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AEROSPACE SAFETY REENTRY
ANALYTICAL AND EXPERIMENTAL PROGRAM
SNAP 2 AND 10A
(INTERIM REPORT)

AEC Research and Development Report

ATOMICS INTERNATIONAL

A DIVISION OF NORTH AMERICAN AVIATION, INC.

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SNAP 2 AND 10A
(INTERIM REPORT)

By
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ATOMICS INTERNATIONAL

A DIVISION OF NORTH AMERICAN AVIATION, INC.
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ABSTRACT

Before an atomic reactor is launched into a satellite orbit about the Earth, the hazards to the population must necessarily be considered in the event that the reactor reenters the Earth's atmosphere and possibly impacts with the ground. This report describes the analytical and experimental effort pursued by the Atomics International Aerospace Safety Reentry Project through February 1963 to investigate these potential hazards as they apply to Systems for Nuclear Auxiliary Power (SNAP) reactors. This work was done as part of the Aerospace Safety Program.

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Frontispiece — SNAP 10A

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I. INTRODUCTION

Before an atomic reactor is launched into a satellite orbit about the Earth, it is prudent to consider what may be the hazards to the population of the Earth when the reactor eventually spirals back into the lower atmosphere and possibly returns to ground level. If the reactor has never been operated at power, there will be no inventory of fission products which could find their way into the life cycle. In this case, the primary danger would be that the reactor, if it returns essentially intact, could attain criticality by environmental influences (e. g. , water reflection) and commence producing significant power. If the reactor has been operating at power in orbit, then it is desirable to melt the fuel material and divide it into small particles. Thus, it will be dispersed in the upper atmosphere so widely that no measurable increase in the existing radioactivity at ground level will occur.

An analytical and experimental program to investigate the hazards that may potentially be caused by reentry of SNAP reactors upon descent from satellite orbits has been pursued at Atomics International for the past three years. Early exploratory studies^{1,2,3} indicated that the available reentry heating during orbital decay was more than adequate to effect complete reactor meltdown and fuel ablation. These studies were, however, based on preliminary aerodynamic calculations.

Recent studies, the results of which are reported in this document, include considerations of the local heating rates on the various components of the reactor and vehicle as well as detailed calculations of the transport of heat within the components. Local heating rates have been estimated for the case of zero angle of attack, and plans have been made for theoretical and experimental investigations of heat rates at other angles of attack. The thermochemical behavior of the fuel alloy in simulated reentry environments has been investigated experimentally in an arc-heated gas jet, and more tests have been planned.

Although the analysis of SNAP reentry has not been completed, it is possible at this time to make the following conclusions:

- a) It is virtually certain that the reactor vessel will melt open before it impacts with the ground.
- b) It is highly probable that the core assembly will be released from the vessel and the individual fuel elements dispersed before impact.

c) Fuel element meltdown and upper atmospheric dispersal is uncertain with the current reactor design.

The remainder of this report is devoted to the presentation of the analyses which lead to these conclusions.

II. BASIC CONSIDERATIONS

A. ORBITAL VEHICLES AND OPERATIONAL CONSIDERATIONS

The current reentry program has been based primarily on the SNAP 10A - Agena D reactor and vehicle combination shown in Figure 1. Future designs of vehicles, as yet undefined, will be studied in this program when information on them becomes available.

The SNAP 10A will remain attached to the Agena during the entire orbital lifetime. Therefore, the dynamics of reentry depend on the dynamic characteristics of the entire vehicle. Current estimates of the mass, center of mass, moments of inertia, and aerodynamic coefficients of the vehicle at the time of reentry are presented in Appendix A. The values given there are tentative, based partially upon engineering estimates, and they may be revised before completion of the first flight system. It is expected, however, that the changes will not be of such magnitude as to invalidate the fundamental conclusions of this report.

Among the several operational factors which may influence reentry safety are the following:

- 1) The vehicle may fail to achieve a satisfactory orbit of long enough life to make it safe to start the reactor. In this case there will be no radioactive fission products to present a hazard, and the problem will be to ensure that the reactor cannot go critical upon impact or subsequent to impact.
- 2) The cooling system may be filled with NaK at the time of reentry, especially if the vehicle has been in a short-life orbit. After many years in orbit, it is highly probable that the system will have lost its integrity because of meteoroid puncture or other effects and that the NaK will have evaporated; but since this cannot be guaranteed at any particular time, the possible presence of NaK at reentry must be considered.
- 3) It is postulated that there is no programmed or command control over any functions on the reactor or vehicle at the time of reentry. Break-up of the reactor and release of the fuel elements must be accomplished by aerodynamic forces and heating alone.

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Figure 1. Typical Orbiting Vehicle Carrying
a SNAP 10A Auxiliary Power Unit

B. DESCRIPTION OF REACTOR FEATURES PERTINENT TO REENTRY

The major components of the presently conceived SNAP 10A reactor are shown in Figure 2. The SNAP 2 reactor is practically identical to the SNAP 10A reactor, except for the lack of the thermoelectric pump and fins on the head end.

The beryllium external reflector is made in two halves which are held together by a steel band wrapped around titanium standoff posts. There are six silver-brazed lap joints in the band which are designed to fail at a temperature of 1250°F, releasing the reflector so that built-in springs can eject it in two halves.

The thermoelectric pump, on the SNAP 10A, comprises a support frame with a permanent magnet surrounding the coolant outlet pipe. Some of the heat in the coolant is conducted through thermoelectric elements and is radiated to space by four aluminum fins. Reentry burnup of this pump is important only in that it shelters the top head of the reactor vessel from some of the aerodynamic heating.

The coolant pipe provides a dual path for the NaK coolant after it leaves the pump. There is a filling tube at the top of this pipe and six thermal switches are installed along the pipe, as shown in view D of Figure 2. If the reactor enters head first, it is probable that the first breach of the coolant system will occur in this pipe. Burnoff of this pipe is required if the top head of the reactor vessel is to be removed and the fuel elements ejected.

The reactor vessel has an extended lip around its upper edge. The top head is held in place by a fusion weld around this lip. If the reactor enters head first this weld is intended to be heated more rapidly than other parts of the top head and vessel. Springs inside the top head exert a force of approximately 350 lb, which is expected to help break the weld and eject the top head. If the reactor does not enter head first, the sides of the vessel are expected to melt and expose the reactor core. After enough of the vessel has melted, the top head is expected to be separated by inertial forces and the core assembly released.

The presence of NaK coolant in the system at the time of reentry may have a critical effect upon the mode of failure. Calculations have shown that the coolant will be in the liquid state while the reactor is in orbit, even in the absence of heat from radioactive decay. The heat absorbed from solar and Earth radiation will be sufficient to keep the coolant temperature above

A FUEL ELEMENT

B REACTOR CORE

COOLANT OUTLET PIPE
NaK FILLING TUBE
THERMAL SWITCHES
THERMOELECTRIC PUMP

C REACTOR WITH REFLECTOR

D ASSEMBLED REACTOR

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Figure 2. Components of the SNAP 10A Reactor

the freezing point of NaK-78, provided that the outer surfaces of the reflector and vessel have a ratio of thermal absorptivity to emissivity greater than unity. This criterion will be met, both when the reactor is new and after it has been eroded by micrometeoroids.

