	Cylinder, 3 D x 4-1/2 L	3/8	7.38	83.5	K	0.267	<u>59.12</u>
									174.96
									<u>Total Weight, 7 Capsules</u>
II	1	7	Cylinder, 2-1/2 D x 5 L	1/4	4.70	57.3	Na	0.442	32.90
	2	7	Cylinder, 2-1/2 D x 5 L	3/8	6.15	75.0	K	0.174	43.40
	3	7	Cylinder, 2-1/2 D x 5 L	1/2	7.50	91.5	Rb	0.207	52.50
	4	7	Cylinder, 2-1/2 D x 5 L	5/8	8.46	103.0	Cs	0.144	<u>59.50</u>
									188.30

*All dimensions in inches and weights in pounds.

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Fig. 3a. Group 1, Phase I, Spherical Capsules--Sodium Fill

Fig. 3b. Group 2, Phase I, Cylindrical Capsules--Cesium Fill

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2. Telemetry Canister

The telemetry canister has been instrumented to measure, record and transmit canister wall temperature during re-entry. It is a hollow cylinder made of titanium alloy (5% Al, 2.5% Sn) 9.5 inches in diameter, 19.5 inches in length, and 0.292 inch in wall thickness. Re-entry temperature will be measured by 23 thermistors distributed on the exterior of shell; three thermistors mounted on the inside wall of the shell will permit determination of thermal gradient in the wall. Inside the titanium shell is a 1-inch insulating layer of Temporell 1500, a mixed and poured insulation which is cured to the solid state. It provides thermal protection for the electronic systems contained within the canister. A fiberglass cylinder 1/8 inch in thickness is fitted to the inside diameter of the insulation to provide annular support and alignment for the electronic components mounted within it. Preformed MIN-K insulation pads 1 inch thick, backed by titanium alloy retainer plates, close both ends of the cylinder. Transmitter antennas are attached to each retainer plate. The antennas are further protected from the re-entry heat pulse by ceramic domes held in place by a titanium alloy retainer ring threaded into the shell. The ceramic domes and retainer rings are covered with a 100-mil thickness of AVCOAT II, an ablative material used, in this case, to prevent thermal shock to the ceramic domes. Charring of the coat during re-entry does not adversely affect transmission of data.

The internal electronic and electrical systems of the canister include a battery power supply, voltage oscillators, power amplifiers, a commutator, a magnetic tape recorder and reproducer, and a transmitter. This equipment is turned on by an electrical signal just prior to re-entry. A programmer housed in the scientific passenger pod is the signal source. All sensed temperatures are transmitted twice to the receiving station, the first transmission occurring the instant the temperature is sensed and the second transmission occurring after a 30-second delay. The delay is effected through the use of a recorder/reproducer which stores all sensed temperature data for 30 seconds prior to playback. This system prevents loss of data as a result of transmission blackout, which will occur when an ion sheath is generated in the atmosphere by the canister on re-entry. The ion sheath persists for approximately 20 seconds. The transmission frequency is 244.3 megacycles.

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The telemetry canister weighs approximately 85 pounds, 40 pounds of which comprises shell, insulation and end plates. Its ballistic coefficient ($W/C_D A$) is about 70.3. Six units have been built. Two will back up four test firings now scheduled; they will then be available for supplemental tests should they be required. Figures 4a and 4b show the instrumented canister shell. A schematic showing canister construction is presented in Fig. 5; a block diagram of the electrical system appears as Fig. 6.

The theoretically predicted temperatures at several significant points on the telemetry capsule are shown in Fig. 7. For the analysis, the cylindrical capsule was assumed to be tumbling and spinning during a lofted re-entry. The temperature profiles were determined after applying the heating rate of a lofted trajectory to a two-dimensional model in the digital heat transfer code. During the maximum-heating period, the temperature at the telemetry capsule corner (B of Fig. 7) will exceed that on the surface of the capsule (A of Fig. 7) by approximately 600° F. This theoretical estimate of the temperature profile remains to be verified by actual flight test data.

3. Ejection Mechanism

The test specimen ejection mechanism, Fig. 8, consists of five eject cylinders and associated hardware. Five units have been built by AFSWC and are incorporated with the structure of the scientific passenger pods. Four of the eject cylinders will contain the four groups of test specimens; the fifth eject cylinder will contain the telemetry canister. Lock pins retain the test articles under spring tension within the eject cylinders. At predetermined times, electrical signals from the pod programmer ignite squibs which actuate solenoids. The solenoids retract the lock pins, thus allowing ejection of the test specimens and telemetry canister. The velocity of ejection will be approximately 4 fps (± 2 fps). In Fig. 9, the sequence of loading the test specimens within the ejection mechanism is shown. The arrangement of the lead capsules on the ejection cartridge is illustrated in Fig. 9a. Figures 9b and 9c show the loading of the specimens within the cylinder; Fig. 9d, the loading of the telemetry canister.

4. Mockup Test Generator--Phase III

Phase III testing is no longer planned, because the required launch vehicles are not presently available. Two different test specimen designs were developed, but neither was built. Model A, Fig. 10, was

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View a

View b

Fig. 4. Telemetry Capsule Showing Transducers and Insulation

Fig. 5. Re-entry Telemetry Capsule (500-lb pod)

Fig. 6. Telemetry Canister Electrical System

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Fig. 7. Expected Temperature Profile--Telemetry Capsule

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Fig. 8. Capsule Ejection Mechanism End View

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Fig. 9a. SNAP Re-entry Test Assembled at Fuel Storage Area

Fig. 9b. Installation of Flare Capsules into Pod

Fig. 9c. Installation of Flare Capsules into Pod

Fig. 9d. Installation of Canister in Pod

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All dimensions in inches.

Generator B

Generator A

Fig. 10. Phase III Specimens

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designed for burnup on a ballistic trajectory similar to that planned for Phase I and Phase II testing. Sodium simulates the fuel core of the generator. Testing was intended to check the precision with which the altitude of fuel core exposure can be predicted analytically for a heterogeneous package. Model B, Fig. 10, was designed for burnup on an orbital re-entry simulated by a powered re-entry of a vehicle such as Pershing or Scout. Model B is a more sophisticated design than Model A in that it more closely approximates operational generator design. The flare material is encased in Haynes-25 alloy, a material used for fuel encapsulation in operational generators. The use of this material is the principal reason for the requirement of higher integrated heating of the powered re-entry. A successful test would verify that the techniques used to predict burnup of heterogeneous packages are adequate. In addition, it could be shown that the destruction of operational hardware is possible.

C. ANALYTICAL PROCEDURES

The method of computing the re-entry behavior and ultimate fate of a body requires the simultaneous solution of trajectory, aerodynamic heating and heat transfer parameters. A more detailed description of the methods used in this study is contained in Appendices B, C and D. Aerodynamic heating is primarily a function of the altitude, velocity and shape of the body. These parameters also determine re-entry trajectory. The equations of heating and equations of motion were programmed for the IBM-7090 to produce a printout of the trajectory and heating rate as functions of time. The aerodynamic heat flux as a function of time was then used as an input to the IBM-7090 heat transfer code (see Appendix C) to determine the temperature gradient through the body and the rate of ablation of the body. Figure 11 shows the aerodynamic heat flux for the spherical test specimen of Phase I. The heating rate was computed for a spinning sphere in laminar flow. The altitude of burnup was calculated to be 192,000 feet.

During re-entry, ablation significantly alters aerodynamic parameters such as ballistic coefficient. To account for this effect, an iterative routine is followed by the IBM-7090. Flight parameters are computed, and ablation is calculated, to a point where the body change is significant. New flight parameters and heat flux data are then computed for the partially ablated body and used as inputs to the trajectory and heat transfer codes. The procedure is continued until the body is completely ablated or until it impacts at the end of its trajectory. All lead test capsules are predicted to be completely ablated. Table 3 gives the minimum and maximum predicted altitudes of burnup for each test specimen of Phases I and II. The minimum altitude is obtained if the re-

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Fig. 11. Stagnation Heating Rates--Three-Inch Diameter Sphere, Phase I Testing

TABLE 3
Predicted Altitudes of Test Specimen Burnup

Test Phase	Weight (lb)	Drag Coefficient, $W/C_D A$	Dimensions (in.)	Alkali	Re-entry Angle (deg)	Burnup Altitude (ft x 10 ⁵)	
						Minimum	Maximum
I	3.51	77.5	Sphere 3, dia 3/8, wall	Na	-16.3	2.09	2.09
					-19.3	2.02	2.02
					-22	1.98	1.98
					-24	1.92	1.92
					-27	1.87	1.87
I	4.6	86.0	Cylinder 2, dia 5, length 3/8, wall	Cs	-16.3	1.88	2.04
					-19.3	1.80	1.98
					-22	1.73	1.89
					-24	1.71	1.88
					-27	1.67	1.85
I	6.37	77.7	Cylinder 2.5, dia 5, length 3/8, wall	Rb	-16.3	1.78	1.99
					-19.3	1.75	1.93
					-22	1.71	1.88
					-24	1.65	1.85
					-27	1.59	1.76
I	7.38	83.5	Cylinder 3, dia 4.5, length 3/8, wall	K	-16.3	1.72	1.99
					-19.3	1.64	1.39
					-22	1.58	1.84
					-24	1.55	1.79
					-27	1.50	1.65
II	4.7	57.3	Cylinder 2.5, dia 5, length 1/4, wall	Na	-16.3	1.72	1.88
					-19.3	1.66	1.83
					-22	1.59	1.77
					-24	1.54	1.71
					-27	1.49	1.66
II	6.15	75.0	Cylinder 2.5, dia 5, length 3/8, wall	K	-16.3	1.78	1.99
					-19.3	1.75	1.93
					-22	1.71	1.88
					-24	1.65	1.85
					-27	1.59	1.76
II	7.5	91.5	Cylinder 2.5, dia 5, length 1/2, wall	Rb	-16.3	1.69	1.85
					-19.3	1.63	1.80
					-22	1.57	1.74
					-24	1.50	1.69
					-27	1.47	1.64
II	8.46	103.0	Cylinder 2.5, dia 5, length 5/8, wall	Cs	-16.3	1.60	1.75
					-19.3	1.50	1.69
					-22	1.45	1.65
					-24	1.39	1.58
					-27	1.35	1.55

entering body both tumbles and spins; the maximum altitude results if only spinning is assumed. Since the heat flux depends on the trajectory, and since the exact trajectory to be achieved in actual test cannot be predetermined, the burnup altitudes (maximum and minimum) were calculated for re-entry flight path angles from -16.3 to -27 degrees. This range was selected because the Atlas vehicle will fly either a lofted (-24 -degree) or standard (-19.3 -degree) trajectory. A dispersion of ± 3 degrees is associated with each of these R&D mission trajectories.

D. DATA ACQUISITION AND REDUCTION

Test data will be gathered by means of photography (filtered, unfiltered, color, black and white, and spectrographic), photometry, telemetry and radar. Geophysics Corporation of America, under subcontract to the Martin Company, recommended the photographic and photometric equipment which will be used (Ref. E-4).

1. Photographic Data

The time, location and re-entry velocity at which the test specimens are penetrated (flare material ignition) will be observed and recorded by time-indexed photographic equipment in the re-entry area. The tests will be observed from aircraft provided by both AFSWC and AMR, from Ascension Island, and possibly from the DAMP ship. The camera equipment located at these stations will include K-24 aerial cameras, BC-4 star cameras, 16- and 35-mm cinecameras, and a Hulcher 70-mm sequence spectrometer. Exposed spectrometer and filtered K-24 camera films will be used to differentiate among test specimens on the basis of the characteristic wavelengths of light emitted by the several alkali materials. Unfiltered K-24 cameras will record re-entry events along with a star background for precise triangulation of the events. The BC-4 camera will remain open throughout the re-entry, recording the streaking of burning bodies. A chopper will obscure the aperture for a precisely known period of time approximately four times per second, causing breaks in the streaks recorded in the film plate. From the knowledge of time, streak length and camera orientation (star background), test specimen re-entry velocity will be determinable.

2. Photometric Data

One two-barrel photometer will be hand-trained on the re-entry events. The photometer will be time-indexed to a visicorder in order to record a real time history of relative light intensities existing during re-entry burnup of test specimens. One barrel of the photometer will be filtered for preferential observation of light of $7800 \pm 50 \text{ \AA}$, the principal rubidium line; the other barrel, for light of $8521 \pm 50 \text{ \AA}$, the principal cesium line. When the rubidium- and cesium-filled test

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specimens are penetrated, light of the appropriate wavelengths will peak in intensity. Time values then obtained will be correlated with re-entry radar data to determine burnup altitude. These data will serve as an independent check of altitudes determined by camera triangulation techniques. The Barnes Engineering Corporation, under contract to AFSWC, will provide and operate both the photometer and the Hulcher 70-mm spectrometer.

