NASA Conference Publication 10116

Nuclear Propulsion Technical Interchange Meeting

Volume II

Proceedings of a meeting held at NASA Lewis Research Center
Plum Brook Station
Sandusky, OH
October 20-23, 1992

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NUCLEAR PROPULSION TECHNICAL INTERCHANGE MEETING

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SYSTEMS MODELING

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NUCLEAR PROPULSION TECHNICAL INTERCHANGE MEETING

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PREFACE

Robert R. Corban
Nuclear Propulsion Office
NASA Lewis Research Center

The Nuclear Propulsion Technical Interchange Meeting (NP-TIM-92) was held at NASA Lewis Research Center's Plum Brook Station in Sandusky, Ohio on October 20-23, 1992. Over 200 people attended the meeting from government, Department of Energy's national laboratories, industry, and academia. The meeting was sponsored and hosted by the Nuclear Propulsion Office at the NASA Lewis Research Center. The purpose of the meeting was to review the work performed in fiscal year 1992 in the areas of nuclear thermal and nuclear electric propulsion technology development.

These proceedings are an accumulation of the presentations provided at the meeting along with annotations provided by the authors. All efforts were made to retain the complete content of the presentations but at the same time limit the total number of pages in the proceedings.

I would like to acknowledge the help and support of a number of people that have contributed to the success of the meeting:

(1) Daniel S. Goldin, NASA Administrator, for taking the time to eloquently contribute to the meeting as our keynote banquet speaker,
(2) the Session Chairmen, for organizing excellent technical content for their sessions and keeping the sessions on-time,
(3) the authors, for describing their results and accomplishments,
(4) our host, Robert Kozar and his dedicated staff at the Plum Brook Station, for providing an excellent facility for the meeting and an commendable tour of their world-class test facilities.
(5) and finally to all the "behind-the-scenes" people that were so instrumental in making the technical interchange meeting a success - especially Bonnie Kaltenstein and Jean Roberts, whose excellent organization and orchestration of the meeting was the key to its success.

NUCLEAR THERMAL PROPULSION

SYSTEMS MODELING

VOLUME 2.

N93-26952

Overview of NASA/DOE/DOD Interagency Modeling Team & Activities

James T. Walton
NASA Lewis Research Center

Outline

- Background
- Team Mission
- Team Objective
- Strategy Development
- Future Direction
- Concluding Remarks

Team Mission

- Integrate State-Of-The-Art Computation Resources With Experimental Knowledge Base To Produce Simulations Of NTP System Performance.

- Provide Users With Variety Of System Models To Aid Design and To Reduce Testing, Cost And Time To Regain Flight Ready Status.

- NASA/DOE/DOD Team Uses Unique Capabilities Of Each Member And Assures Appropriate Peer Review.

The purpose of the interagency modeling team is to integrate state-of-the-art computational resources and techniques, with the current knowledge base, to produce simulations of NTP system performance. The end products will provide users with a variety of validated and/or verified system models to assist in designing and to reduce the testing, cost, and time to reach a flight ready status. This vision can be best achieved by a NASA/DOE/DOD team which can use the unique capabilities of each team member and assure joint support for the resulting models.

Team Objective

- To Develop Five Distinct Computer Programs To Simulate NTP System Performance.

- Each Program Differs In The Level Of Detail And Capability.

A computer model of NTP systems is required for several reasons. First, a parametric NTP model can to predict system performance for several engine configurations on a consistent basis. In other words, a common tool is required to compare the configurations on level grounds; performance numbers for each configuration exist from a variety of sources. Second, a parametric NTP model is required to generate configuration performance data for input into mission analysis codes. Third, a parametric model is required to provide state-point input conditions to the system component designers and analysts. Fourth, an NTP system model is needed to evaluate the effect on performance of system design perturbations (i.e., sensitivity studies). Fifth, an advanced model can evaluate the performance of a given system through startup and shutdown transients. Sixth, a detailed transient model of the experimental engine is required for linkage to the facility model to determine engine-facility interactions. Last, an advanced NTP model can be connected to a control system in order to exercise the control system prior to its integration with hardware. To realize the vision and meet the needs defined above, the objective of the interagency team will be to develop five distinct computer programs, each varying in the level of detail and capability, to simulate NTP system performance.

Team Objective (cont.)

- Level 1 Model - Parametric Steady-State Analysis Tool.
- Level 2 Model - Near-Team Transient Analysis Program.
- Level 3 Model - State-Of-The-Art Transient Analysis Tool With Integrated Fluid Mechanics And Reactor Dynamics.
- Level 4 Model - Transient Model Calibrated To Test Or Flight Engine.
- Level 5 Model - Real-Time Transient Engine Simulation.

The Level 1 model is envisioned to be a relatively simple parametric system model. The primary focus of this program will be to analyze the performance of a variety of configurations. This program is expected to analyze steady-state performance and to require a run time on the order of minutes. The target user market for this program includes mission analysis, component modeling and concept evaluation teams.

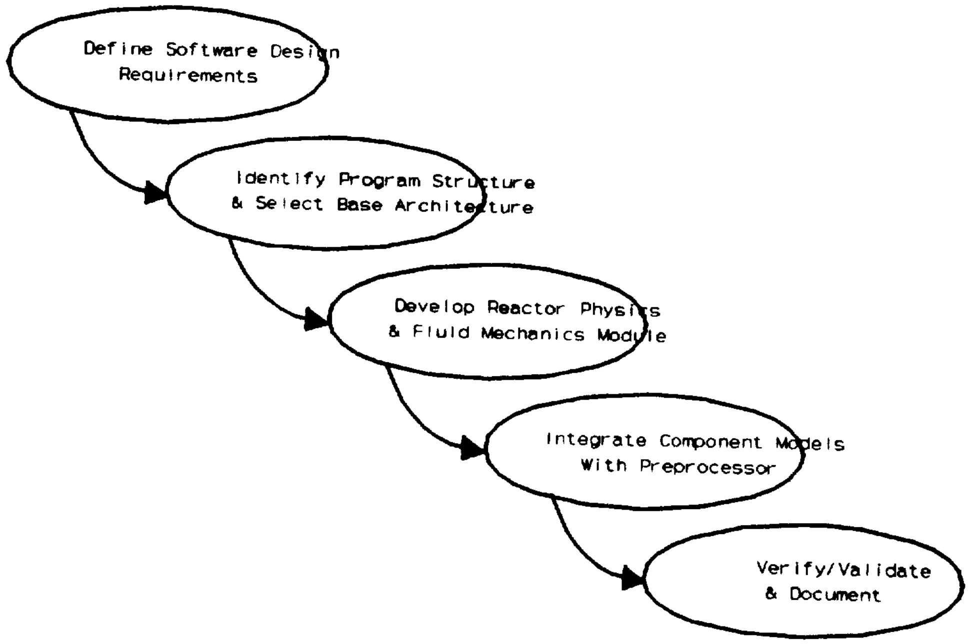
The Level 2 model is envisioned to be a near-term, detailed, transient system analysis program. It may use an existing base architecture program and will be capable of modeling system startup and shutdown as well as system feedbacks and oscillations. The program should be capable of handling control drum rotations, turbopump assembly (TPA) startup, stress analysis, decay heating, and detailed nozzle heat transfer analysis accounting for neutron/gamma heating. The target user market for this program includes component modeling groups and concept evaluation teams.

The Level 3 model is envisioned to be a state-of-the-art, detailed, transient system analysis program. It is anticipated that this program will have neutronic criticality and power density analysis integrated into the base architecture or will provide a means for easy information transfer through coupling. This model will include two-phase and multi-dimensional flow capability. The model will also include shock-capturing numerics to allow simulation of severe accident conditions.

The Level 4 model is envisioned to be a modified version of the Level 3 program tuned to model the experimental or flight engine. The target user market for this program includes component modeling groups, control system developers, and engine performance analysts.

The Level 5 model is envisioned to be a real-time, transient simulation model of the experimental or flight engine. The target user market for this program includes engine operator training groups and flight engine performance review teams.

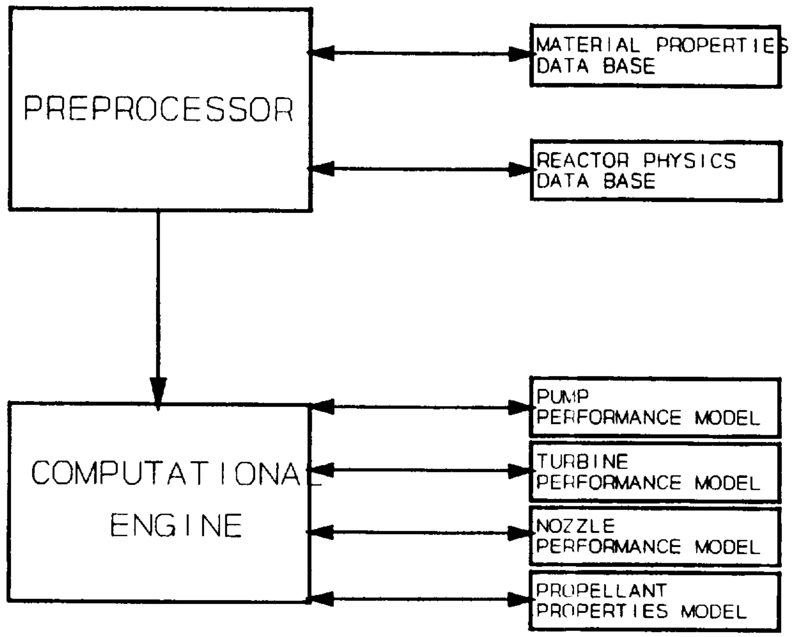
System Modeling Strategy



The strategy for developing each system model is similar and is divided into general tasks as shown above. The strategy begins by working with the users to define their needs in the Software Design Requirements Document and with the identification of the program structure. The subsequent tasks merely reflect the means to assemble the structure and meet the requirements; these tasks evolve from the selected program structure.

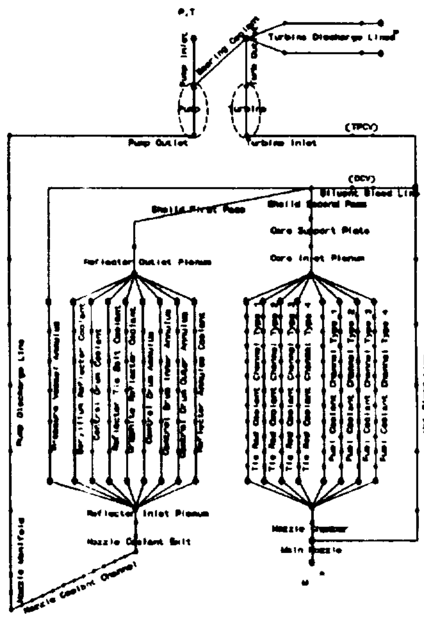
To date, work has focused on the Level 1 System Model. The Software Design Requirements Document has been compiled and the program structure has been identified. A base architecture program has been selected, SAFSIM. While the reactor physics and turbomachinery data bases are under development, the Level 1 model is currently being validated with test data from the NERVA project.

Level 1 Model Structure



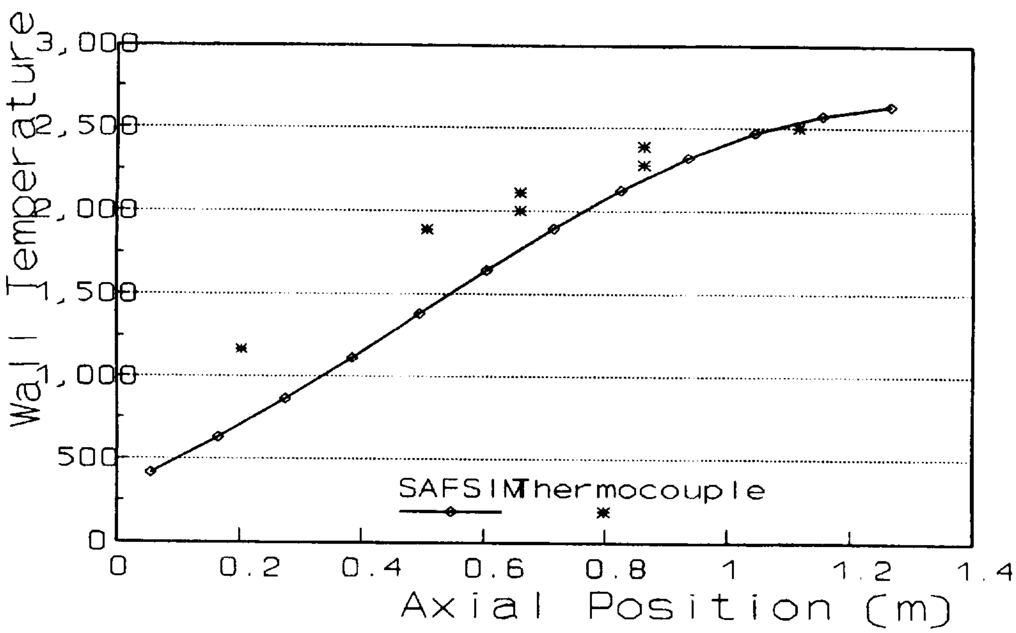
The base architecture (computational engine) for the Level 1 model is a general fluid mechanics program. Therefore, the input file contains all geometry specific information. Thus, the size is quite extensive. An input preprocessor will be used to develop the input files for the user.

Level 1 Model Validation



Concurrent with the development of the databases and component models, the Level 1 model structure is currently being validated with experimental data from the NRX-A4/EST test. Shown above is the schematic flow diagram used to model the NRX-A4/EST. A full-power, steady-state data point was selected for comparison from the EP-IV test run.

Level 1 Model Validation (cont.)



The selected results are from the validation effort are shown above. This figure presents a comparison of measured versus analytical fuel channel wall temperature. The thermocouples were imbedded in the fuel channel wall and, therefore, are expected to be slightly higher.

Level 1 Model Validation (cont.)

	EP-IV	SAFSIM	%Change
Pump Inlet Line			
Mass Flow Rate (kg/s)	36.55		
Pressure (MPa)	0.4208		
Temperature (K)	21.22		
Pump Outlet Line			
Mass Flow Rate (kg/s)	35.38	35.41	00.08
Pressure (MPa)	6.36	6.45	01.42
Temperature (K)	29.	24.3	-16.21
Nozzle Inlet Manifold			
Pressure (MPa)	6.42		
Temperature (K)	24.3		
Reflector Inlet Plenum			
Pressure (MPa)	5.14	5.26	02.33
Temperature (K)	84.4	76.4	09.47
Core Inlet			
Mass Flow Rate (kg/s)	32.8	32.8	00.00
Pressure (MPa)	4.67	4.86	04.07
Temperature (K)	127.	127.	00.00
Tie Rod Exit			
Mass Flow Rate (kg/s)	2.	2.1	05.00
Ave. Temperature (K)	362.		
Fuel Exit			
Mass Flow Rate (kg/s)	30.8	30.7	-00.32
Ave. Temperature (K)	2400.		
Nozzle Chamber			
Pressure (MPa)	3.91		
Temperature (K)	2298.	2301.	00.13
Reactor Power (MW)	1149.4		

A direct comparison of state points shows good agreement except for the pump outlet temperature. The pump efficiency model will be modified to correct this discrepancy.

Future Direction

- Further Develop Data Bases & Component Models For Level 1 System Model.

- Define Requirements & Develop Level 2 System Model.

- Exercise Level 2 Model To Aid Level 3 Definition.

- Initiate Early Development Of Integrated Reactor Physics, Fluid Mechanics & Heat Transfer Program For Level 3 Base Architecture.

The development of the Level 1 model data bases and component models will be a continuing effort. Once completed, the overall model will be documented and a graphical user interface will be developed.

Within the next few months, the development of the Level 2 system model Software Requirements Document will begin. An operational version of this model is needed as soon as possible to provide a test bed for sensitivity studies to aid the Level 3 model definition.

Concurrent with the development of the Level 2 model, initial activities will commence for the Level 3 base architecture.

Concluding Remarks

- An Interagency Effort Was Initiated To Develop Models For Predicting NTP System Performance.

- Models Support Evaluation Of Conceptual Designs And Provide A Diagnostic Tool For Ground Tests.

- Verified & Validated System Models Will Aid In Achieving Man-Rated, Space-Qualified Nuclear Thermal Propelled Vehicles Faster, Cheaper and More Safely.

An interagency NASA/DOE/DOD effort was initiated to develop several models for predicting the performance of nuclear thermal propulsion systems. These models are being developed to support the evaluation of conceptual designs and to provide a diagnostic tool for understanding system tests. Once verified and validated, these system models will aid in regaining the flight-ready status of nuclear thermal propulsion vehicles faster, cheaper, better and more safely by verifying design configurations and minimizing full-scale ground tests.

ENGINE MANAGEMENT DURING NTRE START UP

Mel Bulman
Dave Saltzman
Aerojet Propulsion Division

NP-TIM-92

NASA Lewis Research Center
Plum Brook Station

October 22, 1992

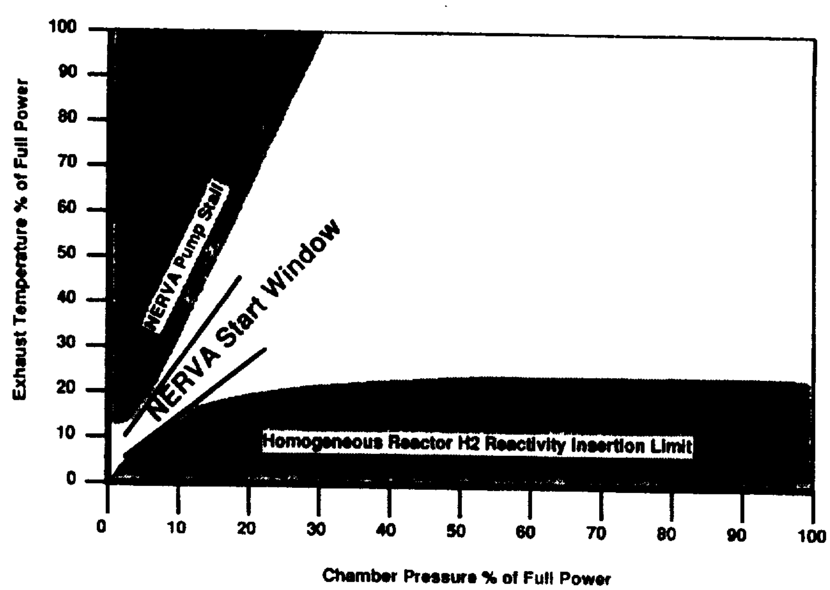
GenCorp Aerojet • Energopool • Babcock & Wilcox

TOTAL ENGINE SYSTEM MANAGEMENT CRITICAL TO SUCCESSFUL NTRE START UP

- Reactor Power Control
  - Hydrogen Reactivity Insertion
  - Moderator Effectiveness (Reactor Spectrum)
- Reactor Cooling
  - Moderator Cooling Loop
  - Fuel Assembly Thermal Shock
- Propellant Feed System Dynamics
  - Pump Characteristics
  - Feed System Pressurization
- Engine Performance
  - Propellant Expended at Low I_{sp}

GenCorp Aerojet • Energopool • Babcock & Wilcox

NERVA Type Engines Have A Narrow Start Window



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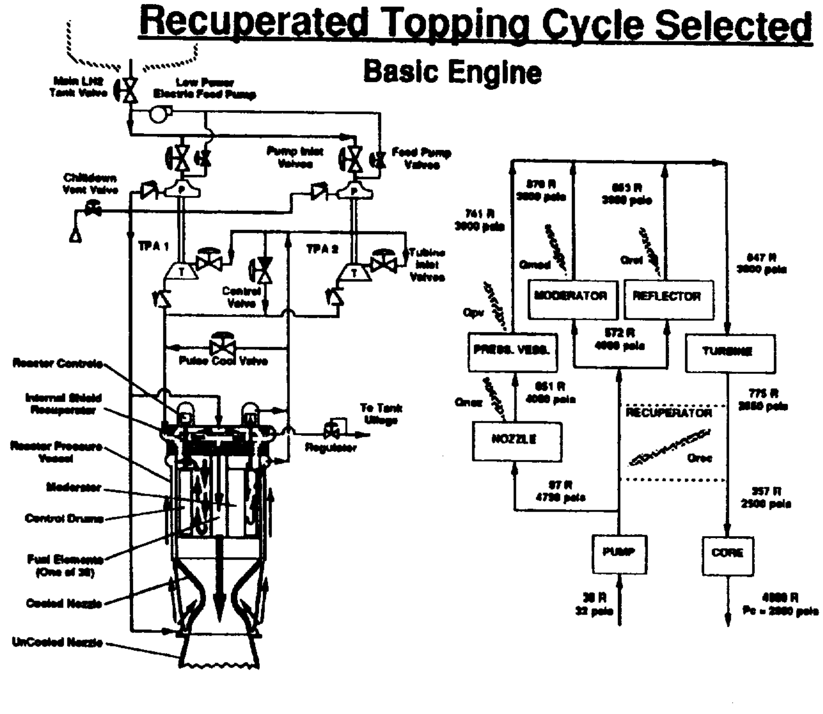
MJB 10/15/92

Recuperated Topping Cycle Selected

Basic Engine



GenCorp Aerojet • Energopool • Babcock & Wilcox

REACTOR POWER CONTROL SUPERIOR WITH HETEROGENEOUS MODERATOR

- More Efficient Fuel Design
- More Efficient Moderator Design
- Less Sensitive to Hydrogen reactivity Insertion
- Reactor Time Constants Longer With more Thermalized Neutrons

GenCorp Aerojet • Energopool • Babcock & Wilcox

HETEROGENEOUS REACTOR COOLING MORE EFFECTIVE

- Moderator Cooled by Separate Loop
  - Fuel Assemblies Can Be Cooled up to Low Power Levels with Moderator Cooling Loop
- Fuel Assembly Inlet Temperature Controlled by Moderator Loop
  - Propellant Preheated in Moderator Loop
  - Recuperator Prevents Large Swings in Propellant Flow or Inlet Temperature (Avoids Thermal Shock)

GenCorp Aerojet • Energopool • Babcock & Wilcox

OUR PROPELLANT FEED SYSTEM DYNAMICS ARE EFFICIENTLY CONTROLLED

- Engine Prestart Conditioning
  - Pumps Chilled In
  - Reactor Warmed
  - Feed System Pressurized
    (Reduces Inrush Dynamics)
- Aerojet Pumps are Designed with Greater Stall Margin
- Our Recuperated Cycle Greatly Aids The Start up
  - Ample Thermal Power Accelerates Bootstrap
  - Provides Thermal and Hydraulic damping
  - Isolates Fuel Assembly from Feed System
- Our Integrated Controller can Choose the Optimun path to Full Power, Balancing:
  - Isp Loss
  - Fuel Element Thermal Shock

GenCorp Aerojet • Energopool • Babcock & Wilcox

INTEGRATED NTRE START SEQUENCE

- Engine Prestart Conditioning
  - Pump Chill In
  - Moderator Loop Pressurization with TPA Chill H_2 (First Start Only)
  - Closed Loop Engine Warm Up (First Start Only)
  - Engine Now on Standby Mode for Starting
- Start
  - Spin Start TPAs with Warm Presurized H_2 From Moderator Loop
  - TPA Acceleration Dominated by Engine Thermal Mass (Power for Approx. 10 Starts in Recuperator Alone)

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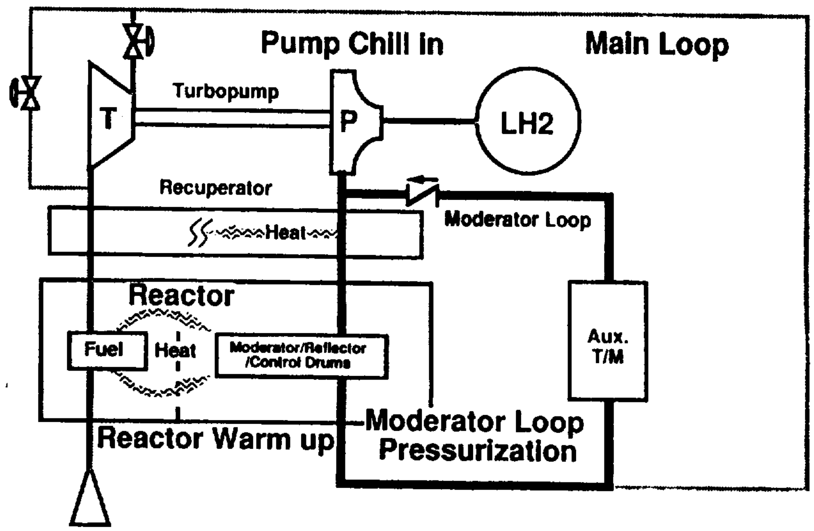
Moderator Cooling Loop Key to Efficient NTRE Starting

Integrated controls

Engine Prestart Conditioning



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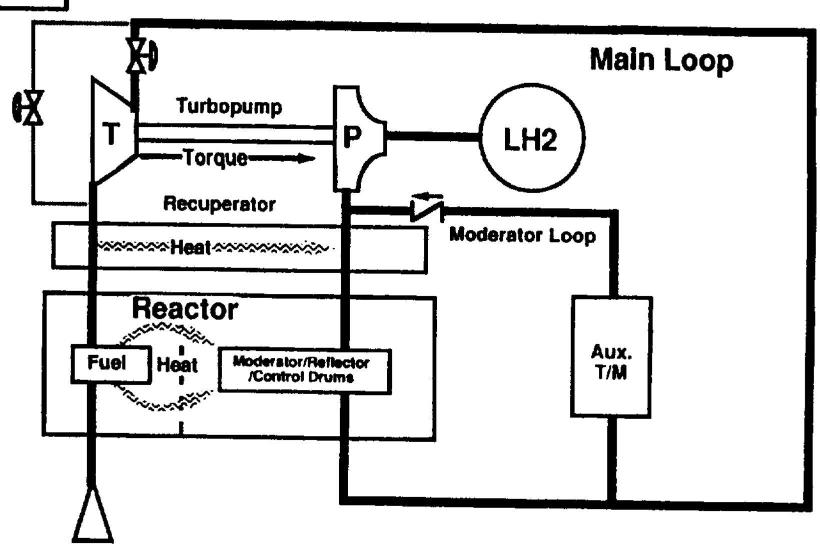
Moderator Cooling Loop Key to Efficient NTRE Starting

