JUN 17 1964

~~CONFIDENTIAL~~

UNCLASSIFIED

~~RESTRICTED DATA~~

THIS DOCUMENT CONTAINS RESTRICTED DATA AS
DEFINED IN THE ATOMIC ENERGY ACT OF 1954
ITS TRANSMITTAL OR THE DISCLOSURE OF ITS
CONTENTS IN ANY MANNER TO AN UNAUTH-
ORIZED PERSON IS PROHIBITED.

Sandia Corporation

Contractor for U.S. Atomic Energy Commission
ALBUQUERQUE, NEW MEXICO LIVERMORE, CALIFORNIA

1 4698

DISTRIBUTION OF THIS DOCUMENT IS UNLIMITED

DISTRIBUTION OF THIS DOCUMENT IS UNLIMITED

~~CONFIDENTIAL~~

DISCLAIMER

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

DISCLAIMER

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

~~CONFIDENTIAL~~

M-3679 (34th Ed.) C-44-d
NUCLEAR TECHNOLOGY-AEROSPACE SAFETY

UNCLASSIFIED

SC-RR-64-65

SAFETY ANALYSIS
OF THE OPERATIONAL SNAP-9A
SYSTEM
(Title Unclassified)

MASTER

William Buckalew, 7411
David Byrne, 7421
Robert Halsey, 7411
Sandia Laboratory, Albuquerque

Exempt from CCRP Re-review Requirements
(per 7/22/82 Duff/Caudle memorandum)
NK 3/1/85

Classification cancelled (or changed to)

March 1964

by authority of TID-1387 (Suppl.)

by EB DTIE, date 10/20/75

Reviewed: H. E. Hansen
H. E. Hansen, 7411
Mission Director

Approved: V. E. Blake
V. E. Blake, 7410
Nuclear Safety Programs Manager

NOTICE
This report was prepared as an account of work sponsored by the United States Government. Neither the United States nor the United States Energy Research and Development Administration, nor any of their employees nor any of their contractors, warranty, express or implied, or assumes any legal liability or responsibility for the accuracy, completeness or usefulness of any information, apparatus, product or process disclosed, or represents that its use would not infringe privately owned rights.

ABSTRACT

This report estimates the potential hazards of plutonium fuel release involved in operational application of the SNAP-9A generator in the Transit 5BN satellite. The discussion covers only those contingencies possible after the system is placed on the launch pad, including launch-pad aborts and re-entry from orbit. It analyzes the probable consequences of a release of plutonium fuel in various mediums.

~~RESTRICTED DATA~~

This document contains restricted data as defined in the Atomic Energy Act of 1954. Its transmittal or disclosure of its contents in any manner to any unauthorized person is prohibited.

GROUP 1 - Excluded from automatic
downgrading and declassification

DISTRIBUTION OF THIS DOCUMENT IS UNLIMITED

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

ACKNOWLEDGMENTS

This report was prepared with the cooperation of personnel of the Applied Physics Laboratory of Johns Hopkins University and the Martin Company Nuclear Division, and with the aid of information made available by the Engineering and Test, and Analysis and Evaluation Branches of the USAEC. The following personnel of Sandia Corporation have also made useful contributions: H. W. Church, 5414; J. W. Reed, 5414; R. H. Bishop, 5411; and W. S. Roberts, 5133. B. W. Marshall, 7411, contributed the information regarding the oxidation of tantalum.

~~CONFIDENTIAL~~

TABLE OF CONTENTS

<u>CHAPTER</u>		<u>Page</u>
I.	INTRODUCTION	5
	Description of the Nominal Mission	6
	Description of the SNAP-9A Generator	8
II.	RADIOLOGICAL HAZARDS OF PLUTONIUM FOR VARIOUS RELEASE CONDITIONS . .	10
	Instantaneous Release of Fuel Inventory on the Launch Pad	10
	Fuel Capsule Ruptured Upon Ground Impact	14
	Long-Term Release in Water	15
	Instantaneous Release in Coastal Waters	17
	High-Altitude Fuel Dispersal	18
	High-Altitude Fuel Dispersal on Re-entry from Short Orbit	18
III.	ANALYSIS OF THE PROBABILITY OF FUEL RELEASE UNDER VARIOUS LAUNCH ABORT ENVIRONMENTS	20
	Launch Pad Fire	20
	Impact by Flying Debris Following Fragmentation of the Launch Vehicle in a Detonation of the Propellant Fuels	20
	Shock Overpressures Resulting from Detonation of the Propellants	23
	Fuel Capsule Impact	26
	Impact and Burial of Fuel Capsules in Various Soils	31
	Re-entry from Short Orbit	32
IV.	SUMMARY AND CONCLUSIONS	54
APPENDIX--	Sample Calculation of Aerodynamic Heating on the Original Configuration of the 5BN/SNAP-9A Under Assumption of Worst Conditions for Fuel Dispersion (i.e., Rearward Re-entry)	57

LIST OF TABLES

<u>Table</u>		
1.	Terminal Velocities of Fuel Capsules in Various Fall Attitudes . . .	28
2.	Results of Fuel Capsule Impact Tests	29
3.	Results of Fuel Capsule Impact Tests	29
4.	Maximum Surface Temperature (°C) Attained by Capsules Impacted at Several Depths in Soils of Various Thermal Conductivities . .	32

LIST OF ILLUSTRATIONS

<u>Figure</u>		
1.	Vehicle altitude versus range in the early part of the flight . . .	7
2.	Impact point versus thrust cutoff time	8
3.	Satellite assembly	9
4.	Material deposition versus downwind distance	12
5.	Material deposition versus downwind distance	12
6.	Material deposition versus downwind distance	13

~~CONFIDENTIAL~~

LIST OF ILLUSTRATIONS (cont)

<u>Figure</u>		<u>Page</u>
7.	Initial fragment velocity as a function of the ratio of explosive mass to container mass	21
8.	Transit 5BN/SNAP-9A configurations	34
9.	Transit 5BN/SNAP-9A configurations	34
10.	Calculated moment coefficient versus angle of attack (Configuration 1)	35
11.	Parameters resulting from trajectory studies	36
12.	Stagnation point heat transfer rate on a hemisphere in Transit 5BN/SNAP-9A trajectory	36
13.	Transit 5BN/SNAP-9A wind tunnel model (1/6 scale)	40
14.	Model of transit 5BN/SNAP-9A (1/6 scale) in 48-inch hypersonic shock tunnel. Note ceiling probes for measuring pressure and heating rate 8.5 inches above centerline of tunnel. Flow is from left to right.	40
15.	Schlieren photograph of 1/6-scale Transit 5BN/SNAP-9A model at 3° angle of attack ($Re_L = 86180$, $M = 14.51$)	41
16.	Schlieren photograph of 1/6-scale Transit 5BN/SNAP-9A model at 33° angle of attack ($Re_L = 97420$, $M = 14.79$)	41
17.	Measured and calculated normal and axial force coefficients versus angle of attack (Configuration 3: generator-first orientation)	42
18.	Measured and calculated moment coefficient versus angle of attack (Configuration 3: generator-first orientation)	43
19.	Measured and calculated static margins versus angle of attack (Configuration 3: generator-first orientation)	43
20.	Measured and calculated normal and axial force coefficients versus angle of attack (Configuration 3: cone-first orientation)	44
21.	Measured and calculated moment coefficient versus angle of attack (Configuration 3: cone-first orientation)	44
22.	Measured and calculated static margins versus angle of attack (Configuration 3: cone-first orientation)	45
23.	SNAP-9A generator re-entry temperature profiles	46
24.	Temperature profiles of the fuel capsules with the fuel in three possible locations	48
25.	Surface temperature and oxygen flux to capsule surface for capsule release at 270,000 feet	50
26.	Event chart for re-entry from a short-lived orbit	52
27.	Heat transfer rate ratio for Point I versus local hypersonic viscous interaction parameter	54
28.	Heat transfer rate ratio for Point III versus local hypersonic viscous interaction parameter	

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

SAFETY ANALYSIS OF THE
OPERATIONAL SNAP-9A SYSTEM

CHAPTER I. INTRODUCTION

The SNAP-9A generator designed and built by the Martin Company for the AEC will supply power for the U.S. Navy Transit 5BN navigational satellite system,¹ consisting of four satellites equally spaced in longitude so that the signals they transmit can be received at any point on the surface of the earth.

The first two of these four navigational satellites--a Transit 5BN powered with the SNAP-9A, and a solar-powered Transit 5E--will be launched by a single Thor-Ablestar booster system from Vandenberg Air Force Base in a southerly direction over the Pacific Missile Range. This mission is intended to establish the satellite in a circular polar orbit with an inclination angle of $90^\circ \pm 0.5^\circ$ and an altitude of 600 ± 50 nautical miles. Such an orbit would have a lifetime of approximately 900 years, at the end of which the Pu^{238} isotope fuel would have decayed almost completely to U^{234} . After successful injection into orbit, the satellite systems will be separated from the Ablestar booster, and the Transit 5BN will be separated from the Transit 5E.¹

The SNAP-9A generator will be capable of providing 20.45 watts of electrical power after 5 years of operation in space. The thermal energy (525 watts) required to generate this power is provided by 16,175 curies (963.3 grams) of Pu^{238} metal, contained within six tantalum-lined Haynes alloy capsules.

Since the release of 16,175 curies of plutonium into the biosphere could under certain conditions result in a radiological hazard to a segment of the population, the safety goals observed by the Martin Company during design of the system were complete containment of the fuel in the event of any accident in the biosphere, and complete burnup and high-altitude dispersal of the fuel during re-entry from orbit. Nevertheless, as a double check, it was considered necessary to investigate the various conceivable modes of fuel release, because release at such locations as the launch pad or its surrounding land mass in particular would result in radiological hazards. The analysis reported in this paper had as its object to determine and outline as clearly as possible the level of risk or uncertainty associated with each part of the operational sequence of the Transit 5BN/SNAP-9A system.

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

The approach used was to postulate releases of SNAP-9A fuel at various kinds of locations in the biosphere in order to determine what the attendant radiological hazards would be. The question of whether or not any of the releases postulated was possible was ignored in this initial phase of the study. Later, and then only in cases where it was found that a release would result in a radiological situation, was the probability of release itself investigated. Specifically, the possible points of fuel release considered were:

1. The launch pad
2. The land mass surrounding the launch area
3. Shallow coastal waters
4. Altitudes above 200,000 feet.

On the results of this initial investigation, which showed that radiological situations could result from fuel release in the vicinity of the launch pad, the following conceivable agents of fuel release were studied:

1. High temperatures resulting from a fire involving the missile propellant
2. Shock overpressures resulting from a detonation of the missile propellant
3. Projectiles resulting from fragmentation of the missile
4. Impact of the fuel capsule on a hard surface after becoming separated from the generator.

An additional fuel release mode investigated, and in this case a desirable one, was dispersal in the event of premature re-entry from orbit. For this mode, the generator would have to be destroyed and the fuel released at high altitudes.

This report is not concerned with the integrity of the SNAP-9A generator before it is placed on the launch vehicle, since up until that time, the environment of the generator can be carefully controlled. In any case, such adverse environments as the generator might conceivably experience before being placed on the launch vehicle would probably be less severe than those which could develop from accidents during the launch sequence.

Description of the Nominal Mission

The general description given above explained that the Transit satellite will be launched from the Pacific Missile Range in a southerly direction into a circular polar orbit with an orbital inclination angle of $90^\circ \pm 5^\circ$, at an altitude of 600 ± 50 nautical miles. In more detail, the mission plan provides for injection into orbit at 82 degrees south latitude in the Eastern Hemisphere after the following sequence of launch events. The Thor main-stage burns for 148 seconds, after which separation of the main stage occurs. Main-stage separation is followed at $t = 155$ seconds by ignition of the Ablestar stage. At $t = 449$ seconds, the first burning of the Ablestar stage ends. A coast period ensues, continuing until $t = 2305$ seconds when the Ablestar motor is reignited to burn for 7 seconds. Injection into orbit occurs at $t = 2312$ seconds.

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

It was also explained above that the lifetime of the nominal orbit is approximately 900 years, at the end of which the Pu^{238} isotope fuel will have decayed almost completely to U^{234} . The hazard associated with injecting U^{234} into the atmosphere is negligible regardless of whether or not burnup occurs during re-entry.

Aborts very early in the mission, that is, during the first 50 seconds of main-stage burning, will result in impact of the system either in the launch area or in the surrounding land environs. Aborts after $t = 50$ seconds, and before approximately 10 seconds in advance of cutoff of the first burning of the Ablestar stage (SECO I), will have instantaneous impact points in the ocean. Aborts occurring between SECO I minus 10 seconds and SECO I have instantaneous impact points ranging from Antarctica to Russia.

The sections of this report which follow examine the hazards associated with each of the aforementioned possible aborts. The probability of each is also estimated.

Once SECO I is attained, the system is in an elliptical orbit of a comparatively short lifetime, that is, something less than a year. The second burning of the Ablestar stage injects the system into its proper long-lived, circular orbit.

Two figures included here will be useful in the analyses given in later sections. Figure 1 is a plot of vehicle altitude versus range in the early part of the flight. If an abort occurs during this portion of the flight, any resulting debris will impact in the launch area or its land environs. Figure 2 gives the instantaneous impact point if the thrust of the Ablestar is cut off prematurely during its first burning period. Times indicated are from launch ($t = 0$ seconds).

Figure 1. Vehicle altitude versus range in the early part of the flight³⁴

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

Figure 2. Impact point versus thrust cutoff time³⁴

Description of the SNAP-9A Generator

The main body of the SNAP-9A generator is a cylinder 8.374 inches high and 9 inches in diameter. Six magnesium fins extend longitudinally from the surface of its outer magnesium housing. The heat source, centrally located in the generator, consists of six fuel capsules mounted in a segmented graphite block. Graphite block and magnesium housing are separated by molded Min-K insulation containing 70 pairs of lead telluride thermoelectric elements through which heat flows from the accumulator to the housing. Discs of high density Min-K insulation are used at top and bottom of the fuel block.

For the purposes of thermal and structural analysis, the generator may be described as follows. The outer magnesium housing is a minimum of 3/32-inch thick and has an inside diameter of 7 inches. The graphite heat accumulator is a minimum of 13/32-inch thick and measures 5-1/4 inches outside across flats. The Min-K insulation is 7/8-inch thick around the sides of the graphite heat accumulator and a minimum of 1-3/16-inch thick between the accumulator and both upper and lower closure domes. Titanium washers placed in each fuel capsule cavity above and below the fuel capsule prevent the capsules from penetrating the Min-K insulation.

Each fuel capsule measures 5.732 inches long overall, with a diameter of 0.981 inch. The fuel is contained within a tantalum inner shell, 0.02-inch thick and 4-3/4-inch long, and a Haynes-25 outer shell 0.1-inch thick. The Haynes-25 end plugs are 1/2-inch thick.

The payload package for the Thor-Ablestar vehicle consists of the SNAP-9A generator uppermost, followed by the Transit 5BN satellite to which the generator

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

2 is attached, and, finally, by the Transit 5E satellite to which the Transit 5BN is in turn connected. The Transit 5E is mounted on the launch vehicle. The overall assembly is shown in Figure 3

Figure 3. Satellite assembly

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

CHAPTER II. RADIOLOGICAL HAZARDS OF PLUTONIUM FOR VARIOUS RELEASE CONDITIONS

This chapter presents estimates of the radiological hazards which would be associated with the release of SNAP-9A fuel under various conditions. It should be emphasized at the outset that the intent of this analysis is only to establish the level of the risks which would be involved if such releases were to occur, and that no attempt is made to weigh the probability that any given release mode will occur. A later section of this report (Chapter III) takes up the matter of the probability of fuel release under such conditions as are found in this chapter likely to result in unacceptable radiological situations.

Specifically, the present chapter considers the following release conditions in the order shown:

1. Instantaneous release of the entire fuel inventory in or near the launch pad
2. Long-term release from one fuel element on the launch pad land mass
3. Long-term and instantaneous fuel release in coastal waters
4. High-altitude dispersal of the entire initial fuel inventory

Instantaneous Release of Fuel Inventory on the Launch Pad

Instantaneous release of the entire fuel inventory on the launch pad could, hypothetically, result from detonation of the propellant, from fire, shock over-pressure, projectiles, or some combination of the four. Sutton's equations may be applied for such a release to calculate the downwind deposition and air concentration of plutonium for both lapse and inversion conditions as well as for several release heights.