3. Radar Data

The Atlantic Missile Range will provide two types of radar support from Ascension Island: re-entry trajectory data and aircraft positioning. Nose cone trajectory data will be obtained from a beacon tracking radar. Since the nose cone will appear on time-marked photographic plates with the test specimens, burnup altitude can be deduced from nose cone time-position history as revealed by the radar. The Mod II and Nike-Hercules radars at Ascension will be used to vector observation aircraft to prescribed positions and courses for viewing the re-entry. Since many of the cameras recording test events are fixed-mount equipments, it will be essential that the aircraft be on course at the precise position and time that re-entry is predicted to occur. The radar track of the aircraft's positions during the re-entry will be permanently recorded. This record will be used, with triangulation, to determine the spatial coordinates of test specimen burnup.

4. Telemetry Data

The output frequency of the telemetry canister transmitter is 244.3 megacycles (mc). In the event test shots should fail, additional information telemetered at 256.1 mc from the scientific passenger pod will be of value in trouble-shooting. Information telemetered at 256.1 mc will include a signal of positive pod separation from the Atlas vehicle and a signal of positive ejection from the pod for each group of test specimens and for the telemetry canister.

AMR telemetry receiving stations at Cape Canaveral, Antigua, Grand Bahama and Ascension Island will receive the transmitted frequencies and record them on magnetic tape (AMPEX FR-100). Copies of the tapes will be furnished to AFSWC for data reduction. Data will be reduced in accordance with the work flow diagram of Fig. 12.

5. Meteorological Data

Aerodynamic heat flux determination requires that the density of the atmosphere be considered. The state of the atmosphere about the shock wave generated by the re-entry body must be known. The ARDC standard atmosphere model has been assumed in predicting re-entry burnup altitudes. At the time of Phase I and Phase II tests, the AMR

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Fig. 12. Telemetry Data Reduction

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- will determine actual atmospheric conditions in the re-entry test area to an altitude of 110,000 feet. Either the atmospheric conditions above this altitude must be extrapolated, following the trends in evidence, or the ARDC standard atmosphere model will be assumed when the post-flight analysis of test data is performed. It is recognized that extrapolation of atmospheric data may be a source of error in comparison of test and predicted results. Since these tests will be conducted to verify and improve predictive techniques, however, it is necessary that the most accurate input data be used. It should be noted here that, since there will be no available information on the atmospheric conditions existing when an operational generator re-enters, all safety studies conducted on such systems are sufficiently conservative to obscure the effects of atmospheric variation.

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III. CONCLUSIONS

Upper atmosphere re-entry tests have been designed to reveal:

- (1) Information on the orientation of generator fuel capsules on re-entry.
- (2) Information on the effect of aerodynamic ballistic coefficient on burnup altitude.
- (3) Information regarding the flow regime of test specimens in re-entry flight (i.e., laminar or turbulent flow).
- (4) Information on the ablation rate of re-entering test specimens.

The data acquisition requirements have been adequately defined with respect to:

- (1) Equipments to be used.
- (2) Viewing positions.
- (3) Accuracy required.

The following test hardware has been designed, built and delivered:

- (1) 64 Phase I test specimens.
- (2) 84 Phase II test specimens.
- (3) 6 telemetry canisters.

Prelaunch safety requirements for the handling of alkali materials at the launch site have been defined, and arrangements with Range personnel have been concluded.

Coordination of testing with Convair, AFSWC, OAR, AMR (Cape Canaveral and Ascension Island), BSD and the 6555th Test Wing, Patrick Air Force Base, has been completed.

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APPENDIX A

ANALYSES OF TEST ARTICLES*

Only that hardware designed, built and tested by the Martin Company is discussed: the test specimens (hereafter referred to as flare capsules or capsules) and the telemetry canister.

A. PHASE I AND PHASE II FLARE CAPSULES

Flare capsule configurations are shown in Figs. A-1 and A-2. The capsules, which simulate SNAP components and can be destroyed on a ballistic trajectory, are fabricated of a lead-silver alloy (99.9 Pb+0.1 Ag). This alloy was chosen because of its low melting point and high density and because it has thermal properties similar to those of Inconel X, Hastelloy C, Haynes 25, etc., materials used in operational SNAP systems. The flare capsules are filled with alkali metals, which, on capsule penetration by aerodynamic heating, will react with the atmosphere and cause brilliant bursts of light. The metals selected are sodium, potassium, rubidium and cesium. Since the liquid alkali metals may react with lead and cause premature capsule penetration, the capsules are lined with polypropylene, a high density plastic. The properties of polypropylene are as follows:

Coefficient of expansion	62×10^{-6} in./in. -° F
Specific heat	0.46 Btu/lb -° F
Thermal conductivity	0.95 Btu/hr-ft ² -° F
Heat of fusion	25 cal/gm
Specific gravity	0.9 to 0.91
Young's modulus	4.4 to 4.8×10^5 lb/in. ²
Tensile stress allowable	4300 to 4900 lb/in. ² at 70° F
Yield stress	1700 lb/in. ² at 212° F
Elongation	150%
Stiffness and flexural modulus	114 to 117×10^3 lb/in. ² at 70° F
Reactivity with Cs, Rb, K	Virtually inert

*W. C. Dallam and R. Stankiewicz.

Fig. A-1. Re-entry Capsules, SNAP Hazards Tests, Phase I

Group 1

Group 2

Group 3

Group 4

NOTE:
All dimensions in inches.

Fig. A-2. Re-entry Capsules, SNAP Hazards Tests, Phase II

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All liners and capsules are designed with void space to allow for volumetric thermal expansion. These internal spaces are filled with argon gas, pressurized to 0.050 psia. Argon was used to prevent the presence of oxygen in the vicinity of the alkali metals.

To provide safety assurance during handling, three drop tests were conducted on two full-scale test capsules. From a distance of 45 feet, capsules were impacted on their ends, corners and sides against concrete. Figure A-3 shows capsule deformation after testing. Post-test visual inspection revealed sound parent material and welds in both capsules. Impact produced a deceleration of 1400 g's and resulted in approximately 15% deformation. Figure A-4 is a graph of lead rupture stress as a function of temperature.

During the ascent of the launch vehicle, temperatures inside the SPP will approach 200° F. Accelerations of about 7 to 10 g's are expected. An analysis has shown that the thinnest-walled lead capsule will withstand loads up to 50 g's without deformation.

B. TELEMETRY CANISTER

The telemetry canister was designed to survive high temperature (1440° F) and descent acceleration (50 g's) for the short time of re-entry. These maximum conditions were determined to occur about 40 seconds after the canister descends to 400,000 feet. The dynamic pressure was calculated to be 2550 pounds per square foot (psf). The tensile yield strength of the titanium alloy (92.5% Ti + 5% Al + 2.5% Sn) canister is given in Fig. A-5. Investigation of the bending stress and deflection of the shell under maximum stress conditions yielded the following:

Bending stresses:

$$\sigma_B = \frac{M \bar{y}}{I} = \frac{P d_o L^2 (d_o)}{\frac{\pi}{8} (d_o^4 - d_i^4) 2} = 270 \text{ psi}$$

Deflection:

$$f = \frac{5}{384} \frac{WL^3}{EI} = \frac{5}{384} \frac{P d_o L^4}{E \frac{\pi}{64} (d_o^4 - d_i^4)} = 0.020 \text{ in.}$$

Thermal stresses resulting from aerodynamic heating on the body were also considered. Figure 7 illustrates the temperature distribu-

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Fig. A-3a. Drop Test Capsule 1 (side impact)

Fig. A-3b. Drop Test Capsule 1 (side impact)

Fig. A-3c. Drop Test Capsule 2 (corner and end impacts)

Fig. A-3d. Drop Test Capsule 2 (corner and end impacts)

Fig. A-4. Rupture Stress of Cast Lead Capsules (0.1% silver additive)

Fig. A-5. Tensile Yield Strength (5 Al-2.5 Sn titanium alloy)

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tion along the surface of the canister. The maximum temperature gradient through the wall occurs almost simultaneously with maximum deceleration load. The thermal stresses in the thin-walled cylindrical body were determined to be:

Tangential stress at inner surface:

$$\sigma_{\phi} = \frac{\alpha E (T_i - T_o)}{2(1-\mu) \ln \frac{b}{a}} \left[1 - \ln \frac{b}{r} - \frac{a^2 (r^2 + b^2)}{r^2 (b^2 - a^2)} \ln \frac{b}{a} \right] = 25,800 \text{ psi (tension)}$$

Axial stress at inner surface:

$$\sigma_z = \frac{\alpha E (T_i - T_o)}{2(1-\mu) \ln \frac{b}{a}} \left[1 - \ln \frac{b}{r} - \frac{2a^2}{b^2 - a^2} \ln \frac{b}{a} \right] = 12,700 \text{ psi (tension)}$$

It appears that a stress condition may exist at a time subsequent to the period when the temperature of the telemetry capsule wall reaches its maximum as shown in Fig. 7. The calculated stresses approximate the tensile yield strengths as extrapolated in Fig. A-5; therefore, as a result, minor distortion may result. However, it is expected that this deformation, if any, will have negligible effects on the test program since the data of interest will have been obtained by this time. It is also anticipated that the recorder-reproducer instruments located within the canister will still be able to operate for a sufficient duration to allow transmission.

C. STRUCTURAL ANALYSIS NOMENCLATURE

<u>Quantity</u>		<u>Unit of Measurement</u>
a	= inner radius	in.
b	= outer radius	in.
d	= diameter	in.
E	= modulus of elasticity	psi
I	= moment of inertia	in. ⁴
L	= length	in.
M	= bending moment	in.-lb

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<u>Quantity</u>		<u>Unit of Measurement</u>
P	= pressure	psi
r	= average radius	in.
T	= temperature	° F
u	= Poisson's ratio	
W	= weight	lb
α	= thermal expansion coefficient	in. / in. -° F
σ	= stress	psi
f	= static deflection	in.

Subscripts

i	= inside wall
o	= outside wall
ϕ	= tangential direction
z	= axial direction
B	= bending

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APPENDIX B

AERODYNAMICS *

This section describes the expected trajectories of the test vehicle and specimens and the flight plans of the downrange observation aircraft. It is planned to fly two types of missions with the test vehicle. Both types will give identical ranges, but they will achieve them by different trajectories. The Standard R&D trajectory, specifically for Pod 23², will be fired to an apogee of 3×10^6 feet while the Lofted R&D trajectory (Pods 22¹, 24³ and 25³) will reach apogee at 4×10^6 feet. Two complete sets of re-entry conditions will result. The location of the downrange instrumentation aircraft will depend on the type of mission flown.

The test specimens will be placed within a pod (scientific passenger pod) which is mounted external to the flight missile. The pod is attached well aft of the carrier's tankage and is jettisoned subsequent to nose cone separation. Specifically, the events are as follows:

- (1) The vehicle tankage (with the attached pod) is retrorocketed at approximately 4 seconds after vernier engine cutoff (VECO + 4 seconds).
- (2) If one looks forward at the aft end of the missile in flight, the pod will be seen to be located in the lower right quadrant. The tankage will have a random tumbling rate after nose cone separation of approximately 2 degrees per second. The pod is separated from the spent tankage at a velocity of 5.5 fps down to the right at VECO + 9 seconds. The trajectory the pod follows will eventually carry it to a re-entry position forward of the tankage.
- (3) At VECO + 1188 seconds (for the Standard trajectory) and VECO + 1336 seconds (for the Lofted trajectory), the first group of specimens will be ejected from the pod. The second, third and fourth groups will be ejected at 25-second intervals. The telemetry canister will be ejected 2 seconds after the last group of flare capsules. Ejection times were so determined that all test units would be exposed to the atmosphere prior to reaching 400,000 feet during re-entry.

*W. Hagis, E. E. Markson, C. L. Jedlicka.

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- (4) Prior to its ejection, the telemetry capsule receives a 600-milliampere current for 1 second from the pod programmer. This pulse of current fires 2 squibs which switch on the telemetry system. During ejection of the telemetry capsule, a microswitch is tripped, providing a positive indication that ejection has taken place.

The re-entering test specimens will surround the scientific passenger pod (SPP) and will tend to disperse in a random direction. Thus, a "cone of re-entry" is established, the groups tending to spread aerodynamic forces act on the various shapes. It has been calculated that the diameter of the cone is approximately 3000 feet at 300,000 feet and 3300 feet at 150,000 feet. Figures B-1 and B-2 show the complete trajectory of pod tankage and nose cone from the point of pod separation to near impact of the bodies. This plot is for the nominal trajectory of the Lofted R&D mission. For convenience, the flight paths of the test specimens are represented by that trajectory expected for the SPP. The scales on the plot should be noted. The altitude and range on the abscissa and ordinate apply only to the nose cone. The relative distance of the vehicle's tankage and the SPP are plotted with respect to the nose cone. The inserted scale of altitude and range are to be used for this plot. Thus, at an altitude of 217,000 feet for the nose cone, the SPP is 13,000 feet downrange and 23,500 feet beneath the nose cone while the tankage is 4000 feet downrange and 18,500 feet beneath it.