Integrated controls

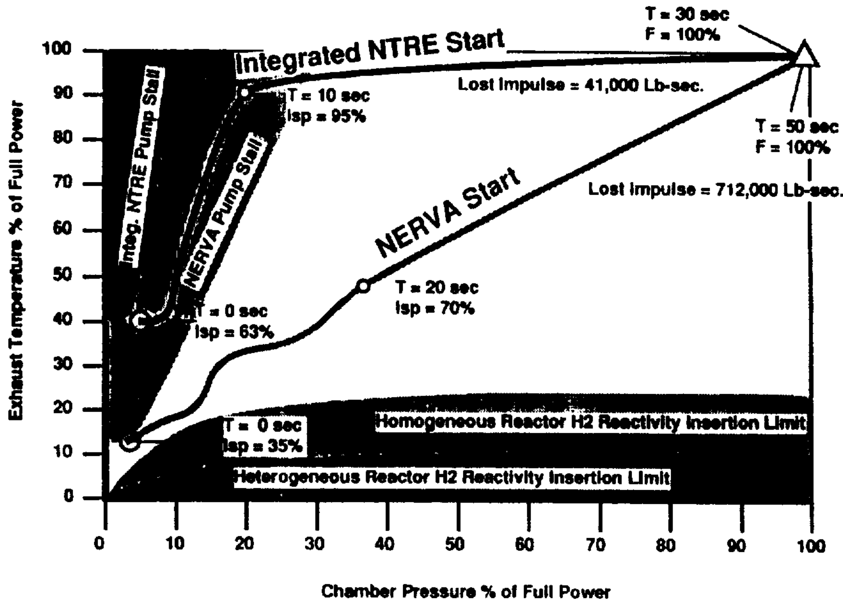
Engine Start



GenCorp Aerojet • Energopool • Babcock & Wilcox

Our Integrated Engine Starts More Reliably

And With Less Impulse Loss than Nerva Type Engines



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MJB 10/15/92

We Are in the Process of Upgrading NETAP

Constructing New Modules for:

- Recuperator
- Moderator
- PBR and CIS Fuel Elements
- Twin 4-Stage TPAs
- Auxiliary Turbo Circulation System

GenCorp Aerojet • Energopool • Babcock & Wilcox

ANALYTICAL SIMULATION IS CRUCIAL TO PROVIDING A LOW RISK ENGINE DEVELOPMENT

- Determine Start Sequence and Operating Limits
  - Valve Phasing
  - Reflector Positioning
  - Thermal Requirements
- Verify Adequate Component Operating Margins Throughout Transient Operation
  - Avoid Pump Stall or Cavitation
  - Reactor Overheating
  - Nozzle Flow Choking
  - Satisfactory Power Balance for Bootstrap
- Establish Control Feedback Requirements

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ACCURATE SIMULATION IS ACHIEVED THROUGH DYNAMIC COUPLING OF PHYSICAL PROCESSES

- TPA Power Balance
- TPA Inertia
- Flow Dynamics and Resistance
  - Method of Charactoristics
  - Volume Filling
- Heat Transfer to Propellant and Components
- Fission Heat Generation / Decay Heat
  - Deposted in Fuel
  - Deposted in moderator
- Momentum, Energy, and Flow Conservation
- Feedback Control Loop

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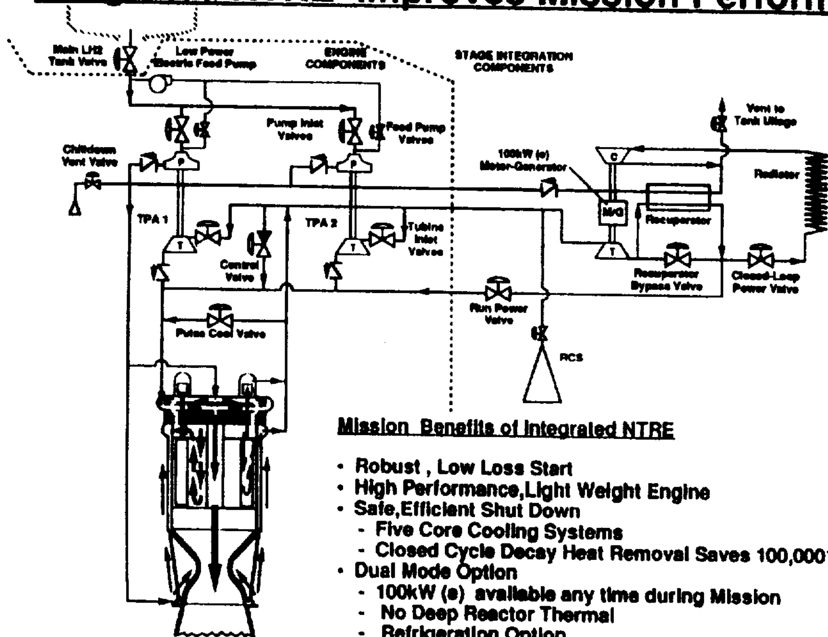
Integrated NTRE Improves Mission Performance



Mission Benefits of Integrated NTRE

- Robust , Low Loss Start
- High Performance,Light Weight Engine
- Safe,Efficient Shut Down
  - Five Core Cooling Systems
  - Closed Cycle Decay Heat Removal Saves 100,000+Lbm IMLEO
- Dual Mode Option
  - 100kW (e) available any time during Mission
  - No Deep Reactor Thermal
  - Refrigeration Option
- OMS & RCS Thrust Available @ High Isp

GenCorp Aerojet • Energopool • Babcock & Wilcox

N93-26954

PARTICLE BED REACTOR MODELING

JOE SAPYTA
HANK REID
LEW WALTON

Babcock & Wilcox

ACKNOWLEDGEMENTS

- **SYSTEM ANALYSES SUPPORTED BY**
  - **SPACE NUCLEAR THERMAL PROPULSION PROGRAM**
  - **B&W INTERNAL FUNDING**
- **PIONEERING WORK FOR PBR APPLICATION TO NTP BY BROOKHAVEN NATIONAL LABORATORY**

View Graph 3 - Acknowledgements

The systems analysis shown in this work was supported by The Space Nuclear Thermal Propulsion Program. The pioneering work on PBR applications to nuclear thermal propulsion systems by Brookhaven National Laboratory is also acknowledged.

PARTICLE BED REACTOR MODELING

- PRESENT THERMAL-HYDRAULIC SYSTEM MODELING TOOLS B&W USES FOR NTP SYSTEMS
- FOCUS ON PARTICLE BED REACTOR TECHNOLOGY AND THERMAL HYDRAULIC METHODS.

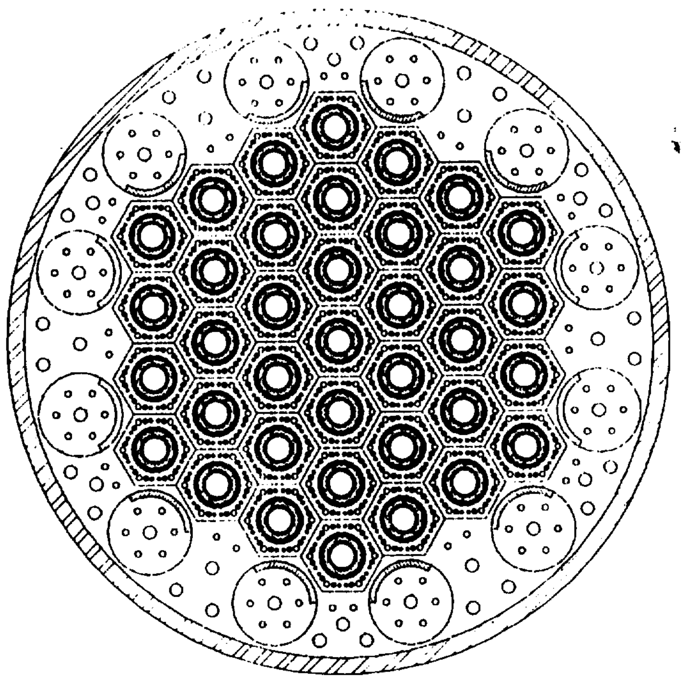
View Graph 4 - Particle Bed Reactor Modeling

The purpose of this discussion is to present to you the thermal-hydraulic system modeling tools B&W uses for nuclear thermal propulsion systems. It will focus on the particle bed reactor technology and the thermal-hydraulic methods used to analyze it. These have received special attention by NASA and others who feel that thermal-hydraulic modeling is a critical issue for nuclear thermal propulsion systems.

The PBR design has received particular scrutiny due to some misconceptions about how flow control is achieved with this technology. I plan to clear up these misunderstandings today.

There will be no discussion of reactor kinetics, reactor physics, or mechanical modeling which are nonetheless important. The presentation will cover some of the challenges of PBR modeling, the Computer codes and physical correlations used, and conclude with some results of analyses and a general philosophy of system modeling.

PBR CORE CROSS SECTION

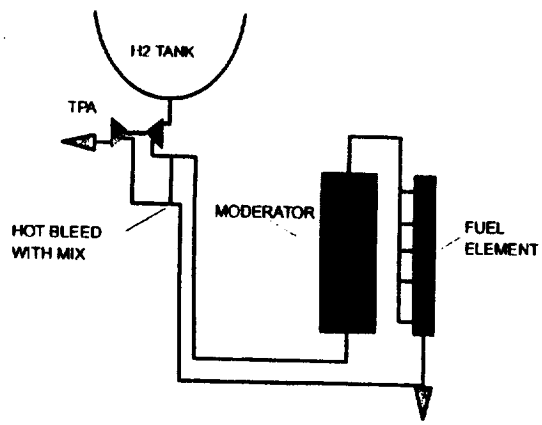


View Graph 5 - Radial Cross Section of Particle Bed Reactor

This view graph shows a radial cross section view of the reactor system we will be discussing today. This system is a generic particle bed reactor system made up of 37 fuel elements as shown by the red circles. The blue area surrounding the fuel elements are hexagonal moderator blocks. Some of the holes shown in the blocks are for propellant flow through the moderator.

This core is surrounded by a reflector and twelve control drums which are in turn surrounded by a pressure vessel. Details of particle bed reactor systems were presented in several papers at this workshop and won't be covered here.

PBR BLEED CYCLE



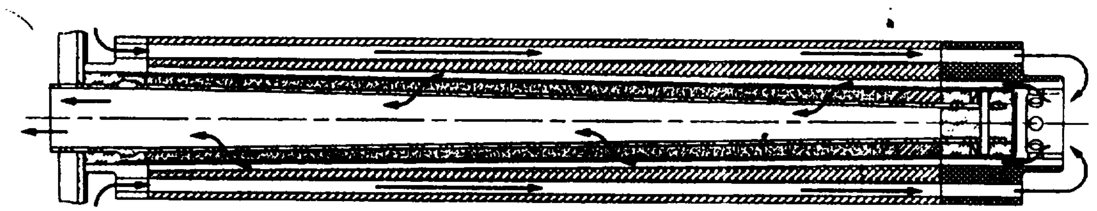
View Graph 6 - PBR Bleed Cycle

Several different flow cycles have been proposed for the PBR rocket system. I will not discuss those here except to describe the flow in the reactor system itself for a hot bleed cycle. In the hot bleed cycle shown here, the propellent is routed through cooling channels in the moderator, reflector and nozzle walls. The propellant may be split between any of the three components, or be separated by plenums and have a single pass cooling loop. Depending on design requirements, flow split and single pass concepts can be used for any combination of the moderator, reflector or nozzle wall flow paths.

The propellant exits the moderator and is collected in a plenum above the core. It is then sent through the fuel element and exits the engine via the nozzle. Target outlet temperatures are nominally very high to maintain high ISP. Mach number is about 0.25 at the outlet.

For purposes of reactor modeling there are three areas which are usually discussed separately since they require different types of computer codes and basic data for evaluation. These include the entire particle bed reactor rocket system, including turbo-pump assemblies. This system modeling will not be discussed here today. The other two areas are fluid flow in the entrance and exit plenums of the reactor system and finally modeling of fluid flow through the particle bed fuel element.

FUEL AND MODERATOR FLOW PATHS



View Graph 7 - Fuel Element Flow paths

This is a view of a particle bed fuel element with flow paths shown by arrows. The red hatched (outer) area is the moderator section; the orange area is the fuel bed and the green areas the innner (hot) and outer (cold) frits that hold the fuel particles.

A typical path has gas entering at the moderator to cool it, then to a plenum at the entrance side of the fuel element, or directly into the fuel element. Orificing of the element can be done at either the moderator entrance or the fuel element entrance.

The gas enters the cold frit which is at the outer annulus of the fuel element, then passes through the fuel bed, and hot frit where it turns and flows out the outlet channel.

Target outlet temperatures are high to maintain high specific impluse. Mach number is approximately 0.25 at the outlet.

PBR MODELING REQUIREMENTS

- **1. FLUID FLOW THROUGH A PARTICLE BED**
- **2. COMPRESSIBLE AND INCOMPRESSIBLE FLOW**
- **3. SINGLE and TWO-PHASE FLOW**
- **4. COUPLES FLUID FLOW and SOLID HEAT TRANSFER**

View Graph 8 - PBR Modeling Requirements

The dynamics of gas flow in this system is dominated by fluid flow characteristics through a packed particle bed. This has been extensively studied along with the application to gas cooled reactors both in this country and Europe.

Since exit Mach number is approximately 0.25, the flow can be treated as incompressible. However, because of the extremely large changes in density in going from the relatively cold inlet temperature to extremely high exit temperatures, thermally expandable flow techniques (fluid density independent of pressure changes) will be required. This can be modeled with the equations used for compressible flow or with a sepatate treatment using equations for thermally expandable flow.

Under normal steady state operation all flow is expected to be single phase, however there are potential accident transients and system cycles where two phase flow would have to be considered.

Computer codes and methods modeling this system will need separate fuel particle and fluid flow modeling to cover the complex thermal-hydraulic dynamics encountered in the fuel bed.

PBR MODELING REQUIREMENTS, Cont.

- RANGE OF SINGLE TO MULTI-DIMENSIONAL MODELING
- TRANSIENT AND STEADY-STATE ANALYSIS

View Graph 9 - PBR Modeling Requirements (Continued)

The computer codes used to analyze the fuel element will need multi-dimensional capabilities. The systems level analysis will use primarily one-dimensional techniques. Both transient and steady state analysis will be required to cover the wide range of operating and accident modes.

CHARACTERISTICS OF PBR and NTP MODELING

- 1. FUEL ELEMENT FLOW-TO-POWER MATCH
- 2. REACTOR FLOW-TO-POWER MATCH
- 3. BED TO COLD FRIT HEATING EFFECTS

View Graph 10 - Characteristics Of PBR and NTP Modeling

The most obvious characteristic of the PBR is flow-to-power matching in the fuel element which must occur to account for axial power distribution and dynamic head in the fuel element exit channel. Other effects like heat conduction from the bed to the cold frit and overall reactor flow-to-power matching to handle radial power distributions and orificing to elements must also be considered.

CHALLENGES FOR PBR and NTP MODELING

- 1. START UP TRANSIENTS
- 2. DECAY HEAT
- 3. THROTTLING CONDITIONS
- 4. ACCIDENT TRANSIENTS
- 5. PRE-TEST PREDICTIONS
- 6. COMPONENT HEATING

View Graph 11 - Challenges for PBR and NTP Modeling

This view graph lists a number of applications of modeling required for a PBR reactor. These also include use of modeling for designing tests and performing post-test evaluations. Examples of system analyses for Decay Heat cooling and Start Up Transients will be presented later.

THERMAL HYDRAULIC COMPUTER CODES

- **1. OTV ENGINE - B&W**
  - **PARTICLE BED FUEL ELEMENT DESIGN SPECIFIC**
- **2. TEMPEST - BATTELLE NORTHWEST**
  - **GENERAL 3-D CFD ANALYSIS**
- **3. SAFSIM - SANDIA**
  - **NETWORK SYSTEMS ANALYSIS CODE**
- **4. SINDA/SINFLO-NASA**
  - **DETAIL THERMAL ANALYZER**

View Graphs 12-21 - Thermal Hydraulic Computer Codes, Code Capabilities and Limitations

The next ten view graphs show the major thermal-hydraulic codes which have been used by B&W for analysis of NTP systems, along with some of their capabilities and limitations. Time doesn't allow a full discussion of these view graphs and the graphs are self-explanatory. Since most of the codes are available in the public domain their names are recognizable to you and won't be discussed. Some of these codes were developed by B&W and are not quite as well known. The primary code in this class was one called OTV Engine. This computer code is used extensively by B&W to provide the nominal fuel element design conditions and specifically to calculate cold frit masking factors that will meter the flow through the cold frit. This code is particularly useful in that it calculates pressure drops due to resistance of the material in the cold frit, and dynamic head effects from gas exiting in the hot channel to provide masking factors which will ensure boundary conditions of constant exit temperature in the exit channel.

You will notice that a wide range of codes are listed here since typically a single code or code system will not provide the combination of capabilities and features desirable for a wide variety of applications. The limitations listed for the major computer codes are a good indication of why a large number of codes are used. In general the one-dimensional network systems analysis codes like SAPSIM will be used for pipe flow and flow splits. The multi-dimensional codes like TEMPEST are used for fuel element analysis.

The SAFSIM Computer Code has been recently obtained from Sandia National Laboratory and has not had significant use by B&W to date, although we are currently in a program to evaluate this code because of its many promising features. This code will be covered by a separate presentation later today. Finally it should be noted that all the codes listed here are single phase. Two-phase capability will be required to analyze off nominal transient and/or accident conditions.

CAPABILITIES FOR PBR/REACTOR APPLICATION

- OTV-ENGINE
  - PROVIDES "NOMINAL" FUEL ELEMENT DESIGN CONDITION
    - SPATIAL FUEL TEMPERATURE
  - PROVIDES "OFF-NOMINAL" DESIGN CONDITIONS

THERMAL/HYDRAULIC CODES, cont.

- 5. ANSYS - SWANSON, INC.
  - DETAIL THERMAL CODE FOR COMPONENT AND LOOP ANALYSIS
- 6. NEST - B&W
  - TRANSIENT ANALYSIS OF COUPLED NEUTRONICS, THERMAL-HYDRAULICS
- 7. ATHENA - INEL
  - 1-D TRANSIENT OR STEADY STATE SIMULATION OF SPACE REACTORS

CAPABILITIES, cont.

- TEMPEST
  - MULTI DIMENSIONAL CFD ANALYSIS
  - ALLOWS ANALYSIS OF ACTUAL DESIGN
  - ADDRESSES COMPLEX THERMAL/FLOW
- SAFSIM
  - REACTOR AND ENGINE SYSTEM
- SINDA
  - GENERALIZED CONDUCTION AND 1-D CIRCUIT FLOW SPLIT MODELING CAPABILITY

CAPABILITIES, Cont.

- ANSYS
  - PERFORMS GENERALIZED DETAIL HEAT TRANSFER ANALYSIS
  - PROVIDES GENERAL COUPLED FLOW/CONDUCTION HEAT TRANSFER FOR SPECIFIED (KNOWN) FLOW REGIONS
- NEST
  - EVALUATION OF SYSTEM CONTROL

LIMITATIONS

- OTV-E
  - STEADY STATE
  - NO REACTOR PHYSICS
  - NO CONDUCTION (gas or solid)
  - NO GENERAL FEATURE CAPABILITY
  - CHANNEL APPROACH TO FLOW (1-D)

LIMITATIONS, cont.

- TEMPEST
  - NO REACTOR PHYSICS
  - LIMITED TO ORTHOGONAL CURVELINEAR GEOMETRICS AT PRESENT
  - TIME STEP LIMITED TO "MATERIAL-COURANT
- SAFSIM
  - TIME STEP LIMITED TO "MATERIAL-COURANT"
  - PSEUDO MULTIDIMENSIONAL (1-D FLOW, NETWORK HEAT TRANSFER)

LIMITATIONS, cont.

- NEST
  - POINT KINETICS
  - QUASI-STEADY FLUID FLOW
- SINDA
  - MODEL DEFINITION IS TEDIOUS
  - FLOW IS INCOMPRESSIBLE
  - NO SPECIFIC PROVISION FOR FLUID FLOW THROUGH PARTICLE BED
  - STEADY STATE

LIMITATIONS, cont.

- ANSYS
  - STEADY STATE FLOW
  - INCOMPRESSIBLE FLOW ONLY
  - LACKS SPECIALIZED CORRELATION CAPABILITY (FRICTION, FILM COEFFICIENT, etc.)
  - PSEUDO MULTI-DIMENSIONAL (1-D FLOW, 3-D HEAT TRANSFER)
- ALL CODES LISTED ARE SINGLE PHASE - WILL NEED TWO PHASE CAPABILITY

PHYSICAL CORRELATIONS

- SPECIFIC CORRELATIONS FOR PARTICLE BED
  - FILM COEFFICIENTS - ACHENBACH
  - FRICTION COEFFICIENT - ERGUN
- FUEL ELEMENT COMPONENTS (COLD & HOT FRITS)
  - MODIFY GENERALIZED CORRELATIONA FOR SPECIFIC APPLICATION BASED ON EXPERIMENTAL DATA

View Graph 22 - Physical Correlations

The next two view graphs provide some information on the second major component of systems modeling - the validity and determination of the physical parameters and correlations used for modeling of the system. These view graphs show well known correlations that have been used in particle bed modeling. They also identify the need for experimental verification of this data. B&W has performed many of the experiments required to verify this data.

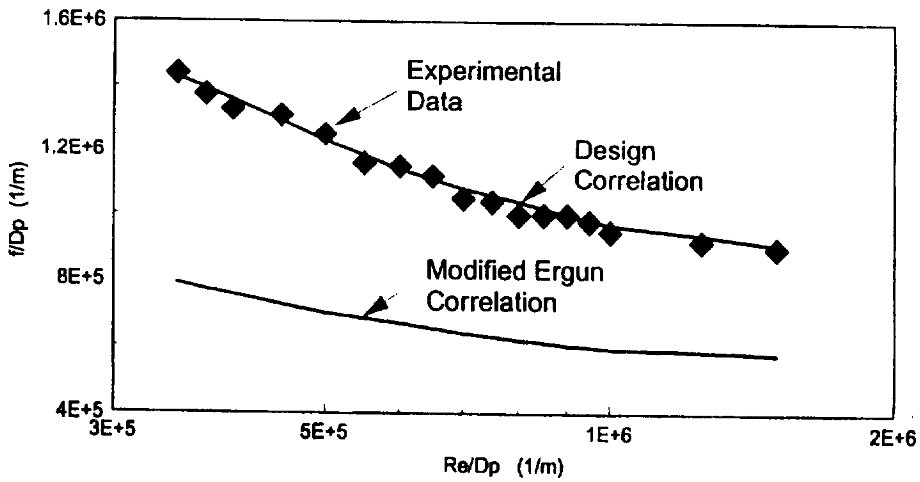
MODIFIED CORRELATIONS

- EXAMPLES
  - MODIFY ERGUN CORRELATION FOR COLD FRIT
  - FRICTION FACTORS FOR BLOWING AND SUCTION FLOW
  - PARTICLE BED CONDUCTIVITY - ZEHNER AND BAUER

View Graph 23 - Modified Correlations

Some examples of correlations that have been modified are shown here. They include friction coefficients for cold frits, friction factors for blowing and suction flow in the entrance and exit annulus of the fuel element and particle bed conductivity.

Comparison of Predicted Friction Factor and Experimental Data



View Graph 24 - Comparison of Predicted Friction Factor And Experimental Data

This view shows a comparison of a predicted friction factor correlation of a outer (cold) frit as compared to the design correlation determined from experimental data taken at B&W's Alliance Research Center. In this case, air was flowed through typical manufactured frits and pressure drop measurements performed. This plot is a measure of the normalized friction factor as a function of Reynolds number. As you can see the design correlation, which has an accuracy of plus or minus 10%, is approximately 30 to 40% higher than the theoretical friction factor and shows a steeper increase with lower Reynolds number.

In addition to tests of cold frit, B&W has used experimental data for friction factors covering blowing and suction flow in the fuel element annulus and have plans for performing tests on particle bed conductivity. As shown on the previous view graph, B&W currently uses the correlation of Zehner and Bauer for particle bed conductivity. This correlation was not developed for PBR applications and therefore will be experimentally verified.

FRIT PRESSURE DROP TESTING WITH H_2, AIR, and N_2

TEST CONDITIONS

P 3.2 MPa

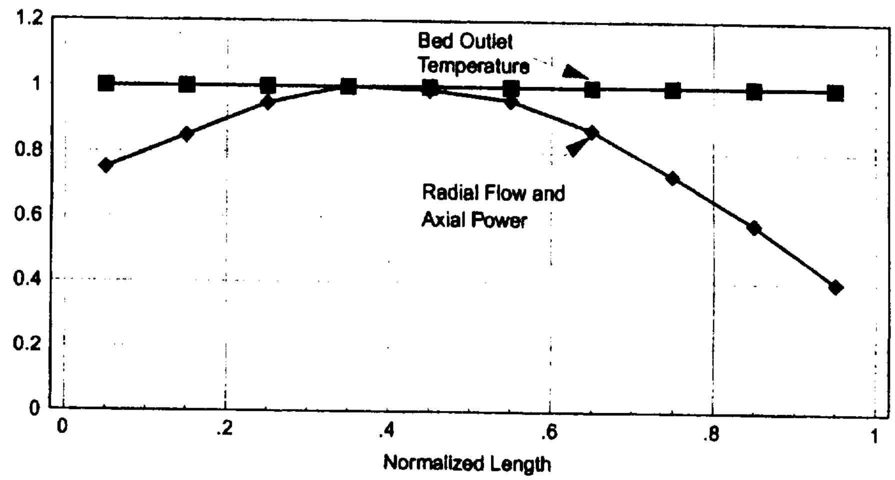
T 294 K

(%)	Re/Dp			Kexp-Kcalc
Gas	(10 1/m)	Kexp (10)	Kcalc (10)	Kcalc
Air	5.08	5.51	5.43	+1.5
Air	5.02	5.36	5.48	- 2.2
H_2	5.38	5.25	5.19	+1.2
H_2	5.38	5.27	5.19	+1.5
N_2	5.04	5.39	5.47	-1.5

View Graph 25 - Frit Pressure Drop Testing

This table shows some results of pressure drop measurements on a outer (cold) frit using hydrogen, air and nitrogen. In this case the data and calculations compare well. It also shows that you can use different gas at the same Reynolds number and get meaningful results.

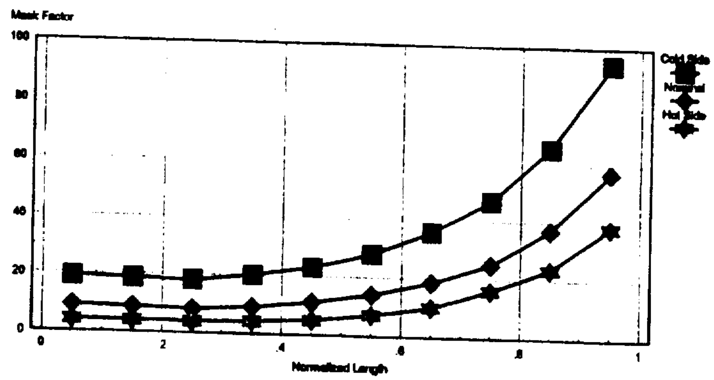
COLD FRIT FLOW-TO-POWER MATCHING



View Graph 26 - Cold Frit Flow-To-Power Matching

Before we get into decay heat cooling, we should show how we control flow to match power at normal operation. The view graph demonstrates the fact that the radial flow into the outer (cold) frit must match the axial power distribution in order to obtain a constant outlet temperature. This metered-flow design is basic to the PBR concept.