The amount of material deposited ($\mu\text{c}/\text{m}^2$) directly downwind at distances from the ground level release point is given by:²

$$W(x,0) = \frac{2Q_0 V_g}{u\pi C_y C_z x^{2-n}} \exp \left[-\frac{4V_g x^{n/2}}{n\pi^{1/2} C_z} \right] \mu\text{c}/\text{m}^2 \quad (1)$$

where:

$W(x,0)$ = deposition $\mu\text{c}/\text{m}^2$
 Q_0 = amount of material released, μc
 V_g = velocity of deposition, m/sec

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

μ = wind velocity, m/sec
 C_y, C_z = diffusion constants, $m^{n/2}$
 x = downwind distance, m
 n = dimensionless stability index.

The values for the parameters n, C_y , and C_z used here for lapse and inversion conditions are listed in the table below:³

Diffusion Parameters

	$\frac{n}{}$	$\frac{C_y}{}$	$\frac{C_z}{}$
Inversion	0.50	0.12	0.05
Lapse	0.25	0.24	0.14

In the case of elevated sources, such as would be found if the material rose to some elevation h as a result of an intense fire, the diffusion equation becomes:²

$$W(x,0) = \frac{2QVg}{u\pi C_y C_z x^{2-n}} \exp - \left[\frac{h^2}{C_z^2 x^{2-n}} \right] \quad (2)$$

where:

h = height of release (meters), and Q is given by:

$$\log \frac{Q}{Q_0} = \frac{b}{m} \left[-\frac{e^{-\xi}}{\xi^m} + \Gamma(-m+1) - \Gamma(-m+1, \xi) \right]$$

where:

$$m = \frac{n}{4 - 2n}$$

$$b = \frac{2Vg}{u C_z \pi^{1/2} (2-n)} \left(\frac{h}{C_z} \right)^{2m}$$

$$\xi = \frac{h^2}{x^{2-n} C_z^2} \quad \text{and}$$

$\Gamma(-m+1, \xi)$ = the incomplete gamma function described as:

$$\Gamma(-m+1, \xi) = \int_0^\xi e^{-t} t^{-m} dt$$

Figures 4, 5, and 6 show the deposition of plutonium in $\mu\text{c}/\text{m}^2$ plotted as function of downwind distance from the release point.

Of more interest, perhaps, is volumetric concentration as a function of downwind distance, since from this, inhalation dose may be calculated. The volumetric concentration is obtained from Eqs. (1) and (2) by dividing out the velocity of deposition. Since the value for the velocity of deposition used here is 0.01 m/sec, increasing the ordinates of Figures 4, 5, and 6 by a factor of 100 corrects them to show volumetric concentration versus downwind distance.

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

Figure 4. Material deposition versus downwind distance

Figure 5. Material deposition versus downwind distance

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

Figure 6. Material deposition versus downwind distance

The inhalation dose, or amount of material deposited in the body for various downwind distances, is given by:

$$D = IV(r)f \quad (3)$$

where:

- D = amount of material deposited in the body (μc)
- I = breathing rate (m^3/sec)
- f = fraction of material inhaled that is deposited in the body
- $V(r)$ = volumetric concentration of plutonium at downwind distance r , $\mu\text{c}\text{-sec}/\text{m}^3$.

Taking the following values,⁴

$$I = 3.47 \times 10^{-4} \text{ m}^3/\text{sec}$$

$f = 0.2$ = fraction of plutonium inhaled that is deposited in the bone, the deposition at 10 miles downwind becomes:

Pu²³⁸ Deposited in the Bone - 10 Miles Downwind

<u>Condition</u>	<u>Release Height (meters)</u>	<u>Deposition (μc)</u>
Lapse	200	0.1
Inversion	50	13.0
Inversion	100	2.1

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

Since the maximum permissible body burden (for occupational exposure) of Pu^{239} deposited in the bone is $0.04 \mu\text{c}$, it appears that a release of plutonium like the one postulated above would create a hazardous condition.

However, for the following reasons, the figures given above are probably overestimates of the amount of material likely to be deposited:

1. Experimental data³ indicate that the equations and parameters used may overestimate the downwind concentration by as much as a factor of 20.
2. Specific irregularities of terrain and their effects upon deposition of material cannot be foreseen, and it is difficult to assess their effect upon downwind concentration.
3. It is assumed that the Pu^{239} is uniformly distributed and that it is in soluble, respirable form.
4. It is doubtful, in the case of lapse conditions at least, whether a fire of the intensity necessary to melt the fuel cladding would boost the material to a height of only 200 meters.

Nevertheless, because (1) the meteorological conditions generally prevailing in the Vandenberg area are inversion, with winds directed toward sensitive areas, and (2) the quantities calculated as likely to be inhaled would result in an excessive lung dose even if the fuel were released in insoluble form, it is likely that release of the SNAP-9A fuel on the launch pad would result in an unacceptable radiological situation. For example, the lung dose 10 miles downwind of the release point, for lapse conditions and a release height of 200 meters, is calculated to be about 400 rem.

Fuel Capsule Ruptured Upon Ground Impact

The second contingency to be considered is an abort resulting in the impact of a fuel capsule on land. It is assumed that the surface impacted is rigid enough to make the fuel capsule rupture and expose a portion of the surface of the fuel itself.

The rate at which plutonium oxidizes is a function both of humidity and of the temperature of the fuel. In some cases the oxide may be tightly adherent to the surface of the fuel and in others it may be loose and powdery.⁵ Oxidation rates have been observed under various temperature and humidity conditions.⁶ Tabulated below are the data for plutonium at a temperature of 75°C , exposed to air with a relative humidity of 50 percent:

Weight gain mg/cm^2 of O_2	Time to achieve weight gain (hours)
0.3	10
5.0	100
100.0	1000

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

Since a weight gain of 1 mg/cm² of oxygen per square centimeter of exposed surface represents a penetration of 4.86 microns into delta-phase plutonium,⁶ the weight of plutonium in the oxide phase may conveniently be computed for the times given above. Assuming that 19 square centimeters of fuel surface are exposed, the following table may be calculated:

<u>Time (hours)</u>	<u>Weight of plutonium oxidized (gm)</u>	<u>Activity of oxidized plutonium (μc)</u>
10	0.05	7.7×10^5
100	0.84	1.3×10^7
1000	16.5	2.6×10^8

Or, as another example, consider the oxidation rate of plutonium exposed to pure oxygen at 315°C. An analytic fit to the oxidation data is given by:⁵

$$\text{mg/cm}^2 \text{ (oxygen)} = 24.6 \log_{10} 4.81 \times t \text{ (hours)}, \quad (4)$$

from which the following results may be computed:

<u>Time (hours)</u>	<u>Weight gain mg/cm² (oxygen)</u>	<u>Weight of plutonium oxidized (gm)</u>	<u>Activity of oxidized plutonium (μc)</u>
10	40	6.7	1.04×10^6
100	64	10.5	1.68×10^8
1000	88	14.5	2.3×10^{10}

From the foregoing data and calculations, it is evident that rupture of a fuel capsule will not cause instantaneous release of the fuel into the atmosphere.

The potential radiological hazard associated with a ruptured, oxidizing fuel capsule is, of course, strongly dependent on the extent to which the oxidized fuel becomes airborne. However, a worst case calculation assuming instantaneous release of 100 hours accumulation of oxidation products indicates that a severe inhalation hazard would exist 1 mile downwind of the point of capsule impact. For example, the lung dose at this downwind distance, assuming inversion conditions, is calculated to be about 220 rem. Because the fuel capsules are relatively small, locating and recovering them once they were impacted would be difficult if not impossible. It is therefore mandatory that the fuel be contained in the event of a missile destruct over the land mass of the launch area.

Long-Term Release in Water

In order to estimate the consequences of a long-term release of Pu²³⁸ fuel into the sea, the following British experimental data were consulted. (These figures represent the results of long-term releases of alpha emitters into the Irish Sea at the rate of 1 curie per 28 days.)

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

British Experimental Data⁷

<u>Total</u>	<u>Weighted mean activity ($\mu\mu\text{c/gm}$)</u>	<u>Maximum permissible level ($\mu\mu\text{c/gm}$)</u>	<u>Fraction of maximum permissible</u>
Seaweed	3.1×10^{-2}	10	3×10^{-3}
Fish	4.1×10^{-3}	30	1.4×10^{-4}
Mussels	3.1×10^{-1}	30	10^{-2}

Assume first that all six of the SNAP-9A fuel elements are completely stripped of their cladding (that is, that the fuel is completely exposed to the sea water), and that corrosion proceeds at a rate proportional to the surface area of the fuel. The total surface area of each fuel element is approximately 44 square centimeters; the corrosion rate of the plutonium is assumed to be 120.6 milligrams per hour. (The actual corrosion rate of plutonium is not well known, but it is assumed that its behavior is similar to that of uranium;⁵ uranium has been observed to corrode in hydrogen-saturated water at 150°F at a rate of 120.6 milligrams per hour.) Under these conditions, plutonium (75% Pu^{238} and 25% Pu^{239}) would release 981.7 curies per 28 days.

Extrapolation of the British data for this release rate would yield the following:

<u>Total</u>	<u>Expected mean activity ($\mu\mu\text{c/gm}$)</u>	<u>Maximum permissible level ($\mu\mu\text{c/gm}$)</u>	<u>Fraction of maximum permissible</u>
Seaweed	30	10	3.0
Fish	4	30	0.133
Mussels	304	30	10.0

Complete stripping of the fuel elements is not likely, however; a more reasonable assumption is that the fuel capsules would rupture and expose the plutonium surface adjacent to the capsule void. This surface is calculated to be 0.43 percent of the total surface; adjusting the values in the second table accordingly yields:

<u>Total</u>	<u>Expected mean activity ($\mu\mu\text{c/gm}$)</u>	<u>Maximum permissible level ($\mu\mu\text{c/gm}$)</u>	<u>Fraction of maximum permissible</u>
Seaweed	13.1	10	1.31
Fish	1.75	30	0.058
Mussels	133.0	30	4.4

The maximum permissible levels shown above are based on a maximum permissible level of soluble plutonium in drinking water of $3 \times 10^{-6} \mu\text{c/cm}^3$.⁷ But on the basis of the more recent value of $5 \times 10^{-5} \mu\text{c/cm}^3$ for plutonium,⁴ the expected activity in seaweed would be less than the maximum permissible level.

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

Instantaneous Release in Coastal Waters

Also to be considered is the possibility of instantaneous release of the plutonium in coastal waters. A model frequently used to describe horizontal diffusion in sea water is given by:⁵

$$S(r,t) = \frac{nM}{2\pi\bar{D}(pt)^2} e^{-r/pt} \quad (5)$$

where:

- $S(r,t)$ = concentration at distance r and time t
- M = total activity released
- $\bar{D}$ = average depth of release
- p = diffusion velocity
- n = parameter to account for restricted diffusion: i.e.
 - $n = 1$, open sea; $n = 2$ when release is near a coast line

In the case of instantaneous release, it is assumed that the fuel is released 1 mile off shore and that the hazard of this type of release is the contamination of sea food. The maximum permissible concentration of plutonium in sea water is sometimes taken as ten times the maximum permissible concentration for drinking water, divided by the concentration factor for sea food. The concentration factor for fish has been given as within the limits of 50 and 13.^{7,9} Choosing the higher value for the concentration factor obtains a value of $10^{-5} \mu\text{C}/\text{cm}^3$ for the maximum permissible concentration of plutonium in sea water.

The time required for the concentration of plutonium to reach a maximum at a point 1 mile from the release point (i.e., at the shore line) is, from Eq. (5):

$$\frac{\partial S}{\partial t} = 0 = -\frac{2}{t^3} + \frac{r}{pt^4}.$$

Substituting the proper values, $t = 22$ hours. Then the concentration at $r = 1$ mile and $t = 22$ hours is, again from Eq. (5),

$$S(r,t) = 1.73 \times 10^{-4} \mu\text{C}/\text{cm}^3.$$

The interim between the time of release at $r = 1$ mile and the time that the maximum permissible concentration is exceeded, is estimated from Eq. (5) to be about 60 hours.

At $r = 0$ (the release point), the interim between the time of release and the time that the concentration exceeds the maximum permissible for fish is about 250 hours. For times of the order calculated above, the maximum permissible concentration of plutonium for flesh fish would probably not be exceeded. Thus it is doubtful that either a long-term or an instantaneous release of the fuel in coastal waters would constitute a hazardous situation.

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

High-Altitude Fuel Dispersal

Assuming that the desired orbit is achieved, about 900 years would pass before re-entry, so that something less than 20 curies of Pu^{238} , 20 curies of Pu^{234} , and 10 curies of U^{234} would be injected into the stratosphere. However, if this orbit were not to be achieved, it is only reasonable to assume that the entire fuel inventory could be injected into the stratosphere.

Elsewhere in this report (p. 51), it is calculated that at 190,000 feet the SNAP-9A plutonium fuel would be dispersed with a particle size not accurately known. (Sufficient energy is available to vaporize the fuel, but the ultimate particle size distribution is in doubt since the means for estimating further particle size reduction are unknown.)

High-Altitude Fuel Dispersal on Re-entry from Short Orbit

In order to obtain a worst case estimate of the air and ground fuel concentration which could result from high-altitude fuel dispersal, assume that the fuel is released as a point source of 3-micron-diameter plutonium dioxide particles.¹⁰ This particle size was chosen because, with a density of 11 gm/cm³, it represents the maximum respirable particle size, and also because inhalation of 3-micron particles results in the maximum lung dose rate. (It is assumed that the fuel would be oxidized, so that the hazard associated with its release would be from inhalation resulting in a localized radiation dose to the lung.)

Church's calculation¹⁰ predicts an ultimate surface plutonium concentration of $1.2 \times 10^{-3} \mu\text{c}/\text{m}^2$ over about 1/36 of the earth's surface. Taking particle fall time and a breathing rate into account, it appears that an inhaled dose of $8.6 \times 10^{-5} \mu\text{c}$ of plutonium is possible.¹⁰ Since each 3-micron size particle contains $2.3 \times 10^{-3} \mu\text{c}$ of plutonium, the concept of using a fall time and a breathing rate to calculate inhaled dose is not applicable. The fact that it cannot be used for this case suggests, however, that inhalation of even one 3-micron particle is remote; but inasmuch as one inhaled 3-micron particle would constitute the greatest possible inhalation dose, it is probably meaningful to make calculations for that dose.

The average lung dose in rem/wk may be calculated from:

$$q = \frac{2.8 \times 10^{-3} \text{ mR}}{\xi} \quad (6)$$

where:

$q = \mu\text{c inhaled} = 2.3 \times 10^{-3} \mu\text{c}$

$m = \text{mass of the lung} = 1000 \text{ gm}$

$R = \text{dose in rem/wk}$

$\xi = \text{effective absorbed energy per disintegration} = 57 \text{ mev for } \text{Pu}^{238}.$

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

Solving for R obtains, for the average lung dose,

$$R = 0.047 \text{ rem/wk.}$$

The maximum ICRP value for occupational exposure is 0.3 rem/wk. One-thirtieth of this value, which is somewhat less than the 0.047 rem/wk calculated above, is suggested as the maximum permissible for the general population. Consider the following conclusion, however, reached by an ad hoc committee convened at the Lovelace Foundation on July 10, 1963, to discuss the biological hazards associated with the use of the SNAP-9A in aerospace applications:

For particles of larger size than 1μ , the probability of deposition in the lung decreases markedly as the size increases, and the mean residence time of a large particle, once deposited in the lung, is expected to be significantly less than in the case of diffuse material. The hazard from the diffuse distribution of Pu^{238} in the lung is considered to be greater than that from a single particle of the same total activity because of the greater number of cells at risk in the case of the diffuse distribution.

In short, although the suggested maximum dose-rate level for the general population would be exceeded initially in the event that 3-micron particles were inhaled, particles of such a size would probably not reside in the lung for more than a few days. Particles of smaller size would, of course, be retained longer; but in the case of smaller particles the amount of active material retained is likewise smaller. On the basis of these data, it may be concluded that high-altitude dispersal of SNAP-9A fuel should be acceptable radiologically.

