MASTER

NUCLEAR POWERED SATELLITE STUDIES

Annual Progress Report
for the Period July 1, 1977 - June 30, 1978

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June 1978

Prepared for
THE U. S. DEPARTMENT OF ENERGY
UNDER CONTRACT NO. EY-76-S-02-4045. A001

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ABSTRACT

Progress achieved during the period July 1, 1977 to June 30, 1978, on U.S. D.O.E. Contract No. EY-76-S-02-4045.A001 is reported. Discussions of several pertinent aspects are included, e.g., schedule, personnel, technology developments, and plans. The reporting period represents the second 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 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.

CONTENTS

	<u>Page</u>
Abstract	iii
1.0 Introduction	1
1.1 Summary of Progress	1
1.2 Major Results	2
1.3 Summary of Plans	3
2.0 Schedule and Personnel	4
2.1 Milestone Chart	4
2.2 Staffing	4
2.3 Plans	7
3.0 Liaison Activities	8
3.1 Presentations	8
3.2 Publications	8
3.3 Point Design Activities with Government and Industry	10
4.0 Technology Developments	11
4.1 Computer Facility Development	11
4.2 Nuclear Powered Communications Satellite Design	14
4.3 Attitude Dynamics Simulation Software	16
4.4 Spacecraft Structural Analysis	19
4.5 Orbital Decay Studies	22
5.0 Plans	25
5.1 Third Quarter 1978	25
5.2 GFY 1979	25
Appendices	
A. An Automatic Attitude Control System for a Dynamic Isotope Powered Satellite	27
B. Sample Orbital Decay Simulation	67
C. Structural Analysis Software	75

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 continued 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 two years of this project several publications have been generated. A number of presentations have been made at technical conferences. Spacecraft designers 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 communications satellites, structural analysis of spacecraft elements (e.g., radiator, heat source assembly, etc.), orbital decay of spacecraft, 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 Group. Two input/output terminals and an interactive graphics plotter permit access to computers around campus and almost anywhere by a telephone line. An extensive array of project-related software has been developed, including a general purpose attitude dynamics simulation program, structural analysis programs, and orbital decay simulations. A preliminary design of a nuclear powered communications satellite was developed to satisfy projected requirements of INTELSAT in the 1980's. This included a comprehensive treatment of the attitude control problems with a dynamic isotope power system onboard. A unique capability exists here to simulate orbital decay and attitude motion simultaneously. We have done this for Skylab and Cosmos 954. 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, 1978. 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, 1978) will provide an opportunity to complete activities underway and begin a new effort related to the changing emphasis of the Department of Energy programs. Present capabilities will permit a quick assessment of orbital decay properties of the General Purpose Heat Source (GPHS) in its various configurations. A number of simulations will be carried out to study attitude and orbital dynamics of these designs. Structural models will be formulated to be later stress analyzed in order to assist in the design of nuclear power sources.

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 79.

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. In February 1978 he received the 1978 Award for Outstanding Achievement in Research in the College of Engineering of the Pennsylvania State University. This award is given annually to one faculty member of the college. Dr. Kaplan was the first Associate Professor ever to receive this honor. Incidentally, as of July 1, 1978, Dr. Kaplan will become Professor of Aerospace Engineering. Dr. James C. Wambold is Associate Professor of Mechanical Engineering and has been on sabbatical leave in Washington D.C., since December 1977. Nevertheless, he has continued

Legend:

- preliminary thinking
- execution
- publication
- presentation

Figure 2-1. Milestone Chart for GFY 78

Figure 2-2. Staffing and Organization as of June 1, 1978

to assist in guiding the development of the structural analysis software. As of September 1, 1978, he will return to the University on a full time basis and to this project at a 10 percent rate. Stephen G. Alexander has just become a senior research technologist, replacing David J. Cwynar, who has completed his master of science work and has taken a position in industry. Mr. Alexander will work full-time, at least through August, 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 analysis and structural models of nuclear satellite components. He will work with Mr. Alexander in developing computer models of the GPHS over the next three months. ECON, Inc., acting as a consultant, has been working with Lockheed to make an economic evaluation of a nuclear powered COASTSAT. This work should be completed soon. Occasionally, undergraduates are hired on a part-time basis to assist with a variety of support tasks.

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, 1978. ECON will be phased out with the completion of the COASTSAT study.

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 79.

3.0 LIAISON ACTIVITIES

3.1 Presentations

A number of relevant presentations have been made over the past two years by the principal investigator. 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 two papers dealing with the use of nuclear power supplies. One of these was presented at the Twelfth Intersociety Energy Conversion Engineering Conference, Washington, D.C., August 28 - September 2, 1977, entitled, "Space Nuclear Power Supply Design in the Shuttle Era." Another one 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." A paper will be presented at the Tenth AIAA/IES/ASTM Space Simulation Conference on orbital decay of satellites, in October 1978.

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.
6. M. H. Kaplan, D. J. Cwynar, and S. G. Alexander, "Simulation of Skylab Orbit Decay and Attitude Dynamics," AIAA Paper 78-1626, to be presented at the 10th AIAA/IES/ASTM Space Simulation Conference, October 1978.

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 Nuclear Satellite Components," 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.

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 past year. 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 is completing economic comparisons of these designs and should have a final report completed soon.

The principal investigator has made a number of trips to various DOE contractors and facilities. For example, he has visited the Los Alamos Scientific Lab in April 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.

4.0 TECHNOLOGY DEVELOPMENTS

4.1 Computer Facility Development

An Astronautics Research Lab/Computer Facility has been developed in the Aerospace Engineering Department. This facility is being utilized for a variety of DOE related studies. To date, two computer terminals and a digital plotter have been acquired. The computer terminals consist of a Dec Writer LA36 hard-copy terminal and a Tektronix 4010-1 video display terminal. The hard copy terminal is extremely useful in the preliminary stages of program development. The video terminal allows a rapid display of data or plots and has interactive graphics capability. Compatible with both of these units is the Tektronix 4662 digital plotter. This device produces high quality plots from digital data as well as the reproduction of images which appear on the Tektronix video screen. This equipment is interfaced with the host computer via an acoustic coupler. Most recently, a VADIC modem was purchased by the Aerospace Engineering Department which increases the video terminal's baud rate by a factor of four. This is extremely useful when a series of images are to be produced on the Tektronix screen. Although primary usage is with the University's IBM 370/168 and PDP 10/40 computers, access is available to any computer system in the country via the acoustic coupler. Capabilities of the simulation lab are briefly discussed below with more details later. The chronological development of the facility and its capabilities is illustrated in Figure 4-1.

Many general usage computer algorithms have been developed during the short period in which the simulation lab has been in existence. One of these is an attitude dynamics simulation program. A variety of attitude control systems are modelled enabling simulation of virtually any type of spacecraft. This rigid body simulation program is currently being modified to model energy dissipation.

Figure 4-1. Computer Facilities Development

Another program which has been developed models orbital decay of almost any satellite or component. This uses the Jacchia 1972 model to calculate the atmospheric density for altitudes above 125 km. Actual solar activity data is continually being obtained from NASA as well as predictions for the future. An orbital decay trajectory can be determined for any spacecraft whose ballistic coefficient is known or can be estimated.

A structural analysis program, SAP IV, is currently being used for spacecraft design applications. The structural integrity of individual elements of the spacecraft are analyzed upon application of a load and vibration environment. Some of the elements which can be analyzed with SAP IV are trusses, beams, plates, and shells, which are the configurations from which spacecraft structures are formed.

A graphics program is also under development to provide a visual display of a spacecraft's attitude motion. Frame-by-frame illustrations of the attitude of the spacecraft are drawn automatically on the Tektronix screen. Attitude angles received from the attitude dynamics program are inputs to the graphics program. A high-speed drawing capability provided by the interface equipment will soon give movie-like illustrations of the spacecraft attitude performance.

A variety of other programs are also available which aid in spacecraft analysis and design. The capability of the simulation lab is continually increasing as additional programs are being developed. Acquisition of additional equipment such as a graphics tablet and hard copy unit is being considered. These units will insure the productive growth of research capabilities in the Astronautics Research Group.

4.2 Nuclear Powered Communications Satellite Design

At 7:45 a.m. Eastern Daylight Time on June 30, 1979, the first orbital flight test of the space transportation system will lift off from Pad 39A at Kennedy Space Center, Florida. This event will ring in the new era of space transportation. New customer requirements, advances in technology, and an increased launch capability will force satellite designers to reexamine alternatives in areas such as power systems. A standardized nuclear powered communications satellite has been conceptually formulated for missions in the 1990's. The projected growth of traffic in the 1980's indicates larger and more powerful communications satellites will be necessary. The system of five satellites, is capable of 80,000 one-way voice channels has been chosen to satisfy our projected demands. The proposed configuration is illustrated in Figure 4-2. It incorporates a large earth pointing face for mounting a variety of high gain antennas. This design has a 4.1 meter diameter with a length of 3.4 meters with antennas stowed. The combination of cylindrical shape and a mass of 6,423 kilograms takes optimum advantage of the shuttle payload bay and the pricing policy.