The reactor core consists of 37 fuel elements and 6 beryllium reflector pieces, loosely held between two grid plates. After the core assembly has been released from the vessel, the individual fuel elements must separate from each other before any significant amount of melting can take place.

The reactor fuel is an alloy of uranium and zirconium, hydrided nearly to the saturation level. Each fuel element comprises a rod of fuel alloy 1.212 in. in diameter and 12.250 in. long, clad with a 0.015-in.-thick jacket of Hastelloy N (nickel alloy). The inside of the cladding is coated with a vitreous material which blocks the escape of hydrogen from the fuel element at reactor operating temperatures. The whole fuel element assembly, as clad, is 1.250 in. in diameter and 12.450 in. long, and weighs 3.30 lb. There is a small pin on each end of the element which is used to hold it in place between the grid plates.

C. ORBITAL DECAY AND VEHICLE MOTIONS

The lifetime in orbit of a satellite vehicle is primarily a function of its mass-to-drag-area ratio ($W/C_D A$) and of the minimum (perigee) altitude of its orbit. The maximum (apogee) altitude has a secondary effect as shown in Figure 3, which was taken from Reference 4. The orbital lifetime for any given altitude may be found by reading the reference value from Figure 3 and correcting it in direct proportion to the ratio of the vehicle $W/C_D A$ with respect to the basic value of 100 lb/ft^2 . The value of $W/C_D A$ for the SNAP 10A-Agena D at orbital altitudes is given in Appendix A as 19 to 127 lb/ft^2 , depending upon its orientation with respect to its direction of motion. While the vehicle is in active operation, it will presumably be stabilized with its longitudinal axis pointed toward the center of the Earth. In this attitude, the value of $W/C_D A$ will be the least and the lifetime will be less than if the vehicle were aligned with its direction of motion.

When, during the orbital decay period, the altitude of the vehicle has decreased to about 1,000,000 ft (~ 160 naut mi), the remaining lifetime in orbit may be measured in hours rather than in years. Figure 4 presents curves of lifetime versus altitude in the range of altitudes between 400,000 and 1,000,000 ft for three

Figure 3. Orbital Lifetimes of Earth Satellites

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Figure 4. Orbital Lifetimes of Satellites in Quasi-Circular Orbits

values of $W/C_D A$. These curves were calculated by the method described in Appendix B. Lifetimes at intermediate values of $W/C_D A$ may be interpolated by direct proportion at any given altitude. Since the orbital period of a near-Earth satellite is approximately 1.5 hours, it is evident from Figure 4 that when its altitude has decreased to approximately 400,000 ft, a satellite having $W/C_D A \leq 100$ will not make another circuit of the Earth. Thus, an altitude of 400,000 ft is frequently used as the starting point for reentry calculations.

The calculations upon which Figure 4 is based were made for an assumed spherical Earth. Actually, the shape of the Earth is an oblate spheroid with the radius at the poles being about 70,000 ft less than at the equator. Hypothetical surfaces of constant density in the atmosphere are oblate also, and thus a satellite in a nonequatorial orbit is subject to periodic fluctuations in aerodynamic drag. These perturbations have very little effect upon the heating and breakup of the vehicle during the final reentry period, but they may have a significant effect upon the vehicle attitudes and oscillations during the early part of the decay period. They may also cause the reentry burnup period to occur in certain latitudes, possibly to the virtual exclusion of other latitudes. Further study of this phenomenon is planned.

After the reactor has been shut down, there will be no power for attitude control, and the orientation of the vehicle will be determined by the balance of the aerodynamic and gravitational forces acting on it. The most general assumption that can be made regarding the attitude at reentry time is that it is completely random. As the vehicle descends, the aerodynamic forces acting upon it gradually increase and the vehicle tends to turn toward an aerodynamically trimmed attitude. Since the damping forces are small, the vehicle will oscillate about the trim position, but the magnitude of the oscillations will normally decrease as the aerodynamic forces become stronger. A study of these oscillations is being made, as described in Appendix C. No results were available at the time of this report.

In the continuum flow regime, the SNAP 10A-Agena D vehicle is not stable when its longitudinal axis is aligned with its direction of motion. This is shown by the curve labeled "Newtonian Theory" in Figure A-3 of Appendix A, where the rate of change of the pitching moment with respect to the angle of attack is positive at angles of 0 and 180°. It appears that the aerodynamic trim angle is about 34° for the undamaged vehicle. After some of the structure has failed due

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Figure 5. Typical Trajectory for Reentry of SNAP 10A Vehicle from a Polar Orbit

to aerodynamic heating, the trim angle will very probably change. With our present lack of knowledge of the structure and of the local heating rates, it is impossible to predict the vehicle attitudes for the entire reentry period. Therefore, these preliminary analyses have been made for the postulated case of zero angle. Further analyses will be made at a later time.

A computer code for calculation of reentering satellite trajectories for an oblate, rotating Earth (RESTORE) has been written and checked. A brief description of this code is given in Appendix D. A representative trajectory calculated by the use of this code has been selected for use in the analyses of reactor breakup presented in Section III of this report. This trajectory, data for which are shown in Figures 5, 6, and 7, is for a polar orbit with initial conditions of $V = 25,694$ ft/sec and $\gamma = -0.05^\circ$ at an altitude of 400,000 ft. The ballistic parameter, $W/C_D A$, was selected as 132 lb/ft^2 , which was considered to be a representative value for free-molecule flow at zero angle and Newtonian flow at the trim angle. The reference heat rates given in Figure 7 were calculated by the methods described in Appendix E.

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Figure 6. Typical Velocity Variation for Reentry of SNAP 10A Vehicle from a Polar Orbit

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Figure 7. Aerodynamic Heating During Reentry of SNAP 10A Vehicle from a Polar Orbit

III. VEHICLE AND REACTOR BREAKUP

A. REFLECTOR RELEASE AND SEPARATION

It is expected that the end-of-life shutdown devices will release the reflector assembly while the vehicle is still in orbit. If it should happen that such release is not accomplished, the heat of reentry will act on the reflector retaining band and melt the fusible joints therein. These joints are made with an alloy of silver, copper, and indium which melts at approximately 1250°F. This temperature will be reached during the reentry period at an altitude between 350,000 and 285,000 ft, depending upon the attitude and rate of tumbling of the vehicle. Figure 8 shows fusible link temperatures for various altitudes during a typical orbital decay trajectory. The methods used for calculation of the heat rates on the band are presented in Appendix F and the thermal model used is described in Appendix G.