COLD FRIT MASK FACTOR
With Azimuthal Power Variations



View Graph 27 - Cold Frit Mask Factor - Azimuthal Variation

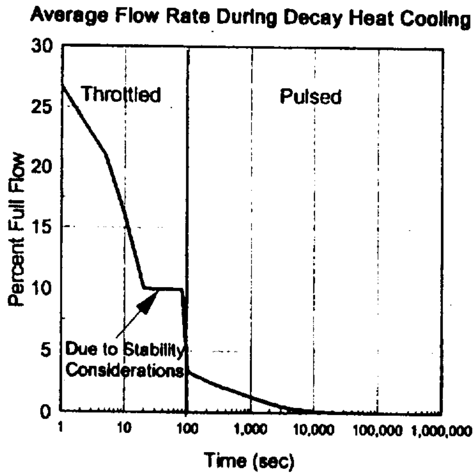
In order to match flow to power in all locations on the outer (cold) frit, friction or masking factors are used to design the frit such that the flow matches the power. In effect assuring that less resistance to flow occurs at the hotter spots.

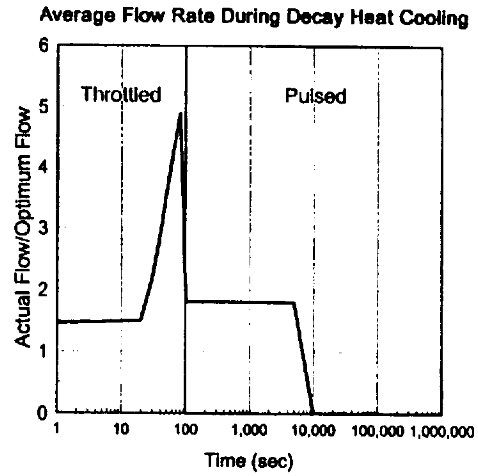
This view graph shows typical masking factor variation along the axial direction of the element. The three curves are for the hot, cold, and nominal (average) power sides of the frit. These differences account for the azimuthal variation around the element produced by the radial change in power in the reactor.

The next segment covers decay heat cooling. Since power, or heat source, distributions change during idlying (decay heat) operation, total flow through the element must account for the fact that the cold frits were masked to match the power at full power operation. This is usually done by supplying excess flow to the element.

The next series of view graphs will show some results of analysis performed for decay heat (idling condition) and start up conditions in a particle bed reactor.

Decay Heat Flow Rate



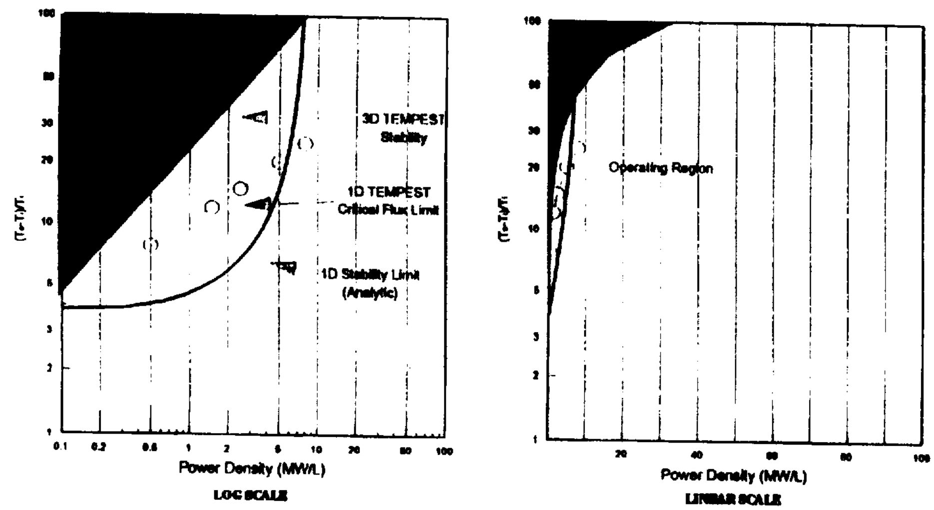


View Graph 28 - Decay Heat Flow Rate

This view graph shows a typical evaluation of the propellant flow rate required after shut-down to cool a particle bed reactor under decay heating caused by the gamma and beta radiation being emitted by nuclear fuel after shut-down. The scenario used for decay heat cooling is to maintain a throttled overflow of propellant for approximately the first 100 seconds after shut-down to insure a cool geometry. The flow is gradually decreased to match the declining power output of the core until the 10% flow plateau is reached. This flow is maintained constant for a while due to stability considerations which I will discuss later. The system then converts to pulse cooling similar to that planned for the NERVA engine. Pulse cooling continues through approximately 10,000 seconds or until the system gets to approximately one to two percent of full power. At this point a long-term closed cycle cooling system would be used to keep the reactor cooled through some type of closed loop system. This system would radiate the small excess heat to space. The view graph on the right is a plot of the actual predicted flow to the optimum flow needed for this process. In this case optimum flow would be that flow needed to exactly match flow to system heat rate. As you can see there is a spike where the actual flow exceeds the optimum flow by approximately five times for a short period of time to accommodate instability limits.

It should be noted that the numbers shown here were obtained with analysis of a single fuel element. They do not account for flow splits in the total system. Also no mechanical analysis were performed to determine the effects of thermal cycling during pulsed cooling.

Stability Regimes



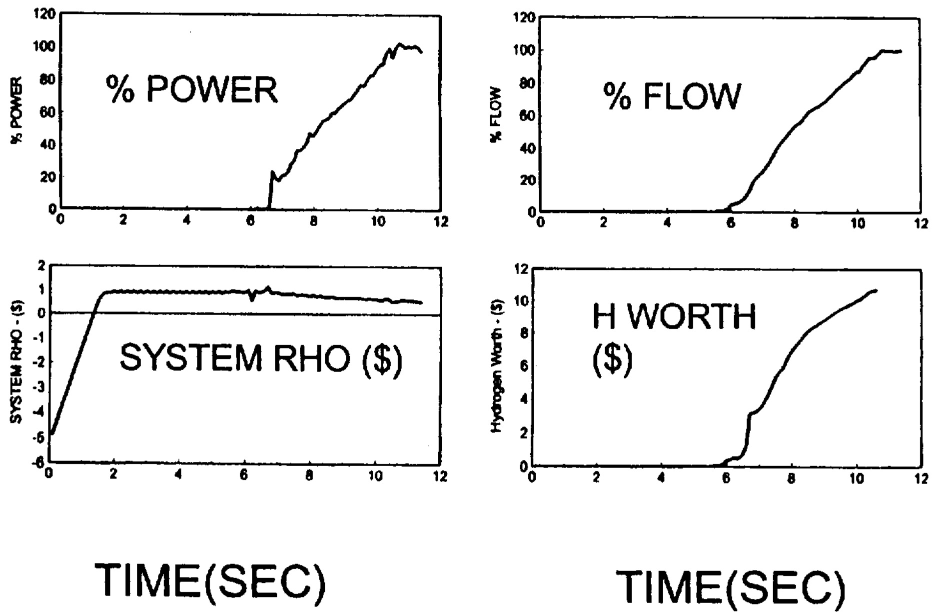
View Graph 29 - Stability Regimes

This view graph shows two presentations of the same data. The one on the left using a log scale for the "x" axis and the one on the right using a linear scale for the "x" axis. The "y" axis is a plot of a instability index developed by Bussard based on NERVA data and applied by Maise of Brookhaven National Laboratory to particle bed systems. This index is the difference between the inlet and the outlet temperature divided by the inlet temperature and here it is shown as a function of power density. If you focus your attention to the view graph on the right, the open area is that region where flow instabilities would not occur. The shaded region is where there are potential flow instabilities. In the case discussed here, the shaded area is only approached during decay heat cooling and is not a factor in the operating regions.

The view graph on the left shows an example of how the unstable region shrinks as one performs more detailed analysis of flow instabilities. The curve shown at the right labeled 1-D stability limit is the analytical result obtained by Bussard. The open circles represent a shift of the one-dimensional stability regime when analyzed with the computational fluid dynamics computer code TEMPEST. The darkly shaded areas show even further movement when a particle bed system is analyzed with three dimensional codes. In this case the area is shown shaded because no sharp boundary exists. Instead we are predicting a gradually increasing probability of flow maldistribution. The actual region of instability would have to be verified by experiment because of these uncertainties. These curves show the advantages of using multi-dimensional analysis on these complex geometries.

We need to note that this is not only a PBR problem - all gas reactors will need to accommodate instability limits at low flow/high delta T conditions.

STARTUP TRANSIENT SIMULATION



View Graph 30 - Start-up Transient Simulation

This view graph gives a representative example of an analysis performed for the start-up of a particle bed reactor. This analysis was done with B&W's NBST computer code system. It was performed to evaluate the unusually high reactivity insertion from flowing cold hydrogen during start-up of the system. In particular it was being used to evaluate the effectiveness of the control mechanisms to mitigate the large insertion of positive reactivity into the system during start-up. These slides show the percent power, percent hydrogen flow, hydrogen worth, and reactivity change of the system versus time over a period of approximately twelve seconds. This analysis shows the system can achieve and maintain design power.

The start up scenario used here is "dry". The reactor is taken critical before hydrogen flow is initiated. As hydrogen starts to flow one set of control elements is moved to overcome the positive reactivity insertion caused by hydrogen flow. Another set of control elements, with different characteristics from the first, is used to control power. The control algorithm controls to a demand startup period while constrained by maximum power versus flow requirements which are shown in this viewgraph.

PHILOSOPHY OF SYSTEMS MODELING

- **THE PROOF OF THE PUDDING IS IN THE TESTING**
- **LEARN FROM EXPERIENCE**
  - **SKYLAB and HUBBLE**
- **SYSTEMS MODELING IS A GUIDE FOR PERFORMANCE AND TESTING. IT IS NOT THE FINAL WORD**

View Graph 31 - Philosophy of Systems Modeling

This is a general attitude or philosophy towards system modeling that says testing is required to verify system operation and subsystem performance (fuel element tests, separate effects test of physical parameters, and separate flow tests through components).

The Hubble telescope had significant problems because it wasn't tested before launch. Skylab was damaged during launch because data from other vehicles was ignored. This is not intended to pick on NASA, there are other industries that have similar tales to tell. These were picked because they are recent or more easily identified by NASA.

SUMMARY

- CHALLENGES OF PBR MODELING AND SYSTEM ANALYSIS
- COMPUTER CODES
- PHYSICAL CORRELATIONS
- RESULTS OF ANALYSIS FOR DECAY HEAT COOLING AND STARTUP
- PHILOSOPHY OF SYSTEMS MODELING

View Graph 32 - Summary

In summary this presentation has covered the characteristics and some challenges of Particle Bed Reactor modeling. It covered the major components of modeling; Computer codes, physical correlations used, a test philosophy, and selected results of decay heat cooling and start-up analyses.

Finally, there was an appeal to all of us to keep in mind the necessity of obtaining experimental data to verify systems performance and systems models.

FINAL THOUGHTS

- NOBODY BELIEVES THE ANALYSIS EXCEPT THE ANALYST
- EVERYBODY BELIEVES THE EXPERIMENT EXCEPT THE EXPERIMENTALYST
  - Seen on NASA wall
- "PAPER REACTORS, REAL REACTORS" Admiral Hyman Rickover - 1953

View Graph 33 - Final Thoughts

In parting, I'll leave you with these words which were seen on a NASA wall poster during a recent visit to the Huntsville Space Center. I have included in the written version of this presentation some excerpts from a paper entitled "Paper Reactors, Real Reactors" written by Admiral Hyman Rickover in 1953. As we all know, the Admiral ran a very successful, man-rated nuclear propulsion program. I won't take the time to read this to you here, but urge you to take a look at this excerpt and remember that times have not changed significantly in the 40 years since this was written. This excerpt can be summarized by saying that "paper reactors always run better than real reactors".

PAPER REACTORS, REAL REACTORS

Admiral Hyman Rickover,
The Journal of Reactor Science and Engineering,
June 1953

An academic reactor or reactor plant almost always has the following basic characteristics: *1) It is simple. 2) It is small. 3) It is cheap. 4) It is light. 5) It can be built very quickly. 6) It is very flexible in purpose. 7) Very little development is required. It will use mostly off-the-shelf components. 8) The reactor is in the study phase.* It is not being built now.

On the other hand, a practical reactor plant can be distinguished by the following characteristics: *1) It is being built now. 2) It is behind schedule. 3) It is requiring an immense amount of development on apparently trivial items. Corrosion, in particular, is a problem. 4) It is very expensive. 5) It takes a long time to build because of the engineering development problems. 6) It is large. 7) It is heavy. 8) It is complicated.*

The tools of the academic reactor-designer are a piece of paper and a pencil with an eraser. If a mistake is made, it can always be erased and changed. If the practical-reactor designer errs, he wears the mistake around his neck; it cannot be erased. Everyone can see it.

The academic-reactor designer is a dilettante. He has not had to assume any real responsibility in connection with his projects. He is free to luxuriate in elegant ideas, the practical shortcomings of which can be relegated to the category of "mere technical details." The practical-reactor designer must live with these same technical details. Although recalcitrant and awkward, they must be solved and cannot be put off until tommorrow. Their solution requires manpower, time and money.

Unfortunately for those who must make far-reaching decisions without the benefit of an intimate knowledge of reactor technology, and unfortunately for the interested public, it is much easier to get the academic side of an issue than the practical side. For a large part those involved with the academic reactors have more inclination and time to present their ideas in reports and orally to those who will listen. Since they are innocently unaware of the real but hidden difficulties of their plans, they speak with great facility and confidence. Those involved with practical reactors, humbled by their experiences, speak less and worry more.

Yet it is incumbent on those in high places to make wise decisions and it is reasonable and important that the public be correctly informed. It is consequently incumbent on all of us to state the facts as forthrightly as possible.

N93-26955

Rocketdyne/Westinghouse Nuclear Thermal Rocket Engine Modeling

October 22, 1992
Jim Glass

Systems Approach Needed for NTR Design Optimization

Nuclear rocket engine systems, like chemical engines, require a systems-oriented approach to the selection and refinement of an optimum design. This approach stresses that all subsystems and components must be optimized or designed together; the goal is to achieve the best possible overall system design.

A well-anchored and validated steady-state design model is required, one which treats all important characteristics and phenomenology of the system elements, together with technology limits and constraints. The program must provide sufficient design detail to fully characterize the engine system, and to provide confidence in the design. The detailed system design file is also passed to the Steady-State Off-Design and Transient models, where it forms the basis of the hardware description needed to initialize the off-design or transient simulation.

Rocketdyne's Steady-State Design Optimization model is based on known and proven methodologies such as those shown. It performs a "rubber engine" conceptual design, and uses scaling only when appropriate. Physical or first-principles component models are preferred. The code performs constrained optimization, with both implicit and explicit constraints. These constraints reflect technology level, risk, reliability, and other limits on the design, and help to ensure that a practical and achievable design is obtained.

Systems Approach Needed for NTR Design Optimization

- All elements of engine system optimized together
  - Reactor
  - Turbomachinery
  - Feed System
  - Controls
  - Nozzle and throat
  - Cooling and heat exchange
- Design model based on anchored and proven methodologies
  - JANNAF Performance Prediction
  - NBS (NIST) Thermodynamic Properties
  - CPIA 246 Expansion Process Losses
- "Rubber Engine" conceptual design versus scaling approach
  - First principles analysis where appropriate
  - Provides design detail
  - Reflects technology level and design constraints
    - Technology year
    - Risk/reliability/cost

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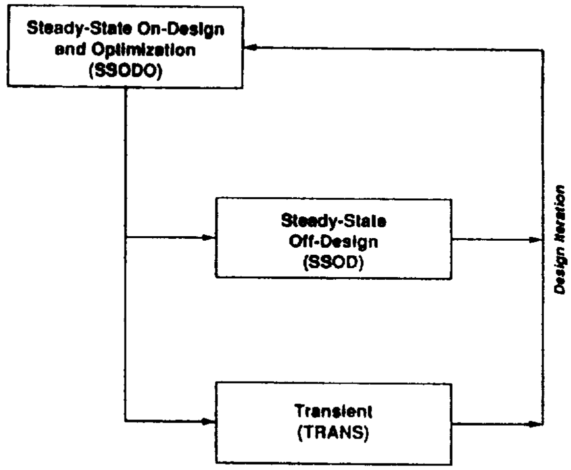
Rocketdyne Division

Generic NTR Engine Power Balance Codes

Rocketdyne's approach to NTR engine system modeling utilizes three separate codes, which are linked by a common hardware description file. The Steady-State Design Optimization program develops an optimized system design, based on user inputs, a schematic description file, and optimization constraints. The output of the design program is a hardware definition file which can be passed to the Steady-State Off-Design code or to the Transient code.

Both of the latter codes (SSDO and TRANS) are off-design models in the sense that they seek to analyze the behavior and response of fixed hardware to changes in control settings, component characteristics, or start/shutdown. The Design Optimization model is an "on-design" model, or "rubber engine" model, which seeks to find the best design operating point to meet user requirements and technology constraints.

Generic NTR Engine Power Balance Codes



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Rocketdyne Nuclear Thermal System Code Heritage/Pedigree

The Rocketdyne NTR system models have been under continuous development at Rocketdyne since 1975, under both company and government funding. These codes form the basis of the company's engine preliminary design capability.

These codes or variants have been successfully utilized to design a variety of flight-type engine systems, including the RS-44, XLR-132, STME, STBE, RSX, and IME engines.

In addition, the codes have been validated by generating "designs" for current and past hardware, including F-1, J-2, SSME, and Russian engine designs.

Rocketdyne Nuclear Thermal System Code

Heritage/Pedigree

- Elements of engine system model under continuous development since 1975.
- Used as preliminary design and optimization tool at Rocketdyne.
- Used to design:

ASE	20,000 lb thrust O2/H2 space engine
RS-44	15,000 lb thrust O2/H2 space engine
XLR-132	3,750 lb thrust NTO/MMH space engine
STME	650,000 lb thrust O2/H2 space transportation engine
STBE	750,000 lb thrust O2/hydrocarbon booster engine
RSX	237,000 lb thrust O2/RP-1 booster engine
IME	30,000 lb thrust O2/H2 space engine

- Validated against current and past hardware:

F-1	Russian RD-170 booster engine
J-2	Russian RD-0120 engine
SSME	Russian RD-701 tripropellant engine

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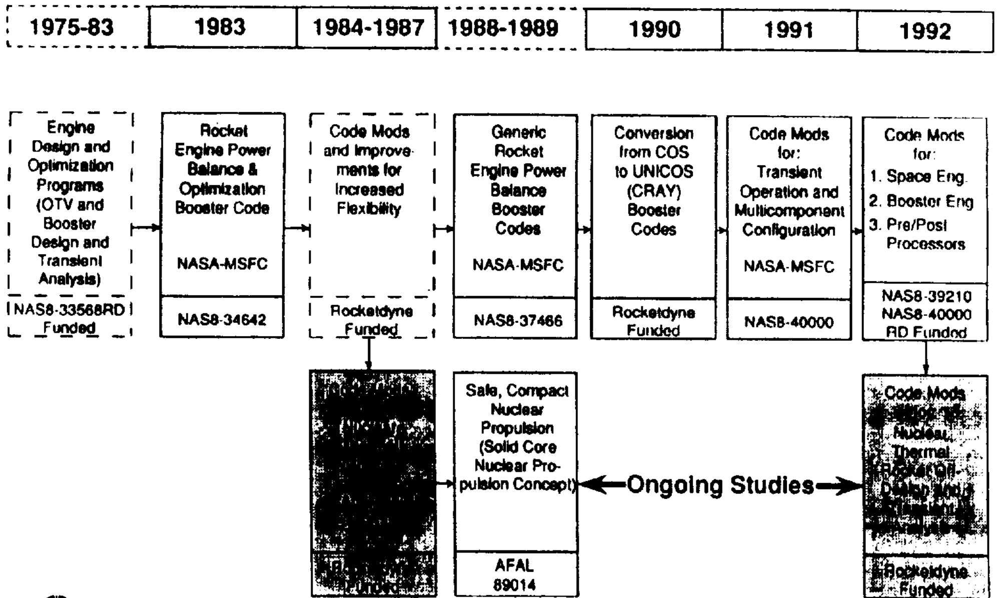
Code History

This chart illustrates the continuous, ongoing effort on the Nuclear Thermal System Model and its precursors. Rocketdyne internal funding has supplemented a series of NASA contracts in development of a robust, validated and flexible engine system modeling code. Recent work (since 1987) has focused on modifications to the code to enable modeling of Nuclear Thermal Rocket systems. A recent Air Force study, the Safe Compact Nuclear Propulsion study, utilized results of the code. Ongoing Rocketdyne in-house studies have also made extensive use of the code results.

Code History



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NTR System Model--Code Features

Key features of Rocketdyne's NTR system model include variable schematic analysis, high-fidelity propellant properties, prismatic core geometry, accurate turbomachinery,heat-transfer, and performance estimation algorithms, and a nonlinear, constrained optimization routine.

The variable schematic capability uses a data-driven approach, in which all design modules and algorithms are contained within a single program, and appropriate modules are called under control of an executive which traverses the input schematic network. This is different from a variable-code approach, in which a new model is generated and re-compiled for each new system configuration. The data-driven approach maximizes code flexibility, does not entail difficulties in traceability of code results, and enables higher-speed modeling (no compile step).

Well-anchored turbomachinery and heat-transfer calculations are included, which improve model accuracy and enhance confidence in the resulting system design.

Use of NBS/NIST and JANNAF propellant and performance methods also increases code fidelity.

The non linear, constrained optimization routine enables comparison of competing candidate system configurations on a common basis; i.e., "best possible" design points for all candidates can be compared.

NTR System Model
Code Features

- **Variable Schematic**
  - Code flexibility
  - Ease of modeling new concepts
  - Fixed code/variable data
- **NBS/NIST Propellant Properties**
  - Accurate energy balance
  - Accurate flow schedule
  - Hydrogen, methane, CO_2, or ammonia propellants
- **Prismatic reactor core geometry**
  - Particle-bed and wire-core may be added
- **NTR-Unique components**
  - Cooled structure
  - Reflector/moderator
  - Nozzle heat load accounting
- **Rocketdyne Turbomachinery Design Routines**
  - Historically-anchored T/M performance and envelope
  - Centrifugal or axial pumps
- **Rocketdyne Heat Transfer Correlations**
  - Accurate prediction of jacket heat loads and ΔP
- **JANNAF/CPIA Performance Estimation**
  - Accurate and rapid delivered performance prediction
  - Accounts for all loss mechanisms (B/L, Kinetics, Divergence)
- **Nonlinear, Constrained Optimization Capability**
  - Minimize or maximize any system variable

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Software Capabilities

The present code is capable of optimizing the system design for Nuclear Thermal Rocket engines in the 10,000 to 250,000 pound thrust range. Key features of the code include the input-controlled variable schematic analyis capability, detailed NBS (NIST) hydrogen properties, a graphic preprocessor (which eases user interaction with the model), and multiple component capability. The multiple component feature enables modeling of engine systems with multiple redundant turbopumps, and design of systems capable of pump-out operation.

Transfer of engine system design information from the design module to the off-design or transient code is possible.

Future (planned) enhancements to the existing models includes incorporation of additional propellants such as ammonia, carbon dioxide, and methane. These propellants have been mentioned as possible alternate propellants, especially for in-situ propellant-based missions. A graphic post-processor is being prepared, which will present the code output in graphical form for ease of interpretation.

Work on the Steady-State Off-Design and Transient codes to incorporate higher-fidelity nuclear elements is planned. The off-design models will also be extended to enable specification of as-measured hardware characteristics (such as pump H-Q maps, turbine maps, etc.).

Software Capabilities

- Current

  Optimize and size engines of 10K to 250K thrust

  Input-controlled variable-schematic capability

  Hydrogen propellant

  Graphic preprocessor

  Multiple component capability: 40 components

  Automatic configuration transfer

  Steady-state design optimization

- Future

  Other propellants: Ammonia, CO_2, CH_4

  Graphic postprocessor

  Steady-state off-design and transient models

  Off-design models will accept actual hardware characteristics

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Steady State Model

The Steady-State Design Optimization model accepts user inputs consisting of general user inputs (thrust, chamber pressure, area ratio, etc.), a schematic definition file, optimization specifications and constraints, and reads data from a knowledge base which provides propellant properties, theoretical performance tables, and other information on components and subsystems.

The major elements of the Steady-State model include a schematic analyzer, component models, optimizer, thermodynamic state computations, and performance calculations.

The **Schematic Analyzer** uses the user-input schematic definition file to develop the interconnections between the engine system elements. The schematic is described in terms of a grid or array of nodes and the connections between the nodes. The schematic analysis routine controls the flow of the program by repeatedly traversing the component/node network until convergence has been obtained.

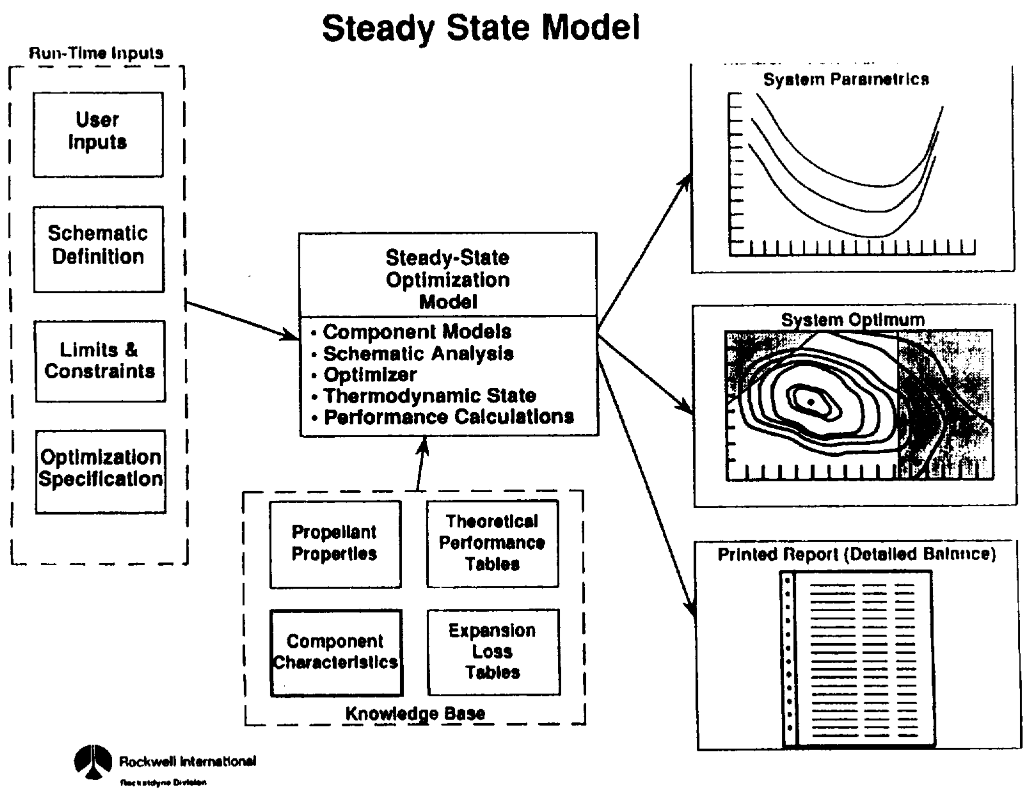
Component models provide algorithms describing the operation, design and sizing of the engine system components, such as turbopumps, heat-exchange elements, reactor, structural jacket, and nozzle.