It can be seen that the nose cone lags every other body during the re-entry. It approaches, and later passes, the other bodies at lower altitudes (~ 120,000 feet). The aerodynamic parameters of the nose cone are such that atmospheric drag has less effect and will not cause as much deceleration as for the SPP and tankage. An aerodynamic coefficient ($W/C_D A$) of 75 was used to calculate the flight path of the SPP for this plot.

The test specimens to be ejected are described in Chapter III of this report. The methods of calculating the aerodynamic heating and heat transfer through the bodies are described in Appendices C and D. The re-entry trajectories for this program were calculated by means of IBM-7090 codes involving both two-dimensional and three-dimensional methods of solution. Because of the possible errors that could be accumulated over a long trajectory, the 3-D program was utilized to establish the spatial relation of the units. An oblate rotating earth was considered.

The following procedure is generally followed during missile firings. During the powered ascent portion of the trajectory, the altitude, velocity and flight path angle are continuously monitored and fed into computers so that impact points can be predicted instantaneously.

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Fig. B-1. Trajectory Profile--Nominal Trajectory of Lofted R&D Mission

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Fig. B-2. Trajectory Profile--Nominal Trajectory of Lofted R&D Mission Low Altitudes

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The thrust of the engines is terminated when the combination of these three parameters is such that a desired point downrange can be reached. Thus, it is possible to have an infinite number of re-entry trajectories, all of which terminate at a common point downrange. Each type of mission can be described by a nominal trajectory with dispersions associated with it.

The Standard R&D trajectory will be at 400,000 feet during re-entry at 1725 seconds ($^{+84}_{-80}$ seconds), at a flight path angle of -19.3 degrees (± 3 degrees), and at a velocity of 21,300 fps. The numbers in parentheses are the dispersions of each parameter.

The Lofted R&D trajectory will have the following parameters at 400,000 feet during re-entry:

Flight path angle	-24 degrees ± 3 degrees
Velocity	21,300 fps
Time	1876 seconds $^{+84}_{-80}$ seconds.

The trajectory, shown in Fig. B-3, represents that for the Lofted R&D series and shows altitude as a function of range.

To encompass the re-entry altitude region between 250,000 and 80,000 feet, the cameras must be able to survey a range field of view of approximately 68 nautical miles. This range dispersion results from trajectory variations stated previously. Theoretically, there should be no dispersion normal to the azimuth of the flight path.

The trajectories shown in Figs. B-4, B-5 and B-6 are for a 3-inch Phase I sphere. These graphs include nominal, slow and fast trajectories for Lofted R&D flights. These curves also indicate the trajectories to be expected for the cylindrical capsules. The earth-relative velocity and flight path angle are shown along with the altitude and average aerodynamic heating rate.

The downrange instrumentation aircraft consists of a C-54 operated by AFSWC personnel and an AFMTC C-54. The equipment aboard each plane is described in Appendix E. The positioning of each aircraft is dependent upon the following parameters:

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Fig. B-3. Altitude Profile of Lead Capsules--Lofted R&D Trajectory

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Fig. B-4. Re-entry History--Spherical Lead Capsule, Group 1, Phase I

Fig. B-5. Re-entry History--Spherical Lead Capsule, Group 1, Phase I

Fig. B-6. Re-entry History--Spherical Lead Capsule, Group 1, Phase I

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- (1) The location of the "flaring" test specimens with respect to the burning vehicle tankage during re-entry.
- (2) The field of view and angle of pointing of the photographic equipment. (All expected "flare" events over the complete dispersion of the trajectories (Fig. B-3) must be enveloped.)
- (3) The pointing of the cameras so as to preclude their viewing the moon.

The full usable viewing angles of the cameras were considered in positioning the aircraft at sufficient distances so that coverage would be assured. It was also deemed advisable not to move the cameras during flight, because of their bulkiness and because there may not be sufficient time for proper positioning following the first sight of a burst.

The AFSWC C-54 (designated the No. 1 aircraft) would fly the flight plans shown in Fig. B-7. Two positions for the aircraft are shown: Position A is for viewing the Lofted R&D trajectory, and Position B is for the Standard R&D trajectory. The relative locations of the test specimens, vehicle tankage and nose cone alone were considered with respect to Items (2) and (3) above.

Aircraft No. 2 (AFMTC C-54) is considered as a backup observation unit; it contains a two-barrel photometer and a high-speed sequence spectrometer (70-mm Hulcher) equipment. Its position during each type of flight is shown in Fig. B-8. The photometer is expected to provide accurate time determinations of the flaring event. The field of view of a photometer of the type used in this program is 2 degrees while the field of view of the spectrometer is 18 x 20 degrees. Thus, a compromise must be established between the aircraft position requirements of the photometers and those of the spectrometer. These No. 2 aircraft optical devices are so mounted that the operator will use the spectrometer as a boresight for the photometer. The operator will aim the spectrometer on the burning re-entering tankage in order that the photometer, with the 2-degree field of view, will capture the flare events in their predetermined positions.

Upon burnout of the vernier engines, the time that the nose cone crosses 250,000 feet during re-entry will be predicted by range computers at the launch site. This time will be communicated to Ascension Island and relayed to the aircraft so that they may be properly positioned with cameras operating when re-entry occurs.

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Fig. B-7. Flight Plan--Aircraft No. 1

Fig. B-8. Flight Plan--Aircraft No. 2

References

- B-1. "Flight Test Plan, Pod 22," General Dynamics/ Astronautics, AE 61-0296, June 13, 1961. CONFIDENTIAL
- B-2. "Flight Test Plan, Pod 23," General Dynamics/ Astronautics, AE 61-0372, July 15, 1961. CONFIDENTIAL
- B-3. "Flight Test Plan, Pods 24 and 25," General Dynamics/ Astronautics, AE 61-0654, September 20, 1961. CONFIDENTIAL

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APPENDIX C

AERODYNAMIC HEATING*

During the descent from high altitude, a re-entering body will encounter flow regimes varying from free molecular flow to continuum flow. In a complete analysis of the aerodynamic heating on the body, the effect of the change in flow field on the heating rates should be included. However, where many re-entry trajectories are to be investigated, it is expedient to simplify the analysis by assuming the flow field to be of the continuum type throughout the descent, provided the error incurred can be considered negligible.

Re-entry of the sample capsules into the sensible atmosphere is assumed to occur at an altitude of 400,000 feet. A typical re-entry trajectory, indicating the variation of velocity with altitude during the descent, is shown in Fig. C-1. Also indicated on this figure are the region of maximum aerodynamic heating, based on continuum flow theory, and the flow regimes expected to be encountered during the descent. The region of maximum aerodynamic heating is defined as the portion of the trajectory where the heating rates exceed 10% of the peak heating rate at the stagnation point.

The approximate locations of the various flow regimes were determined on the basis of the comparison of the local or free stream mean free path and the body radius as defined by Probst (Ref. C-1). These regimes will vary somewhat with changes in the trajectory and the body size. However, the relation between the region of maximum aerodynamic heating and the location of the various flow regimes will be essentially the same. Figure C-1 indicates that continuum flow will be encountered at altitudes below approximately 240,000 feet. Above this altitude, slip flow, transitional flow and free molecular flow will prevail.

References C-1, C-2 and C-3 indicate that the use of continuum theory can be extended into the slip flow regime with some modification to account for such effects as vorticity interaction. However, since the aerodynamic heating in this flow regime is small in comparison to the total heat input to the vehicle, continuum theory, without modification, appears to be adequate. Similarly, in the free molecular flow regime, the heating rates are small--much less than that which continuum theory would indicate. Therefore, the use of continuum theory would tend to overestimate the heating rates of this regime, but the error incurred would be negligible. It appears that the use of continuum theory during the entire descent from 400,000

*G. Nielsen.

Fig. C-1. Typical Re-entry Trajectory

feet would introduce only negligible error. The increased ease of computing the aerodynamic heating during the re-entry counterbalances any error introduced by use of this approach.

The thermochemical state of the gas behind the shock and in the boundary layer can also be an important factor in the evaluation of the heat transfer to the re-entry body. If the flow is in a nonequilibrium or frozen state, heat transfer to the body can be substantially reduced, provided the surface material is noncatalytic (that is, it inhibits recombination of the dissociated gas). However, if the wall material is catalytic, as is the assumed case here, Refs. C-4 and C-5 indicate that, although the heat flow mechanism is different, the total heat transfer rate is substantially the same for equilibrium flow, frozen flow and all intermediate cases.

Another important consideration in evaluating re-entry heating is the possibility of encountering turbulent boundary layer flow with its accompanying increase in heat flux. During the initial phases of re-entry, the Reynolds number is low so that the boundary layer in the stagnation region and over most of the body will be laminar. As the descent continues into the more dense atmosphere with a subsequent increase in Reynolds number, transition from laminar to turbulent flow is possible either in the stagnation region or at some point around the body. Although much theoretical and experimental work has been carried out to determine a method for predicting transition of the flow in the boundary layer, no conclusive results have been attained to date. This is primarily due to the many factors affecting transition. It is known that a favorable pressure gradient and cooling of the boundary layer (heat transfer from boundary layer to the surface) are beneficial in maintaining a laminar boundary layer.

In lieu of more conclusive data, the results obtained by Stetson (Ref. C-6) were used to indicate the probability that transition from laminar to turbulent flow would occur. Reference C-5 indicates that the transition Reynolds number (based on local flow properties at the outer edge of the boundary layer, and the momentum thickness) in the sonic region of a hemisphere varies from 225 to 325 as the stagnation-to-wall enthalpy ratio increases from 9.5 to 29.5. This report also indicates that the transition Reynolds number (based on local internal flow properties and distance from the stagnation point) on the cylindrical portion of the body varies from 350,000 to 750,000 for stagnation-to-wall enthalpy ratios of 3 to 30. Assuming that these results are valid for the configurations of this analysis and taking the conservative approach using the lower value for the transition Reynolds number lead to the conclusion that transition from laminar flow to turbulent flow will occur. However, it also appears that this phenomenon will occur some time after peak laminar heating and after appreciable

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deceleration of the re-entering body. Hence, transition of the boundary was not considered in the re-entry analysis of the test specimens since it occurred long after the predicted altitude of their burnup. In the case of the telemetry canister, which must withstand the effects of aerodynamic heating, the increase in heat transfer due to transition will be small in comparison with the total heat transfer since, as was indicated above, it occurs at a much lower velocity. It has therefore been assumed that transition will have a small effect. It is not considered in this analysis.

Heating of the body by radiation from the hot gases in the shock layer was also investigated. It was found that this mode of heat transfer is of no consequence for the velocities associated with ballistic missile re-entries, as is pointed out by Kivel in Ref. C-7.

The method by which the time variation of stagnation point heating rate and the distribution of heating rates were determined will now be described. The laminar heating rate at the stagnation point on a hemispherical nose for flow in thermochemical equilibrium was computed from the empirical equation given in Ref. C-8.

$$q_{\text{stag}} = \frac{17,600}{\sqrt{R}} \sqrt{\frac{\rho}{\rho_o}} \left(\frac{V}{V_c} \right)^{3.15} \left[\frac{h_{sl} - h_{ws}}{h_{sl} - (h_{ws})_{300}} \right] \quad (\text{Btu/ft}^2 - \text{sec})$$

where

- R = nose radius (ft)
- ρ = mass density
- V = flight velocity (fps)
- h = enthalpy per unit mass.

Subscripts

- o = sea level conditions
- c = velocity required for circular orbit at altitude under consideration
- s = stagnation point
- w = body surface

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l = edge of boundary layer

300 = evaluated at 300° K.

This equation is accurate to within $\pm 10\%$ for velocities up to 26,000 fps and altitudes up to 250,000 feet. Figure C-2, which was obtained from Ref. C-8, shows the correlation between test data and predicted data.

The heating equation given above, with the enthalpy term (term in the brackets) equal to one, has been incorporated into the digital re-entry program used in the present study. This modified equation applies to a highly cooled body whose wall temperature never exceeds approximately 300° K. The re-entry program describes the motion of a point mass about a spherical earth with the 1959 ARDC atmosphere. This program computes the variation with time of the stagnation point heat flux on a hemispherical nose of a highly cooled body. For re-entries in which the wall temperatures can exceed 300° K, the computed heat flux must be corrected by adding the enthalpy term in the heating equation.

Since the heating equation as it appears above is applicable only to hemispherical noses, a correction factor must be employed in evaluating the heat flux to the stagnation point of a flat-faced vehicle. Stoney (Ref. C-9) indicates that, at Mach numbers greater than 5, the stagnation point heat flux on a flat-faced vehicle at zero angle of attack is only one-half that on a hemispherical nose at the same flight conditions. With this correction factor and the heating equation above, heat fluxes were computed for the configuration and test data of Ref. C-10. The correlation between predicted and test data shown in Fig. C-3 was considered to be quite good, with the exception of the heating rate at Mach 13.6 where, for unknown reasons, the test data indicated a reduction in heat flux. The empirical equation was also found to show good correlation with the exact solution of Fay and Riddell (Ref. C-4).