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

CHAPTER III. ANALYSIS OF THE PROBABILITY OF FUEL RELEASE UNDER VARIOUS LAUNCH ABORT ENVIRONMENTS

The preceding chapter examined the radiological hazards associated with the release of SNAP-9A fuel. One conclusion reached was that the release of even a portion of the contents of one fuel capsule on the launch area land mass would probably result in an unacceptable radiological hazard. Although one design criterion for the SNAP-9A generator was that complete containment of the fuel be assured, experimental evidence to prove that this criterion has been met is as yet incomplete. The purpose of this chapter, therefore, is to investigate the events or combinations of events which most threaten to result in release of SNAP-9A fuel on the launch pad or in the launch pad area, including:

1. A fire on the launch pad resulting from ignition of the missile propellants
2. Flying debris, following fragmentation of the launch vehicle as a result of detonation of the propellants
3. Shock overpressures resulting from detonation of the propellants
4. Impact and burial of fuel capsules in various soils
5. Impact of fuel capsules on rigid surfaces.

Launch Pad Fire

The Air Force Special Weapons Center (AFSWC) has performed extensive calculations, with which Sandia Corporation concurs, demonstrating that the fuel capsule cladding could not be breached in any type of launch abort fire even if the fuel capsule were exposed directly to the fire.¹¹ The AFSWC analysis also showed that ignition of the magnesium generator housing could not provide sufficient energy to damage the enclosed capsules. It must therefore be concluded that launch-pad abort fires have neither sufficient intensity nor sufficient duration to cause release of the plutonium fuel to the atmosphere.

Impact by Flying Debris Following Fragmentation of the Launch Vehicle in a Detonation of the Propellant Fuels

The second event investigated is impact of the SNAP-9A generator by flying debris. Theoretically, an explosion might drive parts of the missile into the SNAP-9A generator to the extent not only of penetrating the generator proper but also of breaching the fuel capsules and allowing any ensuing fire to act directly

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

on the plutonium. In order to predict the probability of such damage to the generator, it is necessary to know the velocity and mass of the explosion-propelled fragments.

The velocity that may be imparted to fragments of the launch vehicle can be estimated by assuming a TNT equivalent for detonation of the propellant fuel. The ultimate velocity achieved by fragments of explosives containers, shell cases, bombs, and the like, is dependent on the ratio of the mass of the contained explosive to the mass of the container. One expression for computing the initial velocities of fragments from an exploded cylindrical shell is given by:¹²

$$V_o = (2ER)^{1/2} = \left(\frac{2EC}{m + C/2} \right)^{1/2} \quad (7)$$

where:

V_o = initial fragment velocity
 C = mass of explosive
 m = mass of shell (container)
 $(2E)^{1/2}$ = 8000 fps for TNT; 9000 fps for Comp. B.¹³

This expression for initial fragment velocity is plotted in Figure 7 as a function of the ratio of explosive mass to container mass. Figure 7 shows that, for a ratio of two, the container fragments from exploded TNT have an imparted velocity of about 8000 fps. In the case of the SNAP-9A generator, it can be assumed that the mass of the detonated fuel is much greater than the mass of the fuel container, and, somewhat arbitrarily, that the velocity of the fragments is 9000 fps.

Figure 7. Initial fragment velocity as a function of the ratio of explosive mass to container mass

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

Further, it can be realistically assumed that the top bulkhead of the booster vehicle is the sole source of fragments likely to endanger the integrity of the generator. For the present purpose, the top bulkhead is assumed to be uniformly fragmented into cubes and propelled upward towards the generator by the detonation gases. Since details of the upper bulkhead and possible attendant structural members are not known to the authors, this bulkhead is conservatively assumed to be 0.5-inch thick. In order to damage the generator, its fragments must penetrate the intervening encapsulated electronics with emergent velocities of several thousand feet per second.

Although the range of fragments in various armor materials has been studied extensively, such studies and measurements have not dealt with encapsulants. It is assumed in this calculation that the stopping power of the encapsulating material is similar to that of Celotex, for which experimental data are available.¹⁴ Fragments penetrating Celotex lose velocity in a linear manner in accordance with the following experimentally determined relationship:

$$V/X = 1710 A/M \quad (8)$$

where:

V = velocity of the fragment

X = penetration depth in Celotex

A = projected surface area of fragment that impacts the Celotex

M = mass of the fragment.

Taking the density of aluminum to be 2.7 grams per cubic centimeter, the "shape factor," A/M, for aluminum cubes becomes

$$A/M = 0.774M^{-1/3},$$

and, hence, the above penetration relationship is simply

$$V/X = 1320M^{-1/3}.$$

The mass of the cubical projectiles resulting from fragmentation of the aluminum bulkhead is calculated to be approximately 5.55 grams. Fragments with this mass, and with an initial velocity of 9000 fps, could thus be expected to penetrate 12 inches of Celotex, a thickness considerably less than the thickness of the actual intervening encapsulating material in the SNAP-9A.

It can be concluded, therefore, that even if a detonation of the propellant fuel were to break the missile up into flying fragments, the fragments would not rupture either the capsules or the generator.

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

Shock Overpressures Resulting from Detonation of the Propellants

The shock overpressures which the generator system could experience in a detonation of the propellant fuel may be estimated using the semiempirical equation developed by Smalley and Anderson¹⁵ and the microbarograph readings obtained from the Johnston Island Bluegill Prime abort.

The Smalley and Anderson equation, derived for the most part from pressure measurements taken during the destruction of scaled missiles, is applicable to LOX RP-1 detonations. In the scaled tests, two types of missile failure were simulated: missile fallback, and ripping of the tankage seam as a result of overpressurization. The expression derived for these types of failure is:

$$P = \frac{9862}{Z^3} - \frac{188}{Z^2} + \frac{53}{Z} \quad (9)$$

where:

$$Z = \frac{R}{W^{0.247}}$$

P - average side-on overpressure (psi)

R = distance from the explosion (feet)

W = total weight of propellants (pounds).

Since the SNAP-9A generator is located about 50 feet from the center of the Thor tankage, and since the weight of the Thor propellant is approximately 100,000 pounds, it can be predicted from Eq. (9) that, in the event of a propellant detonation on the launch pad, the SNAP-9A generator would experience an average side-on overpressure of 450 psi. Overpressures of this magnitude, however, are not sufficient to breach it.

Another estimate of the overpressures likely to be created by detonation of the propellant may be obtained from the microbarograph records of the Johnston Island Bluegill Prime abort. The film records of this abort strongly suggest that the mixing of the propellants which took place, promoted by explosive ejection of the heat shield, was so extensive that it probably represents the maximum fuel mixing to be anticipated. A microbarograph located 3597 feet from the launch pad recorded overpressures of 440 and 1720 microbars, respectively, for the explosive ejection of the heat shield and for the detonation of the propellant which followed.

These values of overpressure are considered to be low, however, since the calibration of the instrument was based on homogeneous atmosphere propagation, whereas actually the propagation was upwind and in unstable air. However, the observed overpressures can be adjusted upward, since the explosive charge is known to consist of 50 pounds of TNT. The method followed is to compute the overpressure at 3597 feet from a 50-pound TNT explosion in a homogeneous atmosphere and then correct the two observed overpressures in accordance with the result.

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

The detonation of a 1-pound TNT charge yields an overpressure of 2.55×10^4 microbars at 90 feet from the point of detonation. With distance, the overpressure attenuates as $R^{-1.2}$ (R = distance) for pressures of 2.25×10^4 microbars and less. The distance at which the overpressure from a 50-pound detonation is 2.25×10^4 is obtained from the following expression:

$$R_2 = R_1 \left(\frac{W_2}{W_1} \right)^{1/3} \quad (10)$$

where:

R_2 = distance from detonation of W_2 pounds of explosive necessary to achieve the same overpressure as that observed at a distance R_1 from the detonation of W_1 pounds of explosive.

Hence, for a 50-pound detonation,

$$R_2 = 90(50)^{1/3} = 331 \text{ feet.}$$

Then the overpressure to be expected at a point 3597 feet from a 50-pound TNT detonation in a homogeneous atmosphere is:

$$P = 2.55 \times 10^4 \left(\frac{331}{3597} \right)^{1.2} = 1455 \text{ microbars.}$$

And the fuel detonation overpressure scales up to

$$P = \frac{1455}{440} \times 1720 = 5700 \text{ microbars.}$$

This overpressure extrapolates to an overpressure of 2.55×10^4 microbars at a distance of:

$$R = \left(\frac{5100}{2.55} \times 10^4 \right)^{1/1.2} \times 3600 = 1040 \text{ feet.}$$

The TNT equivalent of the propellant detonation is computed to be:

$$\frac{R_2}{R_1} = \frac{W_2^{1/3}}{1}$$

$$W^{1/3} = \frac{1040}{40}$$

$$W = 1540 \text{ pounds TNT equivalent.}$$

The average side-on overpressure to be expected 50 feet from the detonation of 1540 pounds of TNT may be computed from:

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

$$P = \frac{8240}{Z^3} - \frac{167}{Z^2} + \frac{50}{Z}, \quad (11)$$

where

$$Z = \frac{R}{W^{1/3}},$$

and the other terms are as defined in the Smalley and Anderson expression (Eq. 9). Substitution of the appropriate values into Eq. (11) yields, for the average side-on overpressure 50 feet from the detonation of 1540 pounds of TNT,

$$P = 100 \text{ psi.}$$

The good agreement between the two calculated overpressures, i.e., 450 and 100 psi, strongly indicate that overpressures from detonation of the kind of fuel aboard the Thor will not be capable of breaching the SNAP-9A generator.

The possibility of a detonation of the Ablestar fuel remains to be considered. This fuel consists of 8500 pounds of IRFNA and UDMH--a combination that is hypergolic when mixed and that, hence, should not detonate. Fireball tests performed by the Martin Company on IRFNA-UDMH mixtures have demonstrated partially at least that they do not detonate. In these tests, separate containers of IRFNA and UDMH were explosively ruptured and the spilled contents allowed to flow together and mix. Some fireball data were obtained even though the desired degree of mixing was never achieved; however, in none of the fireball tests were any signs of fuel detonation detected by the passive shock overpressure gages used to instrument the experiments. Although these tests do not demonstrate conclusively the IRFNA-UDMH will not detonate, they do indicate either that an extremely high percentage of mixing is required, or that some unique mode of ignition is necessary in order for detonation to occur.

Assuming that detonation of the Thor fuel would be sufficient to mix and detonate the Ablestar fuel, an estimate can be given for the shock overpressures to which the SNAP-9A generator would be subjected; but it should be remembered that no confidence level can be assigned to such a calculation, since the Smalley-Anderson expression for estimating overpressure versus distance is strictly applicable only to LOX-RP-1 systems.

Applied to the Ablestar fuel, the Smalley-Anderson expression predicts that detonation of the fuel would subject the SNAP-9A generator to an average side-on pressure of approximately 4000 psi. Overpressures of this magnitude would breach the generator but not the fuel capsules, which have a calculated yield strength of about 40,000 psi. Breach of the generator would, of course, be likely to expose the fuel capsules to any ensuing fire, but the AFSWC thermal environment calculations show that such exposure would be of little consequence.

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

Detonation of the Ablestar fuel while the launch vehicle is on the launch pad is considered unlikely, and the possibility of detonation after the vehicle is airborne is considered to be even more remote. Inflight destruct of the vehicle is accomplished by ripping the tankage of the propellant in such a manner that the fuel spills rapidly and that mixing of the constituents is inhibited. In LOX-RP-1 vehicles at least, destruct results in rapid burning of the fuel and no detonation. Since the Ablestar fuel is hypergolic, its behavior in vehicle-destruct is not expected to deviate much from that of the LOX-RP-1.

From the foregoing discussion these conclusions may be drawn:

1. Detonation of some fraction of the Thor fuel is possible in the event of a launch pad abort.
2. The overpressures which would be associated with such a detonation are not sufficient to breach the SNAP-9A generator.
3. Detonation of the Ablestar fuel in the event of a launch pad abort is unlikely.
4. Should an Ablestar fuel detonation occur in a launch pad abort, the associated shock overpressures could breach the SNAP-9A generator but not the fuel capsules.
5. Inflight abort of the Thor-Ablestar would result in burning of the propellants, with no fuel detonation.

Fuel Capsule Impact

Early in the development of the SNAP-9A generator before firm launch trajectories were established, the Martin Company Nuclear Division assumed that land impacts following a launch abort were possible. Consequently, complete fuel containment was established as a design requirement. The most severe conceivable impact surface was assumed to be granite, and Martin conducted tests in which bare fuel capsules were impacted against a granite surface at a test velocity calculated to be the terminal velocity of the fuel capsule plus a margin of safety.

Briefly, these Martin Company tests and their results^{16,17} may be summarized as follows. It was assumed that the fuel capsules would free-fall in a tumbling mode, but that the critical impact attitude would be the end-on attitude. The terminal velocity of a tumbling fuel capsule, plus a margin of safety, was calculated to be 300 feet per second. Accordingly, the fuel capsules were impacted at this velocity end-on against a granite block. No capsule ruptures occurred during the test series.

Sandia Corporation has participated in the fuel capsule impact test program by conducting actual high-altitude drops from an airplane in order to verify the Martin Company's estimate of terminal velocity, and by conducting additional impact tests at measured terminal velocities. In both efforts, the object has been not

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

only to check the Martin Company's findings but also to broaden the impact test program to include locating the velocity at which fuel capsule failure occurs for various impact conditions.

The results of the high-altitude drop tests are summarized in Table 1, which shows that the terminal velocity of a fuel capsule is approximately 250 feet per second at the highest land elevation on the Pacific Missile Range where impact of the Transit 5BN payload could occur.

The Sandia impact test program has included side-on and angled impacts against a granite target as well as additional end-on impacts. Tests have also been accomplished in which a fuel capsule was impacted approximately on its center against the 90-degree angle of a section of angle iron.

The test method used was to fire the target material against the capsule rather than the capsule against the target. Two means of accelerating the target material to the proper speed have been used. Initially, a 155-mm gun was used to accelerate a cylinder tipped with a block of target material against a fuel capsule. When it proved difficult to obtain the correct impact velocity with the gun, however, this method was abandoned in favor of accelerating the target material to the desired impact velocity by means of a rocket sled and then impacting it against the fuel capsule.

The results of the Sandia impact tests are tabulated in Tables 2 and 3. These tables show that the failure point of the fuel capsule is at an impact angle of 45 degrees and an impact velocity of between 225 and 250 feet per second. The tests reported were run with fuel capsules manufactured by Mound Laboratory, duplicates of the actual capsules except that they contained simulated fuel only. Tests were also run at impact angles of 35 and 55 degrees with capsules made by Sandia Corporation. Results of these tests, which are not tabulated, provided comparative data on the severity of angles different from 45 degrees, and indicated that the 45-degree angle of impact produces more damage to the capsule than either 35- or 55-degree impact angles.

A study conducted parallel to the experimental program examined the overall sequence of events necessary for a fuel release resulting from capsule impact. For such a fuel release to take place in the launch area or in the area adjacent to the launch area, the following sequence of events would be required:

1. An abort early in the ascent
2. An explosion or fire, or combination of the two, sufficient to free the fuel capsules from the generator at such an altitude that the fuel capsules would attain their free-fall terminal velocity
3. A capsule impact at the critical angle for capsule rupture
4. An impact on a granite surface.