After insertion into a low parking orbit, geostationary orbit is achieved via an integral Agena liquid rocket engine. The re-start capability of the Agena will allow it to provide both the perigee and apogee velocity increments. During ascent, the spacecraft will be spin-stabilized. On-orbit attitude control is achieved with a double gimbaled momentum wheel providing pointing accuracies of 0.4 degrees in yaw and 0.09 degrees in pitch and roll. The nominal momentum magnitude is 271 newton-meter-seconds with an operational spacecraft mass of 1,497 kilograms. Nearly 2 kilowatts is provided by a Dynamic Isotope Power System (DIPS) on a continuous basis at the beginning of mission. Satellite design life is ten years.

Figure 4-2. Conceptual Design of Isotope Powered Satellite

Details of the design were recently presented at the AIAA Seventh Communications Satellite Systems Conference in San Diego. A paper entitled "A Nuclear Powered Communications Satellite for the 1980's" was co-authored by the principal investigator, D. J. Cwynar, and J. J. Bednarz. Furthermore, Appendix A contains a detailed description of the automatic attitude control system for such a satellite.

4.3 Attitude Dynamics Simulation Software

Any nuclear spacecraft design or mission study must involve a corresponding attitude dynamics analysis. A set of computer programs have been devised that will mathematically simulate a wide range of spacecraft attitude motions and maneuvers. A variety of output capabilities permits the user to make a quick examination of the important parameters.

All of the attitude dynamic simulation software options are based on numerical integration of the attitude equations of motion, Euler's moment equations, for a rigid body. The integration scheme has been checked for stability by comparisons with analog computer solutions. Resulting motion from the integration routine can be described in three different coordinate systems: body fixed axes, orbital plane fixed axes, and inertial axes. The necessity of having three coordinate systems arises from the fact that some relations are more easily derived in one coordinate system than another. For example, Euler's moment equations are easily derived and solved in a body fixed system; whereas, aerodynamic drag moments are most easily handled in an orbital plane fixed system. The appropriate coordinate and rate transformations are employed to transform expressions from one system to another. Figure 4-3 depicts the chronological development of the software.

A range of simulation capabilities has been developed to handle various dynamic situations. The basic problem, torque-free motion, can be easily simulated. This is very useful in determining the relative stability of

Figure 4-3. Attitude Simulation Software Development

different satellite orientations. Two onboard generated torques for attitude maneuvers and control have been incorporated into the simulation program. The generating devices are momentum wheels and attitude thrusters. This program can handle several momentum wheels and/or several thrusters with variations in momentum and thrust moments. Simulations, with or without torques, can be done about any designated body fixed axes.

Certain environmental forces which can cause perturbing torques in satellite attitude motion have been successfully modelled in simulation programs. These are gravitational gradient and aerodynamic drag perturbation torques. For a given orbit, gravitational gradient torques depend on satellite shape and can be modelled analytically. Aerodynamic drag torques are a function of satellite shape, orientation, and local atmospheric density. Density values and variations are calculated using a Jacchia (1970 or 1972) density model. Thus, if expressions or tabulated data are known for the aerodynamic drag moment coefficients, the resulting aerodynamic drag torques can be generated by the simulation program and used along with the gravitational gradient torques in Euler's moment equations to obtain the perturbed attitude motion.

Graphical output capabilities permit important simulation parameters to be quickly plotted either on a screen or in hard copy form. A three dimensional graphics program has been adapted for satellite simulations. This program, using the output of a dynamic simulation program, depicts discrete angular rotations about three axes of a computer drawn satellite.

4.4 Spacecraft Structural Analysis

Structural analysis of spacecraft is based on classical mechanics and strength of materials. Within the last decade, finite element analysis and computer applications have revolutionized structural design methods. The computer program, SAP IV, developed by the Earthquake Engineering Research Center, University of California at Berkeley, is one of the most powerful tools of these new methods.

The basic requirement of spacecraft design is to estimate the reasonable approximation to the minimum-weight structure. The acceleration and vibratory force levels associated with modern boosters have led to a range of structural weights of 15 to 30 percent of the total payload. The reliability of the space vehicle structure is of major importance in the design of the advanced, nuclear-powered systems. The other important requirement of the long-life satellite is the survivability. High survival probability causes increased total mass, due to the increased meteoroid armor on the manifolds and supply lines and the increased satellite wall thickness.

The design of the spacecraft must accommodate all the requirements and limitations resulting from the mission analysis. The tailored structural interfaces will have to be constructed to insure proper attachment to the space shuttle bay in order to obtain the successful operation of future nuclear-powered satellites. On the other hand, the interfaces and the structure of the satellite itself must be able to withstand the random accelerations of launch and the acoustic pressure of the booster, and also, guarantee that the payload will remain secured even in the event of a crash landing.

The employment of computer calculation makes accurate and fast structural analysis possible. The development of an effective computer program for this aspect requires a knowledge of three scientific disciplines: structural

mechanics; finite element analysis; and computer application. The program, SAP IV, is one of the most effective programs designed. The program has the capacity to analyze very large three-dimensional systems as well as small ones. Also, from the complete program, smaller special purpose programs can easily be assembled by simply using only those subroutines which are actually needed in the execution.

Structures composed of a number of different elements of various types and materials can be analyzed for both dynamic and static behavior by SAP IV. The element library of the program includes three-dimensional truss elements, beam elements, solid elements, etc. However, preliminary development of the design program will use only the truss elements, the plane elements, and thin shell elements. As reported previously, SAP IV program has been made compatible with the Pennsylvania State University's IBM-370 digital computer. SAP IV execution is facilitated by the computer equipment acquired within the Astronautics Research Group. Since January 1978 graphic plotting has been made available by a screen terminal and an interactive digital plotter. During the design work the plotter offers a more vivid display than is possible with printout analysis.

For a three-dimensional satellite structure, a graphic program, PICS, recommended by the Laboratory for Environment Design and Planning, College of Arts and Architecture, the Pennsylvania State University, has been investigated. Basically, PICS is organized with an operation control section in its main program which is responsible for calling the various subroutines to perform the instructions input of the program. These instructions are used for operations for generating straight lines, polygonal and circular planes, arcs, and perspective drawings; and for performing rotations and translations of entire sets of lines, planes, and arcs.

The current version of SAP IV for the static and dynamic analysis of linear structural systems has proven to be a very flexible and efficient analysis program and can operate on a variety of large computers. SAP IV is an efficient program per se; however, the pre- and post-processing options can be developed to a greater degree. For the three-dimensional cases, PICS combined with SAP IV, shows great potential for more complicated structural design applications. Appendix C presents an example of SAP IV capability.

4.5 Orbital Decay Studies

Orbital decay trajectories of satellites can be determined by integrating the orbital equations of motion over a given time period. However, with the introduction of time dependent atmospheric density variations around the orbit, the equations of motion must be integrated numerically. In addition, attitude motion of the satellite will dictate the overall aerodynamic drag and, hence, orbital decay rate. Again numerical integration is necessary. But these integrations must be done over a time period of a few months to several years with resulting excessive computer costs and truncation errors.

As an alternative to numerical integration, a simple energy dissipation technique can be used to obtain accurate orbital decay trajectories. For a given time period during which the time varying quantities remain relatively constant, orbital energy dissipation due to drag can be determined. The total orbital energy is then decreased by this amount after each given time period, and the resulting changes in orbital altitude and velocity are calculated. This process is iterated, with the appropriate changes in any time varying quantities, to obtain an orbital decay trajectory.

Time dependence of atmospheric density is caused by variations in solar activity. With the correct predictions in solar activity (based on the 11 year solar cycle), density for a given time and altitude can be determined from the Jacchia (1972) density model. However, determination of an accurate aerodynamic drag coefficient is a problem. If, for a given satellite, part of the actual decay trajectory is known, a drag coefficient can be arrived at by trial and error methods, and the future decay trajectory can be extrapolated. Attempted analytical determinations of satellite drag coefficients, based on free-molecular flow of rarefied gases, have not been successful.

A satellite under active attitude control will have a constant or known orientation with respect to the flow, and the drag coefficient will be constant or will change in a known way. A passive satellite may be oscillating or tumbling and may not have a constant or known orientation with respect to the flow. Therefore, an attitude dynamic study of a passive satellite under the influence of perturbing torques should include a search to determine possible equilibrium orientations and associated drag coefficients. While energy dissipation techniques permit quick and accurate calculations of orbital decay trajectories, results are limited until a comprehensive aerodynamic study provides analytically determined drag coefficients for any satellite geometry. Figure 4-4 offers a chronological overview of this work. Appendix B presents examples of available orbital decay simulation capability.

Figure 4-4. Orbital Decay Simulation Development

5.0 PLANS

5.1 Third Quarter 1978 (Fourth Quarter GFY 1978)

Plans for the remaining three months on the current contract involve several activities. Preparations must be made for the presentation of the paper entitled, "Simulation of Skylab Orbit Decay and Attitude Dynamics," at the October conference of the AIAA/IES/ASTM. This will include examples related to Cosmos 954 and the GPHS. Latest developments and results will be included in this oral delivery. Completion of preliminary work on reentry properties of the GPHS is a major objective. This will include consideration of the various potential configurations. Stable and tumbling situations will be considered. Close coordination with LASL is expected. Simulation and software developments will continue as previously outlined.

ECON, Inc. will be phased out with the completion of economic studies of COASTSAT. Deliberations on potential reactor applications in space will continue until some preliminary selections are made. Continued liaison activities with industry and government will permit direct inputs into ongoing and new program studies.