Similarly, in calculating the heat flux at the stagnation line on a cylinder normal to the flow, a correction factor of $1/\sqrt{2}$, which is indicated in Refs. C-11 and C-12, was applied to the heating rates calculated in the digital program.

The surface distribution of heating rates, in which the local heating rates around the body are defined in terms of the stagnation heating rate, was determined from test data and from the theory of Lee (Ref. C-11). The distribution of heating rates for either configuration, sphere or cylinder, can be shown by a single curve since Lee's theory indicates that, at Mach numbers greater than 5, there is little change in the ratio of local-to-stagnation heating rate with change in Mach number.

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$$q_{\text{stag}} = \frac{17,600}{\sqrt{R}} \sqrt{\frac{\rho}{\rho_o}} \left(\frac{V}{V_{c.}} \right)^{3.15} \left[\frac{h_{sl} - h_{ws}}{h_{sl} - (h_{ws})_{300^\circ K}} \right] \text{ (Btu/ft}^2\text{-sec)}$$

Fig. C-2. Correlation of Theoretical Heating Rates with Test Data on a Hemisphere

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○ Free flight test data

$$q_{stag} = \frac{8800}{\sqrt{R}} \sqrt{\frac{\rho}{\rho_o}} \left(\frac{V}{V_c} \right)^{3.15} \left[\frac{h_{sl} - h_{ws}}{h_{sl} - (h_{ws})_{300^\circ K}} \right] \text{ (Btu/ft}^2\text{-sec)}$$

Fig. C-3. Correlation of Theoretical Heating Rates with Test Data on a Flat-Faced Body

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Test data for the local heating rates on spheres were obtained from Refs. C-5, C-13 and C-14. Since the theory and test data defined the heating rates over only the first 90 degrees of the body, it was necessary to extrapolate the data over the rear portions of the sphere. The heating distribution curve for a sphere is shown in Fig. C-4. The variation in heat flux over the first 90 degrees of a cylinder was obtained from the data of Ref. C-15 and Lee's theory. These data, in conjunction with data on the variation of heat transfer coefficient and temperature recovery about a cylinder obtained from Refs. C-16 and C-17, were used in determining the heating rate distribution about the cylinder. Figure C-5 indicates the variation in heat transfer coefficient and heat flux about the cylinder.

The stagnation line heating rate on a cylinder is also a function of the yaw angle (Fig. C-6), the heating rate being a maximum when the cylinder is normal to the flow. It has been indicated by several authors that the heating rate at the stagnation line will vary as the cosine of the yaw angle (Λ). However, the correlation of heating data from Refs. C-16, C-18 and C-19 indicates that the variation in stagnation line heating rate is more closely approximated by the $(\cos \Lambda)^{3/2}$.

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Fig. C-5. Heat Flux Distribution on a Cylinder

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Fig. C-6. Variation of Heat Transfer Rate with Yaw Angle for a Cylinder

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APPENDIX D*

HEAT TRANSFER ANALYSIS

A. DIGITAL COMPUTATION PROGRAM

The purpose of this section is to describe briefly the high speed digital program used to solve the transient heat transfer problems associated with the re-entry study. The program is entitled "Transient Heating Digital Analysis for Structural Design," authored by D. V. Sallis and E. R. Boogar of the Martin Company.

The program requires a 32,000-word core storage computer and handles arbitrary elements of structure arranged arbitrarily with temperature-dependent material properties. It can handle conduction in three directions, radiation from the surface to infinity, and the complicated process of melting at the outer surface (ablation). Variable specific heat and material phase changes (solid to liquid) are handled through the use of curves of enthalpy as a function of temperature. The system to be analyzed is divided into elements to form a space network describing the problem, and a heat balance is set up for each element by the application of the relevant terms of a general heat balance equation. (Derivation of this equation is found in Section B.) The temperatures are calculated for finite increments in time; and, to improve the accuracy, the heat balance equations are written as functions of the average temperature gradients over a time interval, instead of the temperature gradient at the beginning of the interval. Thus, simultaneous equations are solved for every interval of time. To help solve these equations, an iteration procedure is used with a technique that accelerates convergence, forces convergence in diverging cases, and handles the nonlinearity introduced in the radiation and ablation problems.

B. GENERAL HEAT BALANCE EQUATION

The purpose of this section is to develop and explain the use of the general heat balance equation used in the IBM-7090 digital program.

When the boundary conditions associated with the classical heat transfer problem are functions of both time and position, the transform

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procedures of operational calculus are usually discarded as possible methods of obtaining solutions, because time-varying quantities and the unknown temperature distribution usually occur as products in the equation. Section E shows the derivation of the solution of the one-dimensional transient in a cylinder for the boundary condition of heat flux versus time. The solution is obtained through the use of Hankel transform methods, and the size and complexity of the resultant solution are at once apparent. In addition, this "exact solution" does not consider the problems of varying thermal properties with temperature, the effect of melting of the core material (change of phase), and the problem of melting of the outer surface (ablation). The exact solution is further complicated if multiple layers of material (as is usually the case) are considered. The method usually employed to find approximate solutions in such cases is that of numerical or finite difference procedures. By these methods, the body is divided up into a number of elements and a step-by-step numerical integration of the heat flow equation is made. Since steps in both distance and time must be taken, the procedure is tedious and time-consuming unless a high-speed automatic computing machine is used.

To illustrate the numerical method, consider a plate having all boundaries insulated, except one surface which is exposed to a known heat flux which is a function of time (Fig. D-1).

Fig. D-1. Schematic of a One-Dimensional Heat Transfer Problem

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Heat is considered to flow only in one dimension, from left to right. The body is subdivided into a series of elements although, for illustrative purposes, only three elements are shown. The heat balance is taken at position x .

Rate of heat stored equals the summation of rates of heat flowing into element.

$$\begin{aligned} \frac{\rho v c_p}{\Delta t} (T_{t+\Delta t} - T_t) = & \left[-\frac{1}{2} k_{x-\frac{\Delta x}{2}} A \frac{\partial T}{\partial x} \right]_{x-\frac{\Delta x}{2}} \Big|_{t+\Delta t} \\ & + \left[\frac{1}{2} k_{x+\frac{\Delta x}{2}} A \frac{\partial T}{\partial x} \right]_{x+\frac{\Delta x}{2}} \Big|_{t+\Delta t} \\ & - \left[\frac{1}{2} k_{x-\frac{\Delta x}{2}} A \frac{\partial T}{\partial x} \right]_{x-\frac{\Delta x}{2}} \Big|_t \\ & + \left[\frac{1}{2} k_{x+\frac{\Delta x}{2}} A \frac{\partial T}{\partial x} \right]_{x+\frac{\Delta x}{2}} \Big|_t \end{aligned} \quad (D-1)$$

where

- v = volume of the element
- A = cross-sectional area of element
- T = temperature
- x = position
- t = time
- Δt = time increment
- Δx = position increment
- k = thermal conductivity

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$\dot{q}(t)$ = known heat flux

ρ = density

c_p = specific heat.

Since

$$V = \Delta x (A),$$

taking the limit as $\Delta x \rightarrow 0$ and $\Delta t \rightarrow 0$ gives

$$\rho c_p \left. \frac{\partial T}{\partial t} \right|_{t+\frac{\Delta t}{2}} = \frac{1}{2} \frac{\partial}{\partial x} \left(k \frac{\partial T}{\partial x} \right) \Big|_{t+\Delta t} + \frac{1}{2} \left. \frac{\partial \left(k \frac{\partial T}{\partial x} \right)}{\partial x} \right|_t$$

or

$$\rho c_p \frac{\partial T}{\partial t} = \frac{\partial \left(k \frac{\partial T}{\partial x} \right)}{\partial x} \quad (D-2)$$

This is the governing differential equation for a system considering variable thermal conductivity. In view of the original equation and the relations

$$\left. \frac{\partial T}{\partial x} \right|_{x-\frac{\Delta x}{2}} \approx \frac{T_0 - T_1}{\Delta x}$$

and

$$\left. \frac{\partial T}{\partial x} \right|_{x+\frac{\Delta x}{2}} \approx \frac{T_2 - T_0}{\Delta x}$$

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Eq (D-1) may be written as follows:

$$\begin{aligned} \frac{\rho v}{\Delta t} J_f - \frac{\rho v}{\Delta t} J_i = \frac{1}{2} \left[\frac{k A (T_1 - T_0)}{\Delta x} - \frac{k A (T_0 - T_2)}{\Delta x} \right]_i \\ + \frac{1}{2} \left[\frac{k A (T_1 - T_0)}{\Delta x} - \frac{k A (T_0 - T_2)}{\Delta x} \right]_f \end{aligned} \quad (D-3)$$

where

$$J = c_p \Delta T = \text{enthalpy}$$

and subscripts

i = initial time

f = final time.

The term on the left represents the rate of heat storage in element "zero" and the terms on the right represent the heat conducted into element "zero" from the surrounding elements. This equation could be applied to all elements (except at the boundary), giving a set of equations which must be solved for each time interval Δt . These equations can now be written to include boundary conditions, such as radiation from the outer surface and aerodynamic heat inputs. The general set of equations that develop are similar to the one shown.

$$\begin{aligned} \frac{\rho v}{\Delta t} J_f = \frac{\rho v}{\Delta t} J_i + \frac{1}{2} \left[q_L - q_R - q_{\text{rad}} + q_{\text{aero}} \right]_i \\ + \frac{1}{2} \left[q_L - q_R - q_{\text{rad}} + q_{\text{aero}} \right]_f \end{aligned} \quad (D-4)$$

where

ρ = density of the element

v = volume of the element

Δt = time increment

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- J_f = final enthalpy of the element
 J_i = initial enthalpy of the element
 q_L = heat conducted from the left element
 q_R = heat conducted to the right element
 q_{rad} = heat radiated from the boundary
 q_{aero} = aerodynamic heat input at the boundary.

In summary, the heat balance equations thus formed are written with the enthalpy at the end of a time interval being a function of the initial enthalpy, the initial temperature, the initial temperature of adjacent elements, the final temperature and the final temperatures of adjacent elements. These equations are written for each element in the thermal system, utilizing the applicable terms. Thus is set up a series of equations which must be solved simultaneously for the final enthalpy of each element. The final temperatures can then be determined.

C. HEAT TRANSFER MODEL

The purpose of this section is to explain how the heat transfer model was formed in order to predict the release or burnup time of the flare capsules upon re-entry into the atmosphere. Consider the sketch of a flare capsule, Fig. D-2.

Fig. D-2. Schematic Sketch of a Flare Capsule

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The lead wall, plastic liner and liquid metal sections are divided into annular elements. A complete description of each element of Fig. D-2 (including volume and conducting distances) is read into the heat transfer program (described in Section A), and the program applies the heat balance equation (described in Section B) to each element. The set of simultaneous equations developed is solved by an iterative process which accelerates convergence, minimizing the time required for a solution. The program continues to calculate the time-temperature histories of each element until the required number of elements is ablated to expose the flare material (liquid metal). To understand the handling of the ablation process, consider the characteristic curve of a material that undergoes a change of phase, Fig. D-3.

Fig. D-3. Enthalpy History of a Material

This curve of enthalpy versus temperature is read in for each material of the flare capsule. As aerodynamic heat inputs are introduced at the top boundary, the time-temperature history of each element is calculated by the program. Each element continues to store heat until it has received enough heat to raise it to its melt temperature; it then receives further heat to reach its heat of fusion. At this point, the element is considered ablated (removed) and the boundary inputs are shifted down to the next element. This process continues until all of the required elements have melted off. The process of ablation or melting and removal of the surface material is assumed to have negligible effect on heat transfer to the body.

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D. DETERMINATION OF AVERAGE HEAT FLUX

This section describes how the average heat flux into the flare capsules is determined from the heat flux at the stagnation line. Figure C-5 reveals the heat flux distribution from the stagnation line to a point 180 degrees from the stagnation line. If the cylinder were considered to rotate about its axis, then every point on the surface of the cylinder would have a heat flux that cycled accordingly. The average heat flux at any point would be the integrated average of the curve. Therefore, the average heat flux on a cylinder spinning about its long axis, which is perpendicular to the flow of air, is given by

$$q_{\text{average}} = 0.321 [q_{\text{stagnation line}}]$$

However, in addition to spinning about its axis, the cylinder may also be tumbling end-over-end. In this case, the intensity of the heat flux at the stagnation line is varying according to the angle of yaw. Figure C-6 shows the distribution of the stagnation line heat flux versus angle of yaw. The examples below will serve to illustrate this phenomenon:

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In Case I, at zero angle of yaw, the stagnation line heat flux is equal to 1.0, as would be expected. However, as the yaw angle increases, the intensity at the stagnation line decreases. At 45 degrees (Case II), it equals 0.6 of the value at zero degrees. It continues to decrease until, at 90 degrees (Case III), it is much less than 0.1. (At 90 degrees, this is really only an effective stagnation line value since, at this point, there exists no real stagnation line but, rather, a uniform flux distribution which can be transferred back to a stagnation line value by considering the spinning correction.) Since the stagnation line heat flux can cycle according to Fig. C-6, the average of this cycle, 0.584, is used. Superimposing this varying stagnation line heat flux intensity with the spin factor correction gives the average heat flux into the cycle:

$$q_{\text{average}} = (0.321)(0.584) q_{\text{stagnation line}}$$

$$q_{\text{average}} = 0.1875 q_{\text{stagnation line}}$$

Since much of the work in developing expressions for the heat flux on re-entry bodies has been formulated for heat flux at the stagnation point of a sphere, it would be wise at this point to express the average heat flux on the sides of a cylinder in terms of the heat flux at the stagnation point of a sphere with an equivalent radius. Lee (Ref. C-11) has shown that the heat flux at the stagnation line of a cylinder is equal to 0.707 of the heat flux at the stagnation point of a sphere. Therefore, the necessary relationship is

$$q_{\text{average}} = 0.132 q_{\text{stagnation point of sphere}}$$

for spinning, tumbling cylinder

E. DERIVATION OF ONE-DIMENSIONAL TRANSIENT* ANALYTICAL SOLUTION

In this section, the analytical solution of the transient one-dimensional heat flow problem will be developed. Consider a right-circular cylinder with a known heat flux on the side and the assumptions:

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- (1) No heat losses from ends (one-dimensional).
- (2) Heat flow considered positive when heat is introduced into the cylinder.
- (3) The cylinder has a radius R and a length L.