The **Optimizer** varies selected independent variables (such as pump speed, turbine pressure ratio, or chamber pressure) in order to minimize or maximize a selected object function subject to a set of constraints.

Thermodynamic state computations are performed under control of the schematic analyzer to track the detailed thermodynamic state of the propellant at each engine system station.

Performance calculations are performed in order to develop theoretical and delivered engine and thrust-chamber preformance and associated loss terms based on nozzle geometry, operating temperature, and inlet propellant state.

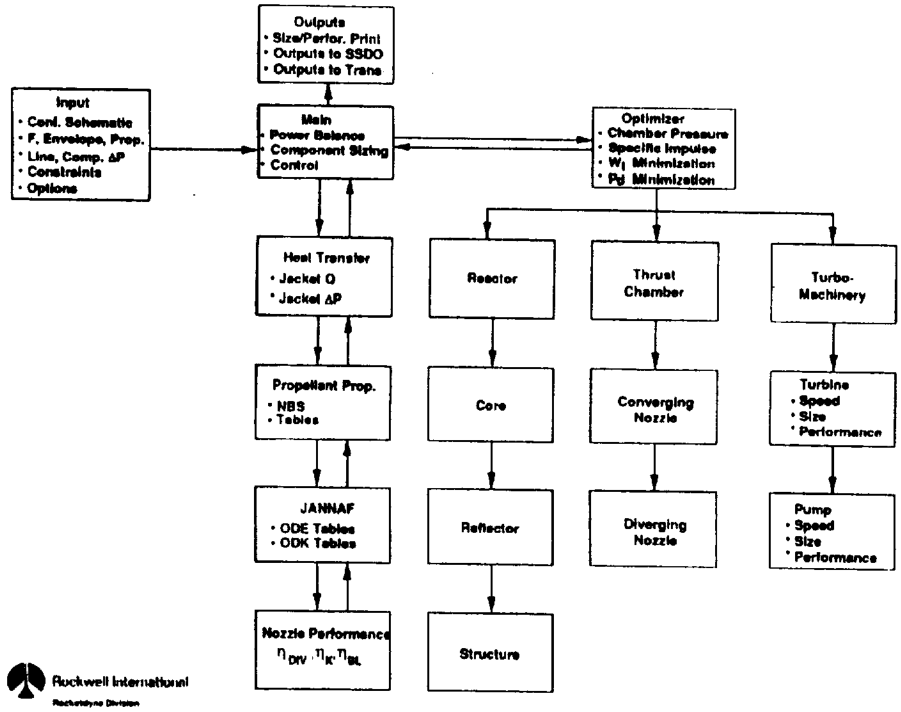
In addition to providing an optimum design point, the model can be operated in a parametric mode to enable generation of parametric curves which describe families of similar system designs. Printed reports and a hardware definition file are also produced.



NTR Engine Optimizer Code -- Logic

This chart illustrates the block-level logic of the Steady-State NTR design code. The figure shows that the main control routine is responsible for driving the schematic analysis and performing component sizing and performance calculations. The optimizer routine is used to maximize or minimize a selected object function by selecting a set of independent variables which control one or more aspects of component or subsystem design.

NTR Engine Optimizer Code - Logic



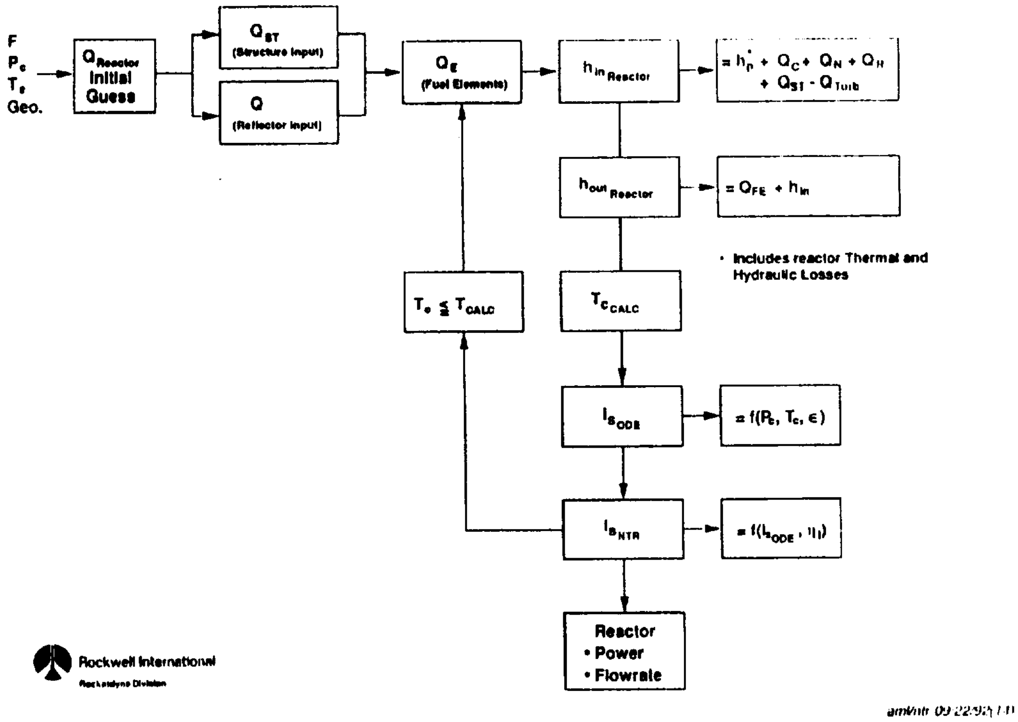
Reactor Power Calculation Logic

The Steady-State code presently contains a lumped reactor model, which essentially treats the reactor as a heat source, but does not perform detailed reactor element sizing. An initial estimate of reactor power (heat) is derived from inputs of thrust, chamber pressure, and desired gas exit temperature. Separate estimates of structure and reflector heat loads are developed based on correlations of detailed heat-transfer analysis.

An initial estimate of the heat load from the reactor is made, from which the reactor exit enthalpy can be computed. The reactor outlet temperature is then computed from the total reactor heat and inlet conditions, and this temperature is compared with the desired exit temperature. If necessary, the reactor heat is readjusted until the exit temperature converges. Once the exit temperature is known, the theoretical specific impulse and C-star can be calculated.

The reactor flowrate is then known, as is net reactor power level.

Reactor Power Calculation Logic



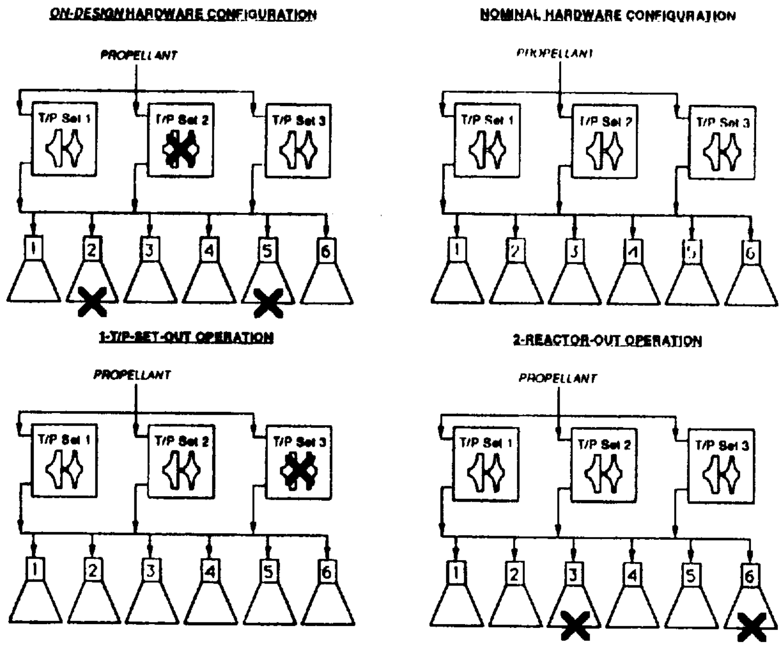
Sample Multi-Component Configuration

Redundant design configuration of NTR propulsion systems is important due to the potential impact of an engine failure on the mission and on the survival of the crew. Design of redundant turbopump sets and/or multiple reactor/thrust chamber sets is attractive because it enables robust propulsion systems which can tolerate a single failure or even multiple failures and continue to operate. Mission success and crew survival can be greatly enhanced by careful application of redundant design philosophy.

The NTR design code is capable of modeling various system configurations which incorporate multiple turbopump and reactor/thrust chamber sets. One possible type is the incorporation of fully-redundant powerhead and reactor/thrust chamber assemblies, which are intended to remain non-operating unless/until one of the operating sets fails. The failed set is then shut down and the "spare" set takes its place. Another possibility is to design multiple powerhead/thrust chambers which are designed to operate in parallel, with no spares. Failure of a turbopump or reactor/thrust chamber would result in shutdown of the entire subsystem with continued operation of the remaining powerheads and reactor/thrust chambers. A third option involves design of multiple turbopump sets, a subset of which (say two out of three) are capable of operating all of the multiple thrust chambers at their design point. A failure of a pump set would still allow on-design operation with the remaining turbomachinery. However, prior to failure, all turbopump sets would operate off-design (throttled or de-rated). Finally, the system can be designed to enable failure of multiple thrust chambers, with the multiple turbopump sets continuing to operate to supply the remaining thrust chamber sets.

Loss of reactors has additional implications: A reactor will continue to produce power from decay heat and from neutron leakage (from adjoining reactors in the engine cluster). Careful consideration of this continued heating must be made from a mission-safety viewpoint. It may be necessary to jettison a failed reactor if the continued heating cannot be adequately controlled and/or suppressed.

Sample Multi-Component Configuration



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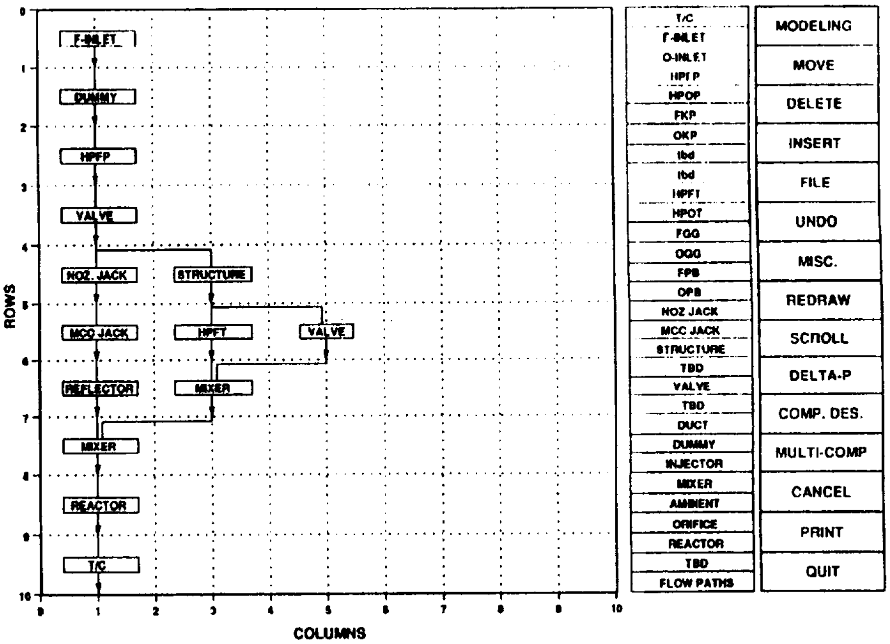
Rocketdyne Division

Configuration Preprocessor

This chart illustrates the graphical pre-processor. The preprocessor presents a grid on the left side of the screen, which is employed by the user to draw the engine components and define their interactions. A main menu (right side of screen) selects modes and operations, and a sub-menu (to left of main menu) presents component choices.

In use, the user selects a component from the sub-menu and then indicates the inlet and exit node locations for the selected component on the schematic grid. By successively adding components, the preprocessor builds an internal representation of the schematic connections, pressure drops, and component characteristics of the desired engine system configuration. When complete, the schematic description and other information is written to an output file, which can then be read by the Steady-State, Off-Design, or Transient codes.

Configuration Preprocessor



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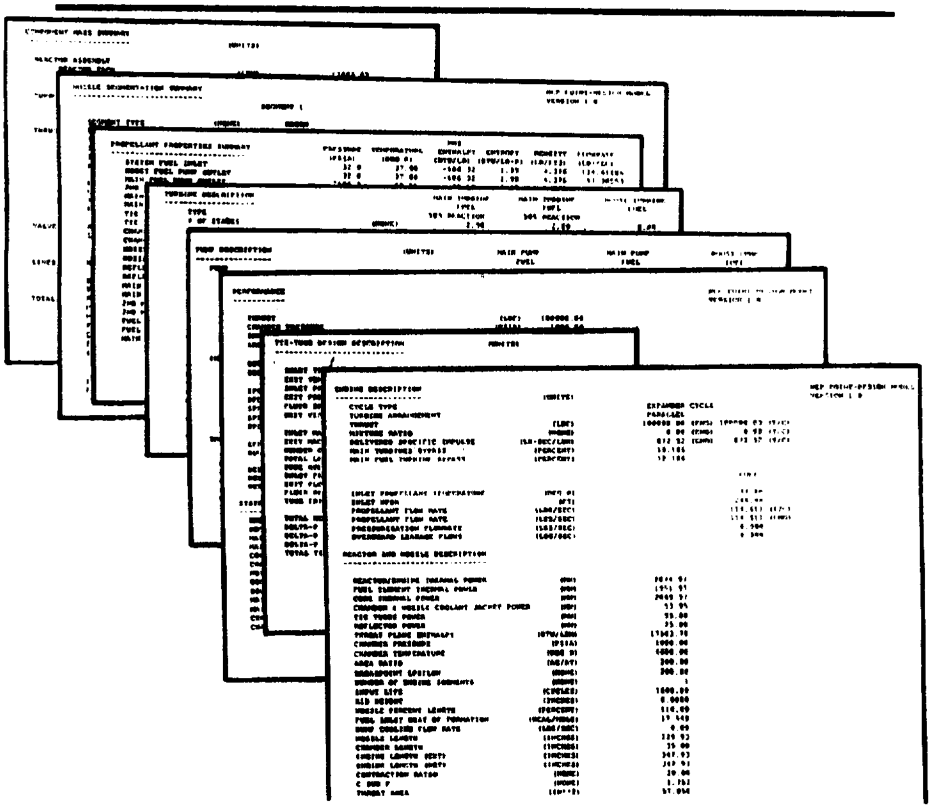
NTR Design Code Output

A typical printout of the Steady-State NTR design code is presented in this chart. As can be seen, the level of design detail available is high. Summary printouts of reactor and nozzle design characteristics, tie-tubes (cooled structure), performance, and turbomachinery design variables are included. A detailed listing of all propellant state properties at each system station is printed, and a system mass estimate is also provided.

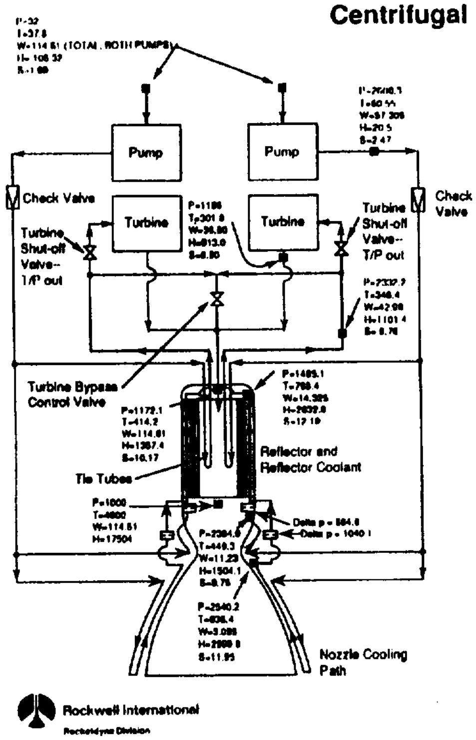
NTR Design Code Output



100K NTR, Expander Cycle, Dual T/P--Centrifugal Pump

This chart illustrates a system design balance performed with the NTR Steady-State Design code. When the graphic post-processor is available, an annotated schematic diagram similar to that shown will be automatically generated by the post-processor.

100K NTR, Expander Cycle, Dual T/P*
Centrifugal Pump



DESIGN VALUES:

PUMP FLOWRATE (TOTAL)	114.6 LB/SEC
PUMP DISCHARGE PRESSURE	2,606 PSIA
NUMBER OF PUMP STAGES	2
PUMP EFFICIENCY	79.71 %
TURBOPUMP RPM	56,500 RPM
TURBOPUMP POWER (EACH)	10,282 HP
TURBINE INLET TEMP	346.4 R
NUMBER OF TURBINE STAGES	2
TURBINE EFFICIENCY	81.71 %
TURBINE PRESSURE RATIO	1.95
TURBINE FLOW RATE (EACH)	38.60 LB/SEC
REACTOR/ENGINE THERMAL POWER	2,075 MW
FUEL ELEMENT TRANSFERRED POWER	1,952 MW
CORE THERMAL POWER (FUEL ELEMENT+TIE TUBE)	2,050 MW
ENGINE THRUST	100,000 LBF
NOZZLE CHAMBER TEMPERATURE	4,600 R
CHAMBER PRESSURE (NOZZLE STAGNATION)	1,000 PSIA
NOZZLE EXPANSION AREA RATIO	200:1
NOZZLE PERCENT LENGTH	110%
VACUUM SPECIFIC IMPULSE (DELIVERED)	872.5 SEC

Heat loads are as follows:	Nozzle-con (total):	35.15 MW
	Nozzle-div (total):	18.80 MW
	Reflector (total):	25.00 MW
	Tie-Tubes (total):	98.00 MW

P = PSIA
T = DEG R
W = LB/S
H = BTU/LB
S = BTU/LB-R

*Note: Flows indicated are for one-half of system.

Future Activities and Capabilities

Future capabilities to the NTR design software are listed in this chart. These enhancements are being added in a series of NASA- and company-funded efforts. The space engine thrust chamber and main pump subroutines are being upgraded to extend the thrust range over which they are applicable. Low pressure boost pump design capability for zero-NPSH operation designs is being added. These two efforts are being funded by MSFC for SEI application. However, the code improvements will also be directly applicable to NTR modeling.

Company-funded efforts will complete the optimization of reactor power, envelope, and weight; the full implementation of the pre- and post- processors, and the full implementation of the transient analysis reactor model.

Future Activities and Capabilities

Activity	Funding	Planned Completion
Low pressure (boost) pump simulation	NAS8-40000	November 1992
Reactor power, envelope, weight optimization model	Rocketdyne	December 1992
Upgrade space engine design optimization	NAS8-39210	January 1993
Enhanced pre/post processors	Rocketdyne	March 1993
Transient analysis model (feed system, thruster, and reactor)	Rocketdyne	April 1993

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Generic NTR Code at Rocketdyne

The Rocketdyne Generic NTR code provides design versatility for all aspects of NTR system analysis (design, off-design, and transient), ease of use and user-friendly features through variable schematic features and pre- and post-processors, and system versatility because it can be operated on a variety of platforms, including VAX, Cray, Alliant, and Sun workstations.

As PC hardware continues to improve, it will soon be possible to port these codes to the PC platform and to operate them with acceptable speed and accuracy.

Generic NTR Code at Rocketdyne

Design Versatility:	**Design Point Optimization**
	Off-Design
	Transients
User Versatility:	**Variable Schematic**
	Pre/Post-Processors
	Auto Configuration Transfer
System Versatility:	**VAX, CRAY, ALLIANT, Sun Workstations**
	Future: Improved PC platforms

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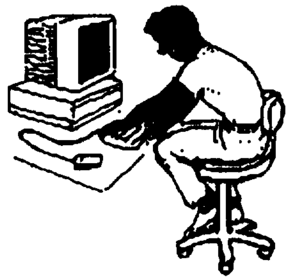
Rocketdyne NTR Model--Summary

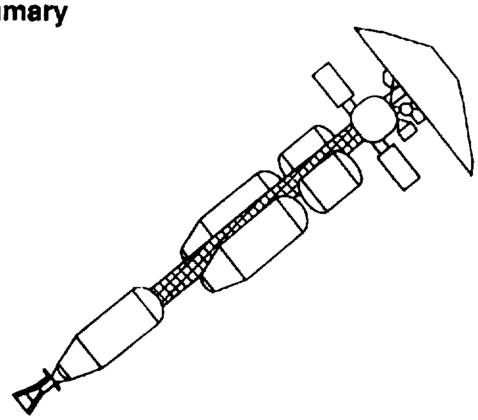
This chart summarizes the essential message of this briefing: Rocketdyne has developed an NTR engine system modeling capability which emphasizes Utility and Fidelity.

Utility is based on the codes' flexibility and versatility, user-friendly features, ease of modification, and documentation.

Fidelity is based on use of first-principles methods, extensive validation against past flight designs and existing hardware, and accurate component and performance algorithms. The codes are adequate for use in preliminary design, screening, and trade studies. With further refinement and deepening, the codes will evolve into full "point-design" models.

Rocketdyne NTR Model
Summary





Utility

- Versatile
- User Friendly
- Easy Modification
- Fully Documented

Fidelity

- First-Principles Analysis Methods
- Flight Engine Validated
- Accurate Component & Performance Algorithms
- Preliminary Design-Level Support

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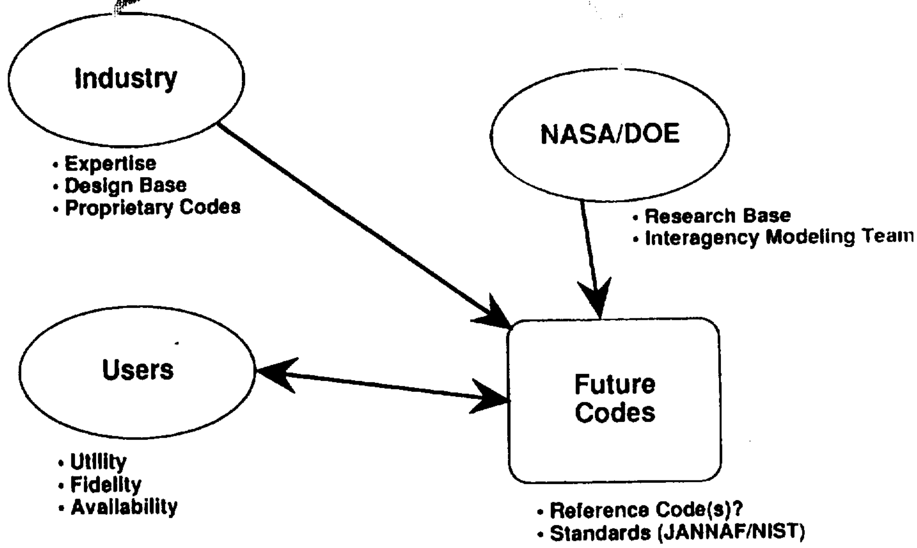
Nuclear Thermal Rocket Modeling Directions

This chart illustrates Rocketdyne's vision of one possible direction in which NTR modeling activities might proceed. We believe that a collaboration among NASA/DOE, end-users, and industry will bring major benefits to the codes and models which are ultimately developed. Industry brings capabilities which compliment and enhance those already in place at NASA centers and national laboratories. Users concerns must be addressed to ensure that the codes developed are usable and meet actual needs. NASA/DOE leadership and direction are critical to successful code development.