After the band fails, the reflectors fall away, assisted by the springs which bear against the bottom of the reflectors. The hinges about which the reflectors pivot are designed to come apart when the reflectors have moved through an angle of 15°. The mating surfaces of the hinges are treated to prevent self-welding during the orbital period. After the hinges have come apart, the reflectors are restrained only by the electrical wires to the control motors, and these will melt through in a short time. Complete release of the reflectors is expected to be accomplished above an altitude of 280,000 ft.

B. FAILURE OF THE COOLANT PIPES

It is highly probable that the coolant system will have been punctured by meteoroids and the coolant (NaK) escaped if the vehicle has been in orbit for many years. However, if the vehicle has been put into a short-lived orbit, accidentally or otherwise, there is a high probability that the coolant system will be filled with NaK at the time of reentry. A conservative analysis must therefore be based on the assumption that the coolant system has not been ruptured prior to reentry.

During a nose-first reentry, the cooling system will be exposed to heat at the following locations: (See Figures 1 and 2.)

- 1) The coolant outlet pipe above the pump
- 2) The coolant lines as they enter the reactor support legs

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Figure 8. Calculated Temperature Histories of Reflector Retaining Band During Reentry

- 3) The manifold at the top of the converter
- 4) The coolant tubes in the converter.

Analyses have thus far been made for heating at the first two of these locations.

The coolant pipe above the pump has been analyzed as a cylinder in cross flow with heat rates estimated as described in Appendix F. The thermal models used for the pipe, both empty and full of NaK, are described in Appendix G. Transient temperatures on the forward side of the pipe are shown in Figure 9. Failure of the pipe occurs at an altitude of approximately 290,000 ft regardless of whether it is filled with NaK or not, provided that the NaK is not flowing.

An analysis of the failure of the coolant pipe with consideration of the flow of NaK has not yet been made in detail, but a qualitative discussion of the possible modes of failure can be made. The deceleration of the vehicle at altitudes above 300,000 ft is less than 0.03 g, and thus there is only a weak tendency for free convection to occur. Thermal gradients through the thermoelectric elements in the pump will tend to cause a circulation of coolant in either direction, depending on the direction of heat flow through the elements. This circulation might be significant during the early heating period, but it would tend to diminish as the temperatures of the fins and the fluid approach each other. Deterioration of the thermoelectrics and the magnet would also tend to diminish the circulation as they became heated.

As the temperature of the NaK in the heated areas rises, the vapor pressure will increase. The expansion compensator inside the converter structure will be driven to its fully extended position. When the pipe is penetrated, there will be a rush of NaK vapor and liquid out of the opening. The decompression of the system will cause some of the liquid to flash into vapor, and the voids formed will allow ablation of the flow system to proceed more rapidly.

C. REMOVAL OF THE THERMOELECTRIC PUMP

The aluminum fins which serve as radiators for the thermoelectric pump will begin to melt at about the same time that the reflector-retaining band is released and will be completely gone when an altitude of 270,000 ft is reached. Figure 10 shows calculated contours of fin meltback distances for various altitudes during the trajectory plotted in Figure 5. The methods used for estimation

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Figure 9. Transient Temperatures in Coolant Pipe Forward of Pump

NUMBERS ON CONTOUR LINES ARE ALTITUDES IN KILOFEET

DOTTED LINES INDICATE NODE DIVISIONS USED FOR THERMAL NETWORK

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Figure 10. Pump Radiator Fin with Contours of Meltback Distances at Various Altitudes During Reentry

of the heat rates to the fins are given in Appendix F, and the thermal model used for calculation of the temperatures is described in Appendix G.

Removal of the pump fins is not an essential part of the reentry breakup sequence; but after the fins are gone, the intensity of the heating on the top head of the core vessel will be increased in the regions formerly in the wake of the fins.

The magnet and other parts of the pump will not melt completely, but they will most probably be thrown off when the coolant pipes and the top head fail. The presence of the magnet has an influence on the flow field over the reactor vessel which tends to reduce the heating in some regions of the top head and increase it in others. The net effect on release of the top head is not deemed to be critical.

D. SEPARATION OF THE REACTOR FROM THE POWER-CONVERSION SYSTEM

The reactor is attached to the converter structure with four hollow support legs which follow the contour of the top of the shield. These legs and the coolant lines within them constitute the only mechanical connection between the reactor

and the power-conversion system. After the reflector halves have been released, the support legs will be exposed to a high rate of heating and the 0.032-in.-thick titanium walls will melt in a few seconds. The coolant lines inside have walls of 0.020-in. stainless steel, and these will melt soon afterward. It is predicted that the reactor will become separated from the rest of the system at an altitude of about 250,000 ft and become a free body, probably tumbling and spinning.

E. CORE VESSEL OPENING

A sectional view of the core vessel lip is shown in Figure 11, with contours of calculated meltback distances at various altitudes during reentry. It is predicted that melting of the lip will begin at an altitude of approximately 290,000 ft, but that the lip will not release until the altitude is 250,000 ft or less. It is interesting to note that the top head melts through at about the same altitude as

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Figure 11. Section of Core Vessel Lip and Top Head with Contours of Meltback Distances at Various Altitudes During Reentry

does the lip. (These calculations were made for a vessel without NaK. The purpose of the lip was to provide a failure point that would not be strongly influenced by the presence or absence of NaK.) Ground tests performed by the Sandia Corporation showed that the top head actually ruptures before the lip releases. It was also found that heating of the sides of the vessel is very important because they are made of thinner material than the top head and lip.

Calculations for a new thermal model with considerations of tumbling and oscillations and with improved estimates of heat rates are being made. Indications are that the vessel will fail by melting and rupture of its lateral surfaces, exposing the core assembly. The analysis of the subsequent release and breakup of the core assembly has not yet been made.

IV. FUEL ELEMENT ABLATION AND DISPERSION

A. RELEASE ALTITUDES AND HEATING RATES

The analysis of fuel element ablation has been carried on concurrently with the analysis of reactor and vehicle breakup. Rather than waiting for the altitude of reactor core release to be calculated, it was decided that several release altitudes would be postulated and the heating and ablation of the fuel calculated for each case. Since the RESTORE code for computation of trajectories was not in operation when this analysis was begun, fuel element trajectories were taken from calculations performed by General Dynamics/ Astronautics as presented in Reference 5. Graphs of altitude versus time for three postulated release altitudes are given in Figure 12. These trajectories are based on the assumption that the ballistic parameter, $W/C_D A$, of the vehicle is 300 lb/ft^2 and the $W/C_D A$ of the fuel element is 20 lb/ft^2 . These values correspond roughly to the assumption that the attitude of the vehicle is axial and the attitude of the fuel element is normal to the direction of motion.

Reference heat rates, calculated by the methods given in Appendix E, are shown in Figure 13. Stagnation-line heat rates on the fuel elements in cross flow are found by dividing the reference heat rates by the square root of 2.0 to correct for the two-dimensional flow field, and further dividing by the square root of the radius of the element in feet. Before any ablation has taken place, the stagnation-line heat rate on the fuel element is 3.1 times the reference value.