The governing differential equation (in polar coordinate form) that describes this system is

$$\frac{\partial^2 T(r, t)}{\partial r^2} + \frac{1}{r} \frac{\partial T(r, t)}{\partial r} = \frac{1}{\alpha} \frac{\partial T(r, t)}{\partial t} \quad (D-5)$$

The boundary conditions can be written:

$$T(0, t) \text{ is finite} \quad (D-6)$$

$$T(r, 0) \text{ is known} \quad (D-7)$$

$$\frac{\partial T(R, t)}{\partial r} = \frac{Q(t)}{2\pi kRL} \quad (D-8)$$

where

T = temperature

r = position radius

t = time

k = thermal conductivity

R = outside radius of cylinder

L = length of cylinder

Q(t) = rate of heat flow into the body (Btu/hr)

α = thermal diffusivity.

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The Hankel transforms of Eqs (D-5) and (D-7) are

$$\frac{R J_0(\xi_i R) Q(t)}{2\pi k R L} - \xi_i^2 \bar{T}(t) = \frac{1}{\alpha} \frac{d\bar{T}(t)}{dt} \quad (D-9)$$

$$\bar{T}(0) = \int_0^R r' T(r', 0) J_0(\xi_i r') dr' \quad (D-10)$$

where the ξ_i terms are the roots of the equation

$$J_1(\xi_i R) = 0$$

Equation (D-9) is a linear differential equation in time with constant coefficients. It can be easily solved by the Laplace transform methods. The Laplace transform of Eq (D-9) is

$$\frac{\alpha J_0(\xi_i R) Q(S)}{2\pi k L} - \alpha \xi_i^2 \bar{T}(S) = S \bar{T}(S) - \bar{T}(0) \quad (D-11)$$

or

$$\bar{T}(S) = \frac{\bar{T}(0) + \frac{\alpha J_0(\xi_i R)}{2\pi k L} Q(S)}{S + \alpha \xi_i^2} \quad (D-12)$$

The inverse Laplace of Eq (D-11) is

$$\bar{T}(t) = e^{-\alpha \xi_i^2 t} \bar{T}(0) + \frac{\alpha J_0(\xi_i R)}{2\pi k L} \int_0^t Q(t') e^{-\alpha \xi_i^2 (t-t')} dt' \quad (D-13)$$

or

$$\bar{T}(t) = e^{-\alpha \xi_i^2 t} \left[\bar{T}(0) + \frac{\alpha J_0(\xi_i R)}{2\pi k L} \int_0^t Q(t') e^{+\alpha \xi_i^2 t'} dt' \right] \quad (D-14)$$

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Taking the inverse Hankel transform, using Eq (D-10), gives the result

$$T(r, t) = \frac{2}{R^2} \sum_i \left\{ \frac{J_0(\xi_i r)}{J_0^2(\xi_i R)} e^{-\alpha \xi_i^2 t} \left[\int_0^R r' T(r', 0) J_0(\xi_i r') dr' + \frac{\alpha J_0(\xi_i R)}{2\pi k L} \int_0^t Q(t') e^{\alpha \xi_i^2 t'} dt' \right] \right\} \quad (D-15)$$

where the ξ_i terms are the roots of the equation

$$J_1(\xi_i R) = 0$$

F. COMPARISON OF ANALYTICAL AND NUMERICAL METHODS

In this section, the results obtained by the analytical solution (Section E) and the solution by the transient heat code (Sections A through D) are compared. The results are based on constant material properties, one-dimensional heat flow, and the absence of radiation at the boundary. A point 3/8 inch from the surface of a 3-inch diameter cylinder was investigated. The results obtained by solving the expressions derived in the previous section are compared with the numerical solution in Table D-1. The temperatures were obtained for different times (0 through 30 seconds). In addition, Column III shows the temperatures obtained if variable properties and radiation at the boundaries (emissivity = 0.8) are included.

It is interesting to note that the average variation is about 2%. This is well within the range of accuracy of the material properties. The probable reason for the greater variation in the constant-property temperatures at the larger time increments is that, for any given accuracy, more terms in the series than were used are required. The variable-property temperatures calculated by the computer are

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somewhat lower than the constant-property temperatures, because radiation at the boundary is included. This has a more significant effect at the higher temperatures.

TABLE D-1
Comparison of Analytical and Digital Computer Results

Time (sec)	<u>Column I</u>	<u>Column II</u> Temperature (° F)	<u>Column III</u>
	Analytical Solution, Constant Material Properties	Computer Solution, Constant Material Properties	Computer Solution, Variable Material Properties
0	100.0	100.0	100.0
5	100.6	101.9	101.8
10	105.8	106.4	104.6
15	115.8	116.9	114.7
20	139.2	141.2	138.0
25	193.7	198.0	190.2
30	318.7	328.7	313.0

G. MATERIAL PROPERTIES

This section outlines the physical properties of the materials used in this program. The values were taken from references listed in the bibliography. In those cases where the values of thermal conductivity were unavailable because they extended beyond the usual temperature range, the following relation was used, according to Ref. D-2, to calculate thermal conductivity:

$$K = 2.61 \times 10^{-8} \left(\frac{T}{\rho}\right) - \frac{2 \times 10^{-17} \left(\frac{T}{\rho}\right)^2}{c_p d} + \frac{97 c_p (d)^2}{MT}$$

DECLASSIFIED ~~SECRET~~1. Physical Properties of LeadGeneral (Ref. D-3)

Density	0.41 lb/in. ³
Heat of fusion	11.3 Btu/lb
Melting point	621° F

Specific Heat (Ref. D-4)

<u>Temperature (° F)</u>	<u>c_p (Btu/lb-° F)</u>
32	0.0306
212	0.0315
392	0.0325
572	0.0335
752	0.0328
932	0.0328
1112	0.0328
1292	0.0328
1472	0.0328

Thermal Conductivity (Ref. D-5)

<u>Temperature (° F)</u>	<u>K (Btu/hr-ft-° F)</u>
-58	22
32	21
392	19

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2. Physical Properties of Potassium (Ref. D-8)

General

Density	0.031 lb/in. ³ *
Heat of fusion	26.4 Btu/lb
Melting point	147° F
*Ref. D-7.	

Specific Heat

Temperature (° F)	c_p (Btu/lb-° F)
-10	0.17*
80	0.177*
145	0.181*
167	0.1956
392	0.1887
752	0.1826
1112	0.1825
1472	0.1884

*Ref. D-9.

Thermal Conductivity

Temperature (° F)	K (Btu/hr-ft-° F)
-99	55.0*
80.6	50.6*
143.6	47.2*
300	26.2
500	25.0
700	23.5
900	22.1
1100	20.6
1300	19.2

*Calculated.

Electrical Resistivity

Temperature (° F)	ρ (microhms)
-103	4.0
32	6.1
131	8.4

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3. Physical Properties of Rubidium

General (Ref. D-8)

Density	0.0553 lb/ in. ³ *
Heat of fusion	11.0 Btu/ lb
Melting point	102° F
*Ref. D-7.	

Specific Heat (Ref. D-9)

<u>Temperature</u> (° F)	<u>c_p</u> (Btu/ lb -° F)
-99	0.069
-10	0.0765
80	0.0842
102	
to	0.0913*
257	

*Ref. D-8.

Thermal Conductivity (calculated)

<u>Temperature</u> (° F)	<u>K (Btu/ hr -ft -° F)</u>
-99	30.9
9.4	30.2
80.6	30.0
122	19.5
167	19.1
212	18.9

Electrical Resistivity (Ref. D-7)

<u>Temperature</u> (° F)	<u>ρ (microhms)</u>
-311	2.5
32	11.6
94	13.4
122	23.15*
167	25.32*
212	27.47*

*Ref. D-8.

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4. Physical Properties of Cesium

General (Ref. D-8)

Density	0.0687 lb/in. ³ *
Heat of fusion	6.76 Btu/lb
Melting point	83° F

*Ref. D-7.

Specific Heat (Ref. D-9)

Temperature (° F)	c_p (Btu/lb-° F)
-100	0.0421
-10	0.0490
80	0.0558
84	0.06*

*Ref. D-8.

Thermal Conductivity (calculated)

<u>Temperature (° F)</u>	<u>K (Btu/ hr -ft -° F)</u>
-99	20
9.4	19.8
80.6	18.4
86	11.75
98.6	11.90

Electrical Resistivity (Ref. D-7)

Temperature (° F)	ρ (microhms)
-305	5.25
32	19.0
80	22.2
85	36.6*
97	37.0*

*Ref. D-7.

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5. Physical Properties of Sodium (Ref. D-8)

General

Density	0.035 lb/in. ³ *
Heat of fusion	49.5 Btu/lb
Melting point	208° F
*Ref. D-7.	

Specific Heat

Temperature (° F)	c_p (Btu/lb-° F)
80	0.289*
207.9	0.305
212	0.3305
392	0.3200
752	0.3055
1112	0.2998

*Ref. D-9.

Thermal Conductivity

Temperature (° F)	K (Btu/hr-ft-° F)
-99	84.5*
80.6	78.4*
206.6	75.0*
300	48.0
400	46.3
500	44.7
600	43.2
700	41.7
1000	37.9
1100	36.7
1300	34.5

*Calculated.

Electrical Resistivity

Temperature (° F)	ρ (microhms)
-103	2.8*
32	4.3*
131	5.4*

*Ref. D-7.

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where

K = thermal conductivity (watts/cm $^{\circ}$ C)

T = absolute temperature ($^{\circ}$ K)

ρ = electrical resistivity (ohm-cm)

c_p = specific heat (cal/gm $^{\circ}$ C)

d = density (gm/cm³)

M = molecular weight, or average molecular weight for alloys.

The report from which this empirical relation was taken claims that, for any metal or alloy (except a semiconductor), whether liquid or solid, the thermal conductivity can normally be estimated within 5 or 10%. The many thermal conductivity correlations which were presented in the report substantiate the claim.

References

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- D-6. Crespi, C., "Material and Design Engineering," January 1958.
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APPENDIX E

PHYSICS OF RE-ENTRY BODIES*

This appendix describes the techniques used in estimating the observability of the re-entering bodies and flares for this program and recommends instruments and equipment for observing and recording these events. The mechanisms of light production by a re-entering body are complex and are not well known quantitatively. However, the values predicted should be sufficiently accurate to permit selection of recording instruments.

A. FLARE MATERIALS AND LIGHT PRODUCTION

The flare materials consist of four pure alkali metals: cesium, sodium, potassium and rubidium. These materials were chosen because they can be easily excited to produce light. Since the spectrum of a pure element is unique, spectroscopic methods can be used for analysis. This is discussed in Section B. During re-entry of a body, the aerodynamic heat melts and vaporizes the solid materials. This vapor is slowed down by individual atomic collisions with air molecules and is dispersed along the trail. A small fraction of these collisions are inelastic and result in ionization and excitation of one or both of the colliding atoms. Tables E-1 and E-2 show the energy required to produce these effects. The amount of energy that can be exchanged by an atom in a single collision is limited by the relative kinetic energy of the pair (Table E-2). The radiation produced by this process consists primarily of line spectra characteristic of the re-entry atoms and does not have a continuous or thermal distribution. This form of energy, called neutral impact radiation, is the common type of radiation observed from natural meteors. Most of the atomic collisions are elastic and produce no radiation. The energy distribution among the vapor trail atoms and in the region behind a shock front approaches equipartition as the process of slowing down continues. The radiations from such regions tend to have a thermal distribution.

In this experiment, the first effect will be the ablation of the lead capsule. Due to the low melting temperature of lead, the surface temperature of the capsules will not be of the magnitude to produce appreciable thermal radiation. Light will be produced from the incandescent region behind the shock (primarily nitrogen band spectra) and from neutral collisions in the wake train. The lead vapor will form a dense wake and will react with oxygen as the trail expands. The radiation produced will be rich in lead spectra.