5.2 GFY 1979

Plans for the coming Government Fiscal Year (GFY 1979) 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 to space applications activities within DOE. In the area of technology development studies related to GPHS, and point design efforts will be major items. Assistance will be offered to NASA, DOD, and commercial users regarding

integration of nuclear supplies in future missions. Further 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

AN AUTOMATIC ATTITUDE CONTROL SYSTEM FOR A DYNAMIC ISOTOPE POWERED SATELLITE*

	<u>Page</u>
A.1 INTRODUCTION	30
A.2 BACKGROUND INFORMATION	33
A.3 EQUATIONS OF MOTION	40
A.4 ATTITUDE CONTROL SYSTEM DESIGN	47
A.5 ATTITUDE MOTION SIMULATIONS	57
A.6 CONCLUSIONS	64
REFERENCES	66

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* Abstracted from PSU Astronautics Research Report No. 78-1, May 1978.

NOMENCLATURE

$\bar{G}$	gravity gradient torque
G_x, G_y, G_z	gravity gradient torque components
$\bar{h}$	total angular momentum of satellite
$\bar{h}_b$	angular momentum of satellite excluding internal momentum devices (momentum of spacecraft body)
h_{bx}, h_{by}, h_{bz}	body momentum components
h_L	longitudinal axis angular momentum of satellite
h_T	transverse axis angular momentum of satellite
h_s	system momentum, h_r plus h_w
$\bar{h}_r$	angular momentum vector of rotor
h_r	angular momentum magnitude of rotor
h_{rb}	nominal bias momentum of the rotor
$\bar{h}_w$	angular momentum vector of the wheel
h_{wb}	nominal bias momentum of the wheel
h_x, h_y, h_z	angular momentum components along satellite axes
h_{yr}	rotor y-component of momentum
H_{wx}, H_{wz}	orbital rate decoupled momentum commands
I_P, I_R, I_Y	pitch, roll, and yaw moments of inertia of satellite
I_w	polar moment of inertia of momentum wheel
I_x, I_y, I_z	satellite principal moments of inertia about its x,y,z axes
k	yaw-to-roll gain ratio
K, K_p	roll and pitch autopilot gains, respectively
M_x, M_y, M_z	misalignment torques
M_P, M_R, M_Y	moments acting about satellite axes
T	disturbance torque
T_r	rotor torque
T_w	wheel torque

T_x, T_y, T_z disturbance torque components

T_{xc}, T_{yc}, T_{zc} control torques

Greek Symbols

α coning angle

γ_x, γ_z gimbal deflection angles

γ_{yx}, γ_{yz} rotor orientation angles referenced from y axis

$\dot{\gamma}_x, \dot{\gamma}_z$ gimbal rates

$\delta_x, \delta_y, \delta_z$ rotor misalignment angles

β_p pitch damping ratio

β_1, β_2 roll and yaw damping ratios, respectively

θ, ϕ, ψ pitch, roll and yaw attitude angles

$\theta_{ss}, \phi_{ss}, \psi_{ss}$ steady-state attitude angles

τ, τ_p roll and pitch time constants, respectively

$\bar{\omega}$ angular velocity of satellite w/r inertial frame

ω_o orbital rate

ω_p pitch natural frequency

ω_r angular velocity of rotor

$\omega_p, \omega_R, \omega_Y$ pitch, roll, and yaw angular velocity components

ω_1, ω_2 roll and yaw natural frequencies, respectively

Ω_n nominal wheel speed

Ω_{ss} steady-state wheel speed

A.1 Introduction

This investigation deals with the designing of an automatic attitude control system for a spacecraft which uses a dynamic isotope power supply. The problem is that this nuclear power supply contains a rotating turbine-alternator which may affect the spacecraft's performance. Figures A-1 and A-2 illustrate the rotating hardware for two dynamic systems under study. At nominal operating speed, these rotating parts create significant angular momentum. The rotating assembly is allowed to deviate from its nominal operating speed, within certain limits, as variations occur within the system. Torques are produced from the resulting rate of change of momentum. The momentum and torques must be compensated or counteracted in order to prevent any unwanted changes in the attitude of the spacecraft.

The problem is three-fold. The first part entails designing of an automatic control system which will maintain the spacecraft's on-orbit attitude within the prescribed limits. This system must compensate for the disturbance torques produced by the turbogenerator as well as environmental disturbances. The second part is to analyze the effects of the spinning hardware on the spacecraft dynamics during its ascent phase, when the automatic control system is inoperative. This is done by simulations to determine if the attitude performance of the spacecraft is affected. An attitude alignment error or severe coning motion would be critical during orbital transfer burns. Finally, several orientations of the DIPS other than the nominal case, which is aligned along the pitch axis, are investigated. Also, simulations are performed to determine the effects of misalignment of the DIPS turbine/alternator rotor with the momentum wheel. Acceptable and unacceptable orientations of the power supply are deduced from this work.

Figure A-1. BIPS Power Conversion System

Figure A-2. KIPS Power Conversion System

A.2 Background Information

There are two DIPS configurations which are currently being evaluated in form of ground demonstration units. Possible flight systems are illustrated in Figure A-3. Both are similar in that they use isotope heat generation to run a turbogenerator. The ground demonstration phase is to be completed in 1978. A flight experiment may take place in the 1980's.

The key component of interest in these systems for this investigation is the turbogenerator. The KIPS turbogenerator rotating assembly has a polar moment of inertia of $0.00383 \text{ ft-lb-s}^2$ (0.00519 N-m-s^2) and operates nominally at 33,700 rpm on orbit⁽¹⁾. An angular momentum of 14.84 ft-lb-s (20.12 N-m-s) is produced. The BIPS assembly, with a rotating moment of inertia of $0.000124 \text{ ft-lb-s}^2$ (0.000168 N-m-s^2), operates at 52,000 rpm having a nominal angular momentum of 0.6761 ft-lb-s (0.9166 N-m-s).⁽²⁾ The work done here is with reference to the KIPS since this has the greater momentum of the two systems and also produces greater torques. The attitude control system developed in this paper for a 2-kW KIPS spacecraft certainly would be sufficient for a similar BIPS spacecraft. The rotation speed and the rotating group moment of inertia for the 1.3-kW system remain the same for the 2-kW system.⁽³⁾

The KIPS will operate at a nominal speed of 33,700 rpm on orbit. A ± 0.05 percent deviation due to on/off power control devices may be expected, as well as a ± 1 percent day-to-day overall speed variation due to sink temperature and heat source variations.⁽³⁾ The maximum expected speed change is 1000 rpm/sec. From this, a maximum torque of 0.4 ft-lb (0.54 N-m) is predicted, which is just the rate of change of angular momentum of the turbogenerator rotating assembly.

Figure A-3. Candidate Dynamic Isotope Systems

During launch and ascent, the turbine and alternator will be operating at what is called "launch overspeed". This higher speed of 50,000 - 56,000 rpm is used to stiffen the jets of the condensor while going through high accelerations, and to help the bearings of the system take the load better. At 56,000 rpm, no power would be delivered to the spacecraft. At 50,000 rpm, 300 W can be derived from the system.

For ascent, the 50,000 rpm case will be assumed since some power will be required for sensors, telemetry, and attitude control equipment such as the counterrotating momentum wheel which will be discussed later. A torque of 0.4 ft-lb (0.54 N-m) will still be assumed to be the maximum internal torque that the nuclear power system will deliver due to speed changes. The angular momentum of the power system at this launch overspeed is 20.05 ft-lb-s (27.18 N-m-s).

With the dynamic power systems being only in the developmental and testing stages, investigation of the effects of the turbogenerator angular momentum on the spacecraft's dynamics has not been carried out substantially. Raab⁽⁴⁾ has done some preliminary analysis on the BIPS for AiResearch in order to determine optimum orientations of the turbogenerator for two types of spacecraft. For a three-axis stabilized spacecraft, Raab pointed out that most will employ internal momentum wheels or control moment gyros to compensate for the periodic torques. These wheels could be biased to cancel out the angular momentum of 0.6761 ft-lb-s (0.9166 N-m-s) of the BIPS rotating hardware. In fact, he states, the compensation would generally be automatic since the sensors will detect spacecraft motion and command wheels to take it out. Therefore, he concludes, no orientation preference can be discerned and the power and weight penalty are negligible.

For a spin-stabilized spacecraft, Raab identifies the turbogenerator axis being transverse to the vehicle spin axis as the worst case. However,

the small amount of momentum would produce coning motion with a half angle of 0.04 degrees for a typical spacecraft. This is well within the 0.1 to 1.0 degree limit nominally specified for spin-stabilized spacecraft. This conclusion does not apply quite so generally to the KIPS rotating group, due to its higher angular momentum.

The dynamic isotope power supplies, as mentioned earlier, are still in the developmental stages where the first flight test is planned for 1981. The spacecraft studied in this investigation is a 2-kW nuclear powered communications satellite which was designed as an earlier project. This spacecraft is typical of many communications satellites which are planned for the 1980's. The details of this design are given in reference 5 and will not be completely reproduced here. Rather a brief description of the spacecraft and the aspects pertinent to the design of the attitude control system will be discussed. Also, insertion and deployment maneuvers will be presented since the work does involve investigation into the effects of the torques produced, by the rotating assembly on the spacecraft, during launch, deployment, ascent, precession, and Earth acquisition. On-orbit attitude is maintained automatically by the attitude control system.