~~CONFIDENTIAL~~

TABLE 1
Terminal Velocities of Fuel Capsules in Various Fall Attitudes

Item	Fall Attitude	Martin (Theoretical)			Sandia (Theoretical)			Sandia (Experimental)		
		$C_D S$ (ft ²)	Weight (lb)	$V_{S.L.}$ (fps)	$C_D S$ (ft ²)	Weight (lb)	$V_{S.L.}$ (fps)	$C_D S$ (ft ²)	Weight (lb)	$V_{S.L.}$ (fps)
Generator w/fins	Tumbling	.802 .936	26.77	155 168	--	--	--	1.65 2.01	27.5	107 118
Generator wo/fins	Tumbling	.438 .500	25.07	205 220	--	--	--	.40 .45	25	218 232
Generator wo/fins	Axial	.338	25.07	250	.354	27	253	--	--	--
Generator wo/fins	Transverse	.660	25.07	179	.421	27	232	--	--	--
Fuel capsule (fuel at one end)	Tumbling	.021 .025	1.3	211 227	.0218	1.3	223	.019 .024	1.270	211 237
Fuel capsule (fuel distributed)	Tumbling	.021 .025	1.3	211 227	.0218	1.3	223	.016 .019	1.217	232 253
Fuel capsule (fuel at one end)	Tumbling	--	--	--	--	--	--	.06 .11	6.446	229 301
Fuel capsule (fuel distributed)	Tumbling	--	--	--	--	--	--	.06 .10	6.480	246 301

~~CONFIDENTIAL~~~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

TABLE 2
Results of Fuel Capsule Impact Tests

<u>Run Number</u>	<u>Configuration</u>	<u>Impact Attitude</u>	<u>Impact Velocity (fps)</u>	<u>Results</u>
M-1-H	Fuel at one end	End-on on fuel end	286	Intact
M-9-H	Fuel at one end	End-on on fuel end	439	Ruptured
M-10-H	Fuel at one end	End-on on fuel end	365	Intact
M-14-H	Fuel at one end	Side-on	287	Intact
M-13-H	Fuel at one end	45°	276	Ruptured
M-4-G	Fuel distributed	End-on	370	Ruptured
M-5-H	Fuel at one end	Side-on	264	Intact
M-5-G	Fuel distributed	Side-on	410	Ruptured
M-6-G-1	Fuel distributed	Side-on	385	Ruptured
M-6-G-3	Fuel distributed	Side-on	349	Ruptured
M-6-G-2	Fuel distributed	Side-on	135	No test

TABLE 3
Results of Fuel Capsule Impact Tests

<u>Capsule Number</u>	<u>Configuration</u>	<u>Impact Attitude</u>	<u>Impact Velocity (fps)</u>	<u>Results</u>
1	Fuel distributed	45°	271	Haynes and tantulum ruptured
2	Fuel distributed	45°	299	Haynes and tantulum ruptured
3	Fuel distributed	45°	276	Haynes and tantulum ruptured
4	Fuel distributed	45°	246	Haynes ruptured
5	Fuel distributed	45°	219	Intact
6	Fuel distributed	45°	245	Haynes ruptured

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

With the aid of studies previously published, probability estimates can be given for all the events in this sequence except the second.

A Pacific Missile Range Study¹⁸ gives the combined probability of a missile abort and the landing of a piece of the destructed missile in a given area of the Pacific Missile Range or the adjacent territory. (Since this study is applicable primarily to those areas beyond a 2-mile radius of the launch site, a separate probability analysis for an impact in the near vicinity of the launch pad is given later in the present discussion, on p. 31.)

Geological survey data¹⁹ details the geology of the Pacific Missile Range and the land adjacent to it, and reports the composition of the exposed surface in these areas. To consider only the amounts of granite surface in the expected impact area does not appear conservative, especially since in the Sandia and Martin Company test series, which both used granite targets, in no way demonstrate that similar results cannot be obtained on surfaces of less compressive strength than granite. Granite itself has a wide range of compressive strength (from 10,000 to 40,000 psi) and accordingly, surfaces defined in the geological survey data¹⁹ as having compressive strength as low as 5000 psi might be considered as potentially hazardous impact areas.

The Pacific Missile Range Study noted above¹⁸ gives its estimates in the form of constant probability lines bounding certain areas in the Pacific Missile Range for this specific mission. These lines, when superimposed on geological maps of the Pacific Missile Range and the adjacent territory, make it possible to read off the probability of a fuel capsule impact in a hard area of the range. The portion of the downrange area exhibiting surface conditions which exceed compressive strength of 5000 psi was measured from geological survey data.¹⁹ When these data for area and location are combined with the contour data on Pacific Missile Range abort probability, it can be estimated that the probability of combined missile abort and debris impact in the hard area is 6.5 in 10,000.

As was reported above, comparison impact tests at 35- and 55-degree impact angles were conducted during the Sandia impact testing program. Since these impacts appeared less severe than those at a 45-degree angle, it may be assumed that a ± 10 -degree spread in impact angle encompasses the angles critical for rupture. During the Sandia drop tests, free-falling capsules were observed to impact with random orientation.²⁰ Therefore, the probability that a fuel capsule will impact at a critical angle is merely 40 degrees out of 360 degrees or 1/9.

The probability that one or more fuel capsules will be released from the generator as a result of an inflight explosion or fire (or a combination of the two) has been discussed at length elsewhere in this report (pp. 20-26). It has been shown that release of the fuel capsule from the generator as a result of one or a combination of these events is highly improbable: it might be conservatively estimated (from the standpoint of safety) as 1/1000.

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

Taking this value into account, the final probability that a fuel capsule will impact on a hard surface, in the area downrange from the launch complex, at an angle to cause capsule rupture, is the product of the individual probabilities as follows:

$$\frac{6.5}{10^4} \times \frac{1}{9} \times \frac{1}{10^3} = 0.7 \times 10^{-7} .$$

As was pointed out above, the Pacific Missile Range Study¹⁸ used so far in this discussion does not define probabilities for missile fragments impacting at points within a 2-mile radius of the launch pad. To define these probabilities a simpler approach may be used.

Using the specified reliability of 95 percent for the Thor,²¹ and assuming that any Thor failure would result in explosion and fire, a conservative estimate of the probability of an abort on or above the launch pad is 5/100. There are no hard geological surfaces in the vicinity of the launch area,¹⁹ and maps of the area show that man-made hard surfaces (i.e., reinforced concrete) constitute less than 10 percent of the surface within a 2-mile radius of the launch pad.

The probability, then, that a fuel capsule will impact a hard surface within this radius is 1/10, and the remaining probabilities for release of fuel capsules from the generator and for impact of the fuel capsules at a critical angle are, as before, 1/1000 and 1/9, respectively. Then the combined probability figure for capsule rupture in the near vicinity of the launch pad is the product of these individual probabilities:

$$\frac{5}{100} \times \frac{1}{10} \times \frac{1}{1000} \times \frac{1}{9} = 0.55 \times 10^{-6} .$$

Impact and Burial of Fuel Capsules in Various Soils

In order for fuel capsules to impact the land mass and penetrate the surface, they must in some manner be separated from the SNAP-9A generator and allowed to free-fall. The only mechanism even hypothetically capable of effecting such a separation appears to be shock overpressures from detonation of the propellants, and it has been demonstrated conclusively that detonation of the Thor propellant is not practically capable of causing separation. This leaves the possibility of a detonation of the Ablestar fuel. For reasons explained earlier in this chapter, it is extremely unlikely that the Ablestar fuel will detonate, particularly during an in-flight abort, but completely conclusive experimental evidence to support this premise is not available; and if the Ablestar fuel should detonate, it appears on the basis of the Smalley-Anderson equation that a shock overpressure sufficient to breach the generator would result.

It is necessary, therefore, to calculate the ultimate temperature which the capsules would attain, freed from the generator and impacted at several depths in soils of various thermal conductivities. Table 4 shows the results of such a calculation.

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

TABLE 4

Maximum Surface Temperature ($^{\circ}\text{C}$) Attained by Capsules Impacted at
Several Depths in Soils of Various Thermal Conductivities

Depth (ft)	$K^1 = 0.00033$	$K^2 = 0.0023$	$K^3 = 0.00063$	$K^4 = 0.0014$
1	2370	322	1240	560
2	2870	390	1500	676
3	3180	431	1670	750
4	3380	454	1770	796
5	3560	483	1870	840
6	3670	496	1930	865

¹ Diatomaceous earth

³ Dry sandy soil

² Average soil

⁴ Sandy soil, 8-percent moisture

From the evidence given in this table, it is apparent that the capsules must impact onto diatomaceous earth or dry sandy soil in order for melting to occur. Since impaction into average soil is much more probable, it can generally be expected that the capsules will not achieve the melting temperature of their cladding.

Re-entry from Short Orbit

The worst hypothetical cases possible in association with re-entry from orbit are (1) the case in which the system re-enters from a short-lived orbit with its fuel capsules damaged, so that upon impacting the ground they shatter and expose plutonium fuel to the air, and (2) the case in which not even a short-lived orbit is attained, so that, unless complete burnup takes place, the system follows a ballistic path and impacts at a considerable distance downrange. In order to determine the probability of these maximum hazard conditions a number of possible re-entry configurations and parameters must be examined.

If the system performs as designed, the Transit 5BN satellite will re-enter alone. However, faulty performance in either of two ways would result in a different re-entry configuration:

1. If the payload failed to separate from the Ablestar vehicle, the re-entry configuration would be the Ablestar vehicle plus the two-satellite payload.
2. If the payload separated from the Ablestar vehicle but the two satellites failed to separate from each other, the connected satellites would comprise the re-entry configuration.

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

In addition, the following parameters must be taken into account: (1) the dynamic and static stability of the re-entering system, (2) the thermal characteristics of the system, and (3) the interactions between materials in the system at elevated temperatures. Finally, it is necessary to assess the probability of a short-lived orbit and the probability that the system would impact a land mass assuming that the capsules did re-enter in a damaged condition.

5 The probability that the Ablestar vehicle will successfully reach SECO I but fail to restart, an occurrence that would place it in a short-lived orbit, has been estimated as 15/1000 by Space-General Corporation, manufacturers of the vehicle.

The reliability of the devices for separating the payload from the Ablestar vehicle is not known. In addition to these devices, there are two others for separating the two satellites; this second pair of devices are independent of each other and of whether the payload has separated from the vehicle. The reliability of all these devices may be reasonably assumed to be 96 percent each. The probability, then, that the Transit 5E and the Transit 5BN will re-enter attached to the Ablestar vehicle would be:

$$\frac{15}{1000} \times \frac{(4)^3}{(100)} = 0.96 \times 10^{-6} .$$

This probability is low enough to eliminate the need for a burnup analysis of this configuration.

The probability that, with short-lived orbit achieved and the payload separated from the Ablestar vehicle, the Transit 5BN will not separate from the Transit 5E is:

$$\frac{15}{1000} \times \frac{(4)^2}{(100)} \times 0.96 = 23 \times 10^{-6} .$$

This leaves a chance of better than one in a million, and is therefore perhaps not quite incredible; but the probability is diminished by adding in the probability for impacting a land mass. Since the ratio of water to land on the earth is 3 to 1, the probability of impacting a land mass under the conditions postulated is:

$$23 \times 10^{-6} \times 1/4 = 6 \times 10^{-6} .$$

On the other hand, the probability that the Transit 5BN satellite will re-enter alone from a short-lived orbit can be calculated as:

$$\frac{15}{1000} \times \frac{(96)^3}{(100)} = 13 \times 10^{-3} .$$

This probability is high enough to warrant examining the condition in more detail.

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

Figures 8 and 9 show the configuration changes through which the SNAP-9A/Transit 5BN system has evolved. The static stability of Configuration 1 was first examined from the standpoint of a theoretical assumption of Newtonian flow. Figure 10 which shows the results of this examination, can be explained as follows: where the moment coefficient, C_m , is negative, the system is stable--that is, tending to return to its initial position; a positive value of C_m , on the other hand, indicates an unstable system--that is, one that is tending to depart from its initial position. This figure shows that when the generator is in the trailing position the system is stable over a large range of angles of attack. Moreover, the moment coefficient in the unstable portion of the range of angles of attack when the generator is in the trailing position is of small magnitude; a small difference between theoretical and actual moment coefficient could place the system in a stable condition over the entire range of angles of attack.

Figure 8. Transit 5BN/SNAP-9A configurations

Figure 9. Transit 5BN/SNAP-9A configurations

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

Figure 10. Calculated moment coefficient versus angle of attack (Configuration 1)

The theoretical calculations, then, showed a good possibility that the system would re-enter with the instrumentation compartment (octagonal section) forward. Since this section contains many electronic components potted in materials with good ablative properties, it was necessary to examine the burnup characteristics of the system closely. The accepted safety criterion for re-entry of this system from orbit is destruction of the fuel (to 1 micron or less) above 100,000 feet.

Re-entry trajectories were calculated assuming the following initial conditions:

Altitude = 400,000 feet

Velocity = 25,600 feet

Re-entry angle = -0.05 degrees

$W/C_D A = 35 \text{ lb/ft}^2$

W = weight in pounds

C_D = dimensionless drag coefficient

A = system reference area (2 ft^2).

The parameters resulting from the trajectory studies are shown in Figure 11. They include altitude, velocity, and acceleration as functions of time. An additional parameter derived from these is the heat flux to the stagnation point of a 1-foot radius curvature. This heat flux is shown in Figure 12. It is a quantity derived from the expression:²²

$$\dot{q}_{hs} \sqrt{R} = 17600 \left(\frac{\rho_{\infty}}{\rho_o} \right)^{1/2} \left(\frac{U}{U_c} \right)^{3.15} \quad (12)$$

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

Figure 11. Parameters resulting from trajectory studies

Figure 12. Stagnation point heat transfer rate on a hemisphere in Transit 5BN/SNAP-9A trajectory

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

where:

ρ_{∞} = free-stream air density, slug/ft³

ρ_0 = sea level air density, slug/ft³

U = vehicle velocity, ft/sec

U_c = circular orbit velocity, ft/sec

This quantity may be used for heat flux to the stagnation point of bodies with radii of curvature other than 1 foot by inserting the value of the radius of curvature (in feet) for R in the expression $\sqrt{R}$. It may be used to give the correct heat flux on a flat-faced cylinder axially aligned to the air stream by substituting four times the cylinder radius for R in the expression $\sqrt{R}$.

Since the SNAP-9A system is composed of flat-sided bodies as well as flat-faced cylinders, a means of investigating the heat flux to these surfaces needed to be established in order to allow a complete investigation of the burnup characteristics of the system. In subsequent investigations, then, heat transfer coefficients to flat plates (assuming laminar flow) were calculated by means of the Blasius flat-plate solution of the boundary layer equations and the modified Reynolds analogy as follows:^{2,3}

$$h = \frac{0.332 k}{x} (P_R)^{1/3} (R_{E_X})^{1/2} \quad (13)$$

where:

h = heat transfer coefficient, Btu/ft²/sec/°R

k = thermal conductivity, Btu/ft/sec/°R

P_R = Prandtl number, $C_p \mu / k$

R_{E_X} = local Reynolds number = $\frac{\rho u x}{\mu}$

C_p = specific heat, Btu/lb/°R

μ = viscosity coefficient, lb-sec/ft²

u = velocity, ft/sec

ρ = density, slugs/ft³

x = distance from leading edge of an ideal flat-plate, ft.

Flow properties were obtained using the reference enthalpy method^{2,4} to average out the effect of the temperature variation throughout the boundary layer. The reference enthalpy used was:^{2,2}

$$i^* = 0.22 i_r + 0.28 i_e + 0.5 i_w \quad (14)$$

where:

i_r = recovery enthalpy = $i_e + \bar{r} \frac{U_e^2}{2}$

i_e = air enthalpy at the outer edge of the boundary layer

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

U_e = velocity at the outer edge of the boundary layer

i_w = air enthalpy at the wall temperature

$\bar{r} = (P_R)^{1/2} = 0.85$ for laminar flow = recovery factor

The heat flux to the wall boundary of a three-dimensional, separated, laminar flow region is equal to 0.56 of the heat flux to a hypothetical wall at the boundary layer-separated region interface.²⁵ The sketch below illustrates this relationship:

Here,

$\dot{q}_{srw}$ = heat flux to a separated region boundary wall

$\dot{q}_{lbl}$ = heat flux through a boundary layer to a hypothetical wall at the dividing streamline.

Temperature profiles were calculated for portions of the vehicle using thin-skin theory²⁵ when applicable, that is:

$$\rho_s c_s \delta \frac{dT}{d\theta} = h (T_{aw} - T) - \sigma \epsilon T^4 \quad (15)$$

where

ρ_s = material density, lb/ft³

c_s = material specific heat, Btu/lb/°R

δ = material thickness

ϵ = material emissivity

σ = Stefan-Boltzman constant = 0.4761×10^{-12} Btu/ft²/sec/°R⁴

T = material temperature, °R

T_{aw} = adiabatic wall temperature, °R

h = heat transfer coefficient, Btu/ft²/sec/°R

θ = time, seconds.

These methods were next applied to an aerothermal analysis of Configuration 1. (A sample calculation showing how the application was carried out is given in the Appendix to this report.) It was conservatively assumed that the angle of attack was a stable 0°, with the instrumentation compartment of the system leading into the re-entry path.

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

The sequence of events for these conditions was calculated to be as follows:

1. The end plate of the satellite will melt at an altitude greater than 300,000 feet.
2. Assuming no heat transfer by conduction from the end plate of the satellite to its side walls, the side walls will reach their melting temperature at an altitude of 245,000 feet. The side walls will gradually recede from the most forward point (that is, adjoining the end plate of the satellite), to the most aft point (that is, adjoining the generator). The most aft point on the satellite will reach melting temperature at 231,000 feet.