B. EFFECTS OF TUMBLING AND ROTATION

It is highly probable that the fuel elements will be tumbling in a random manner upon release from the core assembly. Although the aerodynamic trim angle for a body having the shape of a right circular cylinder is 90° (cross-axial), it is unlikely that the fuel elements will be initially separated at this angle nor that they will be without some initial rotation. A study of the tumbling of fuel elements having various initial conditions has not yet been made, although some preliminary work was done by the AVCO Corporation as reported in Reference 6.

The thermal analyses to date have been made for three assumed modes of fuel element motion. Values of the ratio of average heat input rates integrated

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Figure 12. Representative Fuel Element Trajectories for Various Release Altitudes

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Figure 13. Calculated Reference Heat Rates for Fuel Elements Released at Various Altitudes

over the surface of the element divided by the stagnation-line heat rates are calculated to be as follows:

<u>Mode of Motion</u>	<u>$q_{\text{avg}} / q_{\text{stag}}$</u>
Cross-axial, spinning but not tumbling	0.32 to 0.36
Randomly tumbling	0.28
Tumbling end-over-end	0.22

End-over-end tumbling is highly improbable. Since random tumbling tends to reduce the average heat input rate, a calculation which predicts complete burnup of a tumbling fuel element also indicates that a nontumbling element in the same trajectory would be completely burned.

C. MODE OF FUEL ELEMENT ABLATION

Studies of the results of arc-jet tests of fuel alloys indicate that the hydrogen is driven away from the melting surface, carrying with it a large fraction of the heat. After the hydrogen has left the surface, the temperature of the uranium-zirconium alloy rapidly rises to the melting point. Chemical reactions with the oxygen and nitrogen in the boundary layer take place on the surface. These reactions appear to add heat to the system, but at the same time they cause a crust to form on the surface. The molten metal breaks through this crust and runs back along the surface. Droplets are torn off by aerodynamic forces and are rapidly shattered and burned to small particles. The complete effects of oxidation are not yet understood. The presence of cladding on the fuel element during the early part of the heating period may retard the formation of an oxide crust. Further experimental work in this area is planned.

Analytical studies of fuel element ablation have thus far been based on a simplified thermal model as described in Appendix G. Dehydrogenation and melting of the alloy have been treated as phase changes occurring at temperatures of 2000°F and 3365°F respectively. Studies are being made which are expected to lead to improved thermal models for future analysis. Physical properties of the fuel alloy used in the present calculations are given in Appendix H.

Typical curves of fuel element temperatures versus time after release are given in Figures 14, 15, and 16. Initial temperatures of 200°F and 1000°F have

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Figure 14. Calculated Temperature History of a Randomly Tumbling Fuel Element Released at 298.6 Kilofeet

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Figure 15. Calculated Temperature History of a SNAP 2/10A Fuel Element Released at 252.3 Kilofeet

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Figure 16. Calculated Temperature History of a SNAP 2/10A Fuel Element Released at 204 Kilofeet

been used in these calculations, but the actual temperatures at release will depend upon the altitude of release and on the heating of the core vessel. The curves show that the fuel elements heat at almost uniform temperatures up to the dehydrogenation temperature. The rate of rise is then retarded while heat is going into dissociation of hydrogen. After the dehydrogenation is complete, the temperature rises very rapidly to the melting point. The melting process takes a relatively short time, if it occurs at all. For the three cases presented in the figures, burnup was complete for release at approximately 300,000 ft, partial for 250,000 ft, and none for 200,000 ft.

D. PARTICLE BREAKUP AND FISSION PRODUCT RELEASE

Several investigators outside of Atomics International have studied particle breakup. AVCO⁷ and Astropower⁸ have made experimental studies, while Martin⁹ has made a theoretical study of plutonium particle burnup. Our arc-jet tests have indicated that the runback ablates or burns to submicron sizes, but further analysis and experimental work is required before any definite statements can be made on this subject.

At one time, it was proposed that the radioactive fission products might diffuse from the molten fuel and that the particles of alloy and oxides might be nearly free of radioactivity. Recent study has indicated that only a fractional decrease of the total activity would be experienced if all the gaseous and volatile fission products were stripped out of the fuel. Thus it appears that there is not much to be gained in this area.

E. PARTICLE SETTLING AND DISPERSION

The ultimate objective of ablating the fuel alloy is to so disperse it in the atmosphere that no appreciable increase in the background level of radioactivity will occur at any place on Earth. Studies of fallout of volcanic dust¹⁰ and of bomb debris have indicated that the dwell time of particles in the upper atmosphere may be as long as 10 years. During this time, the long-lived radioactivity in the fuel will not decay appreciably, but it will be attenuated by wide dispersal.

A study of particle fallout has been made by General Dynamics/ Astronautics¹¹ in which it is indicated that particle sizes up to about 50 microns may be tolerable. It is planned that this study will be reviewed and extended using newer data for the trajectories and for the particle generation rates.

V. DESCRIPTION OF EXPERIMENTAL PROGRAM

A. ARC-JET TESTS

The ablation of fuel alloy in simulated reentry environments has been studied by use of a low-density stream of gases heated by an electric arc. Two series of tests were made on zirconium hydride (without uranium) in 1960.^{12,13} The only tests of actual fuel alloy (natural uranium) were made late in 1962.¹⁴ These were in the nature of pilot tests, with limited instrumentation, to determine whether the ablation of zirconium hydride and of two types of fuel alloys differed significantly. The preliminary indications resulting from these tests are that the fuel alloy oxidizes much more rapidly than zirconium and that an oxide crust which retards ablation can form over the material in some environmental conditions.

Plans are being made for conducting a new series of arc-jet tests with complete instrumentation. The latest modification of the fuel alloy will be used and the pressure of the oxygen and nitrogen in the jet will be tailored to match reentry conditions. The effects of the cladding in retarding the formation of the oxide crust will also be investigated.

B. FLIGHT TESTS

An instrumented model of the SNAP 10A reactor has been designed to give data on the heating rates to the pump fins, coolant pipe, fusible link band, top head, and the vessel. Four prototypes of this model have been built, two for preflight qualification tests, one for flight, and one for a second flight if the first one fails. The Sandia Corporation is responsible for the management of these tests, and the test plans are described in two of their reports.^{15,16}

The flight test models will not contain any NaK because it was felt that its presence would unduly complicate the analysis of the heat transfer data. Furthermore, its vapor would have interfered with the transmission of data by radio link. Each model contains 21 thermocouples and 8 switches which will signal events, such as the release of the reflectors and the top head.

The trajectory in which the model will be flown is steeper (-6° at 400,000 ft) than an orbital decay trajectory. This is necessary because of guidance limitations on the Scout booster vehicle to be used in the test. Furthermore, use of a

shallower angle would increase the range of the trajectory and complicate the tracking and recovery problems. Although the variation of heat rate with time will be different in a -6° trajectory, it is felt that the experimental data can be resolved into basic parameters which can then be used in calculations for other trajectories.