Nuclear Thermal Rocket Modeling Directions



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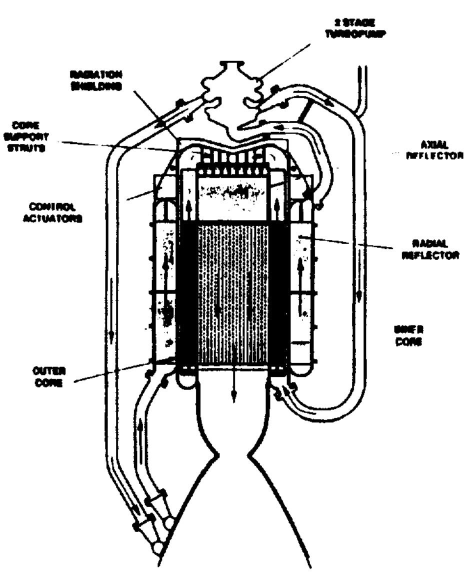
N93-26956

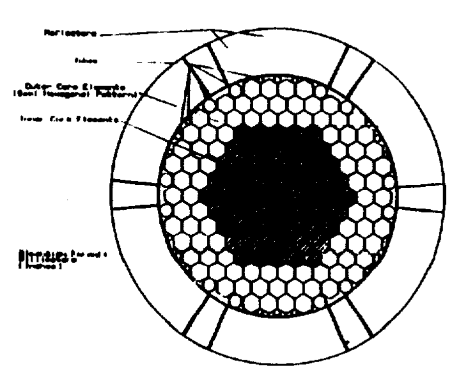
COMPUTATIONAL MODELING OF NUCLEAR THERMAL ROCKETS

Steven D. Peery
Pratt & Whitney
22 October 1992

XNR2000 NTR BASELINE DESIGN

Dual-Pass Cermet Fueled Reactor





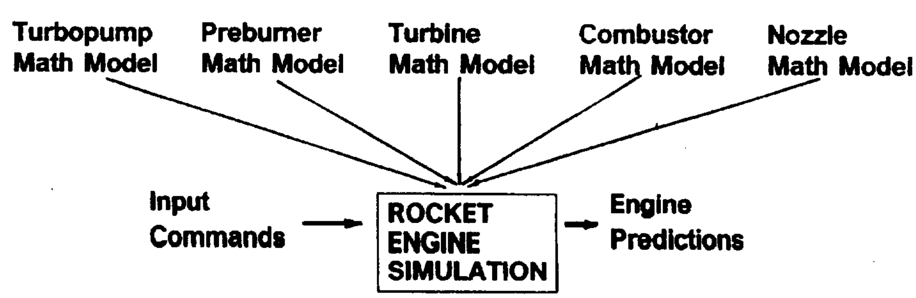
ROCKET ENGINE TRANSIENT SIMULATION (ROCETS) SYSTEM

Developed Under MSFC Contract NAS8-36994

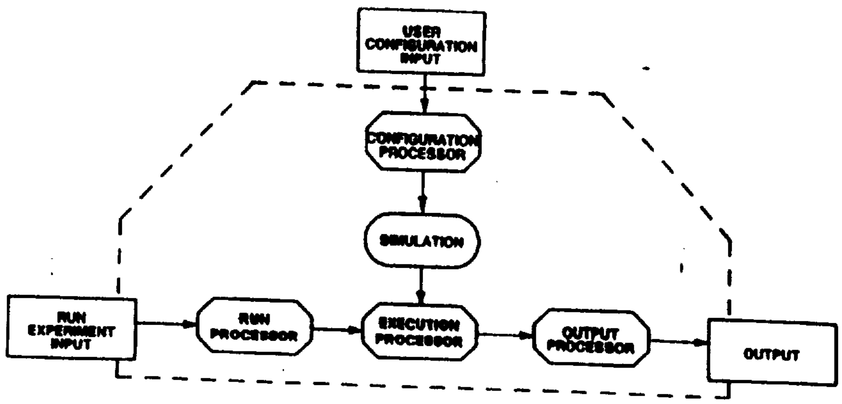
- System Developed To Model Steady-State and Transient Performance of a Wide Variety of Rocket Engine Cycles
- System Has Been Expanded for Nuclear Thermal Rocket (NTR) Concept Studies

ROCETS PERFORMANCE SIMULATIONS COMPOSED OF INTEGRATED COMPONENT MODELS

- Thermal-Fluid Component Models
- Component-by-Component
- Transient and Steady State



ROCETS SYSTEM ARCHITECTURE
SIGNIFICANT FEATURES



ROCETS ENGINEERING NTR MODULES

Component Performance Models

- Reactor (Core, Reflector, Shielding)
- Turbopump
- Turbine
- Plumbing & Valves
- Mixers
- Chamber & Nozzle Cooling
- Nozzle Performance
- Weight

Reactor (Core, Reflector, Shielding):

- Neutronics
  - Diffusion
  - Transport
  - MCNP
- Thermal Fluid CFD
- Properties

ROCETS SYSTEM EASILY ADAPTS FORTRAN ENGINEERING MODULES

CALL LIST

c INTERFACE DATA ⟸ UNIQUE TO ROCETS

c INPUTS/OUTPUTS & DEFINITIONS
c ENGINEERING DESCRIPTION
c TEST CASE & REF TO QTP RESULTS
c SUB-MODULES NEEDED
c HISTORY

FORTRAN
CODE

<u>UNIQUE TO ROCETS</u>

- o Converts Fortran Parameter Names to ROCETS Nomenclature
- o Outside Code Becomes Operational in ROCETS by Adding This Interface
- o "Comment Cards" So Module Can Independently Operate (Decoupled From ROCETS if Required)

ROCETS NTR REACTOR MODULE

Fluid Thermodynamic Model

Reactor Module Input

- Propellant inlet conditions
- Propellant flow rate
- Desired exit temperature
- Calculated radial and axial power profiles
- Fuel element geometry

Reactor Module Output

- Required reactor power
- Propellant thermophysical properties throughout reactor
- Reactor temperatures

ROCETS NTR TURBOMACHINERY MODULE

Hardware Modeling and Clean-Sheet Design Capability

Turbopump Module

- Sets speed based on Nss
- Determines power and size for requested headrise
- Calculates efficiency and pump design parameters

Turbine Module

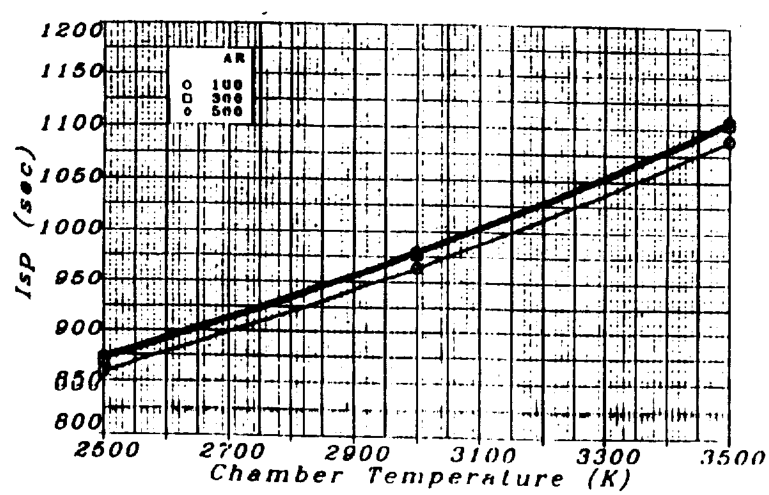
- Determines size and exit conditions for required power
- Limits wheel speed to stay within stress limits
- Calculates efficiency and turbine design parameters

ROCETS NTR NOZZLE PERFORMANCE MODULE

2-DK with Finite Rate Chemistry and Boundary Layer Analysis

Determines Delivered Nozzle Performace and Contours for Both High and Low Pressure Concepts

5 - 1500 psia Pc
2500 - 3500 K Tc
25 - 500 AR



XNR2000 ENGINE PERFORMANCE

Thrust = 25,000 lbf (Baseline)
T/W = 5.3
Isp = 900.0 sec

PROPELLANT FLOW ENGINE STATION CONDITIONS

Station Location	Pressure (psia)	Temperature (Deg K)	Flow (lbm/s)	Enthalpy (Btu/lbm)	Density (lbm/ft**3)
Engine Inlet	26.7	20.6	14.0	-108.0	4.38
Pump Inlet	25.7	20.6	14.0	-108.0	4.38
Pump Exit	2179.3	34.7	14.0	13.0	4.56
Nozzle Coolant Inlet	2157.6	34.8	8.4	13.0	4.55
Reflector Coolant Inlet	1932.6	103.1	28.1	440.9	1.77
Turbine Inlet	1901.6	226.9	11.8	1343.7	0.80
Turbine Exit	1218.2	207.2	11.8	1199.9	0.58
Outer Core Inlet	1108.9	210.4	27.8	1221.6	0.52
Inner Core Inlet	956.3	1659.4	27.8	8865.0	0.06
Chamber	765.9	2668.7	27.8	18188.3	0.03

REACTOR CHARACTERISTICS

Two-Pass Design		
Inner Core Diameter	11.5	in
Outer Core Diameter	18.1	in
Reflector Diameter	32.2	in
Pressure Drop	344.1	psia
Max. RX Fuel Temp.	2880.0	K
Outer Core Fuel Mt'l	Mo-UO2,90	
Inner Core Fuel Mt'l	W-UO2,61	
Power Density	9.41	MW/l
Total Power	510.4	MW

NOZZLE CHARACTERISTICS

Nozzle Area Ratio	200.	
Throat Area	18.8	in**2
Exit Dia.	5.8	ft
Nozzle C*	16443	ft/s
Nozzle Length	10.6	ft
Total S.A.	22524	in**2
Regen. Construction	Cu Tubes	
Rad. Construction	Cb Sheet	

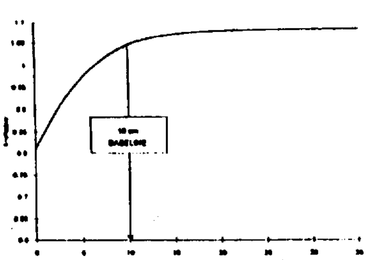
PUMP CHARACTERISTICS

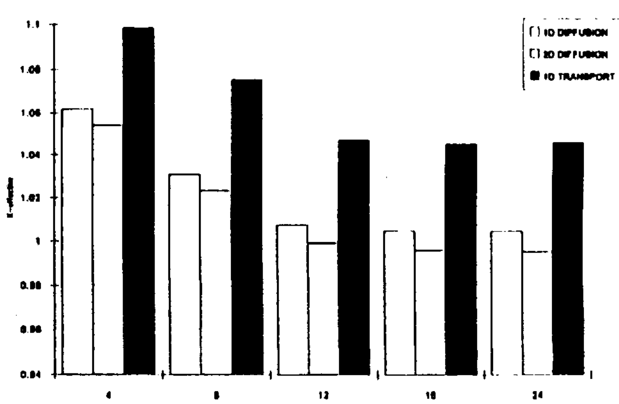
Overall Efficiency	73.2	%
Head Rise	69,018	ft
NPSH Avail.	302.9	ft
Speed	71,323	RPM
Power	2403.2	HP
Vol. Flow Rate	1379	gpm
Stg I Flow Coeff.	0.114	-
Stg II Flow Coeff.	0.113	-
Stg I Head Coeff.	0.521	-
Stg II Head Coeff.	0.521	-
Utip 1	1460.	ft/s
Utip 2	1460.	ft/s

TURBINE CHARACTERISTICS

Inlet Temperature	226.9	K
Inlet Pressure	1901.6	psia
Mass Flow	11.8	lbm/s
Overall Efficiency	85.4	%
Speed	71,233	RPM
Pressure Ratio	1.56	-
Inlet Flow Parameter	0.125	-
Overall Velocity Ratio	0.54	-
DH Actual	143.8	Btu/lb
AN**2(E-08)	193.	
Mean Dia.	4.66	in

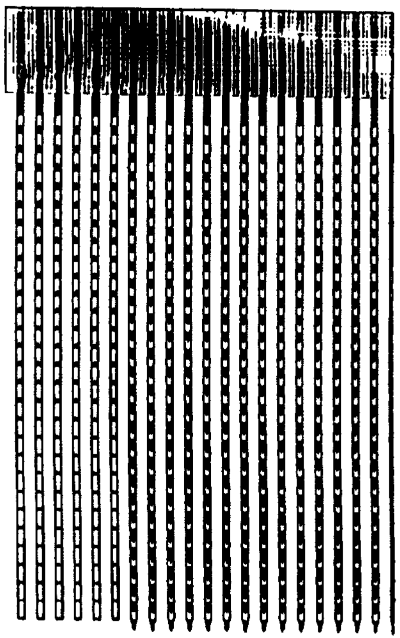
DETAILED REACTOR ANALYSIS CONDUCTED OUTSIDE OF SYSTEM PERFORMANCE EVALUATION

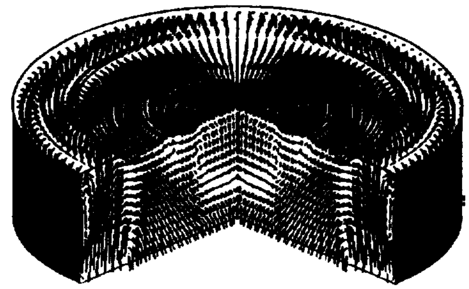




TFS PREDICTED FLOW DITRIBUTION

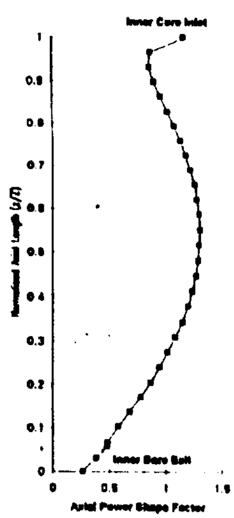
CFD Benchmarks Reactor Engineering Module

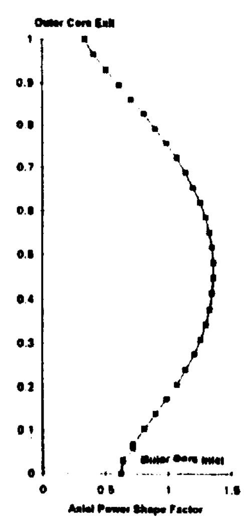


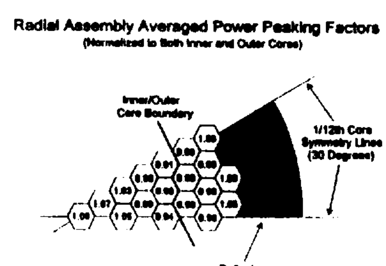


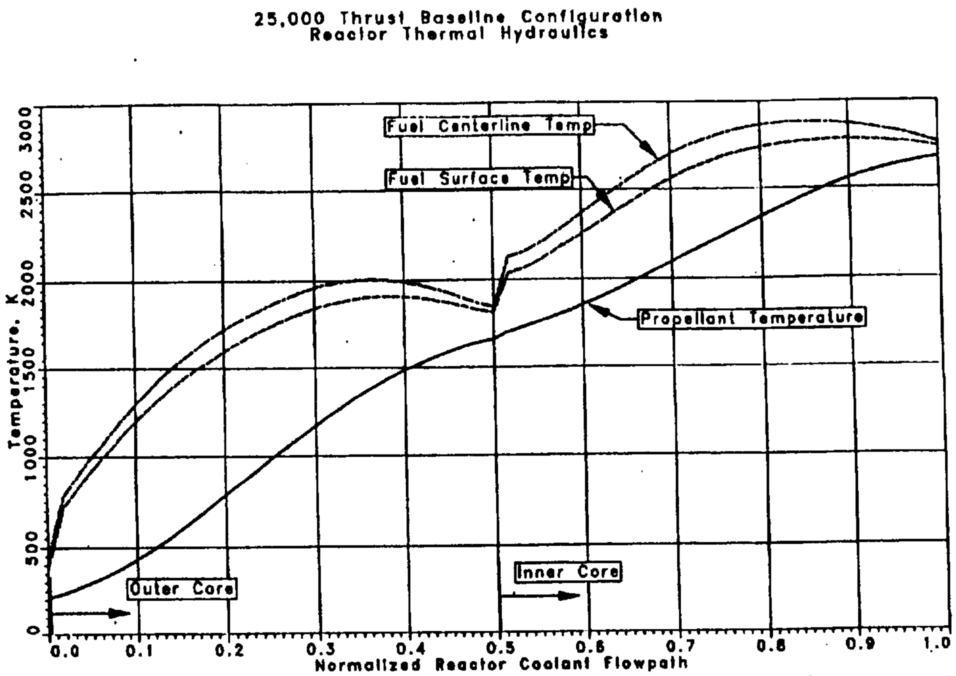
PREDICTED REACTOR POWER PROFILES

Input for Reactor Engineering Module



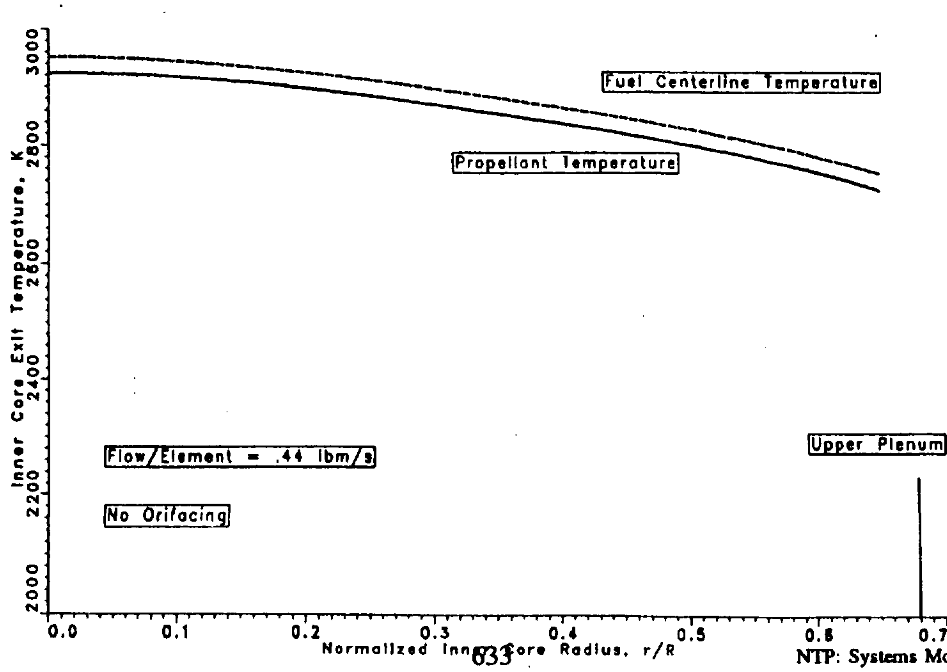


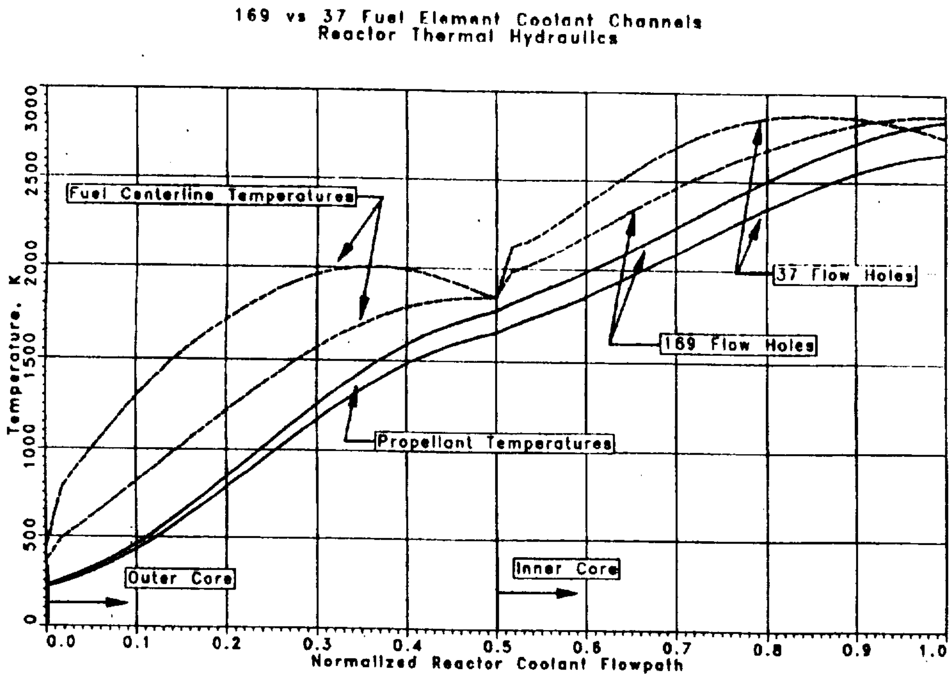


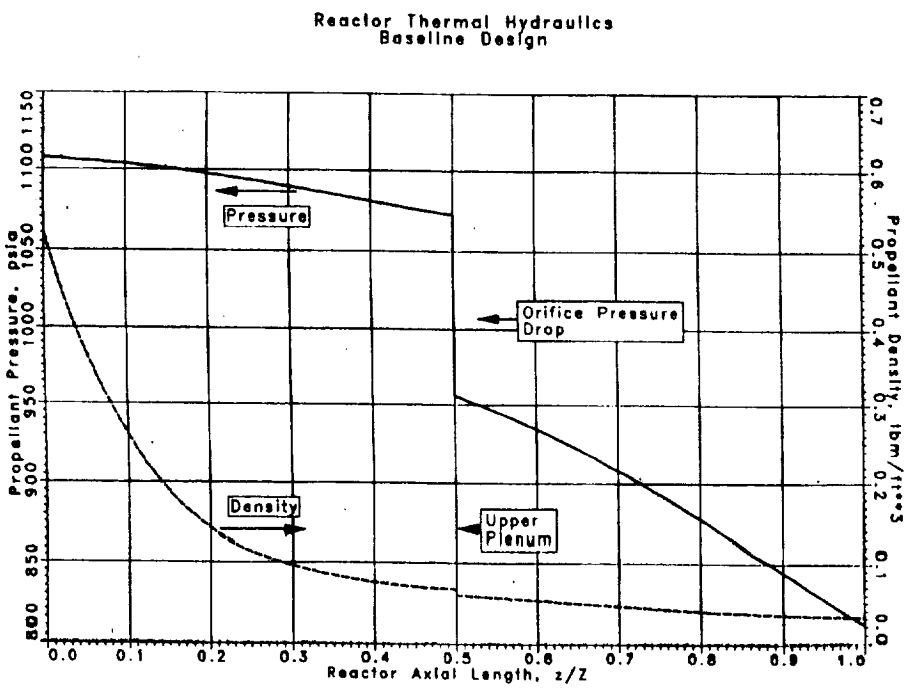


XNR2000 INNER CORE EXIT TEMPERATURE DITRIBUTION

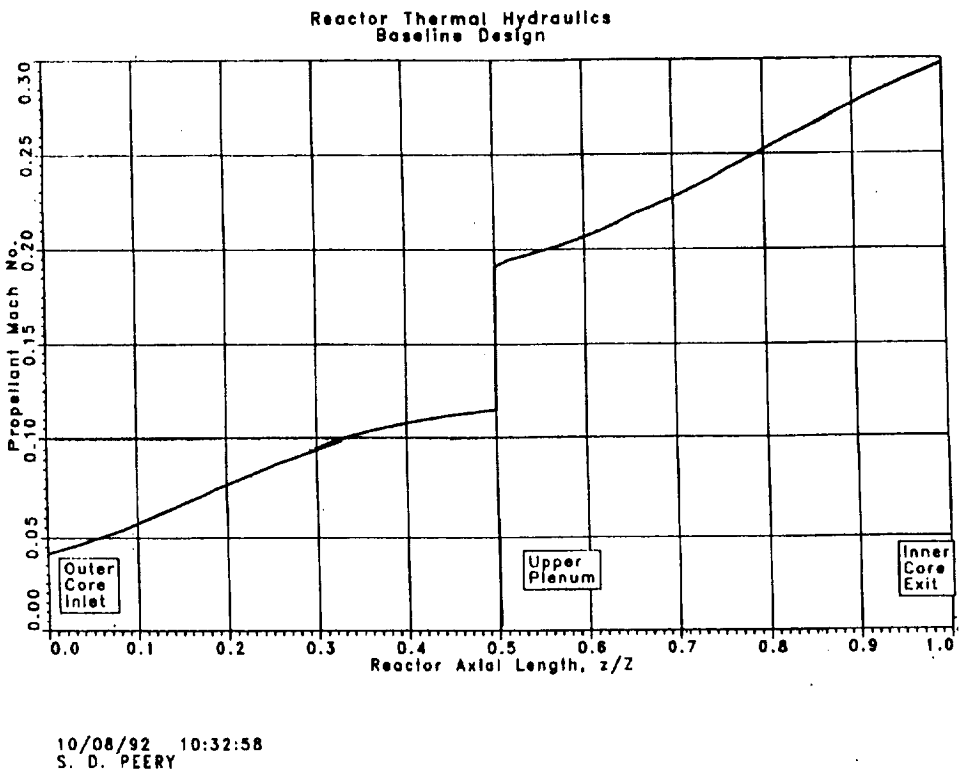
Accounting for Radial Power Distribution

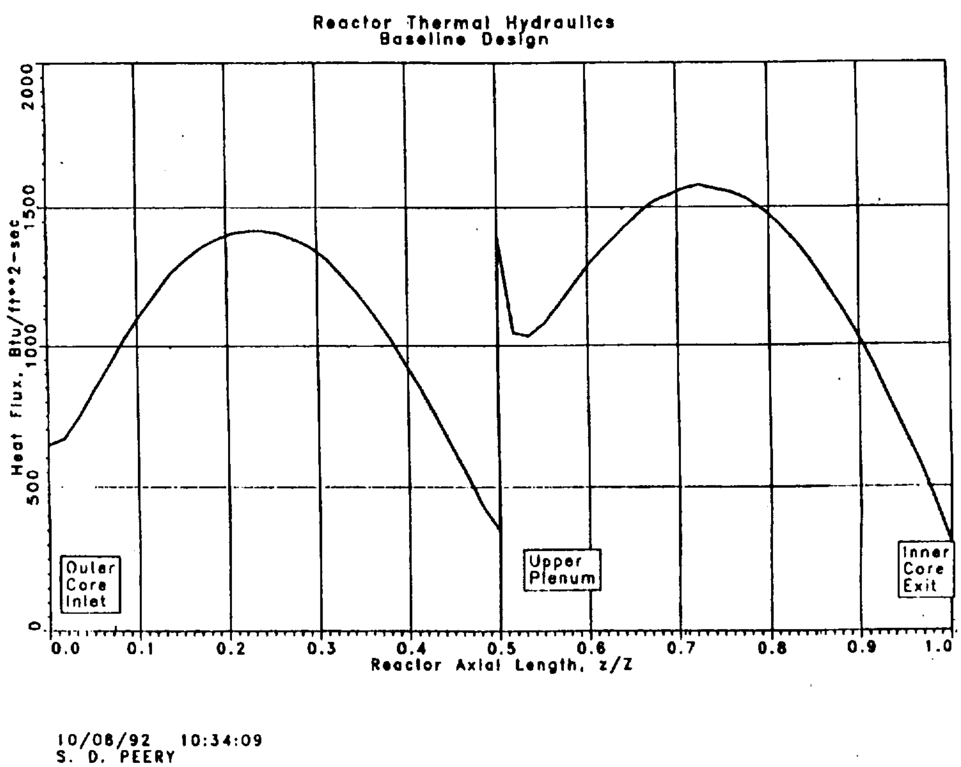






10/08/92 10:27:44
S. D. PEERY


Reactor Thermal Hydraulics
Baseline Design
Propellant Mach No.
0.00
0.05
0.10
0.15
0.20
0.25
0.30
Outer Core Inlet
Upper Plenum
Inner Core Exit
0.0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0.9
1.0
Reactor Axial Length, z/Z
10/08/92 10:32:58
S. D. PEERY


Reactor Thermal Hydraulics
Baseline Design
Heat Flux, Btu/ft**2-sec
0
500
1000
1500
2000
Outer Core Inlet
Upper Plenum
Inner Core Exit
0.0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0.9
1.0
Reactor Axial Length, z/Z
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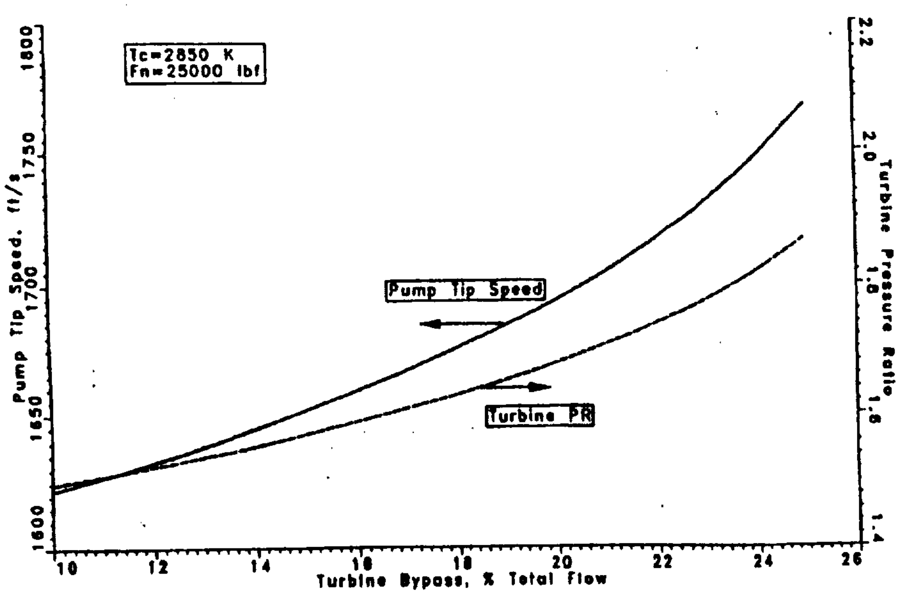
S. D. PEERY