This sequence of events had ramifications which were difficult to assess, however. For example, once the satellite end plate had melted, the interior of the instrumentation section would be exposed to the stagnation point heat flux. Furthermore, because of the hot ram air, the side walls of the satellite would receive heat from its interior. What this might contribute to the melting of the side walls had not been considered. Moreover, if the contents of the instrumentation compartment began to melt, the system's center of gravity (cg) would shift accordingly, until at some point the system would turn end for end and the generator would be facing into the air stream. Calculation could not show that, if this happened at 231,000 feet, the destruction of the fuel capsules and the dispersal of the fuel would be assured.

Obviously, an additional increment of altitude to accommodate burnup of the fuel capsule was necessary. This could be gained if the re-entry system could be made to re-enter with the generator forward (so that release of the fuel capsule would take place earlier). Adding a stabilizing surface or body to the instrumentation-compartment end of the system would cause it to do so. Fortunately, there was an adapter body between the Transit 5BN and Transit 5E satellites which was to have been jettisoned after the two satellites were separated, and it was decided simply to retain this body for the stabilizing purpose. The design of the Transit 5B system is shown, changed accordingly, as Configuration 2 in Figure 8.

Before much analytical work was done on this configuration, however, a design change was made on the adapter body, resulting in Configuration 3 (also shown in Figure 8). Test plans for shock-tunnel tests to determine force, moment, and heat transfer coefficients on the re-entry system had been scheduled on Configuration 2 but, fortunately, it was possible to change the test model configuration to conform to Configuration 3 without undue delay or additional expense.

Shock-tunnel tests on Configuration 3 were performed at the Cornell Aeronautical Laboratory. Figures 13 through 16 show one of the models used in these tests, the model mounted in the test section, and two Schlieren photographs taken during runs. The Appendix to this report gives the results of measurements of the heat transfer coefficients and compares them with calculated values of these coefficients.

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

Figure 13. Transit 5BN/SNAP-9A wind tunnel model (1/6 scale)

Figure 14. Model of Transit 5BN/SNAP-9A (1/6 scale) in 48-inch hypersonic shock tunnel
Note ceiling probes for measuring pressure and heating rate 8.5 inches above centerline of tunnel. Flow is from left to right.

~~CONFIDENTIAL~~

CONFIDENTIAL

Figure 15. Schlieren photograph of 1/6-scale Transit 5BN/SNAP-9A model at 3° angle of attack ($Re_L = 86180$, $M = 14.51$)

Figure 16. Schlieren photograph of 1/6-scale Transit 5BN/SNAP-9A model at 33° angle of attack ($Re_L = 97420$, $M = 14.79$)

CONFIDENTIAL

~~CONFIDENTIAL~~

The results of the force and coefficient measurements are summarized in Figures 17 through 22. Figures 15 and 18 show the static margins derived from these measurements. The data are shown as reduced by Cornell Aeronautical Laboratory for the center of gravity at Sta. 13.5. The data for center of gravity at Sta. 16.8 were obtained by transferring the data at Sta. 13.5 to Sta. 16.9. Although the exact position of the system's center of gravity is not known, the two positions considered should bound it.

However, a comparison of the calculated data with the measured data as plotted in Figures 17 through 22 does not show good agreement as far as magnitudes are concerned. More disturbing, the same trend of the data is not apparent in the case of moment coefficients and static margins for the generator-first orientation of Configuration 3. A study of calculative procedures will be made in the future to determine whether it is possible to predict force and moment coefficients more closely, or whether wind-tunnel tests will always be necessary where knowledge of system stability is an important consideration.

Figure 17. Measured and calculated normal and axial force coefficients versus angle of attack (Configuration 3: generator-first orientation)

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

Figure 18. Measured and calculated moment coefficient versus angle of attack (Configuration 3: generator-first orientation)

Figure 19. Measured and calculated static margins versus angle of attack (Configuration 3: generator-first orientation)

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

Figure 20. Measured and calculated normal and axial force coefficients versus angle of attack (Configuration 3: cone-first orientation)

Figure 21. Measured and calculated moment coefficient versus angle of attack (Configuration 3: cone-first orientation)

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

Figure 22. Measured and calculated static margins versus angle of attack (Configuration 3: cone-first orientation)

Shock-tunnel data show that Configuration 3, even with the center of gravity placed most optimistically, is only slightly stable in the generator-first orientation at angles of attack up to about 50 degrees. In the cone-first orientation, it has a still more pronounced instability. Although the dynamic stability characteristics of the system have not been studied, the static stability characteristics make it reasonable to expect the system to re-enter in the generator-first orientation and with considerable oscillatory motion.

Initial heat transfer calculations were made assuming the most conservative re-entry orientation of the system consistent with the results of the shock-tunnel tests: that is, generator-first re-entry, at 0° angle of attack.

Because the way in which the SNAP-9A generator is constructed makes several modes of disassembly possible, some of the details of its construction should be recalled here. (With the details of the generator's construction in mind, the modes of generator disassembly may be discussed more conveniently.) The lid of the generator body, then, is a bolted-on magnesium casting. Underneath it is a molded form of Min-K insulation. Between the insulation, and over each fuel capsule cavity, is a stainless-steel foil. Beneath these foils are the segmented carbon blocks which, collectively, form a heat accumulator. The carbon blocks in turn

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

are also surrounded by Min-K insulation. The outer side case of the generator is of magnesium, and the bottom of the generator is similar in construction to the top.

Temperature profiles of the various parts of the vehicle were computed according to thin-skin theory, assuming a 0° angle of attack during re-entry. The profiles of several points are illustrated in Figure 23. The heat transfer rates for Points I, II, and III were calculated assuming laminar-separated flow. For Point I, the theoretical calculations were verified by the experimental evidence (see the Appendix to this report). However, Point III heat transfer rates, calculated by laminar-separated flow theory, were found to be substantially lower than those measured during the wind-tunnel tests (see Appendix, p. 57). The reason for this discrepancy is believed to be an increase in heat rate caused by a boundary-layer shock-wave interaction.

Figure 23. SNAP-9A generator re-entry temperature profiles

Since the experimental data indicated that the theoretical computation of the temperature profile of Point III had been too conservative (see Appendix, Figure 28), it was necessary to alter the time at which Point III attained melting temperature. The majority of the experimental values for Point III were higher than the theoretical line of Point I, so it was arbitrarily assumed that Point III would reach melting temperature at the same time as Point I, and that at that time, the entire generator shell would fail and fall away. With this assumption, the altitude at which the

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

generator shell would melt was calculated to be approximately 270,000 feet: that is, 770 seconds from the initial altitude of 400,000 feet. The generator cover, which would melt before the generator shell because it receives stagnation point heating, would reach melting temperature at approximately 390,000 feet: that is, 80 seconds from the assumed starting altitude of 400,000 feet.

At this point, assuming a 0° angle of attack, the interior of the generator would be exposed. If it is further assumed that the Min-K insulation at the top of the generator would be removed by aerodynamic forces, then the stagnation point heat flux would act on the interior materials of the generator. It will be remembered that the materials in the generator are stainless steel, titanium, Haynes-25 alloy, tantalum, graphite, and plutonium. How these materials may interact when heated in contact with one another is largely unknown, but it may be postulated that some of them, when melted, may combine, flow into the gaps between the graphite segments, and solidify, locking the graphite segments together. Such an interaction could protect the fuel capsules and prevent the fuel from being dispersed at a sufficiently safe altitude.

Since not much information on these interactions is available, a test program was initiated to obtain the necessary basic information. Once the specimens from this test, which was conducted under static conditions, had cooled, it was found that some bonding had taken place between the materials of which the generator is constructed. Whether bonding could or would occur under dynamic conditions is not known. One potentiality, of course, is of particular interest: that is, the possibility that the titanium washers might remain in place and so shield the fuel capsules from heating effects. If this happened, the exterior of the fuel capsule would not reach melting temperature while the capsule was contained in the heat accumulator block. Temperature profiles of the fuel capsules for this condition are shown in Figure 24. In this figure, the fuel is shown in three possible locations within the capsule: Configuration 1A is the normal fuel position, Configuration 2A is its most likely position once heating has started, and Configuration 3A is the position which would result in the highest capsule temperatures.

The foregoing discussion explains why an exact description of the sequence of disassembly of the Transit 5BN system and the SNAP-9A generator is not possible. In any given number of re-entries of this system, probably no two disassembly sequences would be exactly alike. However, the fuel capsules will probably be released when the generator shell is destroyed: that is, at an altitude of 270,000 feet.

Once the fuel capsule is released, the altitude increment necessary for melting the Haynes-25 outer shell can be calculated as follows. Assuming a tumbling, spinning cylinder:¹⁶

$$\dot{q}_{\text{cyl}} \text{ (surface averaged)} = \dot{q}_s \text{ (sphere-stagnation point)} \times 0.132, \text{ where sphere and cylinder have the same radius}$$

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

$\dot{q}'_s$ (sphere with 0.49-inch radius) = 4.95 $\dot{q}'_s$ (sphere with 1-foot radius)
 $\dot{q}'_{cyl}$ (cylinder with 0.49-inch radius) = 0.658 $\dot{q}'_s$ (sphere with 1-foot radius)
Initial temperature of fuel capsule = 1000°F
Weight of Haynes-25 shell = 0.68 pound
Weight of tantalum = 0.15 pound
Weight of plutonium = 0.47 pound
Specific heat of Haynes-25 = 0.092 Btu/lb/°F
Specific heat of tantalum = 0.034 Btu/lb/°F
Specific heat of plutonium = 0.034 Btu/lb/°F
Melting temperature of Haynes-25 = 2570°F
Surface area of capsule = 18.46 in.² = 0.128 ft²

Figure 24. Temperature profiles of the fuel capsules with the fuel in three possible locations

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

To bring entire capsule to 2570°F:

$$\text{Btu} = 1570 \times 0.68 \times 0.092 + 1570 \times 0.62 \times 0.034 = 98.2 + 33.1 = 131.3.$$

On a Q/ft^2 basis:

$$\text{Btu} = 131.3/0.128 \times 0.658 = 1560 \text{ Btu required.}$$

Integrating the $\dot{q}$ curve from 270,000 feet, 1560 Btu are obtained after approximately 27.3 seconds, and at an altitude of 264,000 feet.

At this point the tantalum liner is assumed to be exposed. An estimate of the time necessary to dispose of the tantalum cannot be based on the time required for pure tantalum to reach its melting point, because the rate of radiant heat loss involved in this situation is such that pure tantalum never can reach its melting temperature.

To obtain information on the mechanism whereby tantalum might be disposed of during the re-entry, a series of tests was performed in a hyperthermal arc-powered plasma tunnel. The data from these tests is being reduced and it will be made available in a forthcoming report. The tests demonstrated that, in the presence of oxygen and during simulated re-entry conditions, an oxide of tantalum is formed which melts at a much lower temperature (approximately 3400°F) than pure tantalum.

On the basis of these test results, it is possible to predict when the entire tantalum liner will be consumed. Assuming oxide formation, and melting, the important considerations for tantalum burnup become surface temperature, the flow of oxygen mass to the surface, and time. Enough oxygen must be available at the surface to form oxide for the entire thickness of the tantalum and, in addition, it must be available during the time that the tantalum surface temperature is equal to or above the melting temperature of the oxide.

A computer prediction, shown in Figure 25, has been made assuming capsule release at 270,000 feet. This computation indicates that the tantalum will reach 3400°F in 40 seconds, at approximately 260,000 feet. (This temperature is conservative, since it is based entirely on aerodynamic heating and neglects energy released by the oxidation process, whereas the arc-tunnel tests previously mentioned definitely indicate that surface temperature is directly dependent on oxygen flow.) To determine the amount of oxygen required to oxidize the 20-mil thickness of tantalum, assuming formation of Ta_2O_5 , the following analysis was used:

$$t = \frac{4}{5} \frac{(\text{At Wt})_{\text{TA}}}{\rho_{\text{TA}}} \frac{1}{(\text{At Wt})_{\text{ox}}} \int_{\theta_1}^{\theta_2} W_{\text{ox}} d\theta \quad (16)$$

where

t = tantalum thickness, feet

$(\text{AtWt})_{\text{TA}}$ = atomic weight of tantalum = 181 gm/mole

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

$(AtWt)_{OX}$ = molecular weight of oxygen 32 gm/mole

ρ_{TA} - tantalum density, 1034 lb/ft³

θ = time, seconds

W_{OX} = mass flux rate of oxygen at tantalum surface, lb/ft²-sec

then

$$t - \frac{0.020}{12} = \frac{4}{5} \frac{181}{1034} \times \frac{1}{32} \int_{\theta_1}^{\theta_2} W_{OX} d\theta$$

or

$$\int_{\theta_1}^{\theta_2} W_{OX} d\theta = 0.38 \frac{lb}{ft^2}$$

This, then, is the amount of oxygen that must strike the surface of the capsule during the period of time when the surface temperature is equal to or higher than 3400°F.

Figure 25. Surface temperature and oxygen flux to capsule surface for capsule release at 270,000 feet

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

Again, tests performed in the hyperthermal tunnel demonstrated that the oxide will melt and that in melted condition it is readily stripped away. No surface formation appeared that might inhibit or preclude continuation of the oxide melting process.

Heat transfer to a body undergoing re-entry may be expressed by

$$q_s = C_H \rho V (H_s - H_w) \quad (17)$$

or, for cold wall heating,

$$q = C_H \rho V H_s \quad (18)$$

where:

- q_s - aerodynamic heating, Btu/ft²/sec
- C_H = Stanton number, dimensionless, $\frac{h}{\rho V c_p}$
- V = free-stream velocity, fps
- H_s = total gas enthalpy, Btu/lb
- ρ = free-stream air density, lb/ft³
- h = local film coefficient, Btu/ft²/°F/sec

Then the rate at which the gas strikes the surface of the material to release its energy is

$$C_H \rho V = q_{av}/H_s = 0.133 \frac{q_s}{H_s} \quad (18)$$

And, for air, the rate at which oxygen strikes the surface of the material is:

$$(C_H \rho V)_{ox} = 0.21 (C_H \rho V) = 0.21 \frac{(0.133 q_{av})}{H_s} \quad (19)$$

This relationship is shown in Figure 25 as computed for a capsule release at 270,000 feet. The integral of this curve represents the total amount of oxygen striking the capsule surface per square foot. With the integration process beginning at 40 seconds, enough oxygen will have come into contact with the surface after 190 additional seconds to form an oxide of the 20-mil thick tantalum wall. This oxide will melt and be swept away, with the entire process being completed by 190,000 feet.

At this time and altitude the plutonium fuel is exposed, and if a large excess of energy is simultaneously available, consumption of the fuel is assured. A convenient way to explore this possibility is to look at the energy available and assume a conservative amount of it to be converted to heat. The kinetic energy of the plutonium is $1/2 mV^2$ or 0.12×10^6 foot-pounds. The heat equivalent of this

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

energy is 11,700 Btu. Assuming that 1 percent of this energy is available for conversion to heat to be applied to the plutonium, 117 Btu would be available.

The heat of vaporization of plutonium is 600 Btu/lb. To vaporize the 0.47 pound of plutonium present would require 282 Btu. The preceding calculations for disposal of the Haynes-25 alloy and the tantalum assumed that the plutonium was brought to a temperature of 3400°F. The boiling point of plutonium is 5855°F. To bring the 0.47 pound of plutonium from 3400° to 5855°F, neglecting radiation losses, would require 39 Btu. The amount of heat necessary to vaporize the plutonium, then, would be 639 Btu. Since a 1 percent energy conversion factor is not too conservative, assuming continuum flow, complete vaporization of the plutonium cannot be shown on this basis. Heat contributed by possible oxidation processes is not included because the reaction of plutonium in an oxygen-poor atmosphere is not known.

Another investigation assumed 1000-micron particles, an initial velocity of 23,000 feet per second, and continuum flow. On the basis of these assumptions, particle trajectories were run and heat inputs were computed. The total available heat was found to be 682.8×10^{-4} Btu per particle. If the 1000-micron particles are assumed to be plutonium metal with an emissivity of 0.6, it can be shown that the plutonium temperature will not exceed its boiling point, and therefore, will not vaporize from aerodynamic heating alone.