The nuclear powered communications satellite used in this investigation is a 13.5-foot (4.1-m) diameter disk-shaped spacecraft with a main body length of about 7 feet (2.1-m). Figure A-4 is a sketch of this spacecraft which is designed to provide global services into the 1990's. The spacecraft was designed to be launched by the Space Shuttle to a low orbit and then boosted to its geostationary orbit with its integral Agena liquid rocket engine. Expected lifetime of this on-orbit 3,300-lb (1497-kg) satellite is ten years. KIPS provides the 2-kW of electrical energy to the payload and systems. The radiator, which surrounds the outer shell of the spacecraft,

Figure A-4. Conceptual Design of Isotope Powered Satellite

can be seen in this figure. Calculated moments of inertia for several phases during the flight are given in Table A-I.

Table A-I. Satellite Principal Moments of Inertia

MOI AXIS	LAUNCH (ft-lb-s ²)	ASCENT (ft-lb-s ²)	ON-ORBIT (ft-lb-s ²)
I_x Roll	6464 [8764]*	3105 [4210]	1758 [2383]
I_y Pitch	6563 [8898]	3096 [4197]	1749 [2371]
I_z Yaw	10.083 [13,670]	4554 [6174]	2346 [3181]

*[N-m-s²]

Figure A-5 is a cutaway sketch showing several internal components and subsystems. Attitude control for this three-axis stabilized vehicle is achieved using a double-gimbaled momentum wheel with nominal momentum along the pitch axis. Roll and yaw control torques are generated by two gimbal actuators. Pitch control is achieved by changing the wheel speed. Antenna pointing accuracies of 0.4 degrees in yaw and 0.09 degrees in roll and pitch are specified as mission objectives.⁽⁵⁾ The details of the designing of the attitude control system will come later.

North-south stationkeeping is achieved using 1-mlb (4.4-mN) ion thrusters. One-pound (4.4-N) hydrazine thrusters are used for spin-down and east-west stationkeeping. Two-pound (8.9-N) hydrazine thrusters are used for momentum dumping, ascent maneuvers, and a limited amount of additional velocity control.

Figure A-5. Spacecraft Cutaway Sketch

The sequence of events for the launching of this spacecraft follows. After the parking orbit is achieved via the Space Shuttle, the payload bay doors are opened. Orbiter orientation is established to satisfy spacecraft axis alignment for spin-up. The spacecraft is ejected from a spin table which provides gyroscopic stiffness by spinning the vehicle up to 30 rpm. After a safe drift distance is achieved, the preparations for the perigee burn begin. If the attitude of the spacecraft is not disturbed, the perigee burn is performed as the equator is passed. A reorientation maneuver takes place during ascent in order to achieve the proper orientation for the ground commanded apogee burn. This will inject the spacecraft into geostationary orbit. An attitude acquisition maneuver and final station adjustment take place. Attitude control will now be automatic through the use of an earth sensor and gimbal logic.

A.3 Equations of Motion

The equations of motion for a spacecraft with a double-gimbaled momentum wheel (DGMW) aligned along the vehicle y axis will now be developed. The derivations follow that of Mork⁽⁶⁾, which assumes a rigid body with internal momentum. However, a fixed gimbal wheel is added to the spacecraft to represent the rotating turbine blades and generator rotor of the dynamic isotope power supply. From this point on, the fixed wheel will be referred to as the rotor.

Figure A-6 illustrates the on-orbit attitude of the spacecraft with a more detailed illustration of the DGMW and rotor. The x axis of the body nominally points along the spacecraft velocity vector, the y axis is normal to the orbital plane, and the z axis is toward the geocenter. These are generally referred to as the satellite roll, pitch, and yaw axes, respectively,

Figure A-6. On-Orbit Attitude of Spacecraft

while the spacecraft is maintaining its on-orbit attitude. Small rotations about these axes are expressed by the spacecraft attitude angles, ϕ , θ , ψ , which refer to the roll, pitch, and yaw angles, respectively. This X Y Z reference frame, which is used for attitude control reference, rotates with respect to the inertial frame at the orbital rate, ω_o . Hence, it is called the "orbital reference frame." Notice the gimbal angles of the suspended DGMW about the body x and z axes are identified by γ_x and γ_z . Generally, these gimbal angles will be physically restrained to a maximum of 20 degrees.

The total angular momentum of the system, referred to the body coordinate system, is the sum of the body momentum plus the wheel and rotor momentum, i.e.,

$$h = h_b + h_w + h_r \quad (A-1)$$

The body momentum components,

$$\begin{aligned} h_{bx} &= I_x \omega_x \\ h_{by} &= I_y \omega_y \\ h_{bz} &= I_z \omega_z \end{aligned} \quad (A-2)$$

constitute the first term in equation (1) as

$$h_b = h_{bx}i + h_{by}j + h_{bz}k \quad (A-3)$$

I_x , I_y , and I_z represent the moments of inertia and ω_x , ω_y , and ω_z are the angular velocities of the body about the respective axis.

The momentum components due to the wheel are

$$h_w = h_{wx}i + h_{wy}j + h_{wz}k \quad (A-4)$$

Assuming gimbal deflections γ_x and γ_z ,

$$\begin{aligned} h_{wx} &= (\cos\gamma_x \sin\gamma_z)h_w \\ h_{wy} &= -(\cos\gamma_x \cos\gamma_z)h_w \\ h_{wz} &= -(\sin\gamma_z)h_w \end{aligned} \tag{A-5}$$

with h_w being the angular momentum of the wheel with respect to inertial space.

Since the rotor is fixed along the y axis, then only one non-zero component of angular momentum exists, i.e.,

$$h_{yr} = -h_r \tag{A-6}$$

The negative sign denotes the nominal angular velocity being about the negative y axis.

The momentum components due to the body, DGMW, and rotor can now be summed about the x, y, and z axes to yield the system momentum components:

$$\begin{aligned} h_x &= I_x \omega_x + (\cos\gamma_x \sin\gamma_z)h_w \\ h_y &= I_y \omega_y - (\cos\gamma_x \cos\gamma_z)h_w - h_r \\ h_z &= I_z \omega_z - (\sin\gamma_z)h_w \end{aligned} \tag{A-7}$$

An expression for the total angular momentum of the system is that given as equation (1), which is now rewritten as

$$h = h_x i + h_y j + h_z k \tag{A-8}$$

signifying momentum components about the body axes. These components will be used to develop the equations of motion which directly follows.

Euler's equations for a rigid body are given by⁽⁷⁾

$$\bar{T} + \bar{G} = \frac{d\bar{h}}{dt} = \left[\frac{d\bar{h}}{dt} \right]_b + \bar{\omega} \times \bar{h} \quad (A-9)$$

where $\bar{G}$ represents the gravity gradient torques, and $\bar{T}$ are the external disturbance torques such as those due to solar pressure, thrust misalignment, and magnetic interaction. The angular velocity term, $\bar{\omega}$, is the spacecraft angular velocity with respect to the inertial coordinate system. Three equations of motion of the spacecraft about its x, y, and z axes are now obtained from this expression:

$$\begin{aligned} I_x \dot{\omega}_x + (\cos\gamma_x \sin\gamma_z) \dot{h}_w - (\sin\gamma_x \sin\gamma_z) \dot{\gamma}_x h_w \\ + (\cos\gamma_x \cos\gamma_z) \dot{\gamma}_z h_w + \omega_y (I_z \omega_z - h_w \sin\gamma_z) \\ - \omega_z (I_y \omega_y - h_w \cos\gamma_x \cos\gamma_z - h_r) = T_x + G_x \\ \\ I_y \dot{\omega}_y - (\cos\gamma_x \cos\gamma_z) \dot{h}_w + (\sin\gamma_x \cos\gamma_z) \dot{\omega}_x h_w \\ + (\cos\gamma_x \sin\gamma_z) \dot{\gamma}_z h_w - \dot{h}_r + \omega_z (I_x \omega_x \\ - h_w \cos\gamma_x \sin\gamma_z) - \omega_x (I_z \omega_z - h_w \sin\gamma_x) = T_y + G_y \\ \\ I_z \dot{\omega}_z - (\sin\gamma_x) \dot{h}_w - (\cos\gamma_x) \dot{\gamma}_x h_w + \omega_x (I_y \omega_y \\ - h_w \cos\gamma_x \cos\gamma_z - h_r) - \omega_y (I_x \omega_x + h_w \cos\gamma_x \sin\gamma_z) \\ = T_z + G_z \end{aligned} \quad (A-10)$$

Three rotations through the Euler angles ψ , θ , and ϕ will relate the body rates to the inertial coordinate system. A typical Euler transformation is performed⁽⁷⁾. The resulting equations are linearized about a nominal orientation having the x axis pointed in the direction of the spacecraft

velocity vector, the z axis toward the geocenter, and the y axis along orbit normal. This orientation had appeared earlier in Figure A-6. Assuming small angles, the resulting linearized equations appear as

$$\begin{aligned}\omega_x &= \dot{\phi} - \omega_o \psi \\ \omega_y &= \dot{\theta} - \omega_o \\ \omega_z &= \dot{\psi} + \omega_o \phi\end{aligned}\tag{A-11}$$