*C. Reed (Portions of the data reported here were provided by Geophysics Corporation of America, under subcontract.)

TABLE E-1
Significant Properties of the Re-entry Materials

<u>Material</u>	<u>Density (gm/ cm³)</u>	<u>Melting Point (° C)</u>	<u>Heat of Fusion (ergs/ gm x 10⁷)</u>	<u>Heat of Vaporization (ergs/ gm x 10⁷)</u>	<u>Surface* Tension Liquid (dynes/ cm²)</u>
Lead	11.35	327.4	34.7	858	438 at 400° C
Sodium	0.971	97.5	113	4,220	200
Potassium	0.87	62.3	61.2	2,077	86
Oxygen					
Nitrogen					
Rubidium	1.53	38.5	25.50	888	
Cesium	1.89	28.5	15.75	611	
Lithium	0.534	186	662	20,370	

*"Liquid Metals Handbook," NAVEXOS P-733 (rev.), June 1952.

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TABLE E-1 (continued)

<u>Material</u>	<u>Atomic Weight</u>	<u>First Ionization Potential (electron volts)</u>	<u>Principal Spectrum Lines</u>	
			<u>(Angstroms)</u>	<u>(electron volts)</u>
Lead	207.21	7.38	4057.8 9674	3.06 1.29
Sodium	22.99	5.12	5890 5895 8183 8194	2.11 2.105 1.52
Potassium	39	4.32	4044 4047 7665 7699	3.07 3.07 1.62 1.615
Oxygen	16.00			
Nitrogen	14.008			
Rubidium	85.44	4.16	4202 4215.6 7800.2 7900.5	2.96 2.95 1.595 1.57
Cesium	123.91	3.87	4555 4593 8521 8943	2.73 2.70 1.46 1.39
Lithium*	6.94	5.36	6103 6707	2.02 1.85

*Not used because of its relatively high heat of vaporization and ionization potential.

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TABLE E-2
Available Ionization Energies

<u>Material</u>	<u>Velocity (cm/sec x 10⁵)</u>	<u>Kinetic Energy of Flare Atom, E_o (electron volts)</u>	<u>Relative Energy of Impact with Nitrogen, E_r</u>		<u>First Ionization Potential</u>
			<u>N₂</u>	<u>N</u>	
Sodium	6	4.29	2.36	1.62	5.12
Potassium	6	7.28	3.05	1.93	4.32
Rubidium	6	15.92	3.93	2.24	4.16
Cesium	6	24.8	4.3	2.36	3.87
Lithium	6	1.29	1.03	--	5.36
Lead	6	38.6	4.6	2.44	7.38
Oxygen	--	--	--	--	13.55
Nitrogen	--	--	--	--	14.48

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The alkali metals have lower melting points than lead (see Table E-1) and will be liquid at the time of release. The exact circumstances of release are rather unpredictable, depending on the uniformity of capsule ablation. Generally, uniform ablation is expected. The alkali metals, upon their release from the capsules, will be broken up into droplets and vaporized. Vaporization will proceed rapidly and a dense vapor trail will form. The subsequent slowing down and thermalization of the vapor atoms through collision processes is expected to produce a brilliant light. Chemical energy from the formation of alkali oxides will appear during expansion and dilution of the trail and will contribute to the production of light. The spectral nature of this light is expected to consist primarily of the bright line spectra of the alkali metals with a background of band spectra and thermal radiation.

In the following estimate, three areas of light production are considered:

- (1) The region behind the shock of a representative capsule.
- (2) The vapor train of the capsule.
- (3) The vapor train of the flare materials.

For the first case, the results of a known re-entry experiment are extrapolated to fit the case of the capsule. In the second and third cases, a method developed by Öpik (Ref. E-1) for estimating the light intensity of meteors is applied.

1. Thermal Radiation from Behind the Shock

This source of light is relatively unimportant for the purposes of the experiment, and an adequate estimate can be obtained by comparison with a known case. A summary of Project Trailblazer, a NASA experiment, is given:

(1) Re-entry body	5-inch hollow metal sphere
(2) Re-entry velocity	Approximately 7×10^3
m/sec
(3) Altitude observed	140,000 feet
(4) Estimated visual magnitude	-1 to -2

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The Task 7.3 re-entry bodies are somewhat smaller than the 5-inch sphere and should appear at a higher altitude. The re-entry velocity is in the neighborhood of 6×10^3 m/sec. A rough scaling-down of the magnitude to fit these conditions yields a visual magnitude near zero.

2. Radiation from the Vapor Trail

The method of Öpik (Ref. E-1), as presented by Whipple (Ref. E-3), calculates luminosity through use of a luminous efficiency coefficient, τ . The coefficient is the fraction of the total energy dissipated by the vapor that appears in the form of visible radiation as it is slowed down by the collision process. The value of τ depends on the initial velocity of the vapor and is given by $\tau = \tau_0 V_0$, where V_0 is the initial velocity and τ_0 is a constant for a particular case. Unfortunately, τ_0 depends partly on the material composition of the vapor and, to some extent, on the density of the vapor stream. The τ_0 values for the alkali metals and Pb are unknown but should lie near the values observed for aluminum and the brighter visual meteors. The following analysis makes use of the τ_0 value for aluminum recently observed by R. D. McCrosky (Ref. E-2). This observed value includes whatever light is produced by the burning of aluminum vapor at the altitude of 80 km.

The relations used are:

The luminosity equation (Ref. E-3):

$$I = \frac{\tau}{2} \frac{dM}{dT} V^2 \quad (E-1)$$

where

I = luminosity (ergs/sec)

τ = luminous efficiency = $\tau_0 V$

V = velocity (cm/sec)

$\frac{dM}{dT}$ = rate of ablation (gm/sec).

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The ablation equation (Ref. E-3):

$$\frac{dM}{dT} = \frac{\gamma \rho_a \sigma V^3}{2h} \quad (E-2)$$

where

- γ = fraction of dissipated kinetic energy transferred into the body as heat
- h = heat required to dissipate (ablate) 1 gram of the material (ergs/gm)
- σ = instantaneous cross section of the body (cm²)
- ρ_a = atmospheric density (gm/cm³).

The equation of visual magnitude of a source at a distance of 100 km:

$$M_V = 24.6 - 2.5 \log_{10} I \quad (E-3)$$

where

- M_V = visual magnitude
- I = luminosity (ergs/sec) (visual).

a. Luminosity of the lead capsule vapor trail

Assumed values of the parameters are:

- ρ_a = 0.317×10^{-6} gm/cm³ (altitude, 200,000 feet)
- σ = 7.9 cm² (This value was obtained for a 2-inch diameter sphere of 3/8-inch wall thickness, where equal increments in radius are ablated in equal times. The average radius during ablation is 1.59 cm (cross-sectional area of 7.9 cm²).)

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$$\gamma = 0.16 \text{ (evaluated according to Öpik (Ref. E-1))}$$

$$h = 34.7 \times 10^7 \text{ ergs/gm.}$$

Substituting these values into Eq (E-2) yields

$$\text{Average } \frac{dM}{dT} = 119 \text{ gm/sec.}$$

The luminosity is calculated from Eq (E-1), using the following parameter values:

$$V = 6 \times 10^5 \text{ cm/sec}$$

$$\tau_{0_{Al}} = 9 \times 10^{-10} \text{ sec/cm, value for aluminum obtained from Ref. E-2}$$

$$\tau_{0_L} = \text{lower limit for natural meteors} = 1 \times 10^{-11} \text{ sec/cm, obtained from Ref. E-2}$$

$$\frac{dM}{dT} = 119 \text{ gm/sec.}$$

Substitution in Eq (E-1) yields

$$I = 1.28 \times 10^{10} \text{ ergs/sec, using } \tau_{0_{Al}}$$

$$I = 1.28 \times 10^8 \text{ ergs/sec, using } \tau_{0_L}.$$

The corresponding values for visual magnitude, calculated from Eq (E-3), are

$$M_V = -0.8, \text{ for } \tau_{0_{Al}}$$

$$M_V = +4.2, \text{ for } \tau_{0_L}.$$

The +4.2 value is considered a grossly conservative lower limit. $M_V = -0.8$ is a more likely value, and probably conservative. The burning lead vapor at 200,000 feet (approximately 60 km) may well

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be a more efficient light source than the burning aluminum at 80 km. Hence, higher values of luminosity could easily occur.

b. Luminosity of flare burst vapor

Tables E-1 and E-2 show that sodium has both a lower relative energy and higher excitation and ionization voltages than the other alkali metals. Therefore, it will be the least easily excited of the flare materials. On this basis, the following calculations, based on sodium, should yield conservative results for the other materials.

It is assumed that the molten sodium breaks up into droplets before appreciable vaporization occurs. According to Öpik (Ref. E-1), molten sodium under the re-entry conditions for this program would break up into droplets of about 200 to 100 microns in diameter. The initial ablation rates for sodium droplets of 200-, 100-, and 50-micron diameters are computed from Eq (E-2), using the following average conditions:

$$\gamma = 0.6$$

$$h = 4.22 \times 10^{10} \text{ ergs/gm}$$

$$V = 6 \times 10^{10} \text{ cm/sec}$$

$$\rho_a = 0.3 \times 10^{-6} \text{ gm/cm}^3$$

$$\sigma = \pi \times 10^{-4}; \frac{\pi}{4} \times 10^{-4}; \frac{\pi}{16} \times 10^{-4} \text{ cm}^2.$$

The calculated ablation rates and the luminosities expected for this program are shown in Table E-3.

TABLE E-3
Ablation Rates and Luminosities for Masses
of Droplets in Free Molecular Flow

Droplet Size (microns)	Ablation Rate (gm/sec x 10 ⁻⁵)	Total Ablation Rate for 15-gm Na (gm/sec)	Light Intensity for $\tau_0 = 1 \times 10^{-11} \text{ sec/cm}$		Light Intensity for $\tau_0 = 9 \times 10^{-10} \text{ sec/cm}$		Average Duration (sec)
			Luminosity (ergs/sec x 10 ⁹)	Visual Magnitude	Luminosity (ergs/sec x 10 ¹¹)	Visual Magnitude	
200	14.46	515	0.556	+2.75	0.5	-2.2	0.0291
100	3.61	1030	1.11	+2.0	1	-2.9	0.0145
50	0.9	2060	2.22	+1.2	2	-3.6	0.0073

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Before definite conclusions can be drawn, it is necessary to give careful consideration to the analytical assumptions. The luminosities are given in two columns in Table E-3. In one case, a τ_0 value of 1×10^{-11} sec/cm was used. This value applies to the lower limit of τ_0 for natural meteoric material (iron and stone), and the luminosities obtained should be grossly conservative when applied to alkali metals. Hence, the low-magnitude value, +2.75, rounded off to +3, is here called the grossly conservative lower limit. The value of τ_0 which would apply to alkali metals is unknown, but should be at least as large as the value $\tau_0 = 9 \times 10^{-10}$ sec/cm observed for aluminum (Ref. E-2). Therefore, the luminosity and magnitude values given in the $\tau_0 = 9 \times 10^{-10}$ column of Table E-3 should more closely approximate the values to be expected. Table E-3 shows that the ablation rate and luminosity increase with decreasing droplet size. These increases, though, are accompanied by a decrease in duration. A short duration time is desirable since it tends to concentrate the light over a shorter length of trail and more closely approximates a stationary point source. Vapors produced by droplets in the head of a dense stream tend to shield those behind. The finite rate of effluence of the material from the capsule is another type of retarding factor. However, these factors are most effective only at the beginning of the process. A relatively slow, continuous release tends to reduce the vapor shielding effect and increase the speed of breakup and vaporization. It is doubtful whether a sporadic release will occur in the final stages of capsule ablation. Ablation is expected to proceed fairly uniformly over the capsule surface. The thin layer remaining at the time of penetration will be practically molten and of insufficient strength to seriously impede the release process.

It is concluded that a visual magnitude of -2 ± 2 , having a duration time of 0.02 to 0.04 second (during vaporization), is a reasonable estimate for the flare bursts. The burning of the alkali droplets at the assumed altitude of 60 km should be a more efficient light source than the burning of aluminum at 80 km (due to the higher concentration of oxygen at 60 km). Hence, the higher values for τ_0 (total) and the luminosities of the alkali flares may reasonably be expected.

Table E-4 summarizes the light intensities expected for both the capsules and the flares. Values for irradiance at 60- and 100-mile distances are given.

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3. Particulate Matter Flare Material

At the start of the program, it was contemplated that daylight re-entry might be expected. Since the brightness of the sky is sufficient to obscure the light produced by re-entry, a dye-type material, rather than the flare-type, was selected to mark penetration. It was determined that finely divided particulate matter or cellular material can be used for observation. When released at high altitudes, these materials tend to float near the point of release, giving ample time for observation and photographic recording of the reflected sunlight. Powders of various colors can be used to mark the termination of ablation of each type capsule under consideration.