C. WIND TUNNEL TESTS

Since the amount of instrumentation that can be installed in a flight test is severely limited, it is necessary for wind tunnel tests to be run for determination of local heat rates on various parts of the reactor and associated system. The Sandia Corporation has run a series of tests in the Hot-Shot tunnel at Tullahoma, but these were only for attitudes of 0 and 4° . It is essential that tests be made with other angles. Only wind tunnel tests or flight tests can determine heat rates on a tumbling or oscillating reactor.

D. MECHANICAL TESTS

The mode of release of the fusible-link band has been tested experimentally. It was found that the band released as expected.

The failure of the lip weld on the core-can lip was tested in the radiant heating facility of the Sandia Corporation. It was found that the central part of the top head melted through before the lip failed. Further tests of the mode of vessel failure and core assembly ejection are required.

No tests have yet been made to determine the mode of failure of the system if it contains NaK. It will be necessary to make an experimental investigation of the burn-through of at least a mockup of the coolant pipes and system, since prediction of the mode of release of the NaK is beyond theoretical analysis.

E. THERMOPHYSICAL PROPERTY INVESTIGATIONS

Properties of the fuel alloy, such as thermal conductivity, specific heat capacity, and hydrogen diffusion rates, have been measured at low and moderately high temperatures.^{17, 18} The Armour Research Foundation¹⁹ has recently made measurements of thermal conductivity, specific heat capacity, thermal expansion, thermal emissivity, rate of oxide formation, heat of combustion, and tensile strength. Their measurements will be factored into future calculations. AI will also make property value measurements, with particular regard to such properties as will be required to improve our thermal model of ablation of fuel alloys, including possible modifications to the alloy.

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APPENDIX A

AERODYNAMIC AND INERTIAL CHARACTERISTICS OF THE VEHICLE

Figure A-1 shows the external profile of a typical vehicle which may be used for early flight tests of the SNAP 10A as it is expected to look while in orbit and during the early part of the reentry period. This sketch was taken from an unclassified Lockheed drawing, SK-111, dated May 1961.

A representative set of inertial characteristics of this vehicle has been adopted for use in reentry analysis because the actual characteristics have not yet been made final nor have they been made available to AI. It is believed that the values used are accurate enough for preliminary analysis and for checkout of our procedures. The values adopted are as follows:

Mass of vehicle, $W = 5000 \text{ lb (155 slugs)}$,

Location of the center of mass, $X_{cg} = 80 \text{ in. aft of the mating plane}$,

Moment of inertia in pitch, $I_p = 15,000 \text{ slug-ft}^2$.

The moment of inertia of the vehicle was estimated by summing the second moments of each of the major components, using estimated and/or known values of the masses of the components, as available.

Aerodynamic drag and pitching moment coefficients are shown in Figures A-2 and A-3. These coefficients have been estimated by the use of aerodynamic theory applied to the configuration shown in Figure A-1. Values are given for two flow regimes. In free-molecule flow theory, it is assumed that a molecule of air rebounding after collision with the vehicle does not collide with any other air molecules in the vicinity of the vehicle. The methods used in this regime are described in Reference A-1. The coefficients found by free-molecule theory are applicable when the mean free path of the air molecules is greater than the size of the vehicle. This condition is satisfied in our case at altitudes above approximately 500,000 ft.

In Newtonian theory,^{A-2} it is assumed that all molecules of air striking the vehicle glide off parallel to its surface. This theory gives good results for hypersonic flow, and the estimated coefficients are applicable at altitudes below 500,000 ft down to altitudes at which the vehicle begins to break apart.

AXIAL STATIONS, REFERRED TO MATING PLANE (in.)

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Figure A-1. Typical Configuration of SNAP 10A Orbiting Vehicle

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Figure A-2. Estimated Drag Coefficients for
 SNAP 10A Orbiting Vehicle

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Figure A-3. Estimated Pitching Moment Coefficients
for SNAP 10A Orbiting Vehicle

The drag and moment coefficients of Figures A-2 and A-3 are based on the outside diameter of the vehicle, 5.0 ft, and on the corresponding cross-sectional area, 19.63 ft^2 . A positive moment is one which tends to increase the angle of attack. Angle of attack is the angle between the principal axis of the vehicle and the relative wind velocity vector. Thus, the data of Figure A-3 show that in the free-molecule regime the vehicle tends to trim to an angle of either 0 or 180° . In the Newtonian regime, the vehicle is unstable in the neighborhood of 0 and 180° and tends to trim to an angle of 34° . The magnitude and direction of the moments could be changed drastically by a shift in the center of mass of the vehicle.

The ballistic parameter or mass-to-drag-area ratio ($W/C_D A$) of the vehicle may be calculated from the data of Figure A-2. In the free-molecule flow regime, the maximum value is 127 lb/ft^2 and the minimum value is 19 lb/ft^2 . If the vehicle is tumbling rapidly, the trajectory will depend upon the mean value, 30 lb/ft^2 , found by integration over all angles from 0 to 360° . In the Newtonian flow regime, the value at zero angle of attack is 1270 lb/ft^2 and the minimum value is 28 lb/ft^2 , with a mean value of 59 lb/ft^2 . The proper values to use in trajectory calculations depend upon the orientation of the vehicle with respect to its direction of motion, and if the vehicle is tumbling or oscillating, the values will vary with time.

There is no valid theory for the transition regime between free-molecule and Newtonian flows. It is believed that this transition regime extends over about one decade in atmospheric density, which is equivalent to about 50,000 ft in altitude.

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APPENDIX B

ORBITAL LIFETIMES OF SATELLITE VEHICLES

Consider a satellite of mass, m , in an orbit about the Earth at an altitude, Z . If we make the simplifying assumption that the Earth is a sphere of radius, R_e , then the radius of the orbit is $R = R_e + Z$. We denote the gravitational constant at the surface of the Earth by the symbol, g_o . Then, considering the inverse square law for gravitational attraction, a balance of the forces acting on the satellite in the radial direction yields the following equation:

$$m \frac{V_\theta^2}{R} = mg_o \left(\frac{R_e}{R} \right)^2$$

or

$$V_\theta = R_e \sqrt{\frac{g_o}{R}} \quad \dots(B-1)$$

where V_θ is the tangential component of the velocity of the satellite, V .