Assuming small gimbal deflections and nominal values of momentum for the wheel and rotor, equation set (A-10) now becomes:

$$\begin{aligned}I_x \ddot{\phi} + [\omega_o^2(I_y - I_z) + \omega_o h_s] \phi + [\omega_o(I_y - I_x - I_z) + h_s] \dot{\psi} \\ + \omega_o \gamma_x h_{wb} + \dot{\gamma}_z h_{wb} = T_x + G_x\end{aligned}\tag{A-12}$$

$$I_y \ddot{\theta} - (\dot{h}_r - \dot{h}_w) = T_y + G_y$$

$$\begin{aligned}I_z \ddot{\psi} + [\omega_o^2(I_y - I_x) + \omega_o h_s] \psi - [\omega_o(I_y - I_x - I_z) + h_s] \dot{\phi} \\ - \dot{\gamma}_x h_{wb} + \omega_o \gamma_z h_{wb} = T_z + G_z\end{aligned}$$

Further simplification is possible by utilizing the gravity gradient expressions⁽⁷⁾

$$\begin{aligned}G_x &= -3\omega_o^2(I_y - I_z)\phi \\ G_y &= -3\omega_o^2(I_x - I_z)\theta \\ G_z &= 0\end{aligned}\tag{A-13}$$

This allows rewriting equation set (A-12) as

$$\begin{aligned}I_x \ddot{\phi} + [4\omega_o^2(I_y - I_z) + \omega_o h_s] \phi + [\omega_o(I_y - I_x - I_z) + h_s] \dot{\psi} \\ + \omega_o \gamma_x h_{wb} + \dot{\gamma}_z h_{wb} = T_x\end{aligned}$$

$$I_y \ddot{\theta} + 3\omega_o^2(I_x - I_z)\theta - (\dot{h}_r - \dot{h}_w) = T_y \quad (A-14)$$

$$I_z \ddot{\psi} + [\omega_o^2(I_y - I_x) + \omega_o h_s]\psi - [\omega_o(I_y - I_x - I_z) + h_s]\dot{\phi} - \dot{\gamma}_x h_{wb} + \omega_o \gamma_z h_{wb} = T_z$$

The small gimbal deflection relationships simplify equation set (A-5) and represent the momentum storage (or supply) available about each axis, assuming bias momentum:

$$\begin{aligned} h_{wx} &= \gamma_z h_{wb} \\ h_{wy} &= -h_{wb} \\ h_{wz} &= -\gamma_x h_{wb} \end{aligned} \quad (A-15)$$

Hence the control torques available can be obtained by taking the derivatives or

$$\begin{aligned} T_{cx} &= \dot{h}_{wx} = \dot{\gamma}_z h_{wb} \\ T_{cy} &= \dot{h}_{wy} = -\dot{h}_{wb} \\ T_{cz} &= \dot{h}_{wz} = -\dot{\gamma}_x h_{wb} \end{aligned} \quad (A-16)$$

The roll and yaw torques are available through gimbal rates while the pitch torque is available through changes in the wheel speed.

Finally, the roll, pitch, and yaw equations can be written in terms of control torques:

$$\begin{aligned} I_x \ddot{\phi} + [4\omega_o^2(I_y - I_z) + \omega_o h_s]\phi + [\omega_o(I_y - I_x - I_z) + h_s]\dot{\psi} \\ + \omega_o \gamma_x h_{wb} + T_{cx} = T_x \end{aligned}$$

$$I_y \ddot{\theta} + 3\omega_o^2(I_x - I_z)\theta + T_{cy} = T_{yr} \quad (A-17)$$

$$I_z \ddot{\psi} + [\omega_o^2(I_y - I_x) + \omega_o h_s] \psi - [\omega_o(I_y - I_x - I_z) + h_s] \dot{\phi} + T_{cz} + \omega_o \gamma_z h_{wb} = T_z$$

For the pitch equation, the term on the right hand side, T_{yr} , is comprised of

$$T_{yr} = T_y + \dot{h}_r \quad (A-18)$$

representing the external disturbance plus the disturbance torque due to the deviation of the rotor of the DIPS from its nominal speed.

Assuming the spacecraft attitude angles, gimbaled wheel angles, and variation of wheel and rotor momentum are small, the pitch dynamics are decoupled from the roll/yaw dynamics. This allows the three-axis control problem to be reduced to a two-axis problem and single-axis control problem.

A.4 Attitude Control System Design

Since the pitch motion is decoupled from the roll and yaw motion, then the design of an automatic pitch control system is simplified. The procedure for designing the pitch and roll/yaw control systems follows that of Kaplan⁽⁸⁾. The linearized pitch equation of motion derived earlier was shown to be

$$T_{yr} = I_y \ddot{\theta} + 3\omega_o^2(I_x - I_z)\theta + T_{cy} \quad (A-19)$$

Here T_{cy} represents the pitch control torque resulting from changes in the speed of the momentum wheel or more specifically, $\dot{h}_w$. The disturbance torque was represented as

$$T_{yr} = T_y + \dot{h}_r \quad (A-20)$$

where $\dot{h}_r$ is the disturbance due to the deviation of the rotor from its nominal speed, while T_y represents the external disturbance torques such as those due to solar pressure, thrust misalignment, etc. A typical pitch control loop is illustrated in Figure A-7. The control law development follows.

Figure A-7. Conceptual Block Diagram for Automatic Pitch Control

A look at the pitch expression, equation (A-19), indicates that no natural damping is available and can only be provided through the control function, T_{cy} . A satisfactory control law which introduces damping through its $\dot{\theta}$ term is

$$T_{cy} = K_p (\tau_p \dot{\theta} + \theta) \quad (A-21)$$

Here, K_p is the gain of the system and τ_p , the response time. The pitch motion can now be represented by

$$T_{yr} = I_y \ddot{\theta} + K_p \tau_p \dot{\theta} + [3\omega_o^2 (I_x - I_z) + K_p] \theta \quad (A-22)$$

If $3\omega_o^2 (I_x - I_z) \ll K_p$, then the pitch motion becomes that of a damped second order system:

$$T_{yr} = I_y \ddot{\theta} + K_p \tau_p \dot{\theta} + K_p \theta \quad (A-23)$$

Taking the LaPlace transform of the above expression, with all initial conditions equal to zero, gives the transfer function

$$\frac{\theta(s)}{T_{yr}(s)} = \frac{1}{I_y s^2 + K_p \tau_p s + K_p} \quad (A-24)$$

Nominally, $\theta_{ref} = 0$ and sensor errors are ignored in the preliminary considerations, thereby allowing the relatively simple block diagram illustrated in Figure A-8. Damping requires the knowledge of angular rates which are not provided by the available sensors. A pseudorate modulator^(9,10) is used in the loop to synthesize angular rates since direct measurement is not possible. This device produces output pulses only as long as the input voltage (or attitude error) exceeds a certain value.

Figure A-8. Implemented Pitch Control Block Diagram

The natural frequency and damping factor of the transformed pitch equation are found to be

$$\omega_p = \sqrt{\frac{K_p}{I_y}} \quad (A-25)$$

and

$$\beta_p = \frac{\tau_p}{2} \sqrt{\frac{K_p}{I_y}} \quad (A-26)$$

For preliminary design purposes, the damping factor is set equal to one to provide critical damping. A value greater than one would slow down the system response while a smaller value would give undesirable overshoot.

With $\beta_p = 1$, equation (A-24) reduces to

$$\frac{\theta(s)}{T_{yr}(s)} = \frac{1}{I_y(s + \omega_p)^2} \quad (A-27)$$

The pitch autopilot gain, K_p , is determined by the allowable steady-state error and the response time. Since the pitch pointing accuracy requirement is specified at 0.09 degrees, then the steady-state errors must be well below this value.

The maximum magnitude of the solar pressure torque acting on the vehicle about its y-axis (or pitch axis) is expected to be 1.94×10^{-5} ft-lb (2.63×10^{-5} N-m). Assuming $\tau_p \ll$ orbital period, then the maximum steady-state error can be determined by applying the Final Value Theorem to equation (A-27) giving

$$\theta_{ss} = \frac{1.94 \times 10^{-5}}{I_y \omega_p^2} \quad (\text{radians}) \quad (A-28)$$

The steady-state error and response time are tabulated in Table A-II for a range of K_p values. Final gain selection will be based on the system response to the rotor torque disturbance, since this provides the largest torque on the spacecraft that the control system must handle.

Table A-II. Pitch Time Constant and Attitude Error Dependence on Gain.

K_p (ft-lb/rad)	τ_p (sec)	θ_{ss} (deg)
0.25 [0.34]*	167	0.00444
0.50 [0.68]	118	0.00222
1.00 [1.36]	84	0.00111
1.50 [2.03]	68	0.00074
2.00 [2.71]	59	0.00056
4.00 [5.42]	42	0.00028
8.00 [10.84]	21	0.00014

* [N-m/rad]

The responses to step and impulsive inputs will illustrate the behavior of the control system. The inverse LaPlace of equation (A-27) gives the system response as a function of time. For an impulsive input

$$\theta(t) = t e^{-\omega_p t} \cdot \frac{T_{yr}}{I_y} \quad (A-29)$$

where T_{yr} is the impulsive torque. For a step input

$$\theta(t) = \frac{1}{\omega_p} [1 - e^{-\omega_p t} (\omega_p t + 1)] \frac{T_{yr}}{I_y} \quad (A-30)$$

where T_{yr} now represents a step disturbance.