These specific conditions were assumed because, at one stage of the SNAP-9A safety analysis, they appeared to be those under which the plutonium would be exposed. Actually, however, the problem is considerably more complicated. Oxidation processes may contribute heat to the particle. However, the heat needed for vaporization is high, and it is difficult to rationalize the vaporization of the plutonium on this basis. On the other hand, minute particles may be in the free molecular flow régime where the energy conversion factor is higher than it is for continuum flow. In addition, the 1000- μ particle size is merely an assumption.

Knudsen number is a convenient means by which to judge the applicable flow régime. It is defined as the ratio of mean free molecular path length to a characteristic body dimension. If the Knudsen number is above 10, the flow régime may be considered free molecular. If the Knudsen number is below 1, the flow régime is continuum. A Knudsen number between 1 and 10 indicates a transition region. At 218,000 feet the Knudsen number for a 1000- μ particle is 1.6, which indicates that continuum flow is the more likely.

As far as the validity of assuming a 1000- μ particle size is concerned, two methods are available for estimating particle size: the Öpik method,²⁶ and a method proposed by John Beal of Cornell Aeronautical Laboratories.³⁵ Öpik's method is probably not strictly applicable for the calculation of particle size at re-entry velocities; but assuming such velocities, his method results in an estimated particle size of 475 μ at 218,000 feet and 23,000 feet per second velocity. For these same conditions, Beal's methods result in a particle size of 950 μ .

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

On the basis of the preceding discussion, some conclusions as to the events most likely may be drawn. Whether the plutonium is exposed early or late appears immaterial. It cannot be shown to vaporize to molecular form on the basis of known characteristics, but it will probably reach the ground as an oxide rather than a metal. In any case, the particle sizes of plutonium reaching the ground will probably be too large for immediate inhalation.

Configuration 4 (Figure 9) is the Transit system as it will finally be flown. The large fins were added to compensate for the aerodynamic effects of a large-diameter antenna placed on the generator end of the system. While this antenna will burn away quite rapidly once it is exposed to the stagnation heat flux, it was realized that it might cause the system to re-enter with the generator in the trailing position and thus delay the release of the fuel capsules. The large fins will also burn away rapidly, but they should survive longer than the antenna. Therefore, the configuration of interest below 400,000 feet will still be Configuration 3.

~~CONFIDENTIAL~~

CHAPTER IV. SUMMARY AND CONCLUSIONS

Safety analyses of the operational SNAP-9A for use on Transit satellite missions have revealed that for some postulated accident conditions the level of risk can be satisfactorily evaluated, while for other conditions, because data are lacking, the level of risk is indeterminate.

In all cases, the hazards associated with SNAP-9A are ones which would follow from a release of the plutonium fuel in a form to present an inhalation hazard (Figure 26). The problem of fuel release for environments resulting from accidents on the launch pad or from early flight aborts has been shown to be less than 1×10^{-6} .

Figure 26. Event chart for re-entry from a short-lived orbit

~~CONFIDENTIAL~~

Re-entry from a nominal 600-mile circular orbit with a lifetime in excess of 900 years may be discounted because in that case the fuel would have decayed almost completely to U^{234} .

A short-lived orbit (less than one month) could result if the Ablestar stage failed to provide second burn. Likewise, if the Ablestar failed to complete first burn or if its thrust cut off prematurely during the last 10 seconds, orbit would not be achieved and there would be an Instantaneous Impact Point (IIP) on Africa. An identical approach can be used to evaluate the hazards associated with both these malfunctions. The probability of no second burn, which would result in a short orbit, has been estimated as 13×10^{-3} . The probability of premature first burn cutoff, however, has not been estimated for lack of data. The expected sequence of events during re-entry is (1) re-entry burnup and disassembly of the generator, (2) release of the fuel capsules, and (3) burnup of the fuel at high altitude.

The following uncertainties remain, however, after this safety analysis:

1. Can material interactions, known to be possible, occur during re-entry in such a way that the graphite heat accumulator blocks do not disassemble?
2. The tantalum liner of the fuel capsules will probably not reach melting temperature; therefore, oxidation must be depended upon for disposal. Since the mechanisms of tantalum oxidation in a re-entry environment are not completely understood, the results of the analysis herein, should be used with care. Test results, however, indicate that the analysis yields conservative results.
3. Breakup of the plutonium fuel after release to the re-entry environment is not completely understood. Estimates of the size of the molten metal droplets range from millimeters to submicron sizes if vaporization occurs. Undoubtedly, the oxidation processes will affect the final particle size, but the mechanisms are not well understood for this environment.
4. If the capsules reach the earth in a damaged condition, so as to expose the fuel upon or after impact, the resulting oxidation processes have been conservatively assumed to produce an airborne aerosol of inhalable size. However, there is little if any data to support this assumption.

In the last analysis, the uncertainties cited above could be eliminated as potential hazards if it could be shown that the production of an inhalable aerosol from either large masses of plutonium metal or from smaller masses of plutonium-oxide would not result in significant airborne concentrations, or if the final fuel form is designed to preclude the formation of inhalable particles. This conclusion follows from the finding of the ad-hoc committee, which declared that the biological hazard from plutonium-oxide results exclusively from inhalation, inhalable particles being those smaller than 3 microns in diameter.

~~CONFIDENTIAL~~

Blank Page

~~CONFIDENTIAL~~

APPENDIX

Sample Calculation of Aerodynamic Heating on the Original Configuration
of the 5BN/SNAP-9A Under Assumption of Worst Conditions
for Fuel Dispersion (i.e., Rearward Re-entry)

~~CONFIDENTIAL~~

Blank Page

~~CONFIDENTIAL~~

APPENDIX

Sample Calculation of Aerodynamic Heating on the Original Configuration of the 5BN/SNAP-9A Under Assumption of Worst Conditions for Fuel Dispersion (i.e., Rearward Re-entry)

The original configuration appeared as follows (half cross section of appropriate roll orientation):

For initial consideration, the points of interest are:

1. Surface (a)
2. Surfaces (b)
3. Points (c), (d), and (e).

The methods used to calculate heat transfer rates for these points and surfaces are outlined below.

Surface (a)

Surface (a) is a flat-face cylinder. The heat transfer rate on the flat face of a flat-face cylinder with axis parallel to the velocity ranges from $\dot{q}_{\text{St.Pt.}}$, the stagnation point heat transfer rate at the centerline, to $1.5 \dot{q}_{\text{St.Pt.}}$ ²⁷ at the lip of the flat face, i.e., Point (f). The increase at the lip is largely due to expansion

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

at that point. However, another body of larger radius, the satellite, is directly behind the cylinder, and its influence at the lip of the cylinder is significant. Assuming that the flow separates at Point (f) and reattaches at Point (g), then the flow turns $20^{\circ}41'$ at the lip instead of the 90 degrees that it would turn without the satellite. The expansion is cut significantly and as a result the heat transfer rate at the lip is lowered. Consequently, the heat transfer rate over the entire flat face is assumed to be the same as that at the stagnation point, $\dot{q}_{\text{St. Pt.}}$. This distribution is slightly conservative (i.e., more energy is actually transmitted to the vehicle), but not enough to affect the outcome. The stagnation point heat transfer rate was calculated from Chapman:^{2,4}

$$\dot{q}_{\text{St. Pt.}} \sqrt{R_{\text{eff}}} = 17,600 \left(\frac{\rho_{\infty}}{\rho_0} \right)^{1/2} \left(\frac{V_{\infty}}{V_s} \right)^{3.15} \quad (20)$$

where:

$\dot{q}_{\text{St. Pt.}}$ = stagnation point heat transfer rate, Btu/ft² sec

R_{eff} = effective radius, ft

ρ_{∞} = free-stream density

ρ_0 = sea level density

V_{∞} = free-stream velocity

V_s = satellite velocity at altitude in question = $\sqrt{gr}$

g = acceleration of gravity at altitude in question

r = radius from center of earth to vehicle.

Surfaces (b)

Surfaces (b) are affected by the assumption that the flow separates at Point (f) and reattaches at Point (g). Chapman gives a description of heat transfer in regions of separated flow which, verbally, is somewhat confusing. His account states that the flow within a separated region which is three-dimensional and bounded on one side by a laminar boundary layer is such that the heat transfer rates to the walls forming the other boundaries of the separated region are equal to 0.56 percent of the laminar heat transfer rate which would be incident on a hypothetical wall at the interface between separated region and boundary layer. This verbal explanation may be clarified, however, by the following illustration:

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

Chapman's results indicated that, under laminar conditions, the three-dimensional laminar separated region would be:

$$\frac{\dot{q}_b}{\dot{q}_{\text{Lam. B. L.}}} = 0.56 . \quad (21)$$

However, because the flow has a tendency to become two-dimensional, this ratio will increase as Point (g) is approached; and in addition, any unsteady pulsation of the flow field could cause great variation in the heat transfer rates to Surfaces (b).

Nevertheless, the three-dimensional-separated flow approach is the most promising for two reasons: (1) it is applicable for the longest time, and (2) it is conservative, i.e., more energy is actually transmitted to the vehicle than it indicates.

In order to calculate a heat transfer rate at intermediate points between Points (f) and (g), an effective length at Point (f) must be determined. The expansion corner at Point (g) is, in effect, a discontinuity through which momentum must be conserved. Consequently:²⁸

$$\rho_{e_1} u_{e_1}^2 \delta_1^{**} = \rho_{e_2} u_{e_2}^2 \delta_2^{**} \quad (22)$$

for an expansion where

ρ_e = density at boundary layer edge

u_e = velocity at boundary layer edge

δ^{**} = momentum displacement thickness

$\delta^{**} = \delta^* + \theta$

where

$$\delta^* = \int_0^\infty \left(1 - \frac{\rho u}{\rho_e u_e} \right) dy, \text{ the displacement thickness}$$

$$\theta = \int_0^\infty \frac{\rho u}{\rho_e u_e} \left(1 - \frac{u}{u_e} \right) dy$$

y = distance normal to surface.

Consequently,

$$\delta^{**} = \int_0^\infty \left(1 - \frac{\rho u^2}{\rho_e u_e^2} \right) dy \quad (23)$$

()₁ = conditions upstream of discontinuity

()₂ = conditions downstream of discontinuity.

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

For compressible flow on a flat plate, the relationship for momentum displacement thickness (laminar flow) is:

$$\delta^{**} = 2.394 \left(\frac{\mu}{\rho u_e} \right)^{1/2} \ell^{1/2} \quad (24)$$

The corresponding equation for a cone is

$$\delta^{**} = 1.38 \left(\frac{\mu}{\rho u_e} \right)^{1/2} \ell^{1/2} \quad (25)$$

The representative length at Point (f) is therefore:

$$\ell_{2f} = 3 \left(\frac{\rho_{e1}}{\rho_{e2}} \right)^2 \left(\frac{u_{e1}}{u_{e2}} \right)^3 \left(\frac{\mu_1^*}{\mu_2^*} \right) \left(\frac{\rho_2^*}{\rho_1^*} \right) \ell_{1f} \quad (26)$$

where ℓ_1 is the representative length for the flow upstream of the discontinuity at Point (g). The flow characteristics and properties which have a ()^{*} superscript are to be evaluated at reference enthalpy i^* as Eckert²⁹ suggested:

$$i^* = 0.22 i_r + 0.28 i_e + 0.5 i_w \quad (27)$$

where:

$$\begin{aligned} i_r &= \text{recovery enthalpy} = i_e + r \frac{u_e^2}{2} \\ i_e &= \text{enthalpy at outer edge of boundary} \\ i_w &= \text{enthalpy at the wall} \\ r &= (P_r^*)^{1/2} \cong 0.85 \text{ for laminar flow} = \text{recovery factor.} \end{aligned}$$

The heat transfer coefficient at the imaginary conic surface between Points (f) and (g) was determined by the modified Reynolds analogy and the results of the Blasius flat plate solution²³

$$h_{f.p.} = \frac{0.332 k^*}{x} \left(P_r^* \right)^{1/3} \left(R_{e_x}^* \right)^{1/2} \quad (28)$$

where:

$$\begin{aligned} k^* &= \text{local thermal conductivity at Point } x \text{ evaluated at the reference enthalpy, } i^*, \text{ Btu/ft-sec-}^\circ\text{R} \\ P_r^* &= \text{Prandtl number at } x \text{ evaluated at reference enthalpy,} \\ i^*, P_r^* &= \frac{C_p^* \mu^*}{k^*}, \text{ dimensionless} \\ R_{e_x}^* &= \frac{\rho^* x v}{\mu^*} = \text{Reynolds number evaluated at reference enthalpy, dimensionless} \end{aligned}$$

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

ρ^* = density evaluated at reference enthalpy, slugs/ft³

μ^* = viscosity evaluated at reference enthalpy, lb-sec/ft²

v = velocity, ft/sec

C_p^* = specific heat at constant pressure evaluated at reference enthalpy, Btu/lb-°R

$x = \ell_2$ + distance from discontinuity along flat plate or, in this case, where the heat transfer coefficient is to be transferred to a cone, the distance from the discontinuity to the point in question back along the cone, ft.

The heat transfer coefficient on the conic section between (f) and (g) may now be calculated since, as a result of the Mangler transformation,²⁵

$$h_c = \sqrt{3} \ h_{f.p.}$$

where h_c is the heat transfer coefficient on the imaginary conic section between (f) and (g).

The heat transfer rates to the Surfaces (b) may now be calculated:

$$q_b = 0.56 \ h_c \ (T_r - T_w) \quad (29)$$

where:

$$T_r = \text{recovery temperature} = T_e \left(1 + r \frac{\gamma-1}{2} M_e^2 \right)$$

T_w = wall temperature

T_e = ambient temperature at boundary layer edge, °R

M_e = Mach number at edge of boundary layer

The computation of $h_{f.p.}$ must be performed using the characteristics of the point in question on the cone when it is to be transformed to the cone.

Points (c), (d), and (e)

The heat transfer rates at Points (c), (d), and (e) may be determined by the flat plate relation (Eq. 11). The x is now determined by adding the distance from Point (g) back to (c), (d), or (e), to the equivalent length at the point of discontinuity, (g), which is

$$\ell_{2g} = \frac{1}{3} \left(\frac{\rho_{e1}}{\rho_{e2}} \right)^2 \left(\frac{u_{e1}}{u_{e2}} \right)^3 \left(\frac{\mu_1^*}{\mu_2^*} \right) \left(\frac{\rho_2^*}{\rho_1^*} \right) \ell_{1g} \quad (30)$$

where

$$\ell_{1g} = \ell_{2f} + \text{distance from (f) to (g)} = \ell_{2f} + 0.236 \text{ ft.} \quad (14)$$

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

Sample Calculation

For a sample calculation, consider the conditions at 250,000 feet where

$$V_{\infty} = 25,000 \text{ ft/sec} - \text{free-stream velocity}$$

$$M = 28.15$$

$$\dot{q}_{\text{St.Pt.}} \sqrt{R_{\text{eff}}} = 85.52$$

$$R_{\text{eff}} = 4 R_{\text{cyl}} = 4 \times \frac{6.5}{12} = 2.167 \text{ ft}$$

$$\sqrt{R_{\text{eff}}} = 1.472$$

$$\dot{q}_{\text{St.Pt.}} = 58.10 \text{ Btu/ft}^2\text{-sec.}$$

In order to determine the properties upstream and downstream of Point (f), it was assumed that the flow is sonic in the vicinity of Point (f), and that it then expands through an angle of $20^\circ 19'$. Then:³⁰

$$i_{\text{total St.Pt.}} = (160.1)(1.9636 \times 10^6) \frac{\text{ft}^2\text{-lb}}{\text{lb-sec}^2} = 314.4 \times 10^6 \frac{\text{ft}^2}{\text{sec}^2}$$

$$P_{\text{total St.Pt.}} = (1080)(4.364 \times 10^{-2} \text{ lb/ft}^2) = 47.13 \text{ lb/ft}^2$$

$$T_{\text{total St.Pt.}} = (32.95)(328.2)^\circ\text{R} = 1.081 \times 10^4 \text{ }^\circ\text{R}$$

$$\rho_{\text{total St.Pt.}} = (18.45)(7.748 \times 10^{-8}) \text{ lb-sec}^2/\text{ft}^4 = 143.0 \times 10^{-8} \frac{\text{lb-sec}^2}{\text{ft}^4}$$

$$Z_{\text{St.Pt.}} = 1.733 = \frac{P_{\text{St.Pt.}}}{\rho_{\text{St.Pt.}} T_{\text{St.Pt.}}}$$