The potential energy associated with a given altitude is found by integration of the work which would be done against gravity in lifting the satellite to that altitude.

$$E_p = \int_{R_e}^R mg_o \left(\frac{R_e}{R} \right)^2 dR$$

or

$$E_p = mg_o \left(\frac{R_e}{R_e + Z} \right) Z \quad \dots(B-2)$$

The kinetic energy is expressed by the equation:

$$E_k = \frac{1}{2} mV^2 \quad \dots(B-3)$$

It will be shown in Figure B-1 that the radial component of the satellite velocity is much smaller than the tangential component for all cases of interest above approximately 400,000 ft. Thus, the assumption can be made that $V_{\theta}^2 = V^2$, and Equations B-1, B-2, and B-3 can be combined to give an expression for the total mechanical energy:

$$E_t = mg_o \frac{R_e}{R_e + Z} \left[Z + \frac{1}{2} R_e \right]. \quad \dots(B-4)$$

If this expression is differentiated with respect to time, we obtain:

$$\frac{dE_t}{dt} = \frac{1}{2} mg_o \left(\frac{R_e}{R_e + Z} \right)^2 \frac{dZ}{dt}. \quad \dots(B-5)$$

The aerodynamic drag, D , on a body having a drag coefficient, C_D , defined for a reference area, A , in an atmosphere of density, ρ , is given by the following equation:

$$D = C_D A \frac{1}{2} \rho V^2. \quad \dots(B-6)$$

Using Equation B-1, the power dissipated due to this drag is:

$$DV = C_D A \frac{1}{2} \rho V^3 = C_D A g_o V \frac{1}{2} \rho \frac{R_e^2}{R}. \quad \dots(B-7)$$

Equating this power to the rate of change of mechanical energy of the body gives an expression for the rate of change of altitude, which is the radial velocity:

$$V_r = \frac{dZ}{dt} = - \frac{C_D A}{m} \rho V R. \quad \dots(B-8)$$

The time required for the altitude to decrease from a level, Z_2 , to another level, Z_1 , can be found by combining Equations B-8 and B-1, again using the approximation $V_{\theta}^2 = V^2$:

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Figure B-1. Radial Velocities for Satellites
in Decaying Quasi-Circular Orbits

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$$\Delta t_{2 \rightarrow 1} = \frac{m}{C_D A \sqrt{g_0} R_e} \int_{Z_2}^{Z_1} \frac{-dZ}{\rho(Z) \sqrt{R}}. \quad \dots(B-9)$$

If we assume that the density of the atmosphere varies exponentially over the range of altitude Z_2 and Z_1 , then we can write:

$$\rho = \rho_1 \exp \left[- (Z - Z_1)/H \right] \quad \dots(B-10)$$

where H is the scale height evaluated for same range of altitude. If we integrate over fairly small steps in altitude, such as 50,000 ft, it is fair to use a mean value for R in the interval. Then the integration in Equation B-9 can be performed, yielding:

$$\Delta t_{2 \rightarrow 1} = \frac{m(Z_2 - Z_1)}{C_D A \sqrt{g_0} R \rho_1 R_e \ln \left(\frac{\rho_1}{\rho_2} \right)} \left(\frac{\rho_1}{\rho_2} - 1 \right). \quad \dots(B-11)$$

The glide angle, γ , can be found from a rearrangement of Equation B-8, with the assumption that $\sin \gamma = \gamma$, which is valid for angles in the range of interest. Thus:

$$\gamma = \frac{V_r}{V} = - \frac{C_D A}{m} \rho R. \quad \dots(B-12)$$

The equations given above have been evaluated using the ARDC-1959 atmosphere* and taking the radius of the Earth as 20,902,286 ft. The value of g_0 was taken as 32.174049 ft/sec². Lifetimes have been plotted in Figure 4, where zero time corresponds to an altitude of 300,000 ft. Radial velocities are shown in Figure B-1 and glide angles in Figure B-2. It should be noted that at low altitudes, especially with low values of $W/C_D A$, the glide angles become large and the assumption that $V_\theta = V$ becomes untenable. However, above 400,000 ft, the assumptions made in the above derivations are valid.

*R. A. Minzner, K. S. W. Champion, and H. L. Pond, "The ARDC Model Atmosphere, 1959," Air Force Cambridge Research Center, AFCRC-TR-59-267, August 1959

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Figure B-2. Instantaneous Glide Angles for Satellites
in Decaying Quasi-Circular Orbits

APPENDIX C

VEHICLE ATTITUDES AND OSCILLATIONS

The general problem of calculating the oscillations of a vehicle with 3 degrees of freedom is quite complex. Some insight on the problem can be gained by limiting the analysis to that of motion about one of the axes. If I_y is the moment of inertia of the vehicle in pitch and α is the pitch angle, then the following equation holds:

$$I_y \ddot{\alpha} = M \quad \dots(C-1)$$

where M is the aerodynamic pitching moment, given by the equation:

$$M = C_m A D \frac{1}{2} \rho V^2 \quad \dots(C-2)$$

where, in this equation,

C_m = the moment coefficient (Figure A-3),

A = the reference area, 19.63 ft²,

D = the reference length, 5.0 ft,

ρ = the atmospheric density, slug/ft³,

V = the velocity of the vehicle, ft/sec.

The moment coefficient is a nonlinear function of α . Thus, Equation C-1 cannot be integrated in closed form, and numerical methods of evaluation are necessary.

A more nearly complete form of Equation C-1 would include damping terms, proportional to $\dot{\alpha}$, but it can be shown that the damping is of negligible order of magnitude in this problem. A type of pseudo-damping is provided by the increase in atmospheric density, which increases monotonically throughout the orbital decay period.

A computer program has been written for effecting the solution of Equation C-1, but it is not yet operational.

APPENDIX D

DESCRIPTION OF TRAJECTORY COMPUTATION CODE

The trajectory computation code, RESTORE, in its present stage of development can be used for the calculation of the motion of a point-mass sphere in three-dimensional space in the vicinity of the Earth. The sphere is subjected to the forces of atmospheric drag and gravity while moving relative to a rotating, oblate Earth having an atmosphere fixed with respect to the Earth. The constants and formulas used for the shape of the Earth and for its gravitational field are those given by Clarke.^{D-1} The atmosphere is assumed to follow the model known as ARDC-1959^{D-2} at all latitudes.

The differential equations of motion of the sphere were written in a rectangular Cartesian system rotating with the Earth. Input and output data are in spherical coordinates (altitude, latitude, longitude), but the internal calculations are done in rectangular coordinates so that singularities at the poles are avoided.

The aerodynamic drag is calculated from either of two options: (1) constant C_D , and (2) C_D varying according to the formula:

$$C_D = C_{D,c} + (C_{D,FM} - C_{D,c}) \exp\left(-\frac{1}{3} \frac{\rho_\infty D}{\rho_s \lambda_s}\right) \quad \dots(D-1)$$

where

$C_{D,c}$ = drag coefficient in continuum flow

$C_{D,FM}$ = drag coefficient in free-molecule flow

ρ_∞ / ρ_s = ratio of density, free stream to stagnation

D = diameter of sphere

λ_s = mean free path of atmospheric molecules in stagnation region.

This formula is from the work of Masson, Morris, and Bloxsom.^{D-3}

The aerodynamic heat rates on the stagnation point of the sphere are calculated by the formulas given in Appendix E.