Equation (A-29) is used with $T_{yr} = 0.4$ ft-lb (0.54 N-m), the maximum disturbance torque expected to be produced by the rotor. For the purpose of avoiding too fast of a response time, a gain of 5.0 ft-lb/rad (6.78 N-m/rad) is selected. This selection seems reasonable because the rotor is not expected to produce torques of such a high magnitude very often, if at all. Also this value is sufficient to maintain the pointing accuracy requirement

at all times. The corresponding natural frequency, ω_p , is 0.0535 rad/sec and the system response time, τ_p , is about 37 seconds. The earlier assumptions that $3\omega_o^2(I_x - I_z) \ll K_p$, $\tau_p \ll$ orbital period, and $\theta_{ss} \ll$ pointing accuracy requirement are validated for this selected gain value.

The on-orbit attitude of the spacecraft has its pitch axis coincident with orbit normal. The momentum wheel is aligned along the pitch axis to provide attitude stiffness and coupling between the roll and yaw axes. Roll/yaw attitude control torques are obtained by driving the roll and yaw gimbals through relatively small angles. A roll sensor is provided to sense errors about the roll axis. The roll/yaw coupling eliminates the need for a yaw sensor, since yaw deviations can be sensed indirectly through roll. However, the coupling effect must be increased artificially by selecting system momentum and gains to make the roll controller sensitive to yaw errors. Nevertheless, this must be done without transforming the roll errors into yaw errors.

The linearized roll and yaw equations derived earlier are

$$\begin{aligned} T_x = & I_x \ddot{\theta} + [4\omega_o^2(I_y - I_z) + \omega_o h_s] \phi + [\omega_o(I_y - I_x - I_z) + h_s] \dot{\psi} \\ & + \omega_o \gamma_x h_{wb} + T_{cx} \end{aligned} \quad (A-31)$$

$$\begin{aligned} T_z = & I_z \ddot{\psi} + [\omega_o^2(I_y - I_x) + \omega_o h_s] \psi - [\omega_o(I_y - I_x - I_z) \\ & + h_s] \dot{\phi} + \omega_o \gamma_z h_{wb} + T_{cz} \end{aligned}$$

It is evident from these equations that larger values of h_s will increase the coupling effects. A condition which can be imposed on the system to increase the coupling effects is

$$h_s \gg [I_x \omega_o, I_y \omega_o, I_z \omega_o] \quad (A-32)$$

The roll/yaw equations now become more simplified:

$$T_x = I_x \ddot{\phi} + \omega_o h_s \dot{\phi} + h_s \dot{\psi} + \omega_o \gamma_x h_{wb} + T_{cx} \quad (A-33)$$

$$T_z = I_z \ddot{\psi} + \omega_o h_s \dot{\psi} - h_s \dot{\phi} + \omega_o \gamma_z h_{wb} + T_{cz}$$

Using equation set (A-15) while remembering that the control torques are the rate of change of momentum, then

$$T_x = I_x \ddot{\phi} + \omega_o h_s \dot{\phi} + h_s \dot{\psi} + [\dot{h}_{wx} - \omega_o h_{wz}] \quad (A-34)$$

$$T_z = I_z \ddot{\psi} + \omega_o h_s \dot{\psi} + [\dot{h}_{wz} + \omega_o h_{wx} - h_s \dot{\phi}]$$

where the control terms are in brackets.

Consider the roll equation where direct measurement of ϕ is not available through the roll sensor. The control terms must be related to ϕ to provide fast and well damped responses to disturbance torques. This will minimize coupling of roll errors into yaw. A control law which fulfills the above requirements is

$$T_{xc} = K\tau\dot{\phi} + K\phi - \omega_o h_s \dot{\phi} \quad (A-35)$$

The first term of this equation introduces damping. The last two terms transforms the coefficient of ϕ in the roll expression, equation (A-34), from $\omega_o h_s$ to K , the system gain. Replacing the terms in the bracket with the control law results in

$$T_x = h_s \ddot{\psi} = I_x \ddot{\phi} + K\tau\dot{\phi} + K\phi \quad (A-36)$$

This equation illustrates a second order system with driving functions T_x and $-h_s \dot{\psi}$. The $-h_s \dot{\psi}$ term is the gyroscopic coupling term which permits yaw errors to appear in roll sensor outputs.

The yaw control law cannot provide for direct damping because there is no yaw sensor. The control law instead uses the roll sensor and a pseudorate modulator which synthesizes the angular rate. Substitution of the control law and several algebraic steps later the desired expressions in matrix form are:

$$\begin{bmatrix} T_x(s) \\ T_z(s) \end{bmatrix} = \begin{bmatrix} I_x s^2 + K\tau s + K & h_s s \\ -kK\tau s - kK & I_z s^2 + \omega_o h_s \end{bmatrix} \begin{bmatrix} \phi(s) \\ \psi(s) \end{bmatrix} \quad (A-37)$$

This satisfied by the system block diagram of Figure A-9.

Control system parameters for $K = 1.788 \text{ ft-lb/rad}$ (2.424 N-m/rad) are summarized in Table A-III. Responses to step disturbances were evaluated as a further means of system verification. For this case, a step disturbance of $5.89 \times 10^{-5} \text{ ft-lb}$ ($7.90 \times 10^{-5} \text{ N-m}$) was used, based on a 0.5 degree misalignment of the north-south thrusters. This torque was used as both the roll and yaw disturbance torque. The steady-state values for these cases are summarized in Table A-IV. Again, the system design appears satisfactory.

The roll/yaw control system is basically designed to control the cyclic solar pressure torques. Unfortunately, estimates of roll/yaw solar pressure torques for this spacecraft were not available. However, similarities with other roll/yaw systems, such as that in reference 8, verifies the competence of this system. With the control system design completed, investigation into the DIPS interactions with the spacecraft during the launch phase can proceed.

Figure A-9. Schematic of Roll/Yaw Control System⁽⁸⁾

Table A-III. Roll/Yaw Control System Parameters.

PARAMETER		VALUE
H_s	System wheel momentum	200 ft-lb-s (271 N-m-s)
K	Roll autopilot gain	1.788 ft-lb/rad (2.424 N-m/rad)
τ	Roll time constant	68.5 sec
k	Yaw-to-roll gain ratio	0.0584
ω_1	Roll dynamics natural frequency	0.0324 rad/sec
β_1	Roll dynamics damping ratio	0.999
ω_2	Yaw dynamics natural frequency	0.0025 rad/sec
β_2	Yaw dynamics damping ratio	0.993

Table A-IV. Roll/Yaw Attitude Error Due to a Step Disturbance

CASE		STEADY-STATE VALUE (DEG)
ROLL	Roll axis disturbance	0.0018
	Yaw axis disturbance	0.0
YAW	Roll axis disturbance	0.0135
	Yaw axis disturbance	0.2314

A.5 Attitude Motion Simulations

The automatic attitude control system presented will maintain the spacecraft's on-orbit attitude requirements during the operational phase of the mission. The system will become active only after the spacecraft has reached synchronous altitude and has acquired its on-orbit attitude. Before this, ground or Orbiter commanded signals are sent to the spacecraft to initiate perigee and apogee burns as well as attitude maneuvers. The momentum and torques produced by rotating hardware of the DIPS were the topics of the investigation. Simulations determined the effects on attitude dynamics during the launch and ascent phase of the mission. Specifically, there are seven phases during the launch of interest: initial launch to low orbit, spin-up and deployment, perigee burn, ascent, precession maneuver, apogee burn, and attitude acquisition.

Equations of motion provided the basis for the simulations. Thrust moments will be zero except when performing spacecraft maneuvers. Torque produced by the rotor is that modeled as being the disturbance torque. The resultant attitude motion, induced by the disturbance torque, was examined to evaluate the severity of its effects on the spacecraft.

Computer simulations of the attitude motion were carried out with the modeled rotor deviations for all aspects of the launch and ascent phase. The rotor torque was modeled as a random variation with a maximum amplitude of 0.4 ft-lb (0.54 N-m) and as a single impulsive torque of the same magnitude. Generally, it was determined that the rotor torque is too small relative to the spacecraft size, and doesn't act long enough to create any attitude problems. The rotor would only be allowed to deviate ± 1 percent from its nominal speed before the controller of the DIPS corrects the error.

Therefore, the maximum torque of 0.4 ft-lb (0.54 N-m) could act a second at most in one direction. If the speed were to drop below the 1 percent limit, the controller would speed it up with a resultant torque acting in the opposite direction. Therefore, a coning angle could not build beyond a maximum limit, however, it would tend to be bounded and oscillate within certain limits. In fact, since the torque is equal to the time rate of change of angular momentum, i.e.,

$$T_r = I_p \frac{d\omega}{dt} \quad (A-38)$$

then the maximum coning angle can be calculated quite easily. This is done for three phases in the flight where the moments of inertia of the spacecraft are different. The results are shown in Table A-V, indicating the insignificant effect of the rotor disturbance torque. At the worst case, a coning angle of less than 0.01 degrees would be induced from this torque.

Table A-V. Coning Angle Due to a Rotor Disturbance Torque.

