NUCLEAR POWERED SATELLITE STUDIES

Annual Progress Report
for the Period July 1, 1978 - June 30, 1979

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ABSTRACT

Progress achieved during the period July 1, 1978 to June 30, 1979, on U.S. D.O.E. Contract No. EY-76-S-02-4045.A002 is reported. Discussions of several pertinent aspects are included, e.g., schedule, personnel, technology developments, and plans. The reporting period represents the third year of activities of a project which is designed to provide continuing support in the area of nuclear space power technology. Important results have been obtained and communicated to the scientific community via publications and presentations. Project personnel have participated in point design efforts, a NASA Space Power Committee, and periodically provide support to Department of Energy activities as the need arises. This report is intended to summarize activities over the past year and provide a basis for continued project support by U.S. D.O.E.

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1.0 INTRODUCTION

1.1 Summary of Progress

The then U.S. Energy Research and Development Administration awarded the Pennsylvania State University Contract No. EY-76-S-02-4045.*000 on July 1, 1976. This takes the form of a special research support agreement which is now being conducted under the guidance of the Advance Systems and Materials Production Division of the U.S. Department of Energy. The project emphasis is on nuclear powered satellite technology and applications. Activities have been established such that continuing support of the Space Applications activity within the Department of Energy has been provided while related new technologies are developed.

During the first year of work some time was spent in part establishing task assignments and locating personnel to staff the project. A variety of activities were identified in order to fulfill project objectives and provide real time support to the Space Applications team. Contacts in industry and government were established for continued interaction throughout the project tenure. Presentations continue to be made to potential users of nuclear power supplies and project staff participate in design reviews and other related briefings as the occasions arise. The technical basis of this project is the development of tools and inputs to be used in evaluating applications of nuclear power to satellites. Considerations include new constraints of the Space Transportation System, projected missions through the 1990's, and technology advances in nuclear power supplies.

Communication of ideas and developments is a key factor in any research project, but it is particularly important here. The principal investigator is continually interacting with industry and government agencies in order to

maintain a liaison with as many interested parties as is practical. New developments are documented and presented to the scientific community in a timely manner. During the first three years of this project several publications have been generated. A number of presentations have been made at technical conferences. Spacecraft designers, NASA, and users have been the audiences. In-house briefings have also been conducted at user and contractor facilities. A number of areas of technology development are being addressed. These include a comprehensive simulation and computation capability, design aspects of nuclear powered synchronous satellites and orbit transfer vehicles, structural analysis of spacecraft elements (e.g., radiator, heat source assembly, etc.), orbital decay of spacecraft, reentry dynamics of isotope heat sources, attitude control of dynamic isotope powered spacecraft, and reactor power station design aspects.

1.2 Major Results

Although details of technical developments are left for section 4 and the appendices, major results are summarized here. A comprehensive computer capability has been established within the Astronautics Research Laboratory. The two input/output terminals and an interactive graphics plotter have permitted the development of an extensive array of project-related software, including a general purpose attitude dynamics simulation, structural and thermal analysis during reentry, and orbit raising and decay simulations. A preliminary design of a nuclear powered communications satellite was developed to satisfy projected requirements of INTELSAT in the 1980's, the results of which are soon to appear in the Journal of Spacecraft and Rockets.

A nuclear/electric orbit transfer vehicle has been conceived and results were presented to NASA as an alternative to the payload limited IUS. This work will be presented at the 14th International Electric Propulsion Conference, Princeton, October 1979. A unique capability exists here to simulate orbital decay and attitude motion simultaneously. We have done this for Skylab and the General Purpose Heat Source (GPHS). This technology can be applied to any spacecraft or component of a satellite.

1.3 Summary of Plans

Plans for further work must be divided according to funding periods. The present contract is valid until September 30, 1979. Activities beyond that time are dependent upon funding availability and negotiations with the Department of Energy. Nevertheless, both periods are discussed briefly in section 5. To summarize, the remaining three-month period (July 1 to September 30, 1979) will provide an opportunity to complete activities underway and begin new efforts related to changing emphasis of Department of Energy programs. Present capabilities will enable completion of GPHS reentry stability studies with a variety of center of mass position options. Work on a preliminary design of a nuclear/electric transfer vehicle which can also provide 100 kWe in geosynchronous orbit will be completed by September 30.

2.0 SCHEDULE AND PERSONNEL

2.1 Milestone Chart

A milestone chart is used to periodically summarize activities and plans. Of course, such a chart can only estimate the timing and extent of any given task. Figure 2-1 depicts the latest summary of activities at the overview level. Details are left for later sections of this report. Projections are provided only to the end of the current grant period. However, an additional chart will appear in the renewal proposal for GFY 80.

2.2 Staffing

Since the beginning of this project staffing has fluctuated in size and talents. Both personnel and assignments have changed during the year. However, appropriate adjustments have been made in order to fulfill primary contract objectives. Figure 2-2 shows the organization chart currently in effect with assignments of individuals outlined. Faculty participants include Dr. M. H. Kaplan, the principal investigator, who has been devoting 40-50 percent of his time, on the average, since the beginning of the project. Dr. Kaplan is responsible for the budget, schedule, and presentations. This is in addition to the technical responsibilities related to spacecraft design and mission analysis. He received the 1978 Award for Outstanding Achievement in Research in the College of Engineering of the Pennsylvania State University. In December 1978, his new book, Space Shuttle: America's Wings to the Future, was released by AERO Publishers. Dr. Kaplan is a Professor of Aerospace Engineering and Director of the Astronautics Research Laboratory. Dr. James C. Wambold, Associate Professor of Mechanical Engineering, has been guiding the development of the structural and thermal

Legend:

- ☒ Report
- △ Meeting
- Briefing

Figure 2-1. Milestone Chart for GFY 79

Figure 2-2. Organization Chart for GFY 1979

analysis software for our reentry studies. He has been spending 10 percent of his time on this project and advising one of the graduate students. Stephen G. Alexander is a Senior Research Technologist and will soon complete his Master of Science work. Mr. Alexander is working full-time on this project and is responsible for overseeing computer facility operation, orbital decay programs, and simulation software. Mr. Lin-Shu Yang is a Ph.D. candidate and has been working on a half-time basis to develop stress and thermal analysis models of nuclear satellite components. Mr. R. M. Trn and Mr. R. M. Fredo are graduate assistants working half-time on the project. Occasionally, undergraduates assist with a variety of support tasks while working on senior theses.

2.3 Plans

The organization and schedule of the project seem to be appropriate and effective in carrying out the tasks assigned. Therefore, there are no plans to change the schedule or basic organization before September 30, 1979. Personnel seem to be appropriately placed with regard to experience, education, and interest. Additional graduate assistants will become active in September, assuming continuation of funding for GFY 80.