Since the Mach number is assumed to be equal to one just upstream of the discontinuity, then:³¹

$$T_{e_1} = 0.8333 T_{\text{total St.Pt.}} = (0.8333)(10810)^\circ\text{R} = 9008^\circ\text{R}$$

$$\rho_{e_1} = 0.6339 \rho_{\text{St.Pt.}} = (0.6339) \left(143.0 \times 10^{-8} \frac{\text{lb-sec}^2}{\text{ft}^4} \right) = 90.65 \times 10^{-8} \frac{\text{lb-sec}^2}{\text{ft}^4}$$

$$P_{e_1} = 0.5283 P_{\text{total St.Pt.}} = (0.5283)(47.13) \frac{\text{lb}}{\text{ft}^2} = 24.90 \frac{\text{lb}}{\text{ft}^2}$$

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

$$u_{e_1} = 49.1 \sqrt{T_{e_1}} = 49.1 \sqrt{9008} = (49.1)(94.91) \frac{\text{ft}}{\text{sec}} = 4660 \frac{\text{ft}}{\text{sec}}$$

$$i_{e_1} = i_{\text{total St. Pt.}} - \frac{u_{e_1}^2}{2} = 314.4 \times 10^6 - \frac{21.71 \times 10^6}{2}$$

$$= (314.4 - 10.65) \times 10^6 = 303.8 \times 10^6 \frac{\text{ft}^2}{\text{sec}^2}$$

$$i_{r_1} = i_{e_1} + r \frac{u_{e_1}^2}{2} = 303.8 \times 10^6 + (0.85) \frac{21.71 \times 10^6}{2} = (303.8 + 9.23) \times 10^6$$

$$= 313.0 \times 10^6 \frac{\text{ft}^2}{\text{sec}^2}$$

Assuming a hot wall of 1000°R where $C_p = 0.25 \frac{\text{Btu}}{\text{lb}^\circ\text{R}}$,

$$i_{w_1} = 0.25 \frac{\text{Btu}}{\text{lb}^\circ\text{R}} (1000^\circ\text{R}) = 250 \times 778 \times 32.2 = (194.5)(10^3)(32.2) \frac{\text{ft}^2}{\text{sec}^2}$$

$$= 3.22 \times 1.945 \times 10^6 \frac{\text{ft}^2}{\text{sec}^2} = 6.263 \times 10^6 \frac{\text{ft}^2}{\text{sec}^2}$$

$$i_1^* = (0.22) i_{r_1} + (0.28) i_{e_1} + (0.5) i_w = (0.22)(313.0 \times 10^6) + (0.28)(303.8 \times 10^6)$$

$$+ (0.5)(6.263 \times 10^6) = (68.9 + 85.1 + 3.13) \times 10^6 \frac{\text{ft}^2}{\text{sec}^2} = 157.1 \times 10^6 \frac{\text{ft}^2}{\text{sec}^2}$$

$$i_1^* = \frac{157.1 \times 10^6 \text{ ft}^2/\text{sec}^2}{(778)(32.2) \text{ ft}^2/\text{sec}^2 - \text{Btu}} = \frac{157.1 \times 10^3}{(0.778)(32.2)} \frac{\text{Btu}}{\text{lb}}$$

$$\frac{Z_{i_1}^*}{RT} = \frac{i_1^* (\text{Btu})}{(0.0636) T_{e_1} (^\circ\text{R})} = \frac{157.1 \times 10^3}{(0.778)(32.2)(68.6) \times 9.008}$$

$$= \frac{157.1}{(0.778)(3.22)(6.86)(0.9008)} = \frac{157.1}{15.6} = 10.1$$

$P_{e_1} \cong 0.01$ atmospheres; hence, the properties in question may be determined from Hansen³² as follows:

$$T_1^* = 9000^\circ\text{R} \quad P_{e_1} \cong 0.01 \text{ atmos.}$$

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

$$\rho_1^* = \frac{P_{e_1}}{RT_1^* Z} = \frac{24.9}{(1716)(9000)(1.359)} \frac{\text{slug}}{\text{ft}^3} = \frac{2.49 \times 10^{-6}}{(1.719)(0.9)(1.359)} \frac{\text{slug}}{\text{ft}^3}$$
$$= \frac{2.42}{2.102} \times 10^{-6} \frac{\text{slug}}{\text{ft}^3} = 1.151 \times 10^{-6} \frac{\text{lb-sec}^2}{\text{ft}^4}$$

$$\mu_1^* = (2.11)(1.074) \times 10^{-6} \frac{\text{lb-sec}}{\text{ft}^2} = 2.27 \times 10^{-6} \frac{\text{lb-sec}}{\text{ft}^2}.$$

The characteristics downstream of the corner were determined using the Prandtl-Meyer expansion theory at the discontinuity where the flow expands through an angle of $20^\circ 19'$. The Prandtl-Meyer function upstream of the point of discontinuity, ν_1 is equal to 0 while the function downstream of the discontinuity, ν_2 is equal to $20^\circ 19'$. Therefore, again from NACA report 1135,³¹

$$M_2 = 1.79, P_{e_2} = P_t (0.1767), \rho_{e_2} = 0.2900(\rho_t)$$

$$T_{e_2} = 0.6095 T_t = (0.6095)(1.081 \times 10^4 \text{ } ^\circ\text{R}) = 6590^\circ\text{R}$$

$$P_{e_2} = (47.13)(0.1767) = 8.33 \frac{\text{lb}}{\text{ft}^2}$$

$$\rho_{e_2} = (0.29) \left(143 \times 10^{-8} \frac{\text{lb-sec}^2}{\text{ft}^4} \right) = 41.5 \times 10^{-8} \frac{\text{lb-sec}^2}{\text{ft}^4}$$

$$u_{e_2} = a_{e_2} M_2 = (49.1) \sqrt{6590} (1.79) = (87.89)(81.18) \frac{\text{ft}}{\text{sec}} = 7135 \text{ ft/sec}$$

$$i_{e_2} = i_{\text{total}} - \frac{u_{e_2}^2}{2} = 314.4 \times 10^6 \frac{\text{ft}^2}{\text{sec}^2} - \frac{50.91}{2} \times 10^6$$
$$= (314.4 - 25.45) \times 10^6 \frac{\text{ft}^2}{\text{sec}^2} = 290.0 \times 10^6 \frac{\text{ft}^2}{\text{sec}^2}$$

$$i_{r_2} = i_{e_2} + r \frac{u_{e_2}^2}{2} = \left[290.0 + (0.85)(25.45) \right] \times 10^6 = \left[290 + 21.6 \right] \times 10^6$$
$$= 311.6 \times 10^6 \frac{\text{ft}^2}{\text{sec}^2}$$

$$i_{w_2} = 6.26 \times 10^6$$

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

$$\begin{aligned} i_2^* &= 0.22 i_{r_2} + 0.28 i_{e_2} + 0.5 i_{w_2} \quad (68.5 + 81.2 + 3.3) \times 10^6 \frac{\text{ft}^2}{\text{sec}^2} \\ &= 153 \times 10^6 \frac{\text{ft}^2}{\text{sec}^2} \end{aligned}$$

$$i_2^* = \frac{153 \times 10^6}{778 \times 32.2} \frac{\text{Btu}}{\text{lb}} = \frac{153 \times 10^3}{25.05} = 6110 \frac{\text{Btu}}{\text{lb}}$$

$$\frac{Z i_a^*}{RT} = \frac{6110}{(0.0686)(6590)} = \frac{0.9271}{0.0686} = 13.5$$

$$T_{e_2} = 6590^\circ \text{R} = 3660^\circ \text{K} \quad P_{e_2} = \frac{8.33}{2116} = 0.004 \text{ atmosphere}$$

$$\begin{aligned} \rho_2^* &= \frac{P_{e_2}}{RT_1^* Z_2} = \frac{8.33}{(1716)(8100)(1.337)} \frac{\text{lb-sec}^2}{\text{ft}^4} = \frac{8.33 \times 10^{-7}}{(1.716)(0.81)(1.337)} \\ &= \frac{8.33 \times 10^{-7}}{1.89} = 4.41 \times 10^{-7} \frac{\text{lb-sec}^2}{\text{ft}^4} \end{aligned}$$

$$\mu_2^* = (1.055)(1.999) \times 10^{-6} \frac{\text{lb-sec}}{\text{ft}^2} = 2.11 \times 10^{-6} \frac{\text{lb-sec}}{\text{ft}^2}$$

Consequently,

$$\ell_{2f} = 3 \left(\frac{\rho_{e_1}}{\rho_{e_2}} \right)^2 \left(\frac{u_{e_1}}{u_{e_2}} \right)^3 \left(\frac{\mu_1^*}{\mu_2^*} \right) \left(\frac{\rho_2^*}{\rho_1^*} \right) \ell_{1f}$$

$$\frac{\rho_{e_1}}{\rho_{e_2}} = \frac{9.06 \times 10^{-7}}{4.15 \times 10^{-7}} = 2.18, \quad \left(\frac{\rho_{e_1}}{\rho_{e_2}} \right)^2 = 4.77$$

$$\frac{u_{e_1}}{u_{e_2}} = \frac{4660 \text{ ft/sec}}{7135 \text{ ft/sec}} = 0.6531, \quad \left(\frac{u_{e_1}}{u_{e_2}} \right)^2 = 0.279$$

$$\frac{\mu_1^*}{\mu_2^*} = \frac{2.27 \times 10^{-6}}{2.11 \times 10^{-6} \text{ lb-sec/ft}^2} = 1.075$$

$$\frac{\rho_2^*}{\rho_1^*} = 0.383$$

$$\ell_{2f} = (3)(4.77)(0.279)(1.075)(0.383) \ell_{1f} = 1.64 \ell_{1f}$$

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

Using the frontal radius as the ℓ_{1f} , it was found that

$$\ell_{2f} = (1.64) \frac{6.5}{12} = (1.64)(0.542) = 0.889 \text{ ft.}$$

proceeding to the heat transfer coefficient,

$$x = \ell_{2f} + \Delta x.$$

Let Δx be half the distance between (f) and (g), or

$$x = 0.889 + 0.118 = 1.007 \text{ ft}$$

$$\begin{aligned} R_{ex}^* &= \frac{u_e \rho_2^* x}{\mu_2^*} = \frac{(7135)(4.41 \times 10^{-7})(1.007)}{(2.11 \times 10^{-6})} \\ &= \frac{(7135)(0.441)(1.007)}{2.11} = (3382)(0.441)(1.007) = 1500 \end{aligned}$$

$$(R_{ex}^*)^{1/2} = 38.7$$

from Hansen³²

$$K^* = (10)(21) \times 10^{-6} \frac{\text{Btu}}{\text{ft-sec-}^\circ\text{R}} = 2.1 \times 10^{-4} \frac{\text{Btu}}{\text{ft-sec-}^\circ\text{R}}$$

$$K^* = 2.1 \times 10^{-4} \frac{\text{Btu}}{\text{ft-sec-}^\circ\text{R}}$$

$$(P_R^*)^{1/3} = (0.595)^{1/3} = 0.841.$$

Thus

$$\begin{aligned} h_{f.p.} &= \frac{(0.332)(2.1 \times 10^{-4})}{1.007} (0.841)(38.7) = \frac{(0.332)(2.1)(0.841)(38.7)}{1.007} \times 10^{-4} \\ &= \frac{22.7}{1.007} \times 10^{-4} = 22.5 \times 10^{-4} \frac{\text{Btu}}{\text{ft}^2\text{-sec-}^\circ\text{R}} \end{aligned}$$

and

$$h_c = \sqrt{3} h_{f.p.} = (1.732)(22.5 \times 10^{-4}) = 39.0 \times 10^{-4} \frac{\text{Btu}}{\text{ft}^2\text{-sec-}^\circ\text{R}}$$

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

According to Eq. (2),

$$\frac{\dot{q}_b}{\dot{q}_{\text{Lam.B.L.}}} = 0.56 .$$

Therefore the heat transfer to Surface (b) is:

$$\begin{aligned}\dot{q}_b &= 0.56 \dot{q}_{\text{Lam.B.L.}} = (0.56) h_c (T_r - T_w) \\ T_r &= T_{e_2} \left(1 + r \frac{\gamma-1}{2} M_2^2 \right) = 6590 (1 + [0.85][0.2][3.20]) \\ &= (6590)(1 + 0.54) \\ &= 10150^\circ\text{R}.\end{aligned}$$

If one assumes the wall to be at a temperature of 1000°R , as was done in computing the reference enthalpy, then

$$\begin{aligned}\dot{q}_b &= (0.56)(3.9 \times 10^{-3})(10150-1000) \\ &= (0.56)(3.9)(9.15) \\ &= 19.9 \frac{\text{Btu}}{\text{ft}^2\text{-sec}}\end{aligned}$$

Now consider the discontinuity at (g). If one assumes that the properties remain constant along the streamlines between Points (f) and (g), then for the discontinuity at Point (g),

$$\begin{aligned}T_{e_1} &= 6590^\circ\text{R} \\ P_{e_1} &= 8.33 \frac{\text{lb}}{\text{ft}^2} = 0.004 \text{ atmosphere} \\ \rho_{e_1} &= 4.15 \times 10^{-7} \frac{\text{lb-sec}^2}{\text{ft}^4} \\ u_{e_1} &= 7135 \text{ ft/sec} \\ i_1^* &= 153 \times 10^6 \frac{\text{ft}^2}{\text{sec}^2} \\ T_1^* &\cong 4500^\circ\text{K}\end{aligned}$$

Subscript 1 refers to the flow upstream of Point (g). It is evident that the flow upstream of Point (g) is precisely the flow downstream of Point (f).

Consequently, notice that the sub-one values are identical to the sub-two values for the discontinuity at (f).

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

$$\rho_1^* = 4.41 \times 10^{-7} \frac{\text{lb-sec}^2}{\text{ft}^4}$$

$$\mu_1^* = 2.11 \times 10^{-6} \frac{\text{lb-sec}}{\text{ft}^2}$$

The flow properties downstream of Point (g) must be determined in order to establish the values of the heat transfer rate at Points (c), (d), and (e). According to Prandtl-Meyer considerations, the flow downstream of (g) has the following characteristics:

$$M_2 = 6.82$$

$$\begin{aligned} P_{e_2} &= (0.2849 \times 10^{-3}) (P_{\text{total St. Pt.}}) = 0.2849 \times 10^{-3} \times 47.13 \\ &= 13.48 \times 10^{-3} \frac{\text{lb}}{\text{ft}^2} \end{aligned}$$

$$\begin{aligned} \rho_{e_2} &= (0.2935 \times 10^{-2}) (\rho_{\text{total St. Pt.}}) = 0.2935 \times 10^{-2} \times 1.43 \times 10^{-6} \\ &= 4.20 \times 10^{-9} \frac{\text{lb-sec}^2}{\text{ft}^4} \end{aligned}$$

$$T_{e_2} = (0.09706) (T_{\text{total St. Pt.}}) = (970.6) (10^{-4}) (1.081 \times 10^4) = 1050^\circ\text{R}$$

$$u_{e_2} = 49.1 \sqrt{1050} = (49.1) (32.4) = 1590 \frac{\text{ft}}{\text{sec}}$$

$$T_{r_2} = T_{e_2} \left(1 + r \frac{\gamma-1}{2} M_2^2 \right) = 1050 [1 + (.17) 46.5] = 1050 [1 + 7.91] = 9360^\circ\text{R}$$

$$T_{e_2} = 1050^\circ\text{R}$$

$$T_{w_2} \cong 1000^\circ\text{R} .$$

After the manner used to compute i^* , one may now compute T^*

$$T_2^* = 0.28 T_{r_2} + 0.22 T_{e_2} + 0.5 T_{w_2} = 2620 + 230 + 500 = 3350^\circ\text{R} (1860^\circ\text{K})$$

$$\mu_2^* \cong 1.23 \times 10^{-6} \frac{\text{lb-sec}}{\text{ft}^2}$$

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

$$\begin{aligned}\rho_2^* &= \frac{P_{e_2}}{RT_2^*} = \frac{13.43 \times 10^{-3}}{(1716)(3350)} = \frac{13.43 \times 10^{-9}}{1.716 \times 3.35} = \frac{13.43 \times 10^{-9}}{5.75} \\ &= 2.34 \times 10^{-9} \frac{\text{lb-sec}^2}{\text{ft}^4}.\end{aligned}$$