TURBINE BYPASS IMPACT ON SYSTEM

Cycle Impact on Component Design

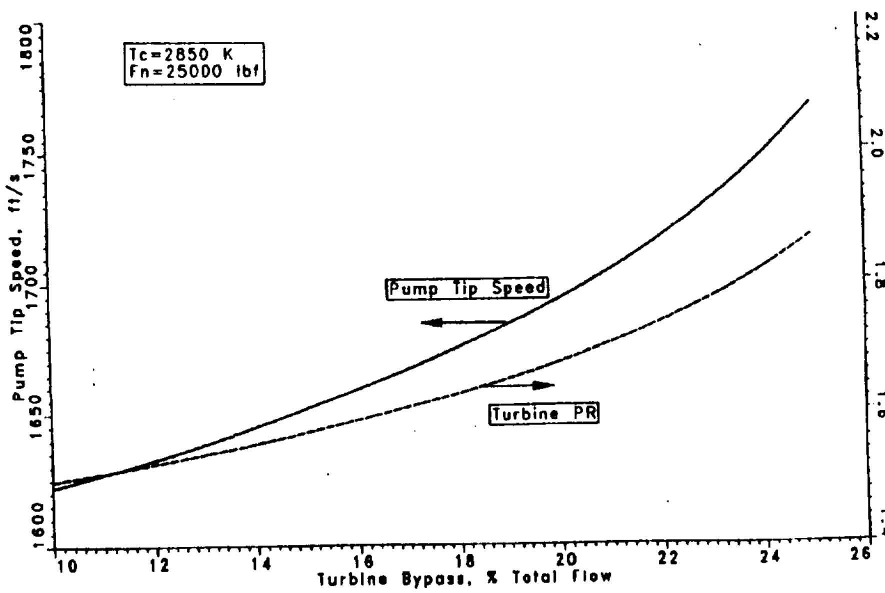


ROCETS NTR ENGINE SIMULATION SUMMARY

- NTR Engine Simulation Computational Models In-Place
- NTR Simulation Is Flexible
- Permits Great Level of Detail
- Permits Incorporation of Test Data
- Open Architecture Allows Continual Model Enhancements
- Permits Parametric NTR System Optimization

TURBINE BYPASS IMPACT ON SYSTEM

Cycle Impact on Component Design



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N93-26957

NTP SYSTEM SIMULATION AND DETAILED NUCLEAR ENGINE MODELING

Samim Anghaie

Innovative Nuclear Space Power and Propulsion Institute

University of Florida

Presented at
Nuclear Propulsion Technical Interchange Meeting
(NP-TIM-92)
October 20-23, 1992

NASA Lewis Research Center
Plum Brook Station

INSPI
University of Florida

NTP SYSTEM SIMULATION &
DETAILED NUCLEAR ENGINE MODELING

Samim Anghaie

Innovative Nuclear Space Power & Propulsion Institute
University of Florida
Gainesville, FL

With Technical Contribution from:

Gary Chen, University of Florida
Jeff Given, University of Florida
James White, University of Florida

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10/20/92

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MODELING AND ENGINEERING SIMULATION OF NUCLEAR THERMAL ROCKET SYSTEMS

- Modular Thermal Fluid Solver with Neutronic Feedback
- Main Component Modules:
  - Pipes, Valves, Mixer
  - Nozzle Skirt
  - Pump, Turbine
  - Reflector, Reactor Core
- Hydrogen (Para- and Dissociated) Property Package
  - $10 \leq T \leq 10{,}000$ K
  - $.1 \leq P \leq 160$ bar
- Models Developed for NTVR, NERVA and XNR 2000
- CFD and Heat Transfer Models for Main NTR Components

10/20/92

A detailed program for modeling of full system nuclear rocket engines is developed. At present time, the model features the expander cycle. Axial power distribution in the reactor core is calculated using 2- and 3-D neutronics computer codes. A complete hydrogen property model is developed and implemented. Three nuclear rocket systems are analyzed. These systems are: a 75,000 lbf NERVA class engine, a 25,000 lbf cermet fueled engine and INSPI's nuclear thermal vapor rocket.

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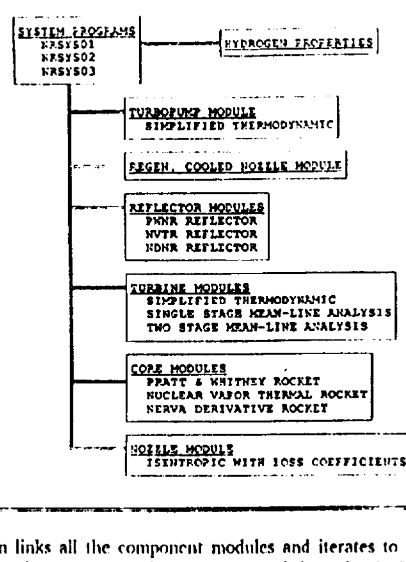
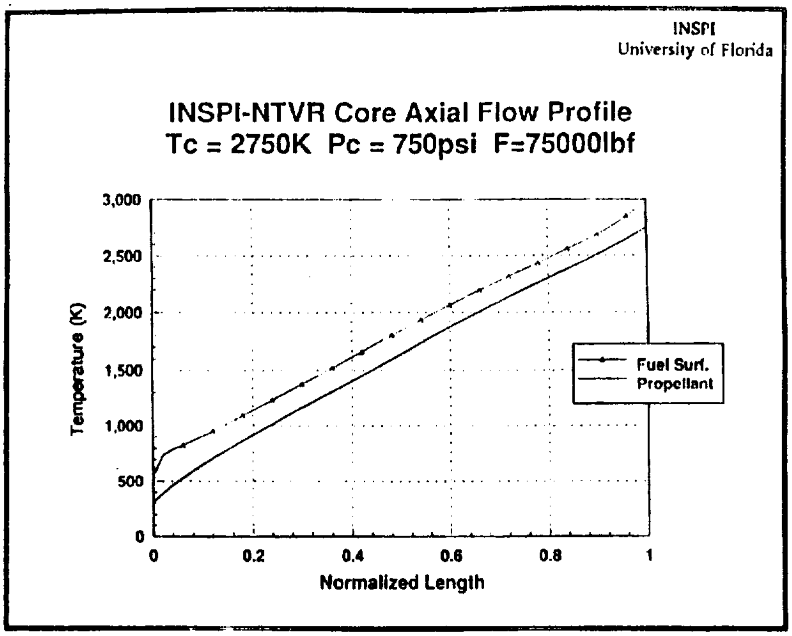
NUCLEAR THERMAL ROCKET SIMULATION SYSTEM

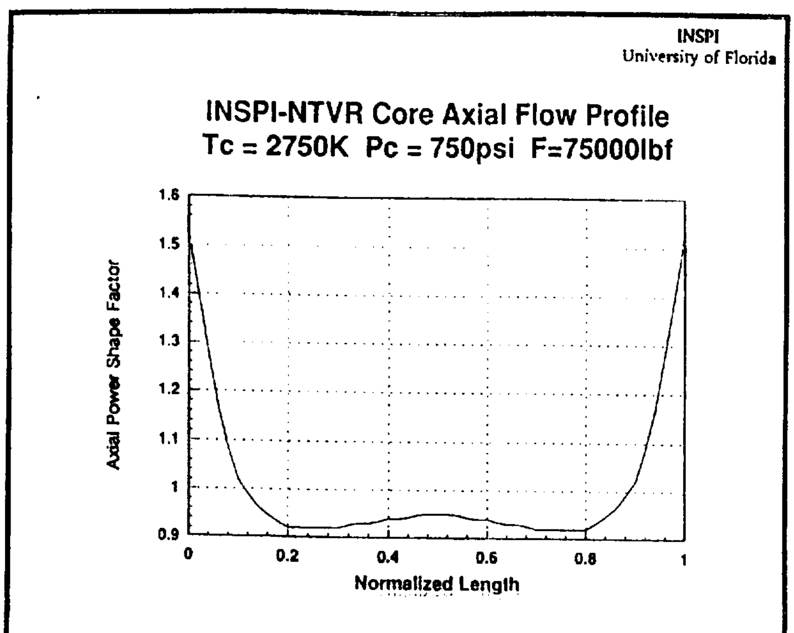


10 20 92

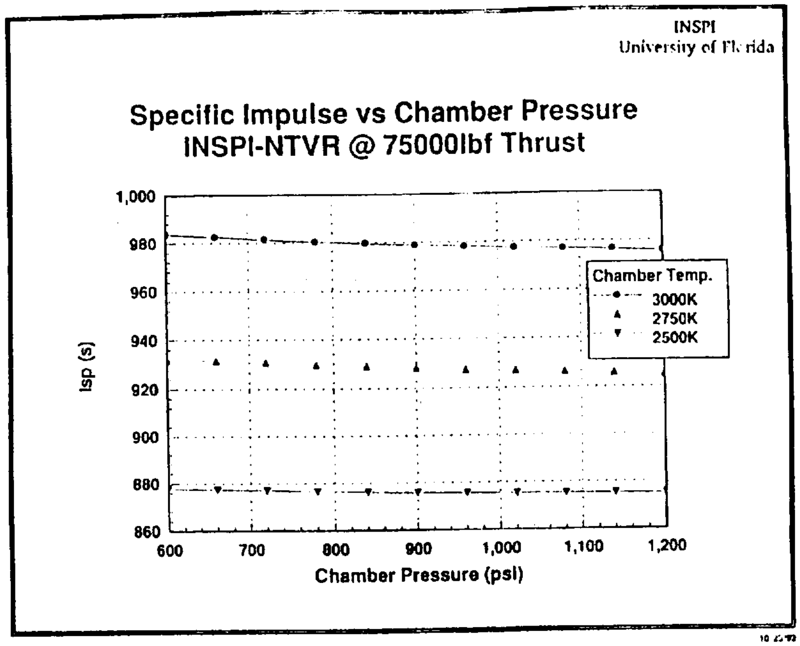
The main program links all the component modules and iterates to arrive at the user specified thrust chamber pressure and temperature and thrust level. Reactor power and propellant flow rate are among outputs of the simulation program. Fuel elements in the core module are prismatic with variable flow area ratio. Each module divides the relative component into N segments.



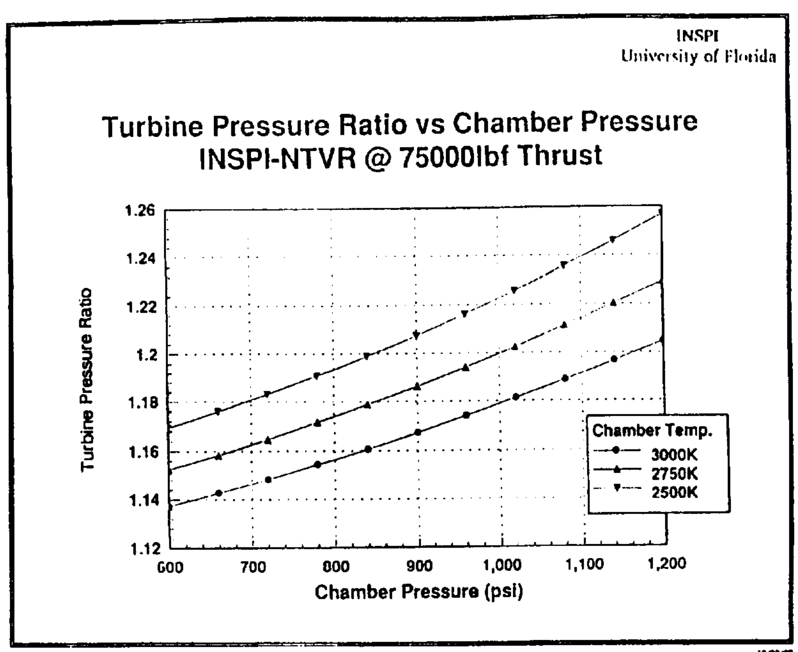
Axial temperature distribution of NVTR fuel surface and propellant in an average power rod. Reactor power is adjusted to achieve the thrust chamber temperature and pressure of 2750 K and 750 psi, respectively.



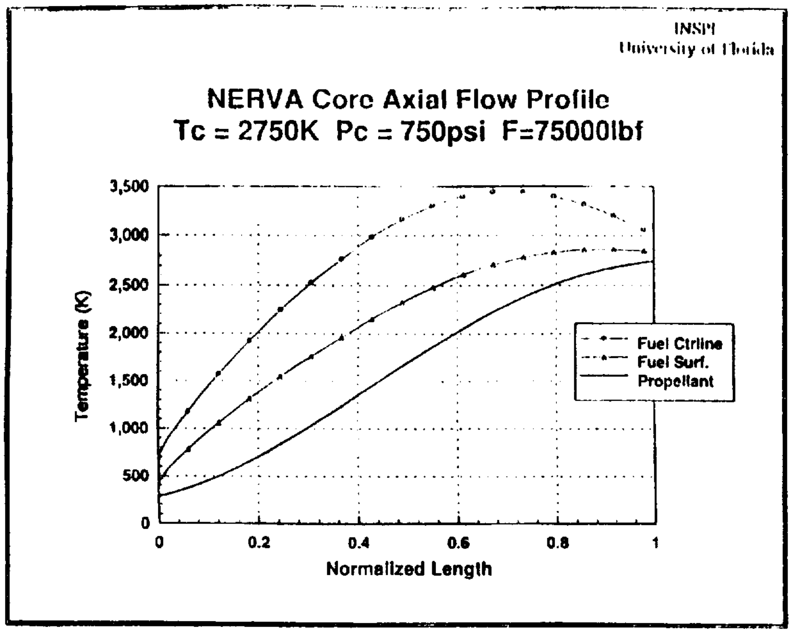
Normalized axial power distribution in C-C composite fuel matrix NTVR, calculated by DOT-2 S_n code. The axial power shape factor is an input for the simulation code.



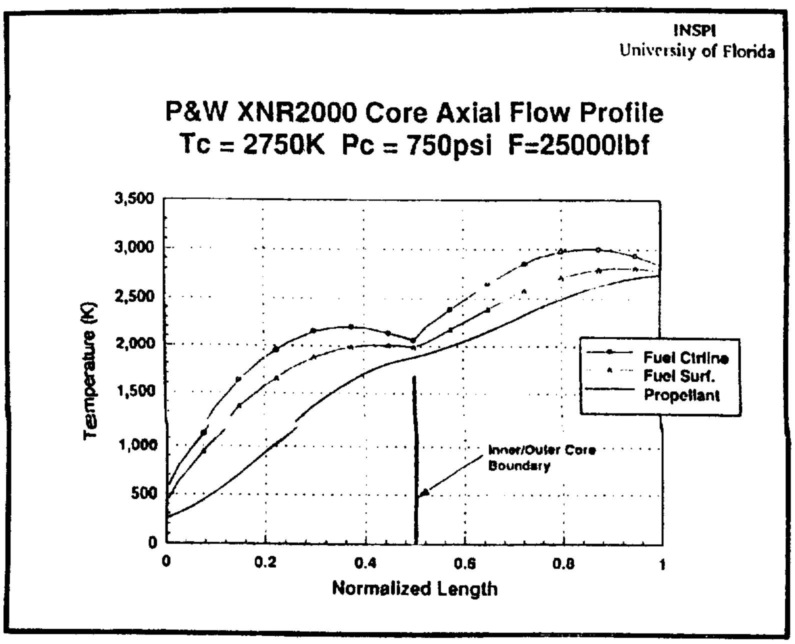
Parametric study of thrust chamber pressure and temperature impact on Isp of NTVR. At higher pressures Isp is less sensitive to thrust chamber temperature.



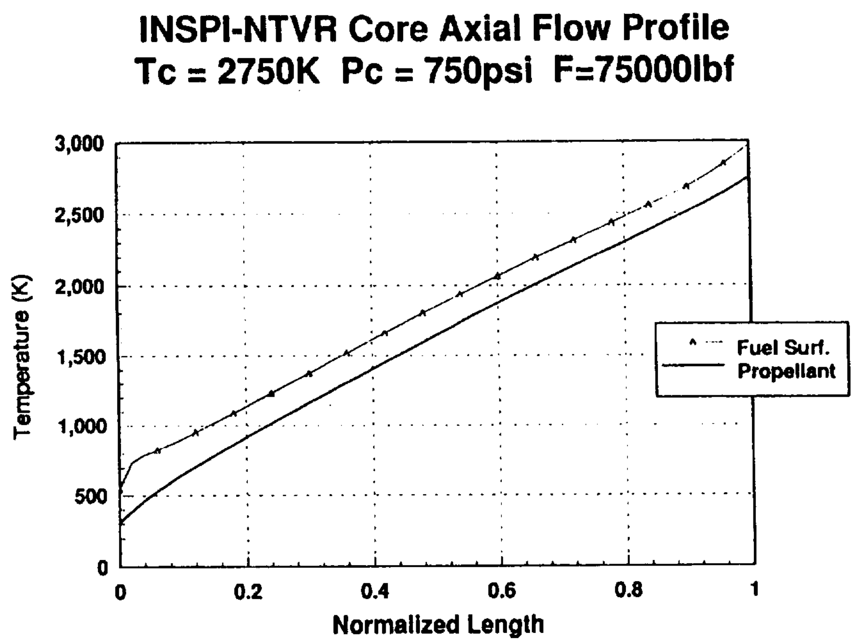
Turbine pressure ratio is sensitive to both thrust chamber pressure and temperature. For thrust chamber pressure of 1200 psi and temperature of 3000 K, the turbine pressure ratio of 1.26 is well within the range of available technology.

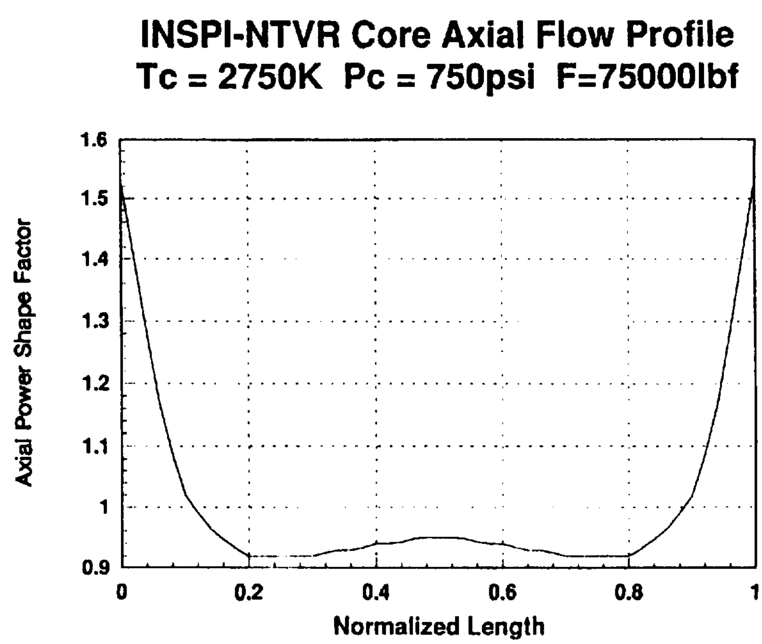


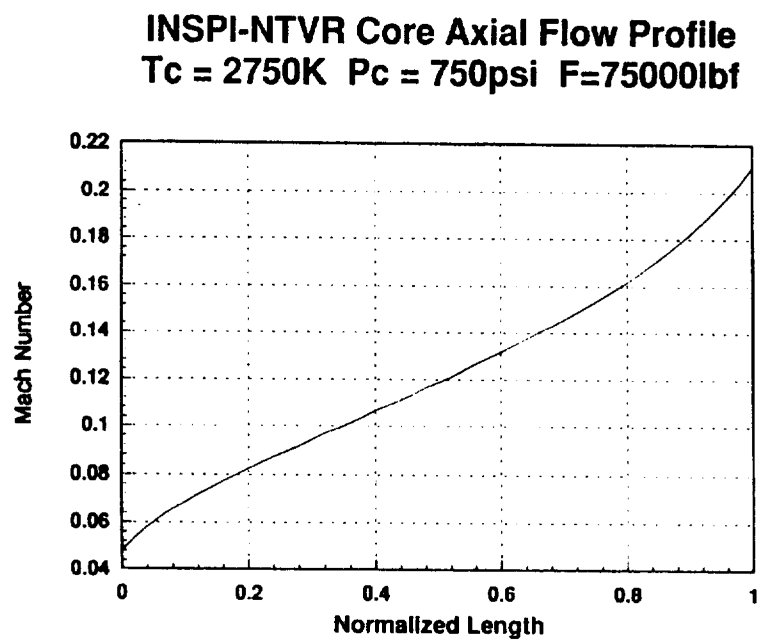
Axial temperature profiles for NERVA-75,000 lbf engine are presented. The maximum fuel temperature is 3490 K at .7 m from the core entrance.

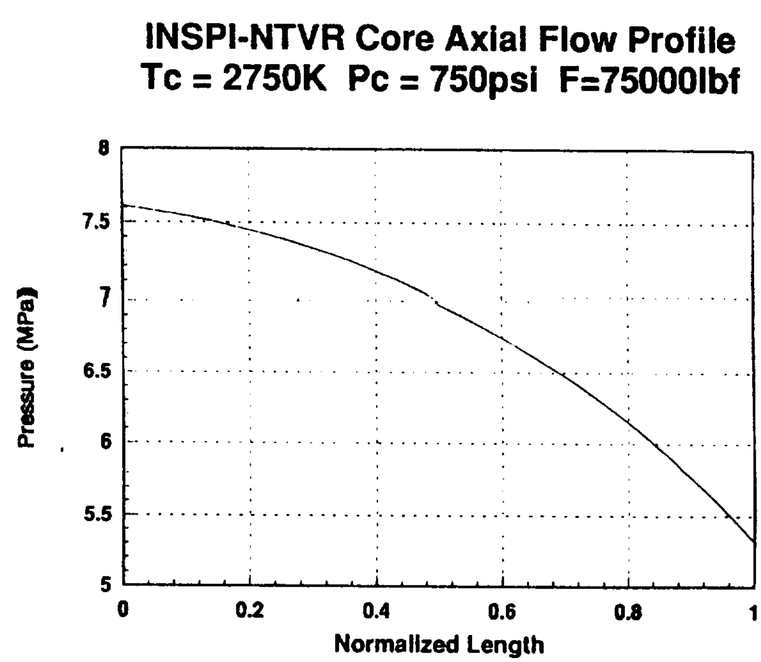


Axial temperature distribution in XNR 2000 core is presented. XNR 2000 features a two path folded flow core fueled with CERMET. The maximum fuel temperature is 3000 K at about 85% from the entrance to the inner core region.