3.0 LIAISON ACTIVITIES

3.1 Presentations

A number of relevant presentations have been made over the past three years by the research team. For example, presentations were given at General Electric and Fairchild Industries on nuclear space power as part of a systems engineering course during the fall of 1977. Other presentations have been made at industrial facilities, and these will continue to be offered when appropriate. A number of radio and television interviews have been aired. Furthermore, the principal investigator presented several papers dealing with the use of nuclear power supplies and related technology. One of these was given at the Seventh AIAA Communications Satellite Systems Conference, San Diego, April 24-27, 1978, entitled, "A Nuclear Powered Communications Satellite for the 1980's." This will be published in the May-June 1979 issue of the Journal of Spacecraft and Rockets. A paper was presented at the Tenth AIAA/IES/ASTM Space Simulation Conference on orbital decay of Skylab, in October 1978. This will be published soon in the Journal of Guidance and Control.

3.2 Publications

Several publications have been generated since the project began. Articles, papers, and reports relevant to this work are listed below.

1. M. H. Kaplan, "Nuclear Powered Satellite Design for Shuttle Launches," AIAA Preprint 77-508, St. Louis, March 1-3, 1977.
2. M. H. Kaplan, "Isotope Powered Satellites for Shuttle Launches," AIAA Journal of Spacecraft and Rockets, Vol. 14, No. 11, November 1977, pp. 703-704.

3. M. H. Kaplan, "Space Nuclear Power Supply Design in the Shuttle Era," Proc. of the 12th IECEC, Washington, D.C., August 28 - September 2, 1977, pp. 1323-1328.
4. M. H. Kaplan, D. J. Cwynar, and J. J. Bednarz, "A Nuclear Powered Communications Satellite for the 1980's," Proc. of the AIAA 7th Communications Satellite Systems Conference, San Diego, April 24-27, 1978, pp. 655-659.
5. M. H. Kaplan, D. J. Cwynar, and S. G. Alexander, "Anticipated Attitude Motion of Skylab for a 1979 Revisit Mission," Journal of Spacecraft and Rockets, July/August, 1978, pp. 219-223.
6. M. H. Kaplan, D. J. Cwynar, and S. G. Alexander, "Simulation of Skylab Orbit Decay and Attitude Dynamics," AIAA Paper 78-1626, presented at the 10th AIAA/IES/ASTM Space Simulation Conference, October 1978.
7. - - - - -, "A Life Cycle Cost Comparison for an Operational Seasat Mission Using Nuclear and Solar Power Supplies," ECON, Inc. Report 78-292, November 7, 1978.
8. M. H. Kaplan and S. G. Alexander, "Skylab Orbit Decay Update and the Eleventh Hour Contingency Plan," AAS Paper 79-032, presented at the AAS Rocky Mountain Guidance and Control Conference, February 1979.
9. M. H. Kaplan, D. J. Cwynar, and J. J. Bednarz, "A Nuclear Powered Communications Satellite for the 1980's," Journal of Spacecraft and Rockets, May/June 1979.
10. M. H. Kaplan, D. J. Cwynar, and S. G. Alexander, "Simulation of Skylab Orbit Decay and Attitude Dynamics," Journal of Guidance and Control, 1979.
11. M. H. Kaplan, R. M. Trn, and D. Buden, "A Nuclear/Electric Orbital Transfer Vehicle for the Shuttle Era," to be presented at the AIAA/DGLR 14th International Electric Propulsion Conference, October 30 - November 1, 1979, Princeton, N. J.

A number of student reports and theses have been completed or are well underway:

Master of Science Theses

D. J. Cwynar, "An Automatic Attitude Control System for a Dynamic Isotope Powered Satellite," (also appears as P.S.U. Astronautics Research Report No. 78-1) May 1978.

S. G. Alexander, "Simulation of Orbital Decay for Satellite and Components," August 1979.

Doctor of Philosophy

L. S. Yang, "Structural Analysis and Design for Nuclear Powered Satellites," 1980.

Undergraduate Project Theses

A. Y. Harrison, "Increasing the Output Power of the Brayton Isotope Power System," February 1978.

J. Trivisonno and C. Morgret, "Spacecraft Motion Simulation Using Digital Graphics," February 1978.

M. Hill, "Heat Transfer Analysis of a Rotating Thin-Walled Cylindrical Radiator and Heat Pipe Assembly for Possible Application as a Waste Heat Rejection Unit for a Spacecraft Powerplant," February 1978.

J. Ipoletta and M. Goodson, "Skylab Orbital Decay Projections," February 1978.

A. Muffoletto, "Thermal and Structural Design for GPHS," May 1979.

3.3 Point Design Activities with Government and Industry

The principal investigator and other project staff members have been involved in preliminary evaluations of nuclear power applications to DOD and NASA missions from time to time. Limited interaction with Fairchild Industries on GPS, GPSCS, and SSS took place during the previous years. Funds left over from GFY 77 were used to initiate work by ECON, Inc., on a future NASA program, COASTSAT. Lockheed carried out a preliminary design effort on both a solar and a dynamic isotope powered version of such a satellite. ECON has completed economic comparisons of these designs and issued a final report on November 7, 1978.

The principal investigator has made a number of trips to NASA, DOE, and contractors. For example, he has visited the Los Alamos Scientific Lab a number of times to discuss studies related to General Purpose Heat

Source reentry into the Earth's atmosphere and applications of nuclear reactors in the 1980's. Of course, visits have been made to the DOE offices in Germantown, Maryland. Further plant visits will be made in connection with design reviews or any special activity as requested of the principal investigator.

The principal investigator was asked in 1978 to join the Ad Hoc Advisory Subcommittee on Power Technology for Future Synchronous Satellites and Platforms of the NASA Advisory Council, Space Systems and Technology Advisory Committee. As part of this activity, a preliminary design of a nuclear/electric orbital transfer vehicle has been formulated. A presentation on this was made to this subcommittee on May 2, 1979 by Penn State and LASL (D. Buden). As a result, the nuclear option will be included in the official recommendations to the NASA Advisory Council in June 1979.

4.0 TECHNOLOGY DEVELOPMENTS

This chapter is intended to provide summaries of each technology area developed over the past year. Detailed descriptions of analyses and simulations are presented when appropriate in appendices. Several tasks have led to separate publications which have been cited in section 3.2

4.1 Satellite Simulation Software

Many programs have been developed to simulate satellite motion under a variety of conditions. These are briefly presented here.