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APPENDIX E

COMPUTATION OF AERODYNAMIC HEATING

A. THE FREE-MOLECULE REGIME

When the density of the atmosphere is so low that the mean free path between collisions of molecules of air is greater than the size of a body traveling through that atmosphere, it is said that the body is in the free-molecule flow regime. If the body is traveling at very high speed, one may consider the molecules of air to be essentially at rest. Thus, the aerodynamic drag and heating may be calculated by consideration of the momentum and energy transfer due to encounters between the body and the molecules. It can be shown^{E-1} that the heating of a surface inclined at an angle, θ , to the direction of motion can be expressed by the equation:

$$Q = a \frac{1}{2} \rho V^3 A \sin \theta \quad \dots(E-1)$$

where

Q = the rate of heat input to the surface, ft-lb/sec,

a = the thermal accommodation coefficient,

ρ = the density of the atmosphere, slug/ft³,

V = the velocity of the body, ft/sec,

A = the area of the surface, ft².

The value of the thermal accommodation coefficient is of the order of 0.9 for encounters between air and most surfaces.^{E-2} An excellent discussion of the physical meaning of this parameter is given by Öpik.^{E-3}

It is convenient to define a parameter for aerodynamic heating which is dependent upon the environment and not upon the specific characteristics of the body which is being heated. Such a parameter can then be calculated along with the trajectory calculations, and local heating rates can be ratioed with reference to this parameter. The reference parameter for free-molecule heating is thus:

$$q_{fm} = \frac{1}{2} \rho V^3 / J \quad \dots(E-2)$$

where q_{fm} is the heat rate per unit projected area of the body, Btu/sec, with a thermal accommodation coefficient of 1.0. The factor, J , is the mechanical equivalent of heat, 778.24 ft-lb/Btu.

B. THE HYPERSONIC, CONTINUUM FLOW REGIME

When the mean free path between collisions of molecules in the atmosphere is much smaller than the size of a body traveling at supersonic velocity through that atmosphere, a shock wave forms around the forward part of the body. The equations for heat transfer to the body in this situation have been worked out by several investigators, notably Fay and Riddell.^{E-4} A semiempirical simplification of Fay and Riddell's equations has been derived by Detra and Hidalgo^{E-5} and has been widely accepted in this field. The simplified equation is:

$$q_{stag} = \frac{N}{\sqrt{R}} \left(\frac{\rho}{\rho_o} \right)^n \left(\frac{V}{V_o} \right)^m \left(\frac{H_\infty - H_w}{H_\infty - H_{300}} \right) \quad \dots(E-3)$$

where

q_{stag} = the heating rate at the stagnation point of a spherical body, Btu/ft²-sec,

R = the radius of curvature of the body surface at the stagnation point, ft,

N = a constant having the value 865.0,

ρ = the density of the local, undisturbed atmosphere, slug/ft³,

ρ_o = the density of the standard, sea level atmosphere, 0.00238 slug/ft³,

V = the velocity of the body, ft/sec,

V_o = a reference velocity, 10,000 ft/sec,

H_∞ = the enthalpy level of the air at the stagnation point, Btu/lb,

H_w = the enthalpy level of air at the temperature of the body surface (wall temperature), Btu/lb,

H_{300} = the enthalpy level of air at 300°K, Btu/lb,

n = an exponent having the value 0.5,

m = an exponent having the value 3.15.

As explained in the section on free-molecule heating, it is convenient to define a parameter to express the heating environment which is not dependent upon the body shape or temperature. Such a parameter is:

$$q \sqrt{R} = N \left(\frac{\rho}{\rho_0} \right)^n \left(\frac{V}{V_0} \right)^m \quad \dots (E-4)$$

where $q \sqrt{R}$ is numerically equal to the heating rate at the stagnation point of a cold (300°K) sphere of 1 ft radius, Btu/ft²-sec. This parameter is called "reference heat rate," and heat rates on bodies of various sizes, shapes, and surface temperatures can be calculated by the use of simple proportion factors applied to this parameter.

If the surface temperature of the body is high, the enthalpy term in Equation E-3 becomes important. Figure E-1 presents values for this term for two assumed surface temperatures over a range of body velocities.

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Figure E-1. Correction Factor for Effect of Body Surface Temperature on Aerodynamic Heating Rate

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Figure E-2. Calculated Aerodynamic Heating Rates at the Stagnation Point of a Sphere of 1-ft Radius Moving at 26,000 ft/sec

C. TRANSITION FROM FREE-MOLECULE TO CONTINUUM FLOW REGIMES

Calculated values for reference heat rates as a function of atmospheric density ratio for an assumed velocity of 26,000 ft/sec are shown in Figure E-2. It may be noted that the curves for free-molecule flow and continuum flow cross at a density ratio of about 4×10^{-7} , which corresponds to an altitude of about 320,000 ft. Below that altitude, the use of free-molecule flow theory would give heat rates that would be much higher than actual. Conversely, the use of continuum theory above that altitude would give falsely high values.

Valid theories do not presently exist for calculation of heat rates in the transition region where the two curves cross. However, according to Probstein^{E-6} the extent of the transition region is probably one decade or less in gas density. Since this region is not particularly important for reentry burnup studies, except possibly for reflector-retaining-band temperature calculations, the expedient has been adopted of calculating heat rates by the use of both theories and selecting the lower of the two values.

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APPENDIX F

ESTIMATES OF LOCAL HEATING RATES

A. FUSIBLE-LINK BAND

Most of the heating of the band occurs in the free-molecule flow regime, since the band is relatively small and it reaches its melting temperature while still at a relatively high altitude. When the vehicle attitude is zero degrees, the projected area is taken to be the edgewise projection of the band. When the vehicle is at 90° , the conservative assumption is made that there is a spinning motion that distributes the heat over the band, and the mean heat rate is taken as $1/\pi$ times the reference heat rate for free-molecule flow. The accommodation coefficient is taken to be 0.9.

B. THERMOELECTRIC PUMP FINS

The fins are assumed to be heated only at their leading edges. Much of the heating occurs in the free-molecule regime. After melting begins, the effective radius of the leading edge is taken as half the thickness of the fin and the stagnation line heat rate is found by dividing the reference value by the root of 2.0 to correct for two-dimensional flow and by the root of the effective radius. The average heat is related to the stagnation heat by the ratio $2/\pi$.

C. COOLANT PIPES

The coolant pipe across the top end of the reactor is analyzed as a cylinder in cross flow with a radius of 0.625 in. and with the reference heat rate divided by the root of 2.0 for the two-dimensional flow effect. There is experimental evidence that the thermal switches along the front of the pipe will cause the heat rates to be increased between the switches, but this effect has been omitted in preliminary analyses.

D. LIP OF CORE VESSEL

In early analysis, the lip was treated as if it were a small cylinder in cross flow. This gave heat rates that were suspected to be too high, so another analysis was made in which the compression in the bow shock from the pump magnet was calculated by ideal gas theory, using the methods of Fay and Riddell.*

*J. A. Fay and F. R. Riddell, "Theory of Stagnation Point Heat Transfer in Dissociated Air," Journal of the Aeronautical Sciences, Vol 25, (February 1958), Equation 63

The ratio of the heat rate on the lip to the reference heat rate was estimated to be 2.1. This ratio is approximately the same as for a hemisphere of 5-in. radius in a free stream, which shows dramatically the effects of the pump and pipe in sheltering the lip. The confidence level of this estimate is very low, and experimental measurements are needed.