From Eq. (13)

$$\ell_{2g} = \frac{1}{3} \left(\frac{\rho_{e_1}}{\rho_{e_2}} \right)^2 \left(\frac{u_{e_1}}{u_{e_2}} \right)^3 \left(\frac{\mu_1^*}{\mu_2^*} \right) \left(\frac{\rho_2^*}{\rho_1^*} \right) \ell_{1g}$$

$$\ell_{1g} = \ell_{2f} + 0.236 \text{ ft} = (0.889) + (0.236) = 1.125 \text{ ft}$$

$$\frac{\rho_{e_1}}{\rho_{e_2}} = \frac{4.41 \times 10^{-7}}{4.20 \times 10^{-9}} = 104.8$$

$$\left(\frac{\rho_{e_1}}{\rho_{e_2}} \right)^2 = 1.097 \times 10^4$$

$$\left(\frac{u_{e_1}}{u_{e_2}} \right)^3 = \left(\frac{7135}{1590} \right)^3 = (90.32)$$

$$\frac{\mu_1^*}{\mu_2^*} = \frac{2.11 \times 10^{-6}}{1.23 \times 10^{-6}} = 1.72$$

$$\frac{\rho_2^*}{\rho_1^*} = \frac{2.34 \times 10^{-9}}{4.41 \times 10^{-7}} = 5.31 \times 10^{-3}$$

$$\ell_{2g} = \frac{1}{3} (1.097 \times 10^4) (9.032 \times 10) (1.72) (5.31 \times 10^{-3}) \ell_{1g}$$

$$= \left(\frac{1.097 \times 9.039 \times 1.72 \times 5.31 \times 10^2}{3} \right) \ell_{1g}$$

$$= (3018) \ell_{1g}$$

$$\ell_{2g} = 3395 \text{ ft}$$

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

$$h_{f.p.} = \frac{0.332 k^*}{x} (P_R^*)^{1/3} (R_{e_x}^*)^{1/2}$$

$$K^* = (3.99)(13.55) \times 10^{-6} \frac{\text{Btu}}{\text{ft}^2 \text{sec}^{\circ}\text{R}} = 5.41 \times 10^{-5} \frac{\text{Btu}}{\text{ft}^2 \text{sec}^{\circ}\text{R}} \quad 32$$

$$(P_R^*)^{1/3} = (0.614)^{1/3} = 0.850$$

$$R_{e_x}^* = \frac{\rho_2^* u_2 l_2}{\mu_2^*} = \frac{(2.34 \times 10^{-3})(1590)(3395)}{1.23}$$

$$= \frac{\overbrace{(2.34)}^{3.72} \overbrace{(1.59)}^{12.6} (3.395)}{1.23} \times 10^3 = 1.02 \times 10^4$$

$$(R_{e_x}^*)^{1/2} = (1.02 \times 10^4)^{1/2} = 101.0$$

$$h_{f.p.} = \frac{(0.330)(5.41 \times 10^{-5})}{3.395} (0.85)(101) \\ = \frac{(0.332)(5.41)(85)(1.01)}{3.395} \times 10^{-6} = 4.51 \times 10^{-7} \frac{\text{Btu}}{\text{ft}^2 \text{sec}^{\circ}\text{R}}$$

While the Prandtl-Meyer theory was used to expand and subsequently compute the representative length and the heat transfer coefficient, the recovery temperature used in calculating the heat transfer rate was the free-stream recovery temperature. This was because the flow could in actuality expand to free-stream conditions.

$$T_{r_{\infty}} = 4.46 \times 10^4 {}^{\circ}\text{R}$$

$$\dot{q}_{f.p.(c,d,e)} = h_{f.p.} (T_{r_{\infty}} - T_w) = (4.57)(4.36) \times 10^{-3} \\ = 19.92 \times 10^{-3} = 0.0199 \frac{\text{Btu}}{\text{ft}^2 \text{sec}}$$

Hence

$$\frac{\dot{q}_{f.p.}}{(\dot{q}_{\text{St.Pt.}})_{\text{Large Radius}}} = \frac{0.0199}{46.99} = 4.23 \times 10^{-4}$$

This figure seemed inordinately small in view of experimental results³³ on cylinders which resulted in

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

$$\frac{\dot{q}_{f.p.}}{\dot{q}_{St.Pt.}} \cong 0.1$$

Consequently, the experimental results were used.

Therefore,

$$\dot{q}_{f.p.(c,d,e)} = (0.1) \left(\dot{q}_{St.Pt.} \right)_{\text{Large Radius}} = (0.1)(46.99) = 4.70 \frac{\text{Btu}}{\text{ft}^2\text{-sec}}$$

Sample Calculation for the Intermediate Configuration of the 5BN/SNAP-9A

The heat transfer rate across the face of the cylinder was assumed to be uniformly the stagnation point rate $\dot{q}_{St.Pt.}$.

The major regions of interest are the flat face of the generator and Points I, II, and III.

If one assumes a Prandtl-Meyer expansion around the corner of the flat face, then the following are the properties downstream of the corner for $V_\infty = 25,000$ ft/sec at 250,000 ft:

$$M = 6.82$$

$$P_e = 13.4 \times 10^{-3} \frac{\text{lb}}{\text{ft}^2}$$

$$\rho_e = 0.20 \times 10^{-9} \frac{\text{lb-sec}^2}{\text{ft}^4}$$

$$T_e = 1050^\circ\text{R}$$

$$u_e = 1590 \text{ ft/sec}$$

$$\begin{aligned} T_{r_\infty} &= T_\infty \left(1 + r \frac{\gamma-1}{2} M_\infty^2 \right) = (328.2) \left(1 + (792)(0.17) \right) \\ &= 328.2 (136) = 4.46 \times 10^4 \end{aligned}$$

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

$T^* = 3350^\circ\text{R}$ - reference temperature

$$\mu_2^* = 1.23 \times 10^{-6} \frac{\text{lb-sec}}{\text{ft}^2}$$

$$\rho^* = 2.34 \times 10^{-9} \frac{\text{lb-sec}^2}{\text{ft}^4}$$

$$K^* = 5.4 \times 10^{-5} \frac{\text{Btu}}{\text{ft-sec-}^\circ\text{R}}$$

$$\left(P_R^*\right)^{1/3} = 0.85$$

$$\left(R_e^*\right)^{1/2} = (1.127)^{1/2} = 1.061 \text{ where } X_I = \frac{4.5}{12} = 0.375 \text{ ft}$$

$$h_{f.p.I} = \frac{0.332K^*}{X_I} \left(P_R^*\right)^{1/3} \left(R_e^*\right)^{1/2}$$

$$h_{f.p.I} = \frac{(0.332)}{(0.375)} (5.41) \times 10^{-5} (0.85)(1.06)$$

$$h_{f.p.I} = (0.885)(5.41)(0.902) \times 10^{-5} = 4.32 \times 10^{-5} \frac{\text{Btu}}{\text{ft}^2\text{-sec-}^\circ\text{R}}$$

Again the free-stream recovery temperature is used in calculating the side wall heat transfer rate. The laminar attached heat transfer rate at Point I is:

$$\dot{q}_{f.p.I} = (4.32 \times 10^{-5}) 4.46 \times 10^4 = 0.1 \times 10^4 = 1.88 \frac{\text{Btu}}{\text{ft}^2\text{-sec}}$$

$$\dot{q}_{\text{St.Pt.}} = \frac{85.52}{\sqrt{\frac{(4)(4.5)}{12}}} - \frac{85.52}{\sqrt{1.5}} = \frac{85.52}{1.225} = 69.81$$

$$\frac{\dot{q}_{f.p.I}}{\dot{q}_{\text{St.Pt.}}} = \frac{1.88}{69.81} = 0.027$$

However, the boundary layer is not attached at I, II, or III because of the step at I. Consequently, the boundary layer is separated at I, II, and III:

$$\frac{\dot{q}_I}{\dot{q}_{f.p.I}} = 0.56 \text{ and } \frac{\dot{q}_I}{\dot{q}_{\text{St.Pt.}}} = (0.56)(0.027) = 0.015$$

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

$$\frac{\dot{q}_{II}}{\dot{q}_I} = \left(\frac{X_I}{X_{II}} \right)^{1/2} = \left(\frac{0.375}{0.75} \right)^{1/2} = (0.500)^{1/2} = 0.707$$

$$\frac{\dot{q}_{II}}{\dot{q}_{St.Pt.}} = (0.707)(0.015) = 0.011$$

$$\frac{\dot{q}_{III}}{\dot{q}_I} = \left(\frac{4.5}{13.5} \right)^{1/2} = 0.577,$$

$$\frac{\dot{q}_{III}}{\dot{q}_{St.Pt.}} = 0.0086$$

The figure for Point III is very unrealistic in view of its proximity to the satellite: experimental results showed the calculated figure for Point III to be as much as an order of magnitude lower than the actual.

The $\dot{q}_I/\dot{q}_{St.Pt.} = 0.015$ was believed to be accurate as the conservative figure for Points I and III, i.e., as the lowest heat transfer rate. The heat transfer rate ratio is plotted in Figures 27 and 28 as a function of the local viscous hypersonic interaction parameter. Several theoretical points are shown, with a curve faired through them. The fact that all but two of the experimental points (obtained in the Cornell Aeronautical Laboratory's 48-inch hypersonic shock tunnel) lie above the theoretical line in Figure 28 proves that the theoretical method is conservative. Similar results for Point III are shown in Figure 28. This result is significant since the same method of calculation was used throughout this study.

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

Figure 27. Heat transfer rate ratio for Point I versus local hypersonic viscous interaction parameter

Figure 28. Heat transfer rate ratio for Point III versus local hypersonic viscous interaction parameter

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

LIST OF REFERENCES

1. The Martin Company Nuclear Safety Unit, SNAP-9A Radioisotope-Fueled Generator Final Safety Analysis for Transit Mission, MND-P-2775-1, August 1962, Confidential.
2. Chamberlain, A. C., Aspects of Travel and Deposition of Aerosol and Vapour Clouds, A.E.R.E., HP/RI261 (1955).
3. Newcomb, C. L., et al, Evaluation of Radiation Hazards Associated with Operations of Nuclear-Powered Space Units at Pacific Missile Range, USNRDL-TR-545, Feb. 1962.
4. Report of Committee II on Permissible Dose for Internal Radiation (1959), Health Physics, Vol. 3 (1960).
5. Reactor Handbook, Vol. I Materials, 2nd Edition 1960.
6. Waber, J. T., "The Corrosion Behavior of Plutonium and Uranium," Peaceful Uses of Atomic Energy, P/699USA (1958).
7. Dunster, H. J., "The Disposal of Radioactive Liquid Wastes into Coastal Waters," Peaceful Uses of Atomic Energy (1958), p. 297.
8. National Academy of Sciences - NRC, Considerations on the Disposal of Radioactive Wastes from Nuclear-Powered Ships into the Marine Environment, Publication 658 (1961).
9. Aten, A. W. H., Jr., "Permissible Concentrations of Radionuclides in Sea Water," Health Physics, Vol. 16 (1955).
10. Church, Hugh W., Biological Hazard Evaluation from Orbit Re-entry Injection of Pu²³⁸, SCTM 225-63(54), July 1963 (U).
11. Pre-operational Safety Study Report of SNAP-9A Radioisotope Fuel Generator, Draft Copy, Headquarters, AFSWC, SWVP-3-024, April 1963.
12. Gurney, R. W., The Initial Velocities of Fragments from Bombs, Shell, and Grenades, BRL Report 405, September 1943.
13. Benson, W. R., The Basic Method of Calculating the Lethal Area of a Warhead, Samuel Feltman Ammunition Laboratories, Picatinny Arsenal, Dover, N.J., Technical Report 2044, October 1954.
14. Roberts, W. S., Missile Trap Calibration - Operation Halo '60, Sandia Corporation Field Report, February 1960.
15. Smalley, W. M. and Anderson, D. E., Explosive Potential of Liquid Oxygen and RP-1 Missiles, Space Technology Laboratories, Inc., January 1959, CRD.
16. The Martin Company Nuclear Division, SNAP-9A Radioisotope-Fueled Generator - Final Safety Analysis for Transit Mission, MND-P-2775-2, March 1963, SRD.
17. The Martin Company Nuclear Division, Safety Test Report for the SNAP-9A Thermoelectric Generator Transit Mission, MND-P-2804, July 1962, CRD.
18. Preliminary Impact Probability Study - Program 435/II (Rough Draft), Pacific Missile Range, U.
19. Geology and Ground-Water Features of Point Arguello Naval Missile Facility, Santa Barbara County California, Geological Survey Water Supply Paper, 1619-F (1963), U.

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

20. Dalby, W. J., Drop Test of SNAP-9A Generator and Capsule, SCDR 63-63, March 1963, CRD.
21. SLV-2 Probability of Survival, Memorandum from L. J. Wells, Aerospace Corporation to W. G. Reynolds, Aerospace Corporation, April 5, 1963, SRD.
22. Chapman, Dean R., An Approximate Analytical Method for Studying Entry into Planetary Atmospheres, NASA TR-R-11 (1959).
23. See, e.g., Schlichting, Herman, Boundary Layer Theory, McGraw-Hill, New York, 4th Edition (1960).
24. Chapman, Dean, A Theoretical Analysis of Heat Transfer in Regions of Separated Flow, NACA TN-3792, October 1956.
25. Truitt, R. W., Fundamentals of Aerodynamic Heating, Ronald Press Co., New York (1960).
26. Öpik, E. J., Physics of Meteor Flight in the Atmosphere, Interscience Tracts on Physics and Astronomy, No. 6 (1958).
27. Bland, W. M., Swanson, A. G., Kalenkiewicz, R., Free Flight Aerodynamic Heating Data at Mach Numbers up to 10.9 on a Flat Faced Cylinder, NACA RM L57K29, January 13, 1958, Confidential.
28. Aerodynamic and Thermodynamic Heating Units, Aerodynamic Heat Transfer Handbook - Volume I, Boeing Airplane Co., BOE-D2-9511, May 5, 1961.
29. Eckert, E. R. G., Survey on Heat Transfer at High Speeds, WADC-TR-54-70, April 1954.
30. Wittliff, C. E. and Curtis, J. T., Normal Shock Wave Parameters in Equilibrium Air, CAL-111, November 1961.
31. Ames Research Staff, Equations, Tables, and Charts for Compressible Flow, NASA Report 1135 (1953).
32. Hansen, Frederick C., Approximations for the Thermodynamic and Transport Properties of High Temperature Air, NASA TR-R-50 (1959).
33. Stoney, William E. and Swanson, Andrew G., Heat Transfer Measured on a Flat Face Cylinder in Free Flight at Mach Numbers up to 13.9, NACA RM L57E13, June 17, 1957.
34. Powered Flight Trajectory Document Program 435/II, Space Systems Division, Air Force Systems Command, SGC-256TA (268).
35. Beal, J. L., Brown, W. R., Haas, F. C., and Pleuthner, R. L., Cornell Aeronautical Laboratory, Inc., Report I08-100, May 1963.

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

DISTRIBUTION:

M-3679, C-44d (34th Ed) (140)
Engineering and Test Branch, Division Reactor Devel., USAEC Hq., Washington 25, D.C. (6)
S. P. Schwartz, 1
R. W. Henderson, 100
E. H. Draper, 1000
C. F. Bild, 1100
R. R. Sowell, 1110
W. M. O'Neill, 1120
J. M. Wiesen, 1440
C. S. Williams, 1442
L. D. Smith, 1500
R. A. Bice, 2000
R. B. Powell, 3000
C. W. Campbell, 4000
R. C. Fletcher, 5000
R. S. Claassen, 5100
J. W. Easley, 5300
T. B. Cook, 5400
J. D. Shreve, 5414
J. W. Weihe, 5420
G. A. Fowler, 7000
L. E. Hollingsworth, 7200
D. B. Shuster, 7400
V. E. Blake, 7410
H. E. Hansen, 7411 (3)
A. J. Clark, Jr., 7412
A. E. Bentz, 7413
J. W. McKiernan, 7419
A. Y. Pope, 7420
A. Y. Pope, 7420, Attn: D. A. Byrne, 7421
J. H. Scott, 7430
W. C. Scrivner, 7600
B. S. Biggs, 8000
E. A. Paxton, 8232-1
B. R. Allen, 3421
M. G. Randle, 3428-1, Bldg. 836
M. G. Randle, 3428-1, Bldg. 880
W. F. Carstens, 3410, Attn: J. M. Zanetti, 3412 (2)
R. C. Smelich, 3427-3 (5)

~~CONFIDENTIAL~~