The Aerodynamic Coefficient Program determines drag, lift and moment coefficients on bodies of any geometry in the free molecular flow regime. For most orbiting bodies, this regime is above 80 miles altitude. The program uses a simplified flow model based on classical free molecular flow theory. Work on determining the aerodynamic force coefficients on a body in hypersonic continuum flow is continuing.

The Orbit Decay Routine uses a discrete orbital energy dissipation technique to simulate the decay of an orbiting body. The idea is that the kinetic energy dissipated due to aerodynamic drag, over a set time increment, is subtracted from the orbit energy to determine a new value of orbit energy. From this new value of energy, a corrected orbit radius can be computed. This procedure is iterated with an appropriate time step to obtain the orbital decay trajectory. Calculation of the aerodynamic drag force involves a predetermined drag coefficient, reference area, and value of local atmospheric density. This local density is computed from the Jacchia (1970) atmosphere model, which accounts for density variations due to altitude, position

(longitude and latitude), solar activity and geomagnetic effects. Actual and future predictions of solar activity data are continually being obtained from NASA and NOAA.

The orbit decay routine has been exercised for computing the decay trajectory for the GPHS fuel module, Skylab and Cosmos 954. The routine can be used for any orbiting body whose drag coefficient is available and whose orbit is circular.

The Orbit Simulation Algorithm can accurately predict the motion of an orbiting body in any type of orbit. It can be used for simulating orbital transfers and orbital decay problems. For example, it can accurately predict the path of the GPHS fuel module during the last few orbits of reentry. The routine involves the numerical integration of three differential equations of motion over a specified time increment. Included in the equations are the forces (aerodynamic, solar pressure, engine thrust, etc.) acting on the orbiting body.

In the Attitude Dynamics Routine, the equations of attitude motion are numerically integrated to determine the motion of a rigid body over a specified period. The routine simulates attitude motion for all body geometries, and it handles aerodynamic, gravity gradient and control torques. This routine has been used to simulate attitude motion for Skylab and the GPHS fuel module.

The attitude dynamics routine and the orbital simulation program have been merged to simultaneously simulate attitude and orbital motion of a rigid body. This program has been used extensively for simulating the motion of Skylab.

4.2 GPHS Reentry Stability

The determination of the GPHS module reentry equilibrium orientation is important because of its influence on the aerodynamic heating of the module. The possible orientations of stability can be found through an analysis of the aerodynamic loading (pressure distribution) on the module. Once the pressures on the module have been computed, the relationship between the aerodynamic moments and orientation angle can be made. From this, statically stable orientations can be found (see Appendix A).

Using the reference system in Figure 4-1, if angular motion of the module is about the z-axis, then the module is statically stable at 74° and 106° as diagrammed in Figure 4-2. If motion is about the y-axis, then the module is statically stable at 45° , also shown in Figure 4-2. The results for motion about the x-axis are very similar, because of the module geometry, to the results for the z-axis.

A shift in the module's center of gravity was considered in order to obtain a more favorable position of static stability. From the many shifts considered for motion about the z-axis, the shift most desirable was that of 5 mm in the y-direction. From the moment plot, there are statically stable positions at 66° , 115° and 270° as diagrammed in Figure 4-3. But, the angular position of 270° is much more stable than the other positions. This means that for damped rotation about the z-axis, the module will very often stabilize at 270° or "face-on."

These results depend on the accuracy of the two-dimensional flow field pressure distributions for motion about each axis. Uncoupling the three-dimensional motion of the module into separate rotations about each axis

Figure 4-1. GPHS Module Reference System

Figure 4-2. GPHS Module Static Stability Orientations

Figure 4-3. GPHS Module Static Stability Orientations for Z-axis Motion with C. G. Shift ($\Delta Y = 5.0$ mm)

and analyzing the two-dimensional flow field can only be considered a rough approximation to the actual conditions. Attempting to propose stable orientations for the general case of three-dimensional random angular motion would be unjustified without computing the three-dimensional flow field. Such computations are currently being considered.

4.3 Early Reactor Mission, Preliminary Design

The need for larger, high power payloads in Geosynchronous Earth Orbit (GEO) initiated the preliminary design of a nuclear/electric orbit-to-orbit transfer vehicle. The proposed vehicle would be taken to low Earth orbit by the NASA Space Shuttle and then work its way to GEO using ion thrusters powered by a nuclear reactor power system. Safety considerations specified that the initial altitude at which the vehicle would begin transfer, must produce a "safe" orbital decay time. The mission then involved an orbital transfer in the form of an ever-widening spiral from the "safe" initial altitude to GEO taking several months to complete.

A major part of the preliminary design study was centered around the selection of a thruster subsystem. The type and number of thrusters used directly influences the amount of payload that can be placed into GEO in a given mission timespan. Choice of a particular thruster subsystem was influenced by Shuttle payload and size constraints and the maximum output of the particular power plant.

Results of this study were put into the form of mission design charts which depict payload ratio as a function of initial vehicle mass and mission time.

4.3.1 Nuclear Reactor Technology

The nuclear power plant components include the reactor, radiation shield, power conditioning equipment and the waste heat radiator. Typical assemblies of these components are shown in Figure 4-4. Current electrical converters have efficiencies ranging from about 12 percent early in their design life to a "steady-state" efficiency of 9 percent. A typical 100 KWe power source must then produce 1110 KWt (thermal).

The particular power plant examined in our study is the Los Alamos TE type with a conical radiator. Data for this power plant is given in Table 4-1. Since the orbital transfer time of the proposed vehicle is a small

Table 4-1. Typical Power Plant

Type: Los Alamos TE type with conical radiator

Lifetime	7 years	
Reactor Power (KWt)	1110	
Specific Mass (We/Kg)	61	
Total System Mass (Kg)	2131	
	<u>Efficiency (%)</u>	<u>Power (KWe)</u>
Early Life	12	130
Steady-State	9	100

portion of the reactor design life, it can be assumed that the vehicle will have at its disposal a power level corresponding to the early life electrical efficiency (12 percent) throughout the transfer. The vehicle then has about 130 KWe for use by the ion thruster subsystem. The system mass given in Table 4-1 includes the mass of the conical type radiator.

Figure 4-4. Typical Power Plant Assemblies

4.3.2 Ion Engine Technology

An important part of the preliminary design study is the choice of the thruster subsystem. Which and how many thrusters we use influences the amount of payload the vehicle can place into GEO in a given time. Four thrusters were compared on the basis of their performance of the desired mission, LEO to GEO. Performance data of these engines are given in Table 4-2.