TABLE E-4
Summary of Light Intensities

Source	Visual Magnitude at 100 km	Duration	Intensity at Source (watts/steradian)	Irradiance at Distances from Source (watts/cm ²)	
				60 mi	100 mi
Lead capsule	0 ± 2 with +4 as a grossly conservative lower limit	About 3 seconds, prior to rupture	7.96 to 340	8.66 x 10 ⁻¹⁴	3.12 x 10 ⁻¹⁴
				to 3.7 x 10 ⁻¹²	to 1.33 x 10 ⁻¹²
All flares	-2 ± 2 with +3 as a grossly conservative lower limit	0.02 to 0.04 second during vaporization; then decay to extinction within 1 or 2 seconds	55 to 2200	5.98 x 10 ⁻¹³	2.15 x 10 ⁻¹³
				to 2.39 x 10 ⁻¹¹	to 8.60 x 10 ⁻¹²

The following materials were investigated:

<u>Material</u>	<u>Color</u>
Aluminum oxide	White
Iron oxide	Red
Chromium oxide	Green
Tungsten oxide	Yellow

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<u>Material</u>	<u>Color</u>	
Cobalt oxide	Blue	
Pemco stain 941	Red	} Industrial ceramic pigments produced by The Pemco Corporation
Pemco stain 931	Yellow	
Pemco stain 205	Green	

The daylight visibility of these materials depends on their spectral reflectivities, particle sizes, concentration in the atmosphere, and orientation with respect to the position of the sun. Density and particle size are important factors in determining the rate of thermal diffusion after release from the capsule. Since the optimum time of observation is when the center of the cloud is optically thin, it is possible that the altitude of release may be difficult to define. The time for expansion of the particulate, when released from the capsule, may be several minutes. During this period, the cloud may have descended to other altitudes along the trajectory. The distance the cloud has moved can be calculated from the following equation:

$$S = \frac{3.08 \rho_p r_p}{k \rho_a} \log_{10} \frac{V_0}{V}$$

where

S = distance traveled

ρ_p = density of the particle

r_p = mean radius of particle

ρ_a = density of the atmosphere

k = drag coefficient--for small, high-altitude particles,
k = 1

V_0 = initial velocity

V = velocity at time t after release

$$V = \frac{4 \rho_p r_p V_0}{k(3 \rho_a V_0 t) + 4 \rho_p r_p}$$

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$$t = \text{time after release} = \frac{4 \rho_p r_p}{3 k \rho_a V_0} \left(\frac{V_0}{V} - 1 \right).$$

Table E-5 shows results for various particle and atmospheric parameters. Presumably, any of the materials listed could produce a cloud detectable from ground level if released in sufficient quantity. The limiting considerations are those of allowable size and weight for flare materials plus capsules, and practical specifications for photographic equipment, etc. Given these conditions, an analysis (Ref. E-4) has determined only two of the materials listed to be suitable for this program. Their characteristics follow.

<u>Material</u>	<u>Color</u>	<u>Average Particle Size (microns)</u>	<u>Theoretical Density (gm/cm³)</u>	<u>Minimum Required for Daylight Viewing (gm)</u>
Al ₂ O ₃	White	0.5 to 1	3.9	200
Pemco red stain GS 941	Red	1 to 2	4.25	600

a. Quantities required

Since the desired particle sizes may not be available, it may be necessary to determine amounts required in other sizes. The visibility of the particulate flares is in direct proportion to the amount of surface area the cloud can present to sunlight. For two equal weights (A and B) of the same material, total particulate surface areas are in inverse proportion to particle diameters. If each cloud is observed at the point where it becomes optically thin, the respective visibilities can be expressed by the same ratio as their total surfaces. This means that two such clouds of 100 grams of 1 micron Al₂O₃ and 200 grams of 2 micron Al₂O₃, respectively, should be equally visible. There is a lower limit of useful particle sizes. The wavelengths of visible light lie in the 0.4- to 0.7-micron range. Particles below this size range tend to reflect all colors equally well; therefore, the true color of the material has little meaning. In the case of Al₂O₃, the reflection is white, in any case, and small micron sizes of 0.5 to 0.3 micron can be used to advantage.

Altitude (ft x 10 ⁵)	Atmospheric Density (gm/ cm ³)	Stopping Distance, S (cm)							
		$\rho = 6$ D = 0.3 x 10 ⁻⁴		$\rho = 6$ D = 0.6 x 10 ⁻⁴		$\rho = 6$ D = 2.0 x 10 ⁻⁴		$\rho = 6$ D = 3.33 x 10 ⁻⁴	
		$\rho = 2$ D = 1 x 10 ⁻⁴		$\rho = 2$ D = 2 x 10 ⁻⁴		$\rho = 2$ D = 2.6 x 10 ⁻⁴		$\rho = 2$ D = 10 x 10 ⁻⁴	
		$\rho = 4$ D = 0.5 x 10 ⁻⁴		$\rho = 4$ D = 1 x 10 ⁻⁴		$\rho = 4$ D = 3 x 10 ⁻⁴		$\rho = 4$ D = 5 x 10 ⁻⁴	
		$\frac{V_0}{V} = 2$	$\frac{V_0}{V} = 10$	$\frac{V_0}{V} = 2$	$\frac{V_0}{V} = 10$	$\frac{V_0}{V} = 2$	$\frac{V_0}{V} = 10$	$\frac{V_0}{V} = 2$	$\frac{V_0}{V} = 10$
1.0	1.78 x 10 ⁻⁵	5.18	17.3	10.36	34.6	31.08	103.8	51.80	173.00
1.5	1.77 x 10 ⁻⁶	52.2	174.0	104.4	348.0	313.2	1044.0	522.00	1740.00 ;
2.0	3.26 x 10 ⁻⁷	283	945.0	566.0	1890.0	1698.0	6670.0	2830.00	9450.00
2.5	4.26 x 10 ⁻⁸	2165	7220.0	4330.0	14,490.0	12,990.0	43,320.0	21,650.00	72,200.00
3.0	3.21 x 10 ⁻⁹	28,800	96,000.0	57,600.0	192,000.0	172,800.0	576,000.0	288,000.00	960,000.00

ρ = particle density (gm/cm³)

D = particle diameter (cm)

$$V_0 = \text{initial velocity (cm/sec)}$$

V = velocity at some time after release (cm/sec)

These tables are included in reference to a test plan where day flares were required. These flares are not required for the present test plan, in which all re-entries occur during the night.

TABLE E-6

Capsule--cylinder; L = 6 in.; D = 3 in.; volume = 20.6 in.³ = 338 cm³

Average Particle Size (microns)	Required Amount (gm)	Capsule Packing Density	
		(gm/ cm ³)	(% of maximum)
4	1000	2.96	76
3	750	2.22	57
2	500	1.47	37.7
1	250	0.74	19
0.9	225	0.665	17.1
0.7	175	0.518	13.3
0.5	125	0.37	9.5
0.3	75	0.222	5.7

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TABLE E-7

Material--Pemco GS-941; density = 4.25 gm/cm^3 , maximum

Capsule--cylinder; L = 5.75 in.; D = 3.75 in.; volume = 35.4 in.^3
 = 580 cm^3

Average Particle Size (microns)	Required Amount (gm)	Capsule Packing Density	
		(gm/cm^3)	(% of maximum)
4	1600	2.76	65
3	1200	2.07	48.7
2	800	1.38	32.4
1.5	600	1.035	24.4
1	400	0.69	16.2
0.9	360	0.62	14.7
0.7	280	0.483	11.4
0.5	200	0.345	8.1

4. Summary of Flare Materials

The particulate flares used for daytime experiments have several disadvantages when compared with the alkali metals required for nighttime experiments:

- (1) It is estimated that 10 grams of pure alkali metal can be photographed during the night while approximately 200 grams of particulate matter is required for daytime observation. Aside from the flare weight increase, the burnup specimens must be larger to contain the particulate material. Thus, the reduction of samples will affect the number of points available for a statistical average of altitude of burnup versus the type of sample.
- (2) The effect of the sun on the observability of the particulate matter is much more pronounced than the effect of the moon on the observability of the alkali. In fact, illumination of the particulate matter would be completely masked if it occurred within 15 degrees of the sun. The amounts of

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daylight materials presented herein as required for the test were determined for twilight observation. Other test conditions would require more material.

These disadvantages ultimately led to a request for nighttime missile launchings. When these were scheduled, the study of particulate matter was terminated and all re-entry flare capsules were designed to contain only alkali metal.

B. INSTRUMENT SPECIFICATIONS

The instruments capable of recording visual events may be broadly classed as photographic and photometric. The camera is of greatest value in recording spatial relations and can also be used to record time of events. The main purpose of the photometer is to provide the precise time of occurrence of an event.

In this test, the desirable characteristics of cameras designed to record spatial relations are as follows:

- (1) Sufficient sensitivity to record the flare events and star background.
- (2) Wide field of view to ensure capture of flare events without need for accurate pointing.
- (3) Large-size film negative to reduce danger of overlapping images and to facilitate data reduction.

The primary camera selected for this is the K-24, a war surplus aerial photography camera having an $f/2.5$ aperture and 7-inch focal length lens. It has an effective field of view of 24 degrees (total field of 37.5 degrees, but with tangential distortion). The film format is 5 x 5 inches. No aiming or movement is required during the test; instead, the aircraft itself is prepositioned. These specifications are ample for obtaining spatial data, and by using multiple camera installations and staggering framing times, three of four K-24's are kept open at all times. Motion picture cameras are included for backup and documentary filming. Camera requirements are listed in Tables E-8, E-9 and E-10.

The photometer used for the test is the two-barrel unit designed and built by Barnes Engineering Corporation. This unit has the following specifications:

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- (1) Photometer head--two comounted photomultiplier tube channels.
- (2) Lens--fused quartz.
- (3) Field of view--2 x 2 degrees (each channel).
- (4) Aperture--25 cm².

Each channel produces a voltage output proportional to the incident light intensity. This signal is recorded on a known time base to provide event times. Each channel is fitted with an interference filter to suppress light produced by objects other than the flares, such as tankage and nose cone. The narrow field of view necessitates that the instrument be carefully aimed at the flare burst region, but this arrangement offers the additional advantage that the tankage and nose cone can be excluded from the instrument's view field to prevent photomultiplier tube saturation.

1. Determination of Position of Flare Burst

It is recognized that more exact positioning of the flare event can best be accomplished by triangulation of two or more observation sites. The reduction of data can be accomplished by standard optical data reduction techniques. Unfortunately, two fixed stations were not available to this program. The scheme of data acquisition was such that only one location would be the primary source of camera data. The position of the primary aircraft, described in Appendix B, and the known plane of re-entry of the nose cone are sufficient information to accurately determine the positions of the flare events. The aircraft would be positioned by radar from Ascension Island to a known location with a quoted accuracy of ± 200 feet. By use of the star background and the known plane of trajectory of the nose cone, triangulation can yield the position of the event.

The information to be obtained from Aircraft No. 2, Ascension Island, the DAMP ship and the other normal complement of downrange stations will be used as backup to the information received from the "one-plane solution." It is expected that the backup data will be as useful as that received from the primary source (Aircraft No. 1). Because of the priority level of this program, the backup data is not to be fully depended upon. It is currently classed as "noninterference"; therefore, the secondary aircraft or DAMP ship could be reassigned prior to the tests. Cloud cover could obscure the re-entry as viewed from the land sites.

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2. Time of the Flare Bursts

The standard time associated with the flare burst can be obtained most accurately through use of the photometer (Instrument 6, Table E-9). Since the rise time of the flare intensity is short (of the order of a few milliseconds), a record of the photometer output versus range time will pinpoint the time of the flare appearance. Similar records of the shutter motions of the cameras will provide an absolute time base. This information will also be necessary in order to locate the position of the aircraft at the event times of interest and to provide a standard time for the events themselves.

3. Identification of the Flare Bursts

The determination of the capsule design producing the light depends on identification of the flare material. The light produced in the spectra of every element is unique and can be analyzed by spectrographic or by filter methods. This analysis involves the use of a dispersive element, such as a diffraction grating, in the path of the light or the use of filters which favorably transmit the wavelengths produced by one element and attenuate those produced by the others.

A careful selection of film and filter will cause a camera to selectively photograph the light produced by a single element. Four K-24 cameras shown as Instruments 1, 2, 3 and 4 in Tables E-8 and E-10 are fitted in this manner. Films, filters and the wavelength sensitivities are shown in Table E-8. In addition to the filtered cameras, a Hulcher 70-mm cinespectrometer is included as Camera 5 in Table E-9. This instrument will reproduce the line spectra of each element and thus provide identification. Also, its frame rate and shutter motions will be recorded on a known time base to provide event times. The photometers, fitted with filters (Instrument 6, Table E-9), can also be used to identify the flares.