Phase of Flight	Coning Angle (deg)
Pre-perigee burn	0.0018
Post-perigee burn but pre-apogee	0.0037
Post-apogee burn	0.0066

The momentum of the DIPS has more of an affect on the attitude of the spacecraft than the rotor speed variations. A coning motion would also be produced here since the rotor axis is transverse to the spin axis of the vehicle during ascent. The half-angle of this coning motion, α , is found by taking the ratio of the angular momentum of the transverse axis of the spacecraft to that of its longitudinal or spin axis:

$$\alpha = \frac{h_T}{h_L} \quad (\text{radians}) \quad (\text{A-39})$$

In this case, h_T is the rotor momentum and h_L is the spacecraft's momentum due to the 30 rpm spin rate. The resultant coning angle from momentum considerations is indicated in Table A-VI, now dependent upon the moment of inertia of the vehicle axis of rotation. Here again, the angle is not large enough to effect alignment for either the perigee or the apogee burn. However, if the momentum wheel is spun-up opposite the rotor to the same value of momentum, the coning angle would be zero. This is true because the net transverse momentum, h_r plus h_w , is zero. Of course, the wheel should be spun up prior to Shuttle release of the spacecraft, preferably on the ground. A smaller spacecraft would require the momentum wheel before Shuttle release to avoid large coning motion. Simulations using smaller spacecraft show this to be true.

Table A-VI. Coning Angle Due to DIPS Momentum.

Phase of Flight	Coning Angle (deg)
Pre-perigee burn	0.036
Post-perigee burn but pre-apogee	0.080
Post apogee burn	0.155

The nominal spacecraft design has the DIPS rotor aligned along the y axis of the spacecraft. A maximum torque of 0.4 ft-lb (0.54 N-m) is produced by the rotor as it deviates from its nominal speed⁽³⁾. The attitude control system was designed to maintain the pointing accuracy requirement for the spacecraft despite the relatively large torque. Eight other DIPS orientations are now considered. The gimbals of the momentum

wheel are biased to cancel out the momentum of the rotor for these orientations. A four-degree gimbal deflection is sufficient to compensate for the rotor momentum when it is transverse to the DGMW, i.e., lying along the x or z axis. Other orientations require less biasing of the gimbals. Although the attitude control system is not redesigned for each case, some conclusions can be drawn using the nominally designed system. Table A-VII summarizes the attitude control system performance for nine specific orientations of the DIPS.

Table A-VII. Attitude Errors for Several DIPS Orientations

Case	Rotor Axis		Maximum Error (degrees)		
			Pitch $\theta_{\max}$	Roll $\phi_{\max}$	Yaw $\psi_{\max}$
I	y (pitch)		0.09	0	0
II	x (roll)		0	0.14	0.08
III	z (yaw)		0	13.00	1.40
IV	yx plane	22°	0.08	0.05	0.03
V		45°	0.06	0.10	0.06
VI		68°	0.03	0.13	0.07
VII	yz plane	22°	0.08	4.88	0.53
VIII		45°	0.06	9.19	1.08
IX		68°	0.03	12.05	1.30

Simulations were performed to determine the effects of the rotor misalignment relative to the DGMW on the on-orbit attitude performance of the spacecraft. A 0.1 degree misalignment was assumed for the analysis. The power supply misalignments were also considered for different orientations in the yx and yz planes. Typically, nine orientations were considered.

Aside from the nominal orientation along the y axis, the power supply rotor was also aligned along the x and z axes. The remaining six orientations are in the yx and yz planes, each 22, 45, and 68 degrees from the y axis. The angular displacement from the y axis toward the x and z axes are signified by γ_{yx} and γ_{yz} and are shown in Figure A-10. For these cases, the DGMW gimbals are biased to compensate for the rotor momentum. Nevertheless, some misalignment may exist.

Altogether, nine different DIPS orientations were considered with a total of fifteen possible misalignment combinations. The results are summarized in Table A-VIII. In all cases, an individual misalignment angle δ_x , δ_y , or δ_z is 0.1 degrees. Combinations were considered in a limited number of cases. Cases I, II, and III represent the DIPS in the nominal orientation. This seems to be the best. When the DIPS is aligned along the z axis (cases IV through VI), the roll and yaw errors are small, however, constant or at least periodic pitch control is needed. Without this control, the pitch angle would continue to grow and the pitch attitude would be lost within minutes. This seems to be the case also if the DIPS rotor were aligned along the x axis. These results correspond to cases VII through IX. The roll and yaw errors are small, however, continual pitch control is a requirement.

The final six cases represent orientations of the DIPS in the yz and yx planes. In cases X through XII, the DIPS is oriented 22, 45, and 68 degrees from the y axis in the yz plane. The results here are similar to those of cases IV through IX. Likewise, the final three cases have small roll and yaw errors induced, but require active pitch control. In this respect, the nominal orientation seems to be the optimum, since the other orientations require direct pitch control.

Legend:

DIPS Rotor

Figure A-10. Rotor Misalignment and Orientation Angles

Table A-VIII. Attitude Errors due to Rotor Misalignments
for Various DIPS Orientations.

Case	Rotor Axis		Angular Misalignment	Maximum error (degrees)		
				Pitch $\theta_{\max}$	Roll $\phi_{\max}$	Yax $\psi_{\max}$
I	y (pitch)		δ_x	0	0.014	0.007
II			δ_z	0	0.007	0.014
III			δ_x, δ_z	0	0.016	0.016
IV	z (yaw)		δ_x	0	0.014	0.007
V			δ_y	∞	0	0
VI			δ_x, δ_y	∞	0.014	0.007
VII	x (roll)		δ_z	0	0.007	0.014
VIII			δ_y	∞	0	0
IX			δ_z, δ_y	∞	0.007	0.014
X	yz plane	22°	δ_y, δ_z	∞	0.003	0.005
XI		45°	δ_y, δ_z	∞	0.005	0.003
XII		68°	δ_y, δ_z	∞	0.006	0.013
XIII	yx plane	22°	δ_x, δ_y	∞	0.005	0.003
XIV		45°	δ_x, δ_y	∞	0.010	0.005
XV		68°	δ_x, δ_y	∞	0.013	0.006

A.6 Conclusions

An automatic attitude control system has been designed for a spacecraft which derives its energy from a dynamic isotope power supply. A double-gimbaled momentum wheel is the heart of the control system. Roll/yaw attitude control is achieved by deflecting the gimbals of the momentum wheel through relatively small angles. Pitch control is available by changing the wheel speed. The pitch system was designed to compensate for environmental disturbances as well as the rotor disturbance, which provided the largest torque of the two. Gain values were selected and control system responses illustrated. The control system appears to be quite adequate for the maintaining the on-orbit attitude of the spacecraft.

The effects of the momentum and torques produced by the rotor during the launch phase were investigated through digital simulations. These effects were found to be negligible for the spacecraft considered. The momentum wheel would be spun up in the direction opposite to that of the rotor in order to cancel out momentum produced by the DIPS rotor. This would eliminate any potential dynamic interactions between the spacecraft and rotor, thereby avoiding any coning motion. As for the rotor disturbance torque, it was found that this could only produce an attitude error with a maximum of 0.01 degrees about the pitch axis.

Several orientations of the DIPS rotor were considered other than the nominal alignment with the y axis. Alignment along the x and z axes, as well as in the yx and yz planes, was investigated. The z axis was found to be an unacceptable orientation, due to its sensitivity to the rotor disturbance torque. The yz plane orientations were also unacceptable. This can be corrected, however, if a yaw sensor is added to the system. The rotor

coincident with the y axis was found to be the optimum orientation. Likewise, the yx plane proved to be satisfactory for the layout of the DIPS. For other than the nominal orientation, the momentum wheel gimbals would be biased to compensate for transverse momentum due to the rotor.

The effects of rotor misalignment on the spacecraft's performance were also investigated for the various orientations. Here again, the y axis was found to be optimum for rotor alignment. Although other orientations were possible, an active pitch controller would be needed to constantly maintain the attitude requirements for those cases. A growth characteristic of the pitch angle was indicated in all orientations except the nominal.

Future investigations should consider the effect of the DIPS produced torques on smaller spacecraft. The satellite used in this investigation is massive compared to most of the spacecraft in orbit today. If the DIPS were used in a lighter satellite, the rotor disturbances would be more pronounced. A method to adequately compensate for the disturbance torque, however, may not be the best solution in this case. Perhaps modification of the DIPS power conversion unit would be a more practical approach in lessening the magnitude of the rotor disturbance torque. Furthermore, compatibility of other control systems with the DIPS is a topic of interest which should also be investigated, for example, a three-wheel or skewed wheel system may be better than a DGMW system for some of the orientations of the DIPS rotor.

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APPENDIX B

SAMPLE ORBITAL DECAY SIMULATION

B.1 Introduction

The Skylab spacecraft is now a passive, low orbiting vehicle which will, if left alone, reenter the atmosphere by the end of 1980. NASA recently announced a plan to use the Shuttle OFT 3 to either reboost Skylab or target a reentry sometime in October of 1979. Recent increases in solar activity, due to the 11 year solar cycle, have caused a corresponding increase in atmospheric density in the altitude range of Skylab. The resulting increase in drag experienced by Skylab changes the original lifetime estimate determined by NASA.

An orbital decay trajectory of Skylab was calculated using orbital energy dissipation methods. The simulation provides for variation in atmospheric density due to changes in altitude and solar activity. A limited consideration of attitude orientation is also taken into account. Results show an increase in decay rate from the original NASA estimate. It is also seen that initiation of general tumbling motion or attitude change will greatly delay reentry and enhance the success of a reboost mission.