Table 4-2. Typical Ion Engine Performance Data

<u>Type</u>	<u>Thrust N</u>	<u>Specific Impulse-sec</u>	<u>Efficiency %</u>	<u>Power Req'd KWe</u>	<u>System Mass Kg</u>
<u>30 cm-diameter HG Bombardment Type Thrusters</u>					
NASA-Lewis	.129	2900	61	3.06	51.36
JPL/Hughes	.370	9000	80	22.68	245.78
NASA-Lewis	.500	9000	80	30.65	325.46
<u>50 cm-diameter HG Bombardment Type Thruster</u>					
Aerospace Corporation	.428	2071	63	7.28	96.34

*Note: All system masses calculated with assumed power processor specific mass of 10 Kg/KWe and electrical efficiency of 90 percent. A fuel tank mass of 12 Kg for each thruster is also included.

The first thruster listed is a typical low thrust, low specific impulse and low efficiency type whose only impressive feature is a low power requirement. The following two thrusters are representative hi-thrust, hi-Isp devices with corresponding high power requirements and system masses. One advantage of hi-Isp thrusters is their inherent low mass flowrate. Also included in the table is a 50 cm-diameter thruster described in a study performed by the Aerospace Corporation. This is a hi-thrust, low-Isp thruster with power and mass characteristics somewhere between that of the previous lo-thrust, lo-Isp and the hi-thrust, hi-Isp 30 cm-diameter devices. The 50 cm-diameter thruster was referred to as a "sized up 30 cm-diameter thruster."

Notice that with a fixed power supply the number of thrusters one can use is limited by their individual power requirements and size. The maximum thrust that is attainable with a given thruster subsystem therefore is also limited.

4.3.3 Mission Constraints/Objectives

Size and Weight Constraints. The vehicle is to be carried to LEO by the Space Shuttle and therefore its maximum size is limited by the dimensions of the Shuttle bay. The maximum length and diameter of the Shuttle bay is 60 ft. and 15 ft., respectively. The maximum weight is a function of the required initial altitude. The Shuttle can deliver 65,000 lbs of payload to 160 nm altitude. Higher initial altitudes require the addition of OMS Kits and thereby reduce the maximum allowable size and weight of the proposed vehicle.

Orbit Considerations. The initial altitude where the vehicle begins transfer to GEO is chosen to give the vehicle a "safe" orbital decay time. For this preliminary study the "safe" decay time was defined as 500 years. Estimates of vehicle size, shape and weight were used to calculate a ballistic coefficient which was then related to the minimum initial altitude for a 500 year lifetime as determined by drag studies (see Appendix B).

4.3.4 Time of Transfer Calculations

What was needed for this study was an easy straightforward approach to calculate the minimum propulsion time for transfer between inclined circular orbits taking into consideration the vehicle mass and available thrust. A closed form solution of the orbital equations of motion was found that was easily programmed and enabled a quantitative evaluation

of the various thruster/power plant combinations. For a specific vehicle powered by "N" number of a given thruster, the time of transfer from the "safe" initial orbit to GEO can be found by:

$$\text{Time of Transfer} = \frac{V_o * \text{Mass}}{N * \text{Thrust}} \sqrt{1 - 2 \left(\frac{V_f}{V_o} \right) \cos(\sqrt{2} \text{ Incl}) \left(\frac{V_f}{V_o} \right)^2}$$

where: Thrust/Mass ratio is assumed constant throughout the transfer

V_o is the velocity in the initial orbit

V_f is the velocity in the final orbit (GEO)

Incl is the inclination change (28.5°)

Once the mission time is established the amount of fuel required and payload can be calculated.

4.3.5 Results

Table 4-3 shows the results of a typical analysis. The mission is defined by an initial vehicle mass out of the Shuttle, an initial "safe" altitude, final altitude (GEO) and the inclination change. All such analyses were performed using power and mass characteristics associated with the Los Alamos TE type reactor power system. From the Table, a 10,000 Kg vehicle using 36 NASA-Lewis thrusters can arrive at GEO in 136 days, 16 hours. The payload delivered to GEO is defined by taking the initial vehicle mass and subtracting from it; the mass of the propulsion system, the mass of the fuel and the mass of the power plant.

For comparison the results obtained for one of the hi-thrust, hi-Isp thrusters has been included. Five of these can place a 6022 Kg payload into GEO in 340 days, 14 hours. The results of our study to date have been summarized in the form of mission design charts. These charts (drawn for each thruster studied) give the payload ratio possible at GEO as a function of mission time and initial vehicle mass (see Appendix B).

DATA

MISSION: INITIAL VEHICLE MASS 10000 KG
 INITIAL ALTITUDE 875 KM
 FINAL ALTITUDE GEO
 INCLINATION CHANGE 28.5 DEG

POWER PLANT: LOS ALAMOS TE TYPE
 OUTPUT POWER 130KW_e
 SYSTEM MASS 2132 KG

THRUSTER: NASA-LEWIS 30 CM-DIA HG BOMBARDMENT TYPE
 THRUST .129 N I_{sp} 2900 SEC EFF .67
 POWER REQD. 3.06 KW_e SYSTEM MASS 51.4 KG/UNIT
 MASS FLOWRATE .454E-06 KG/SEC

NO# OF	SYSTEM	T/M	FUEL	PAYLOAD	TIME	
THRUSTERS	MASS-KG	KM/SEC ²	KG	KG	DAYS	HOURS
36	1848.9	.464E-06	1914.3	4104.9	135	16.3

THRUSTER: JPL/HUGHES 30 CM-DIA HG BOMBARDMENT TYPE
 THRUST .370 N I_{sp} 9000 SEC EFF .80
 POWER REQD. 22.68 KW_e SYSTEM MASS 245.78 KG/UNIT
 MASS FLOWRATE .419E-05 KG/SEC

NO# OF	SYSTEM	T/M	FUEL	PAYLOAD	TIME	
THRUSTERS	MASS-KG	KM/SEC ²	KG	KG	DAYS	HOURS
5	1228.9	.185E-06	616.8	6022.3	340	14.2

Table 4-3. Typical Analysis Data

4.3.6 Summary

It was determined that there exists an optimum value of specific impulse that will produce the maximum payload at GEO. The best thruster for our particular mission is the low specific impulse type for which we have two choices of thrust level. One, the low thrust type represented by the NASA-Lewis device (thrust = .129 N) of which 36 are required. This thruster has the advantage of being a fully tested off-the-shelf item. Or, we may use the high thrust (.428 N) 50 cm-diameter model suggested in the study performed by the Aerospace Corporation. Only half as many of these would be needed if they were to become available. An important consideration is that the 50 cm-diameter thruster would have to be developed.

The maximum payload that can be placed into GEO in six months is from 6700 to 9100 Kg depending upon which of the two thrusters is used.