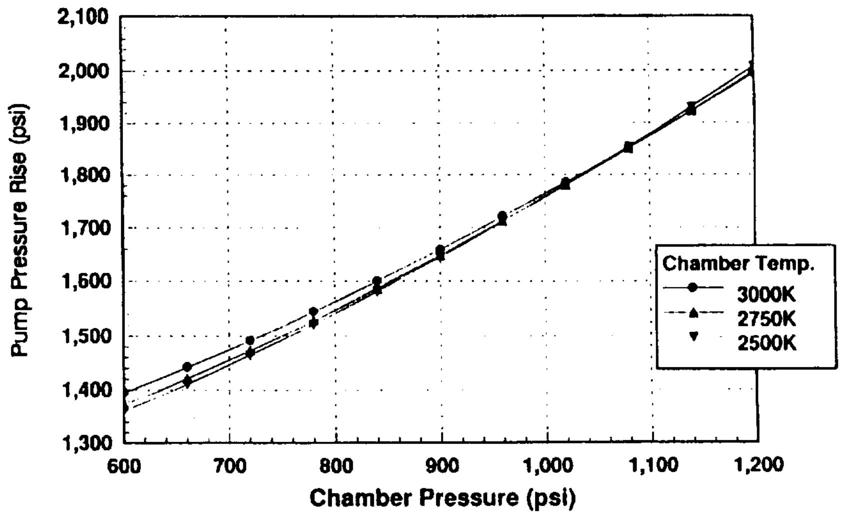

INSPI-NTVR Core Axial Flow Profile
Tc = 2750K Pc = 750psi F=75000lbf
Temperature (K)
3,000
2,500
2,000
1,500
1,000
500
0
0
0.2
0.4
0.6
0.8
1
Normalized Length
Fuel Surf.
Propellant


INSPI-NTVR Core Axial Flow Profile
Tc = 2750K Pc = 750psi F=75000lbf
Axial Power Shape Factor
1.6
1.5
1.4
1.3
1.2
1.1
1
0.9
0
0.2
0.4
0.6
0.8
1
Normalized Length

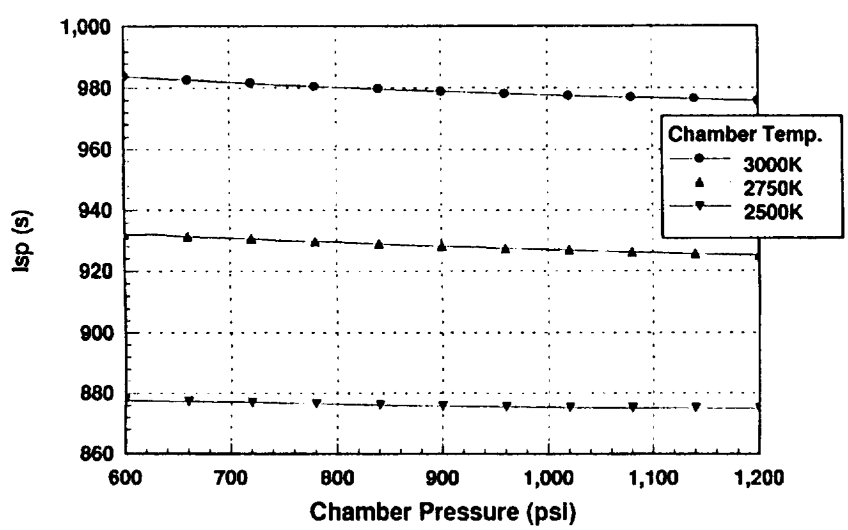

INSPI-NTVR Core Axial Flow Profile
Tc = 2750K Pc = 750psi F=75000lbf
Mach Number
0.22
0.2
0.18
0.16
0.14
0.12
0.1
0.08
0.06
0.04
0
0.2
0.4
0.6
0.8
1
Normalized Length


INSPI-NTVR Core Axial Flow Profile
Tc = 2750K Pc = 750psi F=75000lbf
Pressure (MPa)
8
7.5
7
6.5
6
5.5
5
0
0.2
0.4
0.6
0.8
1
Normalized Length

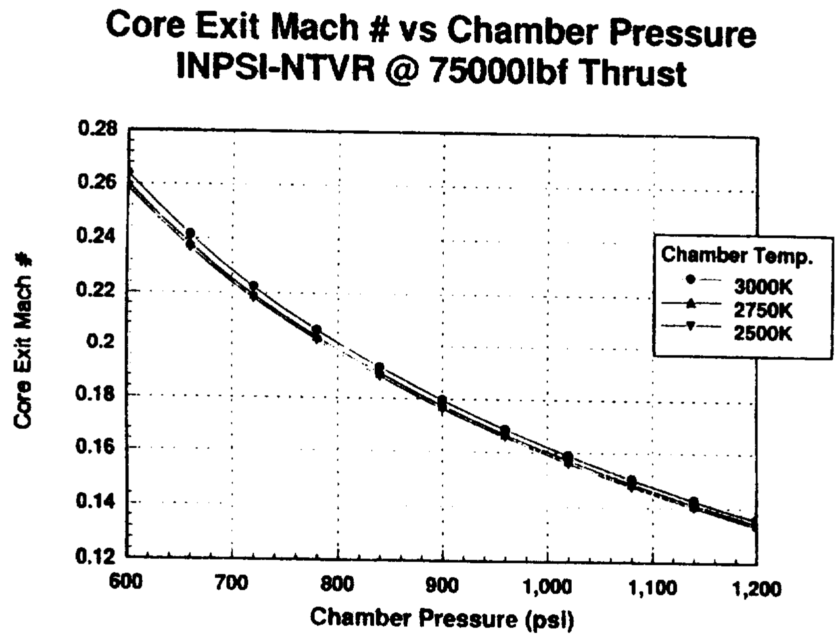
Pump Pressure Rise vs Chamber Pressure
INSPI-NTVR @ 75000lbf Thrust

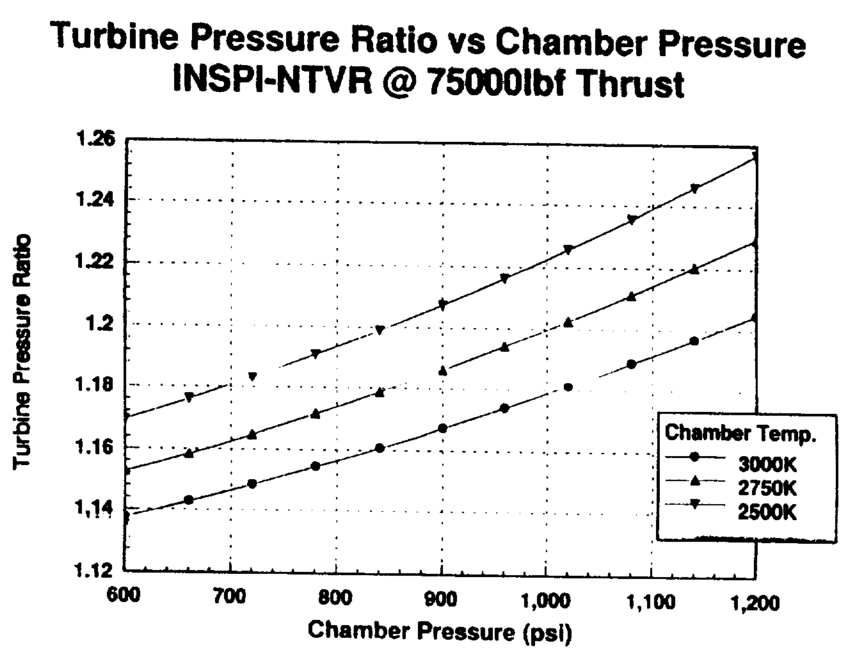


Specific Impulse vs Chamber Pressure
INSPI-NTVR @ 75000lbf Thrust

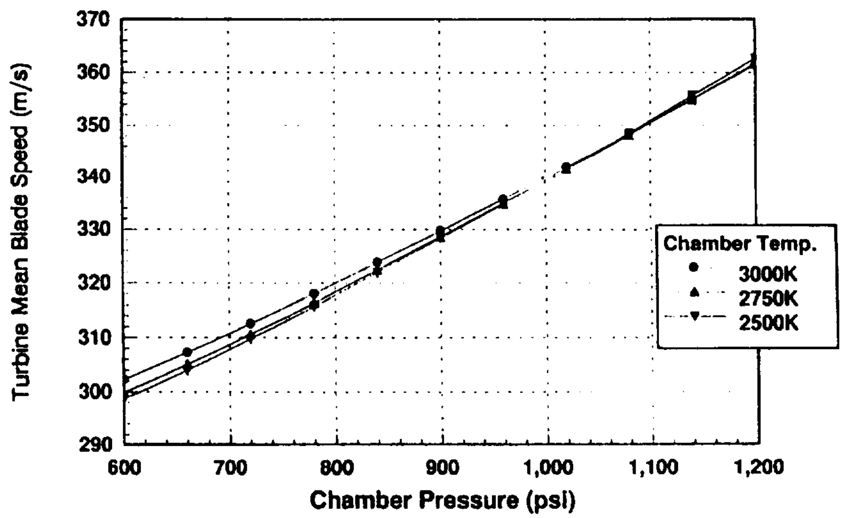


C-2

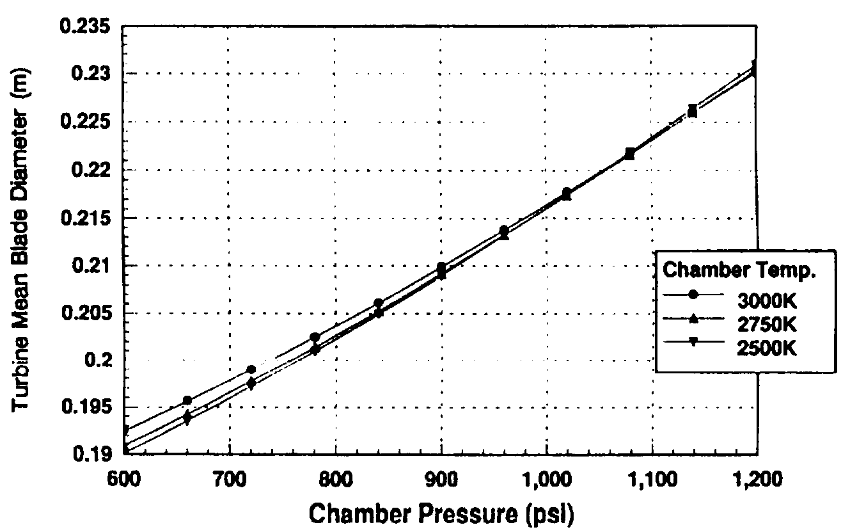

Core Exit Mach # vs Chamber Pressure
INPSI-NTVR @ 75000lbf Thrust
0.28
0.26
0.24
0.22
0.2
0.18
0.16
0.14
0.12
Core Exit Mach #
600
700
800
900
1,000
1,100
1,200
Chamber Pressure (psi)
Chamber Temp.
3000K
2750K
2500K


Turbine Pressure Ratio vs Chamber Pressure
INSPI-NTVR @ 75000lbf Thrust
1.26
1.24
1.22
1.2
1.18
1.16
1.14
1.12
Turbine Pressure Ratio
600
700
800
900
1,000
1,100
1,200
Chamber Pressure (psi)
Chamber Temp.
3000K
2750K
2500K

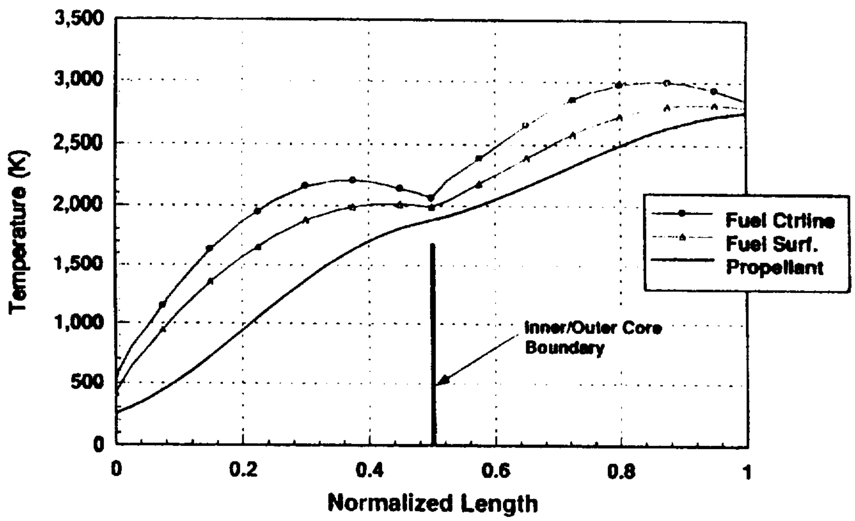
Turbine Blade Speed vs Chamber Pressure
INSPI-NTVR @ 75000lbf Thrust



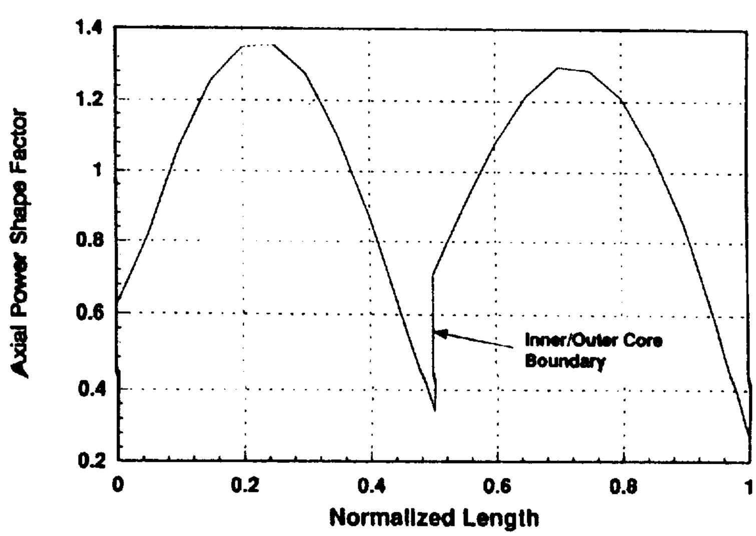
Turbine Blade Diameter vs Chamber Pressure
INSPI-NTVR @ 75000lbf Thrust



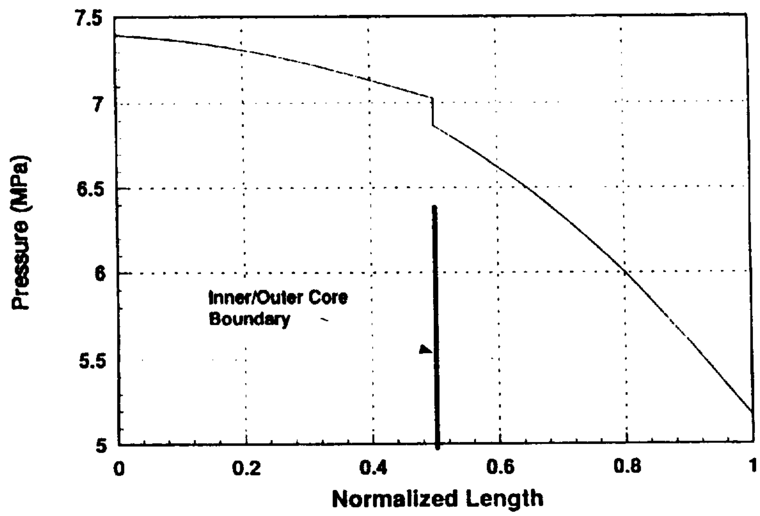
P&W XNR2000 Core Axial Flow Profile
Tc = 2750K Pc = 750psi F=25000lbf



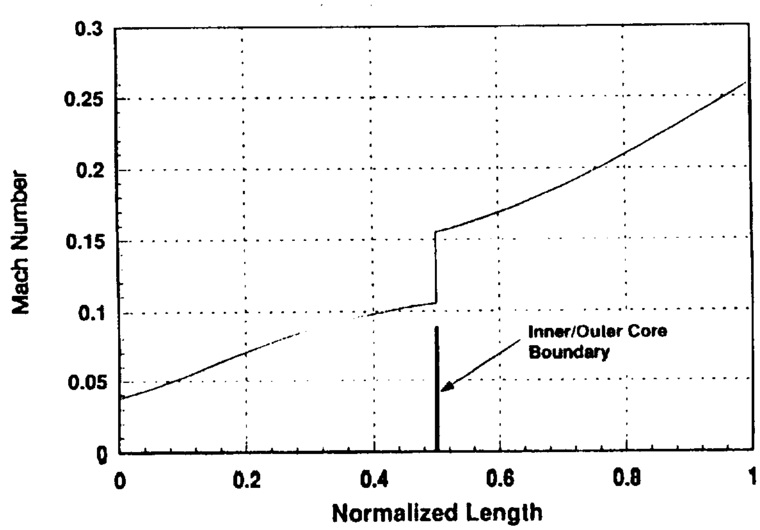
P&W XNR2000 Core Axial Flow Profile
Tc = 2750K Pc = 750psi F=25000lbf



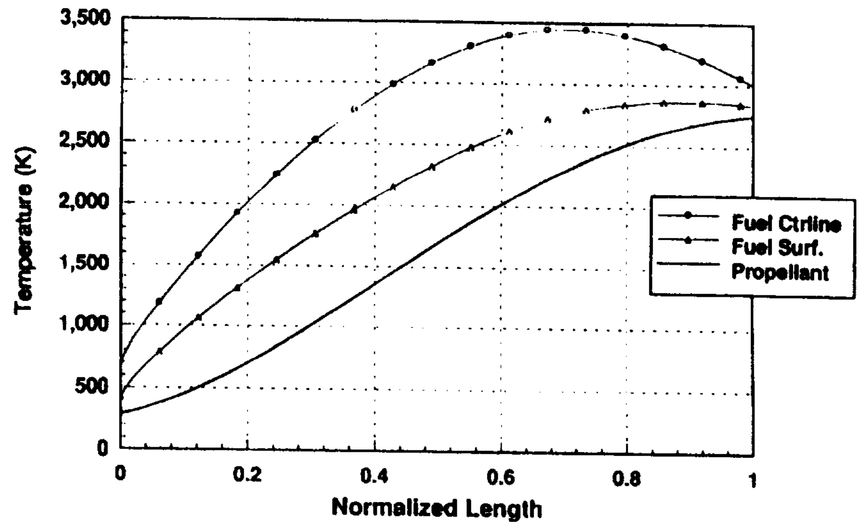
P&W XNR2000 Core Axial Flow Profile
Tc = 2750K Pc = 750psi F=25000lbf



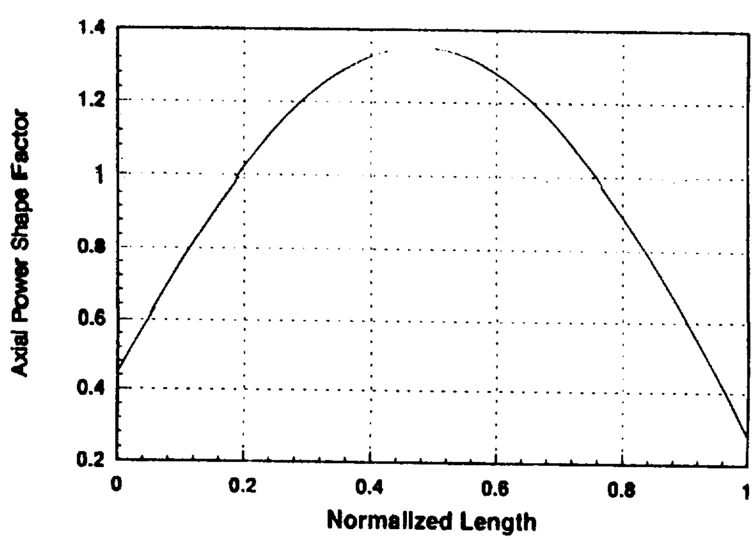
P&W XNR2000 Core Axial Flow Profile
Tc = 2750K Pc = 750psi F=25000lbf



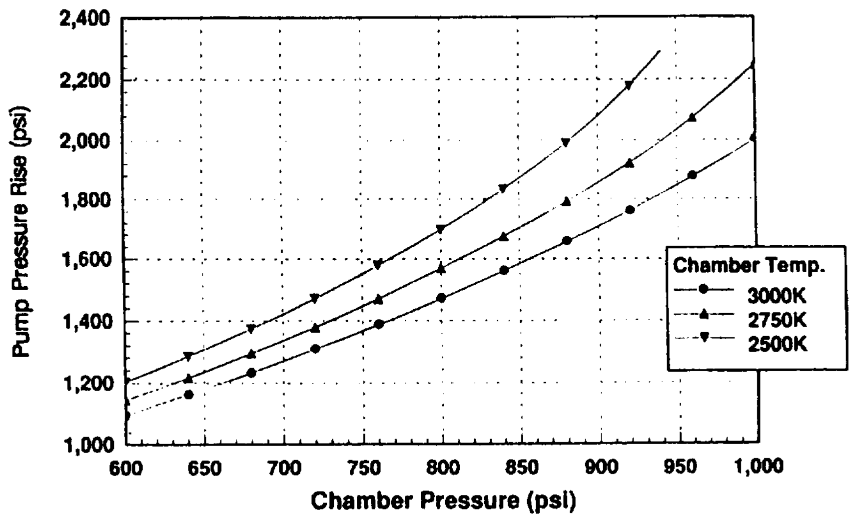
NERVA Core Axial Flow Profile
Tc = 2750K Pc = 750psi F=75000lbf



NERVA Core Axial Flow Profile
Tc = 2750K Pc = 750psi F=75000lbf



Pump Pressure Rise vs Chamber Pressure
NERVA @ 75000lbf Thrust



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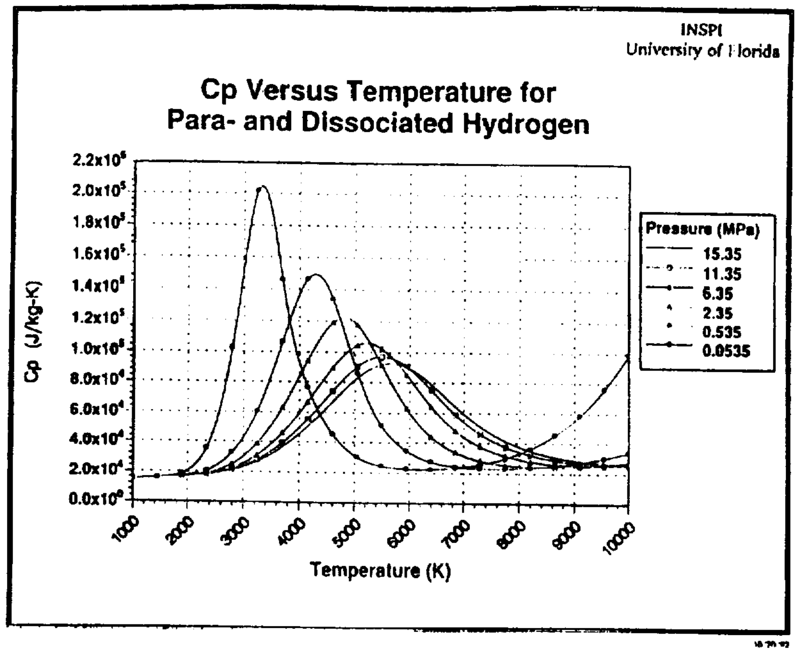
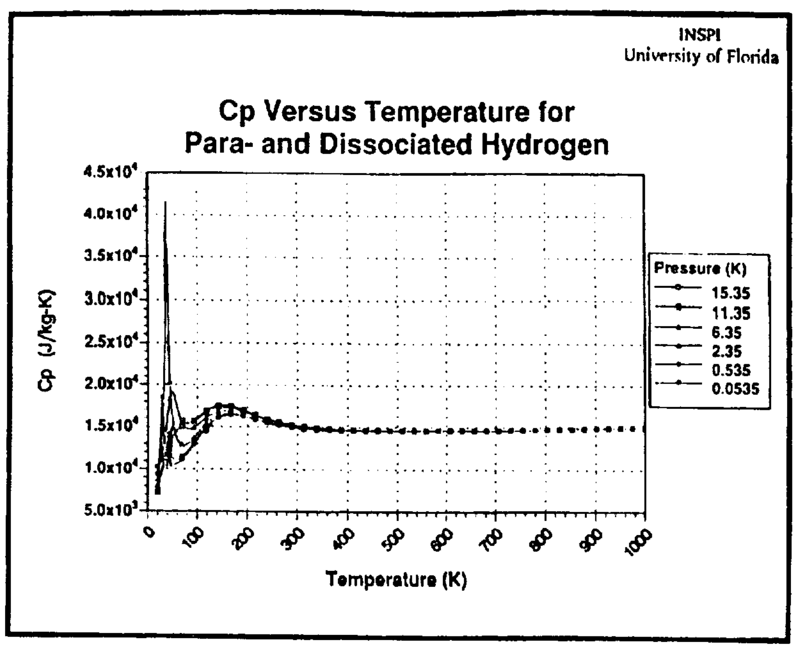
EVALUATION OF PARA- AND DISSOCIATED HYDROGEN PROPERTIES AT T = 10 - 10,000 K

- NASA/NIST Property Package
  (13.8 < T < 10,000 K and .1 < P < 160 bar)
  - Molecular Weight, Density
  - Enthalpy, Entropy
  - Specific Heats, Specific Heat Ratio
  - Thermal Conductivity, Viscosity
- Hydrogen Property Generator Code Features
  - Linear Interpolation
  - Natural Cubic Spline
  - Least Square Curve Fitting with Pentad Spline Joint Functions
- Graphical Representation of Properties

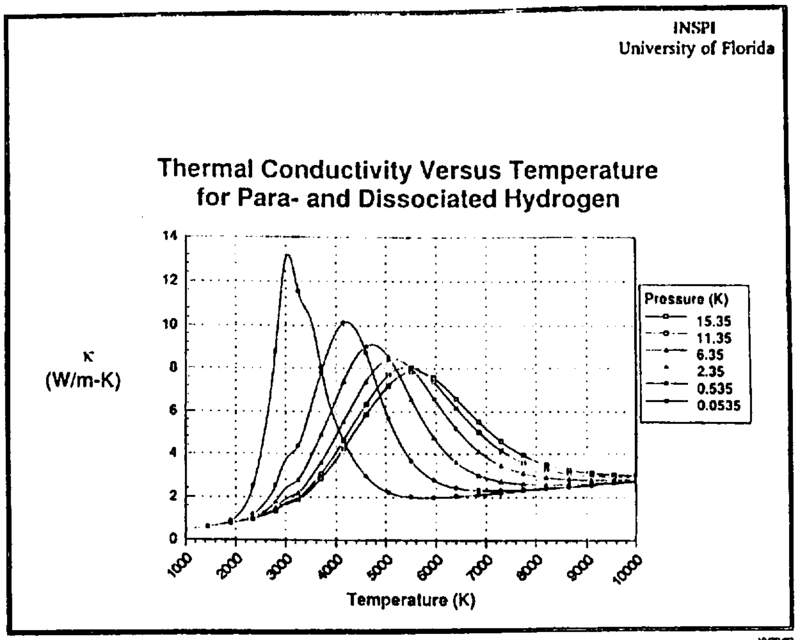
The hydrogen property generator utilizes two interpolation techniques and a least-square curve fitting routine with a pentad spline function which links least-square fitted pieces together. The property generator package is incorporated into the NTR simulation code and also into a system of CFD-HT codes.



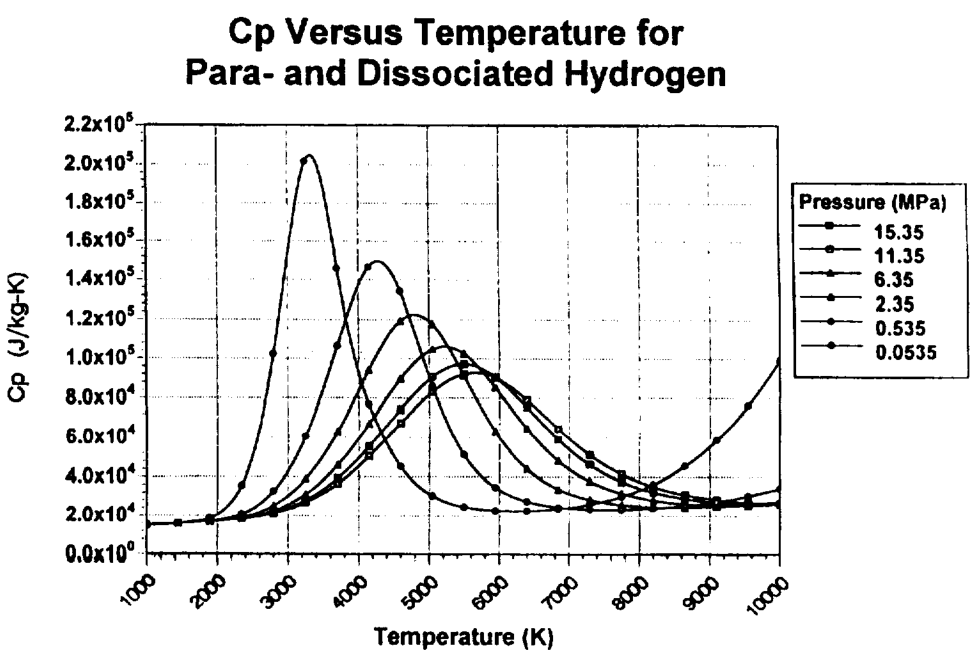
At higher temperatures, the heat capacity data displays smooth behavior. The sharp increase in C_p value at temperatures above 2000 K is due to hydrogen dissociation.