4. Capsule Velocities

Camera 13, Table E-8, as described under Part 1, Position Fixing, is provided with an aperture chopper which will segment the photographed capsule tracks into intervals of known time length. The velocity of the capsule can be computed for any position along its visible track. The following are miscellaneous considerations with reference to the photographic equipment:

- (1) Synchronization of shutters--The shutters in each camera group should be synchronized to minimize dead or index time. For example, the four filter cameras are to be synchronized so that three of them are in the shutter-open position at all times during use.

- (2) Focusing of filtered cameras--The focal length of a lens varies with wavelength; hence, each camera must be carefully focused for the wavelengths passed by its filter.
- (3) Backup and documentary cameras--Some of the cameras listed in the tables are designated for documentary or backup use. These are the motion picture cameras and the unfiltered K-24's.

5. Station Requirements

As indicated in Tables E-8, E-9 and E-10, two aircraft and one ground station will be available. It is possible that additional stations will be committed by the AMR by test dates.

TABLE E-8

Camera Requirements for Main Aircraft
(supplied by AFSWC)

<u>Instrument Number</u>	<u>Instrument Function and Description</u>	<u>Group Function and Description</u>
1	K-24 camera for photography of sodium flares--Uses Royal X pan film and Wratten 15 filter to utilize the Sodium D lines 5890 Å and 5896 Å.	Bank of four cameras synchronized and operated so that at least three are open at all times during use--These cameras are prepointed before use to ensure that the re-entry events fall within their field of view (2-second exposure, 0.4-second index, 5-minute film pack).
2	K-24 camera for photography of potassium flares--Uses high speed infrared film and Wratten 88A filter to utilize the 7665- and 7699-Å potassium lines.	
3	K-24 camera for photography of rubidium flares--Uses high speed infrared film and Wratten 87 filter to utilize the 7800- and 7984-Å rubidium lines	

TABLE E-8 (continued)

<u>Instrument Number</u>	<u>Instrument Function and Description</u>	<u>Group Function and Description</u>
4	K-24 camera for photography of cesium flares -- Uses high speed infrared film and Wratten 87C filter to utilize the 8521- and 8943-Å cesium lines.	Same as on preceding page.
5, 6	K-24 cameras, unfiltered framing rate, 3 per second -- Used as backups to the filter cameras to provide control (unfiltered) photographs.	Camera pair synchronized and index staggered so that one is open at all times during use. Cameras are prepointed from in-flight data before use to ensure that the re-entry events fall within their view fields (1/2-second exposure, 5-minute film pack).
7, 8	P-2 camera pair automatically indexed by timing signals -- Used to obtain star background and position of flare capsule trajectories with respect to the Atlas re-entry vehicle.	
9, 10	35-mm Mitchell motion picture cameras -- Pair operated at 100 frames per second to obtain times of flare ignitions and to provide documentary film. Shutter mechanism index staggered to minimize lead time.	
11, 12	16-mm motion picture cameras -- Operated at 16 frames per second to provide additional documentary film coverage.	
13	One unfiltered K-24 camera -- Operated with 2-second exposure time; chopped at 4 chops/second. Used to provide additional flare time data and to provide data for computation of capsule velocities and trajectories.	

TABLE E-9

Instrument Requirements for No. 2 Aircraft
(supplied by AFMTC)

<u>Instrument Number</u>	<u>Function and Description</u>
5	Hulcher 70-mm cinespectrometer--Used to obtain spectrographs of the re-entry bodies and flare capsules and to provide additional event time data.
6	Two-barrel photometer--Equipped with interference filters for detection and obtaining time of capsule ignition. Filtered for cesium and rubidium 100-A filtered bandwidth.

TABLE E-10

Instruments for Ascension Island
(supplied by AFSWC except as noted)

<u>Instrument Number</u>	<u>Function and Description</u>
1,2,3,4,5,6	Same as in Table E-8.
7, 8	Same as Cameras 3 and 4, Table E-8.
9, 10	Two ballistic cameras, BC-4--Used in same manner as Cameras 1 and 2, Table E-8 (supplied by AMR).
11	One IGOR camera--Used for high resolution pickup of details. Radar trained (supplied by AMR).

NOTES:

- (1) All cameras are required to be time-indexed and are expected to record flare bursts and nose cone on same frames.
- (2) Camera framing to be initiated so that 50% of the film pack will be expended when nose cone reaches 250,000 feet. Time to 250,000 feet to be supplied by range in-flight information.

References

- E-1. Öpik, Ernst J., "Physics of Meteor Flight in the Atmosphere," Interscience Publications, 1958.
- E-2. McCrosky, Richard E., "Observations of Simulated Meteors," Smithsonian Contribution to Astrophysics, Vol. 5, No. 4, 1961.
- E-3. Whipple, Fred L., "Meteors and the Earth's Upper Atmosphere," Review of Modern Physics, Vol. 15, No. 4, October 1943.
- E-4. Geophysics Corporation of America, "Detectability of Point Sources of Light and of Particulate Clouds in the Upper Atmosphere," prepared for Martin Nuclear Division under AEC Contract AT(30-3)-217.

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APPENDIX F

PLASMA ARC TESTS

Burnup tests of scale models of fuel core simulators (flare capsules) have been conducted in the Martin 40-kilowatt plasma facility. These tests were run to verify that the heating rates to which the test article will be subjected in Phase I and Phase II testing will cause penetration. Results of the test were positive. Tests were run at heat fluxes of 52, 85 and 231 Btu/sec-ft². Heating rates on re-entry are not expected to exceed 200 Btu/sec-ft². The tests, apparatus and data collected are presented in the section that follows.

A. PLASMA JET FACILITY

A 40-kilowatt arc-heated plasma facility was used for this work. The arc-chamber and nozzle assembly is shown in Fig. F-1. The primary gas (nitrogen) enters the torch assembly and flows into a chamber where an umbrella-shaped arc is struck between the electrode and the chamber wall. The heated gas, now a plasma, passes into a plenum located upstream of a Delaval nozzle. In this chamber the primary gas is mixed with oxygen, the secondary gas, to simulate air. The gas is then expanded through the Delaval nozzle and impinged on the model mounted in a vacuum tank 1/4 inch from the exit of the nozzle.

A four-sting assembly for mounting the test models was used. This assembly allows consecutive testing of up to four models at constant settings of the facility controls. The central shaft of the assembly is outside of the tunnel housing and can be rotated at increments of 90 degrees. Devices are provided to lock the assembly at the four operating positions.

The gas enthalpy was determined by a heat balance method; the energy applied to the arc chamber was measured with a voltmeter and an ammeter. The remainder of the heat balance apparatus is shown schematically in Fig. F-2. The flow of cooling water to the arc chamber was measured with a calibrated orifice and a manometer. Cooling water temperatures upstream from the control console, and upstream and downstream of the arc chamber, were measured with calibrated chromel-alumel thermocouples and a galvanometer-type potentiometer. The rate of flow of the cooling water through the Delaval nozzle walls was measured by timed weight samples, and its

Fig. F-1. Arc Chamber and Nozzle Assembly

Fig. F-2. Schematic of Heat Balance Apparatus

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temperature rise across the Delaval nozzle was measured with a dial-type thermometer. Nitrogen and oxygen gas flows were measured with ball-type flowmeters. Static pressures in the Delaval nozzle plenum chamber and the vacuum tank were measured with mercury manometers and a barometer.

B. MODELS

1. Heat Flux Calibration Models

A calorimetric method was used to calibrate the plasma jet controls for the required input heat fluxes. The model configuration is shown in Fig. F-3. A small copper cylinder is imbedded in a phenolic nylon holder, and the junction of a chromel-alumel thermocouple is silver-soldered to the rear face of the cylinder. The temperature of the rear face, as a function of time, is measured with a Minneapolis-Honeywell Visicorder (recording oscillograph).

2. Test Models

Instrumented and uninstrumented flare models were used in the program. The latter, duplicates of the instrumented models without thermocouples, were used for visual and photographic studies of the flare models.

The re-entry flare bodies are either cylindrical or spherical, with a wall thickness of $3/8$ inch. A cylindrical flare body with a 2-inch diameter and a 3-inch length was selected as a representative design. A diameter of $3/4$ inch was chosen for the scale flare models to assure uniform heating by the plasma.

It was not possible to use the full-scale volume ratio of lead to flare material in the scaled-down flare models. Therefore, to simulate burnup characteristics of the re-entry body in a plasma arc test, it was necessary to calculate the front wall thickness and flare material volume for the flare model. In the thickness calculation for the leading end of the model, the flare body was assumed to re-enter with its longitudinal axis parallel to the re-entry path, without tumbling. The frontal surface area-to-volume of lead ratio for the leading end of the $3/4$ -inch diameter model was equated to that ratio for the full-scale re-entry body. This equation gave a thickness of $3/8$ inch for the leading end of the model. The diameter ($1/2$ inch) of the flare material and the wall thickness of the cylindrical portions of the flare models were scaled relative to the re-entry body. Equating the full-scale re-entry body's ratio of outer surface area to volume of flare material to that of the flare model gave the length of the flare material. The

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Fig. F-3. Calibration Model

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resultant scale models have a 3/4-inch outer diameter and a volume of 0.147 cubic inch of flare material (Fig. F-4).

The lead containers are mounted on a phenolic nylon holder to lessen the heat conduction to the string assembly. All alkali flare materials were loaded in a dry box under argon. The screw threads between the lead container and the phenolic nylon were provided with an airtight seal. In the instrumented models, there is an axial hole through the nylon holder to allow a ceramic insulator containing a chromel-alumel thermocouple to be inserted. Three instrumented models were used for each burnup test at a constant input heat flux. Each model contained one thermocouple with its junction located at Positions A, B or C as shown in Fig. F-4. The test variables are given in Table F-1.

C. DISCUSSION OF RESULTS

Photographs of the burnup tests were made with a 70-mm camera operating at 10 frames per second and a 16-mm camera running at approximately 500 feet per second. The 70-mm film will be processed as glass slides and color prints. Observation of the flare models during burnup revealed that the front wall of the lead container progressively melted and was blown away by the plasma stream. As predicted, the higher the heat flux, the greater the light intensity from the flare material and the shorter the flash duration. The lead container appeared blue while sodium and potassium were seen as bright orange and faint violet, respectively.

Temperatures measured at the three experimental heat fluxes are presented graphically in Figs. F-5, F-6 and F-7 for each thermocouple junction position (see Fig. F-4). The temperature curves for the flare models showed the following:

- (1) A short time lapse when there was no temperature rise occurred until the heat was conducted through the lead. This lapse increased with increased distance between the thermocouple and the front edge of the lead container.
- (2) The temperature rise was nearly linear at each junction position until burnthrough of the leading end occurred.
- (3) After burnthrough of the container, the temperature rose rapidly to the maximum deflection of the oscillograph.

Two sets of data were taken at each heat flux. The set of curves giving the most consistent data is presented in this report.

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Fig. F-4. Flare Models

TABLE F-1
Plasma Arc Test Variables

Condition	Power Input (kw)	Gas Flow		Plenum Pressure (in. Hg abs)	Tank Pressure (in. Hg abs)	Calibration Heat Flux (Btu/ft ² -sec)	Gas Enthalpy (Btu/lb)
Symbol	Q	(SCFH)	(lb/sec)	P _p	T _p	q̇	h
Values	12.7	125	0.0029	2.68	2.89	52	1985
	15.2	125	0.0029	2.89	3.14	85	2294
	24.4	125	0.0029	3.51	3.14	231	3607

Fig. F-5. Temperature Versus Time for Sodium-Filled Flare Models

Fig. F-6. Temperature Versus Time for Sodium-Filled Flare Models

Fig. F-7. Temperature Versus Time for Sodium-Filled Flare Models

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Several of the temperature curves (thermocouple positions at B and C) in the other tests showed no temperature rise until after container burnthrough. This was attributed to faulty thermocouple installations.

All of the A temperature curves showed a slight temperature drop prior to exposing the thermocouple to the plasma flame (indicated by the rapid temperature rise). This decrease in temperature is believed due to a slight cooling effect as the sodium left the flare models at the instant of burnthrough. The burnthrough time given by both sets of A thermocouple data is plotted as a function of heat flux in Fig. F-8. The experimental data indicate that the burnthrough time is given by $\Delta t \sim (\dot{q})^{-0.77}$. It is felt, however, that this relation is not a precise representation of burnthrough time as a function of heat flux. The temperature data indicate that the initial temperature of the lead was somewhat different for each heat flux. Calculations were made, considering the "sink" effect of the sodium, to correct the burnthrough times for a common initial temperature of 200° F. These calculations gave the relation $\Delta t \propto (\dot{q})^{-0.88}$. If burnthrough time is calculated with the assumption of front face heating only (considering the "sink" effect of the sodium), the result is $\Delta t \propto (\dot{q})^{-1.12}$. Therefore, it is felt that the experimental relation is approximate and indicates only that burnthrough time is nearly inversely proportional to heat flux. A test variable not accounted for was the increased distance of the sample surface from the nozzle as melting progressed; this distance results in a lowered heat flux at the heated surface as ablation takes place.

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Fig. F-8. Model Burnup Versus Heat Flux

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