The conclusion of the preliminary study is that it is technically feasible to design a vehicle which can raise large amounts of payload and power to Geosynchronous altitude and is compatible with the NASA Space Shuttle. In addition the concept can be shown to be more economical than present solar powered vehicles proposed for similar missions.

4.4 Structural and Thermal Analysis Capabilities

One of the newest and most popular numerical techniques which engineers are using is the finite element method. With high-speed digital computers, the finite element method has greatly enlarged the range of engineering problems amenable to numerical analysis. The method of finite elements started some 20 years ago in the study of aircraft structures. It began as an extension of matrix methods for structural analysis, but now it is recognized as a powerful tool in many other problems.

Exploiting the finite element method, the University of California at Berkeley developed the SAP-IV program. The current version of SAP-IV has proven to be a very flexible and efficient program. The linear structural systems to be analyzed in SAP-IV may be composed of combinations of nine different structural elements and perform both static and dynamic analyses.

One important application of finite element method is the evaluation of the General Purpose Heat Source (GPHS), which must sustain reentry and impact intact. Should a launch or onboard explosion take place, these fuel modules would probably separate and follow independent trajectories back into the atmosphere. In addition to the studies of orbital decay and reentry dynamics already mentioned, simulations of stress analysis and thermal heating were developed to simulate the situation during the reentry phase. Bulk graphites and carbon-carbon composites are considered to be the best material candidates for the GPHS due to the reentry requirements. The strength of graphite increases with temperature, while most materials lose strength at higher temperatures. Due to the anisotropic properties, pyrolytic graphite cannot sustain high strength along the direction of layers of carbon atoms arranged in two-dimensional rings. A block made of three-dimensional graphite fibers shows more desirable properties at high temperatures.

Thermal and stress analyses are being carried out by the SAP-IV program. Preliminary development of the GPHS design program uses only plain stress and plain strain elements, three-dimensional elements, and thin shell elements. The SAP-IV program for static analysis has been modified to generate thermal and stress analyses. The displacements from the program can be used in analogy to the temperature input data. The heat flux from

the aerodynamic heating and fuel pellets obtained from the calculation of convective heat transfer, is the only input data required by the program. An example problem has been tested to calculate the temperature distribution and stress distribution in a graphite block. The heat flux distribution on a flat-faced side was used as data input for the program. For simplicity, the heat flux from isotope fuel and the heat transfer from the leeward surface of a reentry member were all neglected. Final results of stress distribution are shown in Appendix C.

5.0 PLANS

5.1 Third Quarter 1979 (Fourth Quarter GFY 1979)

Plans for the remaining three months on the current contract involve several activities. Preparations must be made for the presentation of the paper entitled, "A Nuclear/Electric Orbital Transfer Vehicle for the Shuttle Era," at the AIAA/DGLR 14th International Electric Propulsion Conference, October, 1979. Latest developments and results will be included in this oral delivery. Work on GPHS stability analyses will lead to a technical paper for publication in an appropriate journal. This technology will be applicable to other reentry situations. Close coordination with LASL is expected relative to developments concerning an early reactor mission.

Development of software will continue. Although our capabilities currently match or exceed all others in the area of reentry simulation and vehicle dynamics, we are committed to continued advancement of these technologies. Liaison activities with industry and government permit direct inputs into ongoing and new program studies.

5.2 GFY 1979

Plans for the coming Government Fiscal Year (GFY 1980) will be fully detailed in a renewal proposal. However, it seems appropriate to at least summarize these plans in this report. Activities can be categorized as ongoing development of technology and continued real-time support so space applications activities within DOE. In the area of technology development, studies related to reentry dynamics, point design efforts and Shuttle impact will be major items. Assistance will be offered to NASA, DOD, and commercial users regarding integration of nuclear supplies in future

missions. Some expansion of simulation capabilities and analyses related to nuclear satellites is anticipated. The varied capabilities of the Penn State team can be brought into action to support real-time needs of DOE while continuing to develop new technology and identifying areas of advantage for nuclear space power.

Appendix A: Results of GPHS Reentry Stability Simulations

Problem Description

This study is to investigate the reentry stability characteristics of the GPHS module so that predictions of aerodynamic heating can be ascertained. Since aerodynamic heating is most important when the reentering body is in the continuum flow regime (less than 50 miles altitude), this study will be restricted to continuum hypersonic flow where the laws of classical fluid mechanics are valid.

The prediction of aerodynamic loading is necessary for understanding the attitude stability characteristics of a blunt reentry body. The aerodynamic loading is obtained from the computation of the flowfield pressure distribution about the body. The calculation of blunt body reentry flowfields has been the focus of numerous investigations during the past 15 or 20 years¹⁻⁸. The reentry flowfield includes regions of supersonic and subsonic flow, where viscous effects are considered near the body (boundary layer) and neglected away from the body (inviscid region). Because of the complexity of the flowfield, accurate methods of solution use detailed numerical techniques.

For a blunt, flat-faced, sharp-cornered body like the GPHS module, the flow can be classified as three-dimensional and asymmetric, and produces a flowfield similar to that in Fig. A-1. Although a numerical technique is needed to accurately predict the three-dimensional pressure distribution, a first order approximation can be made by analyzing the entire flowfield in three separate reference planes and thus obtain three distinct pressure

Figure A-1. Blunt Body Reentry Flowfield

distributions. Although this avoids the difficulty in determining the three-dimensional pressure distribution, it may yield unreliable results.

Procedure

From references 9 and 10, we found experimental and computational single-plane pressure distributions on flat-faced bodies in hypersonic flow at 0° , 10° and 20° angle of attack. Distributions for higher angles of attack were extrapolated from this reference data. With pressure distribution available, the aerodynamic moments can be obtained. Since we have uncoupled the initially assumed random three dimensional motion of the module into a separate rotation about each body axis, the moments for angular motion about each axis are computed. Plots are shown for the y and z body axes (Figures A-2 and A-3). A plot for the angular motion about the x-axis is not shown since it would be very similar to the z-axis plot.

A shift of the center of gravity of the GPHS module was introduced in an attempt to favorably alter the stability characteristics of the module. Plots of moment versus angle of attack with center of gravity shifts are shown (Figures A-4 to A-8).