APPENDIX G

THERMAL MODELS FOR HEATING AND ABLATION

A. GENERAL STATEMENT

Transient temperatures in various components of the system are calculated with the aid of a digital computer code, TAP-2, written for use on the IBM-7090 computer. The name of this code stands for "Thermal Analyzer Program," and the code uses finite difference methods for integration of the equations of heat flow in lumped systems. The physical system is assumed to be divided into several discrete sections, called nodes, and the thermal capacity of each node is calculated from the volume and the specific heat capacity of the material. Thermal resistances between each pair of connected nodes are calculated from the heat flow path length, cross-sectional area, and thermal conductivity. Provisions are available for inclusion of radiant heat exchange between surfaces, heat conduction and convection, internal heat generation, and heats of phase transformation. An important feature of the code is that any parameter of the system may be varied as a function of time or of any other parameter according to tabular inputs. This makes it possible to disconnect nodes which have exceeded their melting temperatures and to connect the sources of aerodynamic heat to other nodes as the ablation of the material takes place.

In the analyses of this report, aerodynamic heating is put into the calculations by the use of dummy nodes and resistors. The potentials of the dummy nodes are made to vary in proportion to the intensity of the aerodynamic heating as a function of time. The resistances are assigned values inversely proportional to the local surface areas which receive the heat. Thermal radiation from external surfaces of the physical system is put into the calculations by the use of dummy nodes at sink temperatures and resistors whose values are calculated within the code as functions of the terminal temperatures and of input values of the surface area, emissivity, and shape factor. For radiation to open space, the sink nodes are set at a temperature of absolute zero.

B. FUSIBLE-LINK BAND

A portion of the fusible-link band is shown in Figure G-1 with indications of how it was considered to be divided into nodes. Portions of the standoff posts

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Figure G-1. Portion of Fusible Link Band

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and of the control drum hinge brackets were included because of their possible heat-sink effects. It was found, however, that at zero angle of attack, the posts and brackets were heated more than the band, which is a desirable circumstance.

C. THERMOELECTRIC PUMP FINS

The physical system of Figure 9, with node divisions as shown, was approximated by a lumped-parameter network. Aerodynamic heating was assumed to occur only at the leading edge of the fin, although there would actually be a small amount of heat input to the lateral surfaces of the fin. No allowance was made for heat radiation from other components of the reactor nor for interchange between the fins. Thus, the heat inputs have been taken as lower (more conservative) than the probable values. The thermal properties of the fin material, aluminum alloy, were taken to be as given in Appendix H.

D. COOLANT PIPE FORWARD OF PUMP

The portion of the coolant pipe considered in these analyses lies crosswise to the axis of the vehicle. Two thermal models were considered, one representing the empty pipe and one representing the pipe filled with NaK coolant (not flowing). A diagram showing part of the thermal network used for the empty pipe is shown in Figure G-2. Not shown, but used in the calculation, are radiation paths between each of the possible pairs of nodes which can exchange radiation inside the pipe. As each node melts, heat flow from it to other nodes is cut off, and in some cases new heat flow paths are opened. For example, with reference to Figure G-2, when node 14 melts, dummy node 5 is switched on so that heat flows to the edge of node 15. Also, dummy node 10 is allowed to supply heat to node 18. The external surface of each node, except node 19, radiates to space. In addition, nodes 17 through 19 radiate to a sink at 200°F, representing portions of the pump magnet.

For the case of the pipe filled with NaK, the internal heat flow paths were replaced by paths to three concentric nodes representing the NaK. An analysis of the effects of flow in the NaK has not yet been made.

E. LIP OF CORE VESSEL

The lumped-parameter network approximation to the upper edge of the core vessel, including the lip, is shown in Figure G-3. Circumferential heat flow has

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Figure G-2. Thermal Network for Empty Coolant Pipe

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Figure G-3. Thermal Network for Core Vessel Lip Heating

been neglected in this model, but will be included in later analyses. Also, the upper part of the vessel has been assumed to be empty of NaK on the premise that, by the time the lip melts, the coolant pipes will have ruptured and drained the NaK. Later analyses will include the effects of NaK for at least the early part of the heating period. The properties of the material, stainless steel, were taken as given in Appendix H.

F. FUEL ELEMENT

A circular cross section through the fuel element was considered to be divided into 12 concentric nodes as shown in Figure G-4. The thermal resistance and capacity of the cladding was lumped together with the outermost fuel node. The assumption was made that the fuel element was turning about its major axis rapidly enough to equalize the heat input over its surface. This was done to make the problem one-dimensional and was considered to be justified for preliminary analysis. Comparative calculations for nonspinning rods will be made in the future. The thermal properties of the fuel material were taken to be as given in Appendix H.

POTENTIAL VARIES WITH TIME
TO SIMULATE AERO HEATING

RADIATION HEAT
SINK

CONDUCTANCE VALUES
DEPEND UPON SURFACE
AREAS

* AS EACH NODE ABLATES, CONNECTIONS
TO NEXT NODE ARE SWITCHED ON OR
OFF AS REQUIRED.

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Figure G-4. Thermal Network for Sector of Fuel Element

APPENDIX H

THERMOPHYSICAL PROPERTIES OF REACTOR MATERIALS

A. FUEL ALLOY

The following property values were used as typical of the fuel material:

Thermal conductivity	15 Btu/ft-hr-°F
Specific heat capacity	0.1 Btu/lb-°F
Density	0.216 lb/in. ³
Thermal emissivity	0.4
Melting temperature	3365°F
Heat of fusion	90 Btu/lb
Heat of hydrogen release	610 Btu/lb

B. STAINLESS STEEL

The coolant pipes are made of stainless steel, type 405, and the core vessel is of type 316. The following property values were used for the analyses of this report:

Thermal conductivity (0°F)	8.0 Btu/ft-hr-°F
(600°F)	11.1
(1200°F)	13.6
(1800°F)	16.1
(2790°F)	20.3
Specific heat capacity	0.15 Btu/lb-°F
Density	475 lb/ft ³
Thermal emissivity	0.6
Melting temperature	2790°F
Heat of fusion	120 Btu/lb

C. ALUMINUM

The fins on the thermoelectric pump are made of aluminum, with a coating for high emissivity. Property values used were as follows:

Thermal conductivity (212°F)	118.6 Btu/ft-hr-°F
(572°F)	155.0
(1112°F)	244.5
Specific heat capacity (212°F)	0.225 Btu/lb-°F
(572°F)	0.248
(1112°F)	0.277

Density	168.5 lb/ft ³
Thermal emissivity	0.9
Melting temperature	1218 °F
Heat of fusion	170.2 Btu/lb