B.2 Orbital Energy Dissipation Analysis

A circular orbit approximation was assumed for Skylab. Therefore, the orbital energy per unit mass can be expressed as

$$E = \frac{-\mu}{2r}, \quad (B-1)$$

and the circular orbital velocity is

$$v_c = \left(\frac{\mu}{r}\right)^{1/2}, \quad (B-2)$$

where μ is the gravitational constant of the earth and r is the circular orbit radius [1]. The aerodynamic drag force, D , acting on Skylab can be calculated from

$$D = \frac{1}{2} C_D \rho A_{ref} v_c^2, \quad (B-3)$$

where C_D , ρ , and A_{ref} are aerodynamic drag coefficient, local atmospheric density, and Skylab reference area, respectively. Determination of these parameters will be discussed later. The power dissipated by this drag force is given as

$$P = D v_c = \frac{1}{2} C_D \rho A_{ref} v_c^3 \quad (B-4)$$

Now the incremental change in orbital kinetic energy, dK , for a time span, dt , is

$$dK = P dt \quad (B-5)$$

If for a given time span C_D , ρ , and v_c remain relatively constant, equation (B-5) becomes

$$\Delta K = P \Delta t \quad (B-6)$$

It was found that a Δt of one day is satisfactory. Thus the new orbital energy per unit mass, E' , is

$$E' = E - \frac{\Delta K}{M} \quad (B-7)$$

where M is the mass of Skylab. The resulting changes in orbital radius, r' , and velocity, v_c' , are

$$r' = \frac{-\mu}{2E'}, \quad (B-8)$$

and

$$v_c' = \left(\frac{\mu}{r'}\right)^{1/2} \quad (B-9)$$

This procedure can be iterated day by day to obtain an orbital decay trajectory.

B.3 Density Model

It was found that atmospheric density had to be considered as a function of both altitude and time. Therefore, the Jacchia (1972) density model for altitudes above 125 Km. was used. In program form, this model takes altitude input and time input in the form of solar activity and geomagnetic indices. Actual values for these indices were used from 1974-77; the predicted values for 1978-81 were obtained from NASA-MSFC [2].

B.4 Aerodynamic Drag Model

Previous Skylab attitude dynamics investigation has shown that Skylab is presently oriented in a semi-stable gravity gradient mode with the Apollo docking port away from the geocenter and the solar arrays slightly rolled from the orbit normal [3]. Aerodynamic drag coefficients and Skylab reference area were obtained from NASA-MSFC [4].

The above orientation corresponds to a drag coefficient of approximately 8.0. To reduce this drag coefficient and thus the decay rate, NASA is presently planning to use the remaining attitude thruster fuel on board to put Skylab in a general tumble or to align the longitudinal axis along the orbital velocity vector [5]. The resulting drag coefficients are 5.0 and 2.4 respectively.

B.5 Simulations and Results

Simulations were done on an IBM 370/168 digital computer. The starting date was May 14, 1974 when Skylab was last boosted to an altitude of 438 km. The solid curve in Figure B-1 shows the subsequent orbital decay trajectory for a high drag gravity gradient orientation. The irregularities in the curve during the 1979 period are caused by predicted fluctuations in solar activity. Also shown in Figure 1 are the predicted decay trajectories for low drag orientations initiated on June 1, 1978. It is seen that either of these attitude modifications greatly reduce the decay rate and thus increase the lifetime of Skylab and enhance the possibility of a successful reboost mission.

B.6 Orbital Decay of a GPHS Fuel Module

The capabilities developed for Skylab have also been applied to other low orbiting objects. For example, we considered the General Purpose Heat Source (GPHS) nuclear fuel modules which are currently being tested at the Los Alamos Scientific Laboratory and the Battelle Columbus Laboratories. The modules basically consist of PuO_2 fuel pellets surrounded by graphite heat shields and impact caps. The fuel modules can be arranged on the satellite according to individual satellite design criteria. Upon satellite decay and reentry, the GPHS fuel modules may separate from the satellite and follow their own decay and reentry trajectories.

The orbital decay of Conceptual Design No. 6 GPHS fuel module [6] (Figure B-2) was determined. The aerodynamic drag coefficient is for a flat plate flying flat-face forward in a free-molecular flow environment. The Jacchia (1972) density model provided accurate variation in atmospheric density. The decay trajectory was calculated by determining dissipation

SKYLAB ORBITAL DECAY

LEGEND:

*****ACTUAL DATA

——GRAVITY GRADIENT ORIENTATION, CD=6.8

....TUMBLE MODE INITIATED 6/1/78; CD=5.8

++++LONGITUDINAL AXIS ALLIGNED WITH VELOCITY

VECTOR INITIATED 6/1/78; CD=2.4

Figure B-1. Skylab Orbital Decay Trajectories

FIGURE B-2* ASSEMBLY DRAWING FOR CONCEPTUAL DESIGN NO. 6

* Obtained from Battelle Columbus Laboratories quarterly report [6]

in orbital energy caused by aerodynamic drag. Figure B-3 shows the subsequent orbital decay trajectory of the GPHS fuel module from an initial altitude of 250 km.

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ORBITAL DECAY OF GPHS FUEL MODULE NO. 6

INITIAL ALTITUDE = 250 KM.

DRAW COEFFICIENT = 4.31 AREA = 8.8168 SQ. M.

MASS = 11.45 KG.

FIGURE B-3 GPHS CONCEPTUAL DESIGN NO. 6 ORBITAL DECAY TRAJECTORY

APPENDIX C

STRUCTURAL ANALYSIS SOFTWARE

The current program version of SAP IV for static and dynamic analysis has proven to be a very flexible and efficient tool for linear structural systems. In contrast to the previous versions of the SAP program, the current one offers a new variable-number-nodes thick shell and three-dimensional element, and out-of-core direct integration for time history analysis.

The capacity of SAP IV is controlled by the number of nodal points of the structural system. For each nodal point six boundary conditions, three coordinates, and the nodal point temperatures are required. All nodal point data is retained in high speed storage during the formation of the element stiffness and mass matrices. With the coordinates of all nodal points known and the equation numbers of the degrees of freedom having been established, the stiffness, mass, and stress-displacement transformation matrices for each structural element in the system are calculated.

The nodal point equilibrium equations for a linear system of structural elements can be derived by several different approaches, and all methods yield a set of linear equations of the following form:

$$M\ddot{u} + c\dot{u} + ku = R, \quad (C-1)$$

where M is the mass matrix, c is the damping matrix, and k is the stiffness matrix of the element assemblage; the vectors u , $\dot{u}$, $\ddot{u}$, and R are the nodal displacements, velocities, accelerations and generalized loads, respectively. A static analysis involves the solution of the equilibrium equations

$$ku = R, \quad (C-2)$$

followed by the calculation of element stresses. The load vectors, R , have been assembled at the same time as the structure stiffness matrix was formed. The solution of the equations is obtained using the large capacity linear equation solver, SESOL, which uses Gauss elimination on the positive-definite symmetrical system of equations.

In a dynamic response analysis the solution of the equations

$$M\ddot{u} + c\dot{u} + ku = R(t) \quad (C-3)$$

is required, where $R(t)$ can be a vector of arbitrary time varying loads or of effective loads which result from ground motion. The program can carry out a history analysis or a response spectrum analysis for the solution of the equations. The history analysis can be carried out using mode superposition or direct integration.

The radiator design of KIPS (Kilowatt Isotope Power System) is a very good example of the SAP IV program applications. The physical configuration of the radiator is as shown in Figure C-1. The configuration of the cross section of each radiator strip is shown in Figure C-2. The dimensions employed in the calculation are:

$w = 2.41$ in.	Tube Wall Thickness = 0.035
$t = 0.03$ in.	$T_{fin} = 104^\circ\text{F}$
$l = 0.374$ in.	$T_{fout} = 42.2^\circ\text{F}$
Tube i.d., = 0.18 in.	$k = 118$ Btu/hr.

The nodal mesh of the radiator strip cross section in the SAP IV program data input consists of 40 nodal points and 42 elements which are shown in Figure C-3. The pressure in the fluid pipe is assumed to be 1.07 psi.

The heat of the medium fluid in the pipe passes through the solid wall of the radiator and radiates into space. The inside of the radiator

Figure C-1. Radiator Configuration

Figure C-2. Cross Section of Radiator Strip

Figure C-3. Nodal Mesh for Calculation

strip is made adiabatic by the insulation material. The temperature of the strip cross section is assumed as shown in Figure C-4.

The maximum displacement in the calculation results occurs at element number 1 and 2. The maximum stress happens to be at element number 2 and 4 due to the great temperature gradient. Part of the computer printout results is shown in Table C-1. For design purposes, the selected material should be flexible enough to sustain the maximum deflection and strong enough to tolerate the maximum stress occurring in the radiator strip.

The SAP IV program is a general computer program for the linear static and dynamic analysis of complex structures. With regard to the future of the program, various important improvements are envisaged. Considering additional analysis capabilities, such as the use of consistent mass matrices, the possibility of including geometric and material nonlinearities, etc., are all presently being developed.

Figure C-4. Temperature Distribution

NODE DISPLACEMENTS

NODE NO.	Y - TRANSLATION (in)
2	-0.10382 D - 06
1	-0.10453 D - 06

ELEMENT STRESSES

ELEMENT NO.	MAXIMUM STRESS (lb / in ²)
4	0.92303 D + 01
2	0.84087 D + 01

Table C-1. Calculation Results