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Figure A-2. GPHS Module Moment Plot for Z-axis

Figure A-3. GPHS Module Moment Plot for Y-axis

C. G. Shift: $\Delta x = 0.0$ mm
 $\Delta y = 5.0$ mm

Figure A-4. GPHS Module C. G. Shift Moment Plot for Z-axis

C. G. Shift: $\Delta x = 8.0$ mm

$\Delta y = 8.0$ mm

Figure A-5. GPHS Module C. G. Shift Moment
Plot for Z-axis

C. G. Shift: $\Delta x = 6.0$ mm

$\Delta y = 6.0$ mm

Figure A-6. GPHS Module C. G. Shift Moment Plot for Z-axis

C. G. Shift: $\Delta x = 6.0 \text{ mm}$
 $\Delta y = 6.0 \text{ mm}$

Figure A-7. GPHS Module C. G. Shift Moment Plot for Y-axis

C. G. Shift: $\Delta x = 8.0 \text{ mm}$
 $\Delta y = 2.0 \text{ mm}$

Figure A-8. GPHS Module C. G. Shift Moment Plot for Y-axis

Appendix B: Reactor Powered Orbit Transfer Mission

Power Plant

In all the mission analyses to date the power and weights for the power plant are those associated with the Los Alamos TE type nuclear reactor subsystem¹. An output power of 130 KWe has been assumed with an associated power plant specific mass of 16.4 Kg/KWe. The total mass of the power plant then used in this study is given by:

$$M_p = P_o \times \text{Specific Mass} = 130 \times 16.4 = 2132 \text{ Kg} \quad (\text{B-1})$$

Thruster Subsystem

The most important factor influencing the performance of the nuclear/electric transfer vehicle is the design of the thruster subsystem. The goal of this design is to determine which thruster/combination of thrusters will place the maximum payload into GEO in a reasonable time (less than six months). A study was conducted of four different thrusters and a comparison was made of the maximum payload to GEO in various mission times. Each thruster has its own thrust, specific impulse, mass flowrate, hardware weight and power requirement which influences its performance. Following is an explanation of these various thruster characteristics and how they influence the mission time and maximum payload to GEO.

Power Required to Run a Single Thruster

Given the thruster specific impulse, I_{sp} , the thrust, T and the efficiency of the thruster, η_t , we can calculate the power required to run a single thruster from:

$$P_{req}(\text{KWe}) = \frac{T \cdot I_{sp} \cdot 0.0098}{2 \cdot \eta_t} \quad (\text{B-2})$$

This power requirement is increased when we include the efficiency of the power processing unit, η_p . The efficiency of current power processing units is about 90 percent³. The power required is now

$$P_{req}(KWe) = \frac{T \cdot I_{sp} \cdot .0098}{2 \cdot \eta_t \cdot \eta_p} \quad (B-3)$$

Figure B-1 shows the fraction of power output by the reactor that is ultimately transformed by the thruster into exhaust kinetic energy. Since current power processors have efficiencies of around 90 percent, the overall system efficiency is .90 times the individual thruster efficiency which from Table 4-2 ranges from about .60 to .80 depending upon the value of specific impulse.

Example 1: Power Required

NASA-Lewis 30 cm-diameter HG Bombardment Thruster⁴

Thrust = .50N I_{sp} = 9000 sec η_t = .80 η_p = .90

$$P_{req} (KWe/unit) = \frac{.50 \cdot 9000 \cdot .0098}{2 \cdot .80 \cdot .90} = 30.625 \text{ KWe/thruster}$$

From the power required we can determine the maximum number of thrusters a particular power plant can support.

$$N_{max} = \frac{\text{Power Output}}{P_{req}} \quad (B-4)$$

Using the above thruster then

$$N_{max} = \frac{130 \text{ KWe}}{30.625 \text{ KWe}} = 4.24 = 4 \text{ thrusters}$$

POWER CHARACTERISTICS SUMMARY

Figure B-1. Power Characteristics Summary

Thrust Subsystem Mass

The total mass of the thrust subsystem is a function of the individual thruster mass, M_t , the mass of the power processing unit, M_{pp} and the mass of the fuel tanks, M_{ft} . Other items associated with the thruster mass are neglected since nearly 90 percent of the total subsystem mass is due to the power processor. The mass of the power processing unit is a function of the power required by the individual thruster. The specific mass of the current power processors is about 10 Kg/KWe. Using this value of α_p we can calculate the mass of the PPU from

$$M_{pp} = \alpha_p P_{req} \quad (B-5)$$

The total subsystem mass is then

$$M_{sub} = M_t + \alpha_p P_{req} + M_{ft} \quad (\text{Kg/Thruster}) \quad (B-6)$$

Using the previously calculated values for P_{req} and assuming a thruster mass of 7 Kg with a tank mass (M_{ft}) = 12 Kg the total weight/thruster of the thruster subsystem is

$$M_{sub} = 7.0 + 10 \times 30.625 + 12.0 = 325.25 \text{ Kg/Thruster}$$

If the thruster subsystem contains N thrusters the mass of the sub-system is simply $N \times M_{sub}$.

Payload Mass

The payload is defined as that which is not part of the thruster subsystem, the power supply, and fuel.

$$\text{Payload Mass} = \text{Initial Vehicle Mass} - \text{Fuel} - M_{sub} - M_p \quad (B-7)$$

Note that M_{sub} and Fuel depend upon how many thrusters make up the thruster subsystem.

The mass flowrate, $\dot{m}$, used in calculating the fuel requirement is found from

$$\dot{m} = \text{Thrust}/I_{sp}/9.8 \quad (\text{Kg/sec}) \quad (\text{B-8})$$

For our example thruster

$$\dot{m} = .50/9000/9.8 = 5.67 \times 10^{-6} \text{ Kg/sec}$$

The fuel required to perform a given mission is then

$$\text{Fuel} = N \cdot \dot{m} \cdot (\text{Mission Time}) \quad \text{Kg} \quad (\text{B-9})$$

Repeating the equation for payload then

$$\text{Payload} = M_o - N \cdot (\dot{m} \cdot \text{Time} + M_{\text{sub}}) - M_p \quad \text{Kg} \quad (\text{B-10})$$

Initial Altitude Requirement

The "safe" altitude has yet to be fully defined but for this initial study it is assumed to be that altitude which would give a decay time of 500 years. The required altitude is a function of the vehicle's ballistic coefficient given by:

$$\beta = \frac{M}{C_d \cdot A_{\text{ref}}} \quad (\text{B-11})$$

where M is the vehicle mass

C_d is the drag coefficient (around 3.0)

A_{ref} is some reference area (frontal area of cylinder is 16.4 m^2)

Assuming a constant drag coefficient acting on a given reference area, we can see from Figure B-2 that the ballistic coefficient is a linear function of the vehicle mass. From drag studies then we can relate the ballistic coefficient to the initial altitude needed to give the desired decay time of 500 years.

Figure B-2. Ballistic Coefficient as a Function of Vehicle Mass and Initial Altitude for 500 yr. Life.

Once we have an estimated vehicle mass and its corresponding initial altitude we need to check that this combination is compatible with Shuttle capabilities. From Figure B-3, the Shuttle payload capability as a function of altitude is represented by the dashed line curve. This gives the maximum vehicle mass that can be raised to a given altitude using the Shuttle. The second curve is that formed by the outline of the shaded area and represents the minimum altitude a vehicle of a given mass may "safely" begin transfer to higher orbits. As long as the altitude requirement for safety is within the capability of the Shuttle the mass/initial altitude combination represents a possible mission.

For example: A vehicle of 20,000 Kg must be taken to about 815 Km altitude before it can safely begin transfer to GEO. But, the Shuttle can raise a maximum payload of only 18,000 Kg to that altitude. Therefore a vehicle of 20,000 Kg cannot meet both safety and compatibility requirements.

We can see from Figure B-3 that the combination of safety and Shuttle capability considerations effectively limit the maximum vehicle mass to around 17,500 Kg.

Mission Design Charts

The first design chart (Figure B-4) is for the 10-thrust, 10-Isp thruster⁵ and shows payload ratio as a function of mission time and the initial vehicle mass. All possible combinations of time, initial mass and payload ratio are given in the shaded area of the chart. The limits of this area are defined by three factors. The first is the desired mission time. In this study we have chosen to consider only those missions which will reach GEO in under 180 days (six months). Secondly, there is a limit to the

Figure B-3. Compatibility of 500 Year Orbit with Payload Capacity of the Space Shuttle.

Figure B-4. Payload Ratio vs. Mission Time for 500 Year Initial Orbit to GEO

Maximum number of thrusters for size reasons is 36.
The maximum payload that can be placed into GEO using this thruster is 6700. Kg. This corresponds to an initial vehicle mass of 13200. Kg.

number of a specific thruster we can use for either size or power reasons. For the particular thruster of Figure B-4 that limit is 36 for size reasons. If you will recall the data of Table 4-3, 36 of the NASA-Lewis thrusters on a vehicle with an initial mass of 10,000 Kg can place 4,100 Kg of payload into GEO in just over 135 days. You will find this data point at the intersection of the 10 K initial mass curve and the dashed "maximum number of thrusters" line. The third factor is the minimum vehicle mass we will consider for practical purposes. Any vehicle of about 5,000 Kg or less would probably be raised by methods other than electric propulsion. From the charts we can determine the maximum payload possible using the particular thruster subsystem. On the chart this maximum value is simply the intersection of the "Time Limit" (dashed line) with the maximum number of thrusters curve. So in Figure B-4 we get a payload ratio of around .50 and an initial mass of about 13,000 Kg. The maximum possible payload at GEO is then 6,700 Kg using 36 of the NASA-Lewis thrusters for a transfer time of six months.

Figures B-5 and B-6 show results obtained for the Hi-thrust, Hi-Isp thrusters. It is obvious from the figures that these thrusters are not well suited to our mission. The JPL-Hughes model³ will place a 1600 Kg payload into GEO in 180 days. The corresponding initial vehicle mass is 5300 Kg--nearly impractical. The NASA-Lewis model⁴ (Figure B-6) has a thrust slightly higher than the JPL-Hughes thruster and gives a maximum payload of 1963 Kg with an initial mass again bordering on impracticality (5746 Kg). The final thruster studied, the Aerospace Corporation 50 cm diameter model⁶, produced the results of Figure B-7. The combination of Hi-thrust, and Lo-Isp has the capability to place nearly 11,000 Kg of payload into GEO, but

Figure B-5. Payload Ratio vs. Mission Time for 500 Year Initial Orbit to GEO

The TE reactor can support a maximum of 5 of these thrusters.
The maximum payload that can be placed into GEO is 1600. Kg.
The corresponding initial vehicle mass is 5300. Kg.

Figure B-6. Payload Ratio vs. Mission Time for a 500 Year Initial Orbit to GEO

The maximum number of these thrusters that the TE reactor can support is 4.

The maximum payload that can be placed into GEO using this thruster is 1960. Kg. The corresponding initial vehicle mass is 5700. Kg.

Figure B-7. Payload Ratio vs. Mission Time for a 500 Year Initial Orbit to GEO

The maximum number of these thrusters that the TE reactor can support is 17.

The maximum payload that can be placed into GEO is limited by initial cargo mass constraints of the Shuttle. The maximum initial vehicle mass is the maximum allowed by the Shuttle which is 17500. Kg. This corresponds to a possible payload at GEO of 9100. Kg.

because of Shuttle payload constraints (max. vehicle mass limited to 17,500 Kg), the maximum payload is limited to 9187 Kg.

Figure B-8 presents the maximum payload versus mission time obtained for each of the four thrusters studied. The figure reflects the effect of a significant change in the power processor specific mass. The power processor being the largest contributor to the total thruster subsystem mass. From Figure B-8, a 50 percent reduction in power processor specific mass is shown to produce very little change in maximum payload as evidenced by the difference between the solid and dashed lines.

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Figure B-8. Maximum Payload vs. Mission Time Using Various Thruster Subsystems

The a and b subscripts correspond to results obtained using power processor specific masses of 10 and 5 Kg/KWe respectively.

1.	NASA-Lewis	30 cm-dia. HG	Thrust = .129 N
2.	JPL/Hughes	30 cm-dia. HG	Thrust = .370 N
3.	NASA-Lewis	30 cm-dia. HG	Thrust = .500 N
4.	Aersp.Corp	50 cm-dia. HG	Thrust = .428 N

Appendix C: Structural and Thermal Analysis Capabilities

Introduction

The development of an effective computer program entirely depends on the powerful numerical technique. Using the Finite Element Method (FEM), SAP-IV was developed by University of California at Berkeley and designed for both static and dynamic analysis of linear structural systems¹. The current program version has proven to be a very flexible and efficient analysis tool. This program is coded in FORTRAN IV and operates without modifications on CDC computers and with little modification on the IBM and UNIVAC computers.

Structural systems analyzed by SAP IV may be composed of combinations of a number of different structural elements. The program presently contains the following element types:

- (a) three-dimensional truss element,
- (b) three-dimensional beam element,
- (c) two-dimensional element,
- (d) three-dimensional solid element,
- (e) variable-number-nodes thick shell element,
- (f) thin shell element
- (g) pipe element.

These structural elements can be used in both in static or dynamic analysis.

The capacity of the program depends mainly on the total number of nodal points in the system, the number of eigenvalues needed in the dynamic analysis and the computer used. There is practically no restriction on the number of elements used, the number of load cases or the order and bandwidth of the

stiffness matrix. Each nodal point in the system can have from zero to six displacement degrees of freedom. The element stiffness and mass matrices are assembled in condensed form; therefore, the program is equally efficient in the analysis of one-, two- or three-dimensional systems.

The formation of the structure matrices is carried out in the same way in a static or dynamic analysis. The static analysis is continued by solving the equations of equilibrium followed by the computation of element stresses. In a dynamic analysis the choice is between

- (1) frequency calculations only,
- (2) frequency calculations followed by response history analysis,
- (3) frequency calculations followed by response spectrum analysis,
- (4) response history analysis by direct integration.

To obtain the frequencies and vibration mode shapes solution, routines are used which calculate the required eigenvalues and eigenvectors directly without a transformation of the structure stiffness matrix and mass matrix to a reduced form. In the direct integration an unconditionally stable integration scheme is used, which also operates on the original structure stiffness matrix and mass matrix. This way the program operation and necessary input data for a dynamic analysis is a simple addition to what is needed for the static analysis.

Description of a SAP-IV Calculation

A static analysis involves the solution of the equilibrium equations

$$K_u = R$$

The load vectors R have been assembled at the same time as the structure stiffness matrix and mass matrix were formed. The solution of the equations

is obtained by Gauss elimination on the positive-definite symmetrical system of equations. In the solution, the load vectors are reduced at the same time as K is decomposed. After the nodal point displacements have been evaluated, sequentially the element stress-displacement matrices are read from low speed storage and the element stresses are calculated.

The dynamic analysis of a structural system using mode superposition requires as the first step the solution of the generalized eigenvalue problem

$$K\phi = \omega^2 M\phi$$

where ω and ϕ are free vibration frequencies and mode shapes, respectively. Two different solution procedures are used in the program, a determinant search technique or a subspace iteration solution. The determinant search solution is carried out when the stiffness matrix can be contained in high-speed storage in one block. Therefore, for systems of large order and bandwidth the subspace iteration method is used.

In dynamic response analysis the solution of the equations

$$M\ddot{u} + c\dot{u} + Ku = R$$

is required, where R can be a vector of arbitrary time varying loads. The program can carry out a history analysis or a response spectrum analysis for the solution of the governing equation. The history analysis can be carried out using mode superposition or direct integration. The response spectrum analysis necessitates, of course, first the solution of the required eigen-system.

The most expensive phase in mode superposition analysis is usually the calculation of frequencies and mode shapes. However, once the required eigen-system has been solved for, it can be used to analyze the structure for different loading conditions. Also, in a design process the history or spectrum analysis for the same loading can be carried out economically a few times, for example, to study the stress history in different parts of the structure.

Application to the General Purpose Heat Source

One of the most interesting applications of FEM is the design of General Purpose Heat Source (GPHS), which contains isotope fuel and a graphite block which must sustain reentry and impact without breaking up. Should a launch or onboard explosion take place, these fuel modules would probably separate and follow their own decay and reentry trajectories. It is highly likely that an ejected GPHS will enter into the atmosphere in random tumbling mode. However, symmetrical bodies may have equilibrium orientations about which the moments are all zero. With a view to the consideration of survivability of the module, the tumbling mode is preferred to minimize thermal ablation and aerodynamic heating during reentry.

As part of the reentry dynamics studies, simulations of stress analysis and thermal heating are being developed. Materials such as bulk graphites and three-dimensional carbon-carbon composites are considered to be the best candidates for the reentry requirements. The tensile strength variation with temperature of graphite and the other metals is shown in Figure C-1². Tensile strength of graphite increases with temperature, while most materials lose strength at higher temperatures. The other physical properties of typical graphite, shown in Figure C-2³, also indicate that the

Figure C-1. Strength of Graphite Increases With Temperature

Figure C-2. Typical Strength of Graphite

strength of graphite increases up to 5000°F. For this reason graphite is often used in high temperature structural applications. Owing to the anisotropic properties, pyrolytic graphite cannot sustain high stress along the direction of layers of graphite atoms arranged in a two-dimensional array. A block made of three-dimensional graphite fibers shows more desirable properties at high temperatures⁴. Simulations of heating and stress analysis are very difficult with such materials. However, the finite element procedure is a straightforward tool to handle this problem.

Temperature and stress distribution were determined using the SAP-IV program. The thermal analysis was made possible through the use of an analogy developed between the two-dimensional linear elastic strain equations of solid mechanics and the two-dimensional heat conduction equations⁵. Thus, temperature in heat conduction equations corresponds to the stress distribution in linear elastic strain equations while suppressing the velocity along the transverse direction. Therefore, the SAP-IV program for static analysis was modified to process the thermal analysis and stress analysis together. The displacements obtained are treated as the temperature input data in the second run of the program. The dynamical analysis part has been discarded since the governing equation of transient heat conduction is different from that of structure dynamics.

Elements used in the modified program consist of plane strain and plane stress, trusses, and boundary elements. Thus, the program can be exploited to calculate any two-dimensional heat conduction problem. For simplicity, the finite element mesh used in thermal analysis will be employed in the stress analysis. The final output format for the modified program has been changed to print out the stress, temperature, and displacement distributions. A few statements were added to store the displacement data as the temperature

input in the thermal stress analysis. Heat flux from aerodynamic heating and the fuel pellet, obtained from the calculation of convective heat transfer, is the input data to the program.

Results of Calculations

For conceptual design number 1 of Figure C-3⁶, the temperature distribution along the leading face of the reentry member is shown in Figure C-4. Temperature values along the inside of the fuel cavity and along the left side of the mesh remain constant. In general, values increase linearly from left to right and from bottom to top. The maximum temperature, around 5000°F, happens to be in the upper right-hand corner, corresponding to the front broadface edge of GPHS. The maximum value of the magnitude of temperature gradient happens to be in the same place as the maximum temperature. The portion of the thermal mesh below that in the figure has values and gradients small enough to be ignored relative to the values shown.

The three-dimensional plot of the r-direction stresses is shown in Figure C-5. From this plot, the maximum compression stress occurs at the left corner of the leading edge and the maximum tensile stress occurs at the graphite along the right-hand edge. The plane right in the middle represents the standard surface. Ratios of maximum tensile stress to material tensile strength and maximum compressive stress to material compressive strength are 1.29 and 0.31, respectively. It should be noted that aside from the area of highest tensile stresses most values were well under the stress to failure values of the material. Since this analysis dealt with the worst possible reentry conditions, this design could be an acceptable one.

Figure C-3. Conceptual Design No. 1

Figure C-4. Temperature Distribution of
Conceptual Design No. 1

Figure C-5. r-Direction Stresses of
Conceptual Design No. 1

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