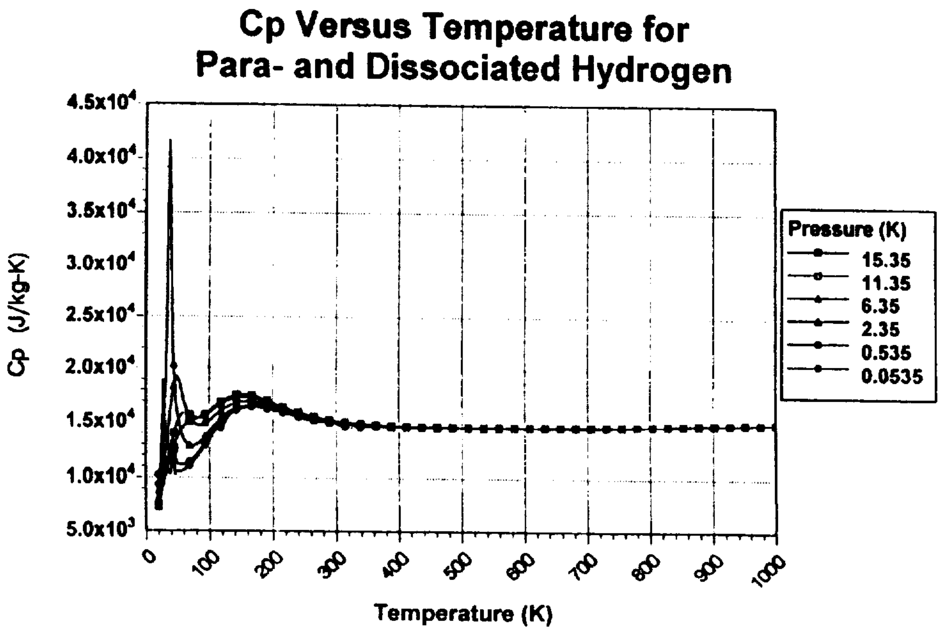


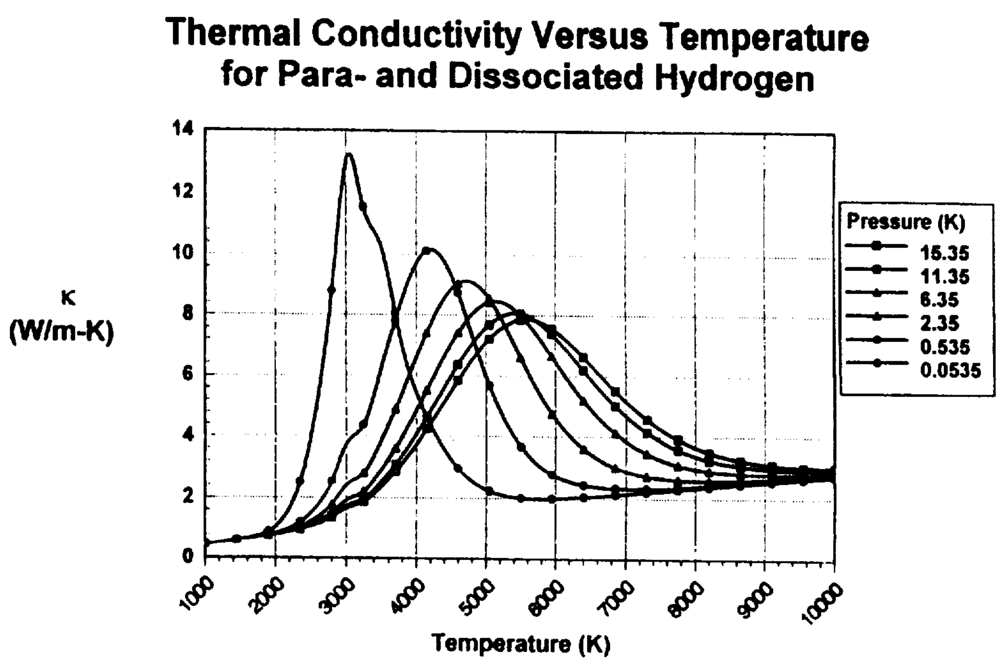
Heat capacity of hydrogen near the critical point shows large gradient and oscillatory behavior. At p = 2.35 MPa the property package indicates a sharp peak for C_p.



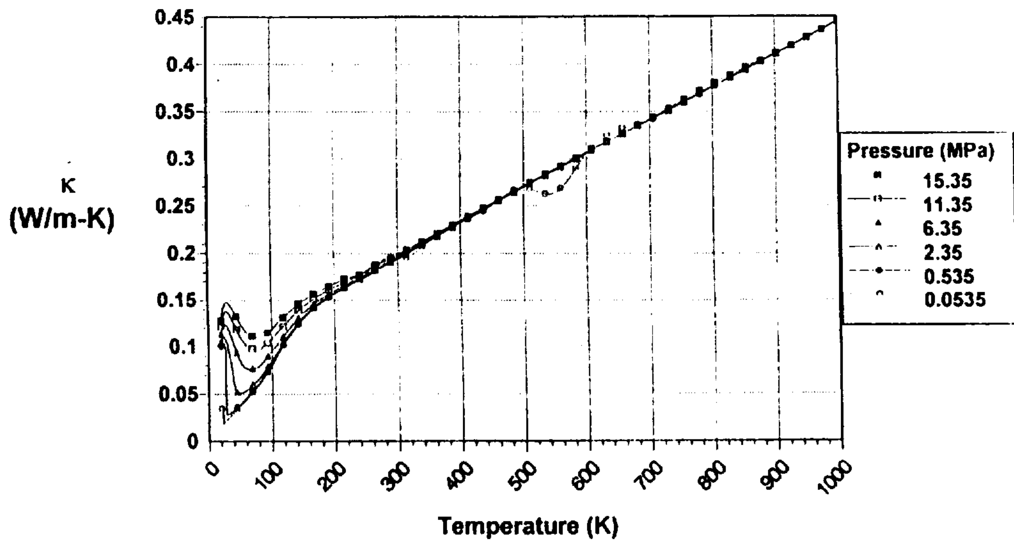
The hydrogen property package is a combination of two subpackages covering the temperature ranges 10 - 3000 K and 3000 - 10,000 K, respectively. The large change of gradients in hydrogen viscosity at 3000 K indicates a non-physical flaw in the model.




Cp Versus Temperature for
Para- and Dissociated Hydrogen
4.5x10^4
4.0x10^4
3.5x10^4
3.0x10^4
2.5x10^4
2.0x10^4
1.5x10^4
1.0x10^4
5.0x10^3
Cp (J/kg-K)
0
100
200
300
400
500
600
700
800
900
1000
Temperature (K)
Pressure (K)
15.35
11.35
6.35
2.35
0.535
0.0535


Thermal Conductivity Versus Temperature
for Para- and Dissociated Hydrogen
14
12
10
8
6
4
2
0
K
(W/m-K)
1000
2000
3000
4000
5000
6000
7000
8000
9000
10000
Temperature (K)
Pressure (K)
15.35
11.35
6.35
2.35
0.535
0.0535

Thermal Conductivity Versus Temperature for Para- and Dissociated Hydrogen



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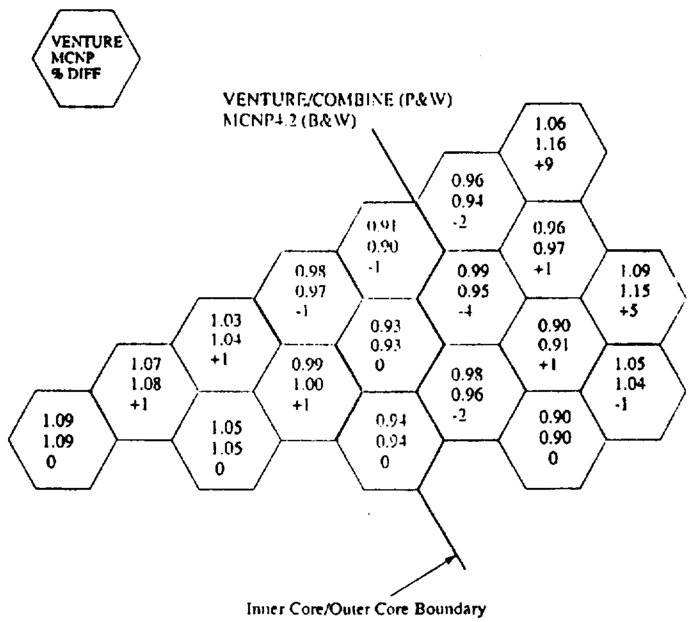
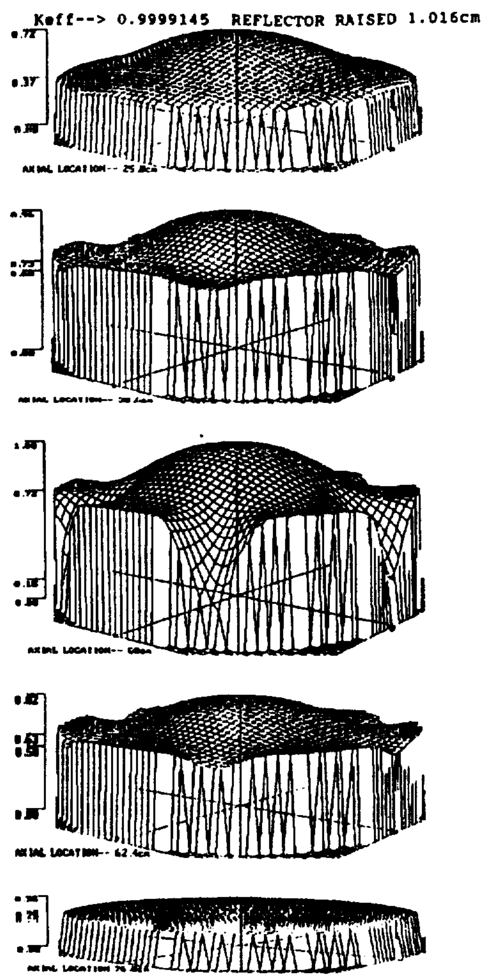
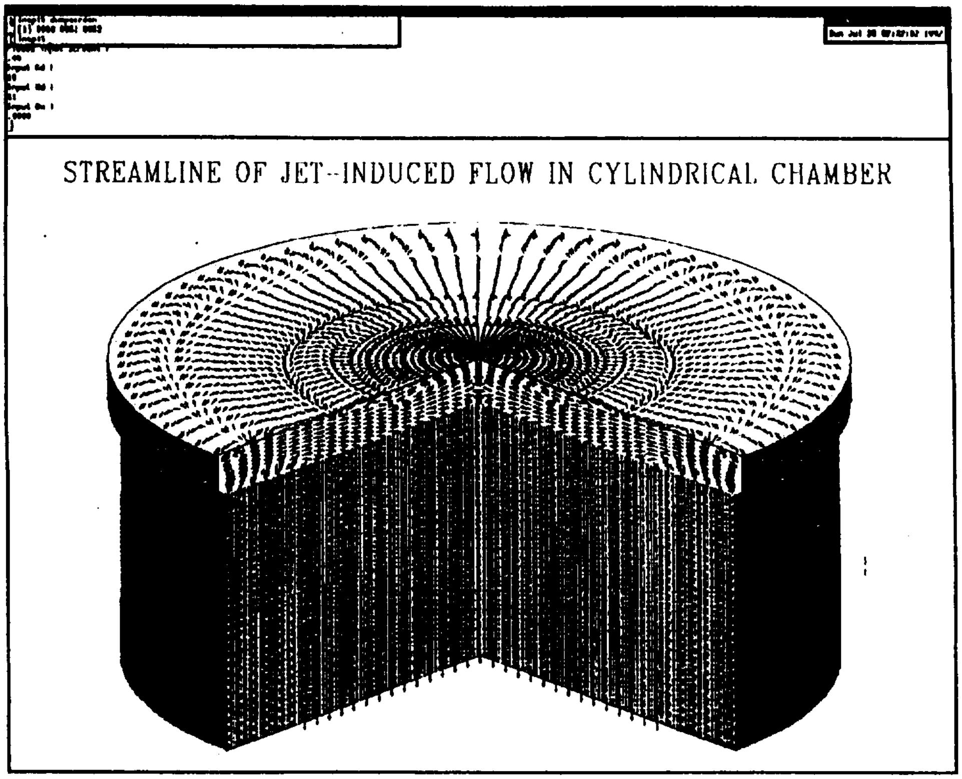
NUCLEAR DESIGN ANALYSIS PACKAGE

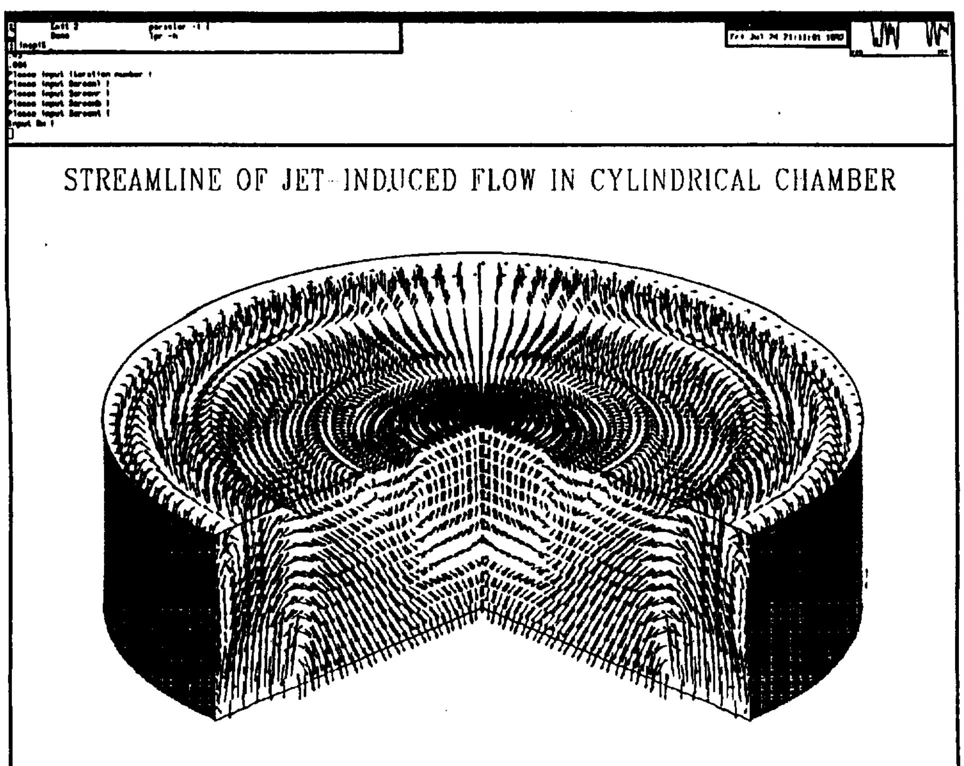
- Multigroup Cross-sections Generated by COMBINE (ENDFB-V)
- MCNP (4.2) for Complex Geometries
- BOLD VENTURE (3-D, Diffusion) for Power Profile and Reactivity Calculations
- ANISN (1-D, S_n) for Analysis of Heterogeneous Boundaries
- DOT IV (1, 2-D, S_n) for Analysis of Reflector
- XSDRNPM (1-D, S_n) TWODANT (2-D, S_n), NJOY, AMPX for Cross-comparison

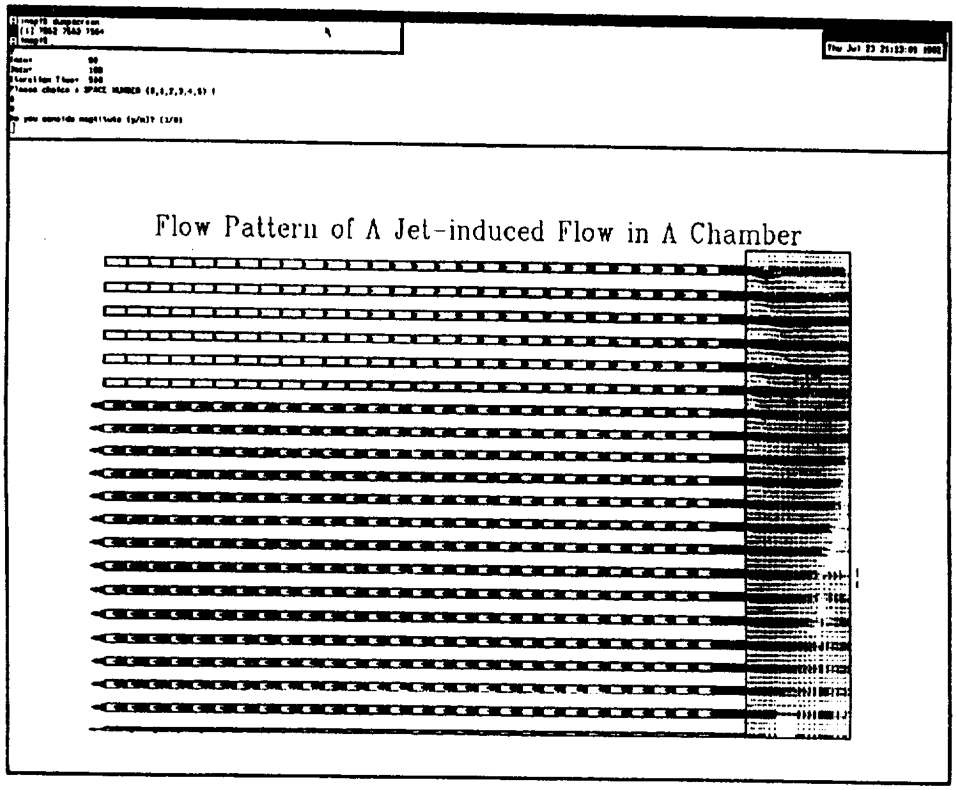
XNR2000 Rodwise Radial Power Distribution (normalized)

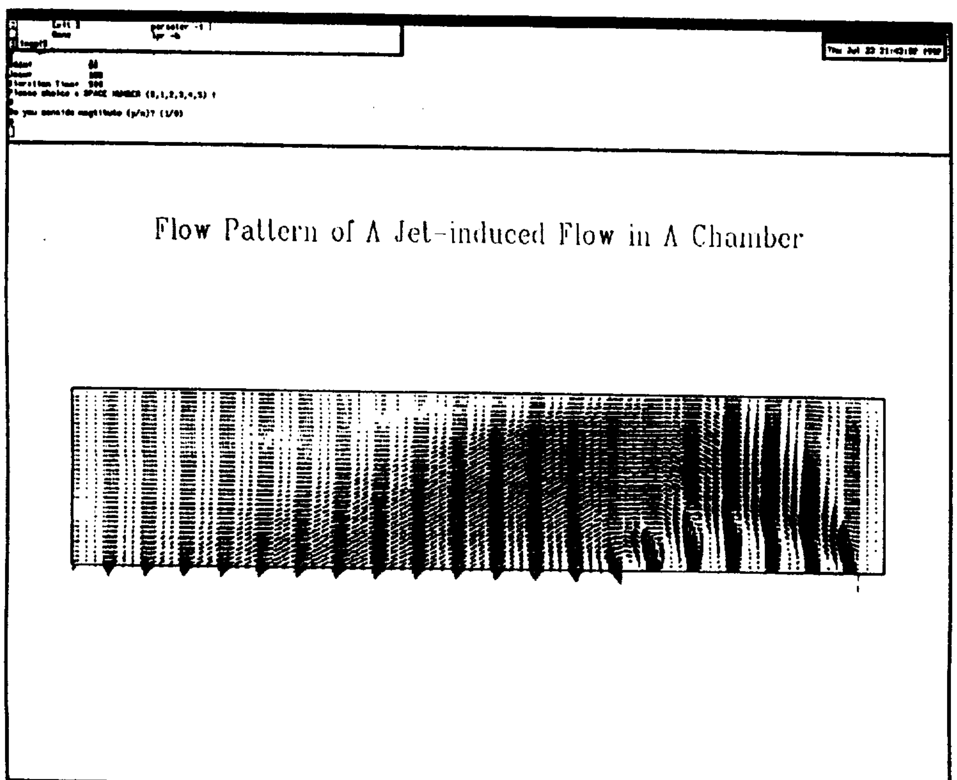


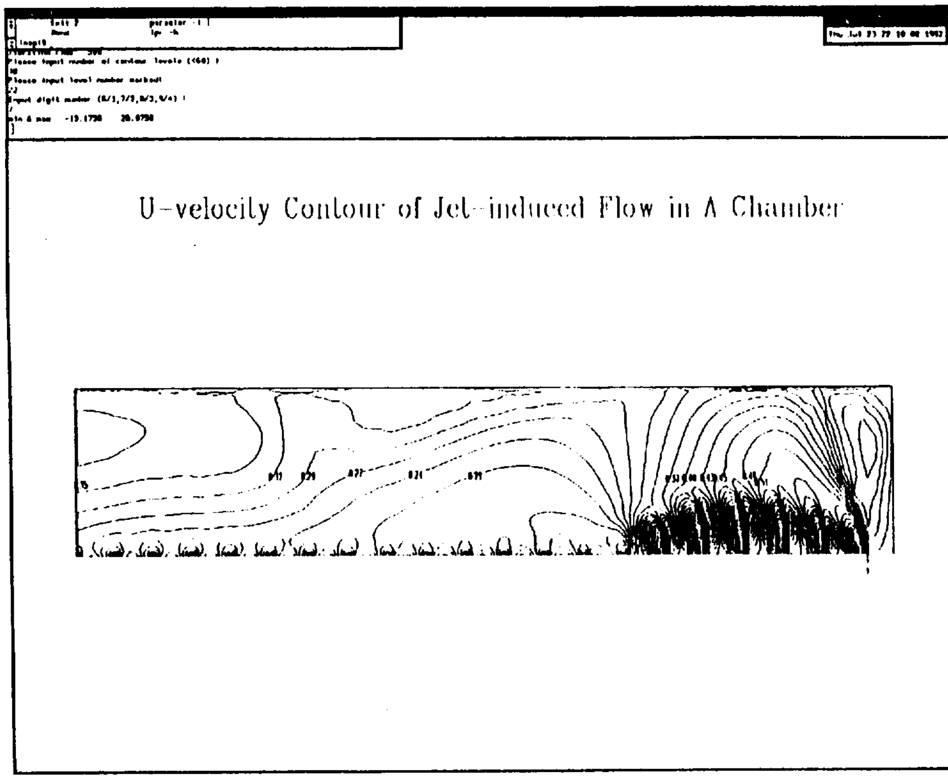


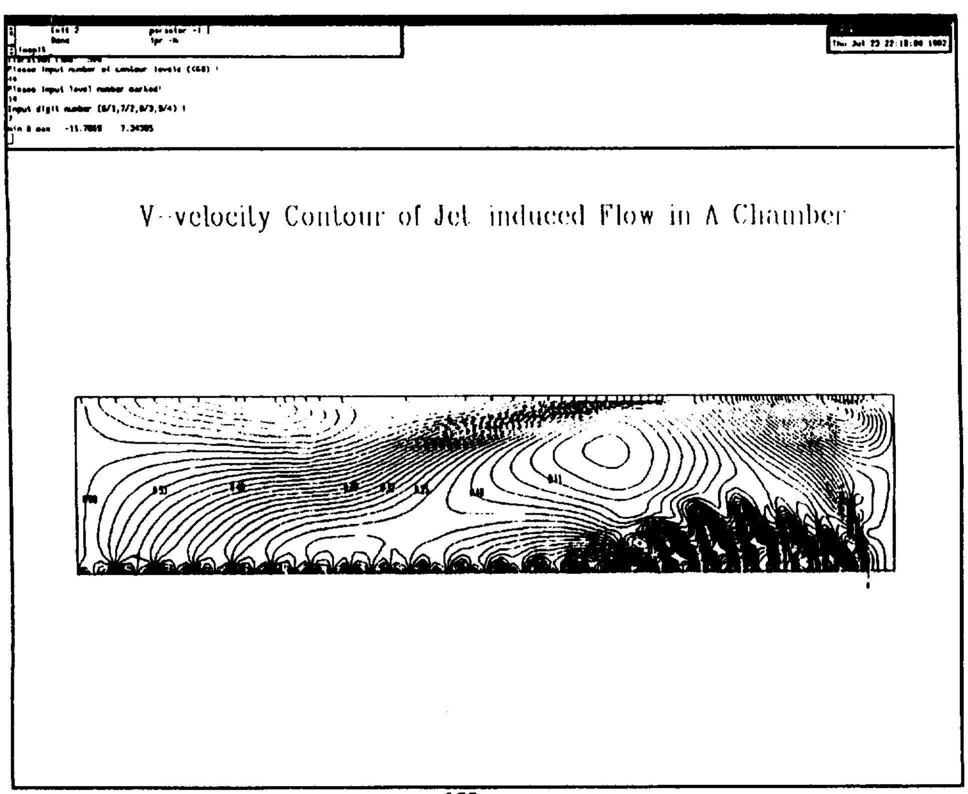

STREAMLINE OF JET-INDUCED FLOW IN CYLINDRICAL CHAMBER

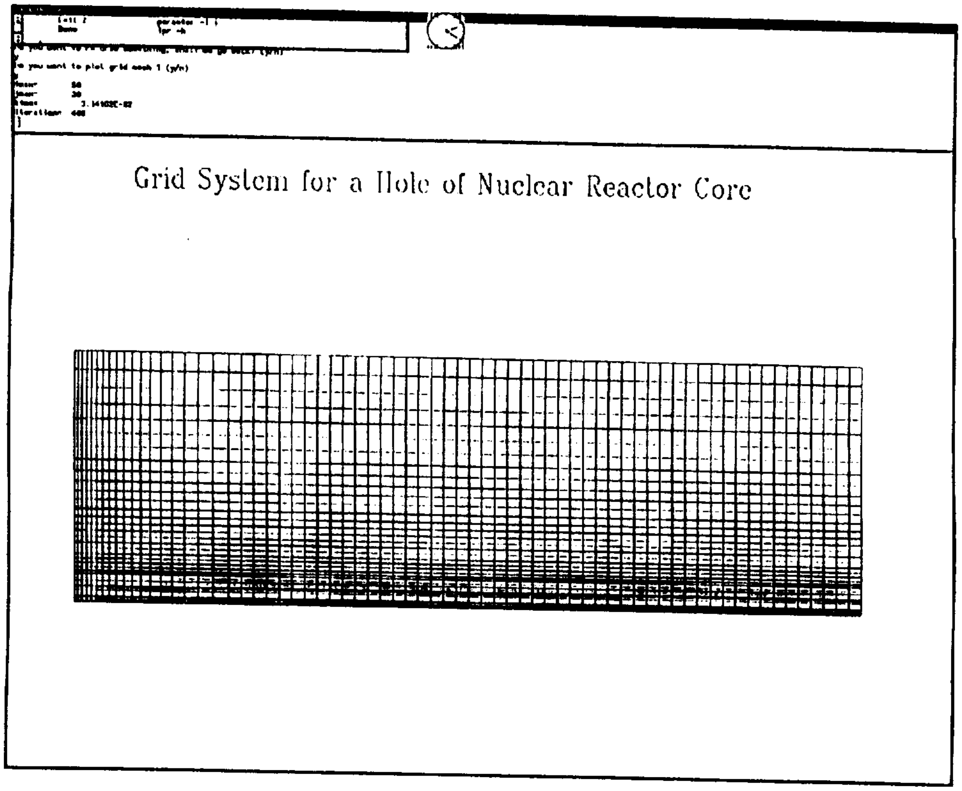

STREAMLINE OF JET-INDUCED FLOW IN CYLINDRICAL CHAMBER

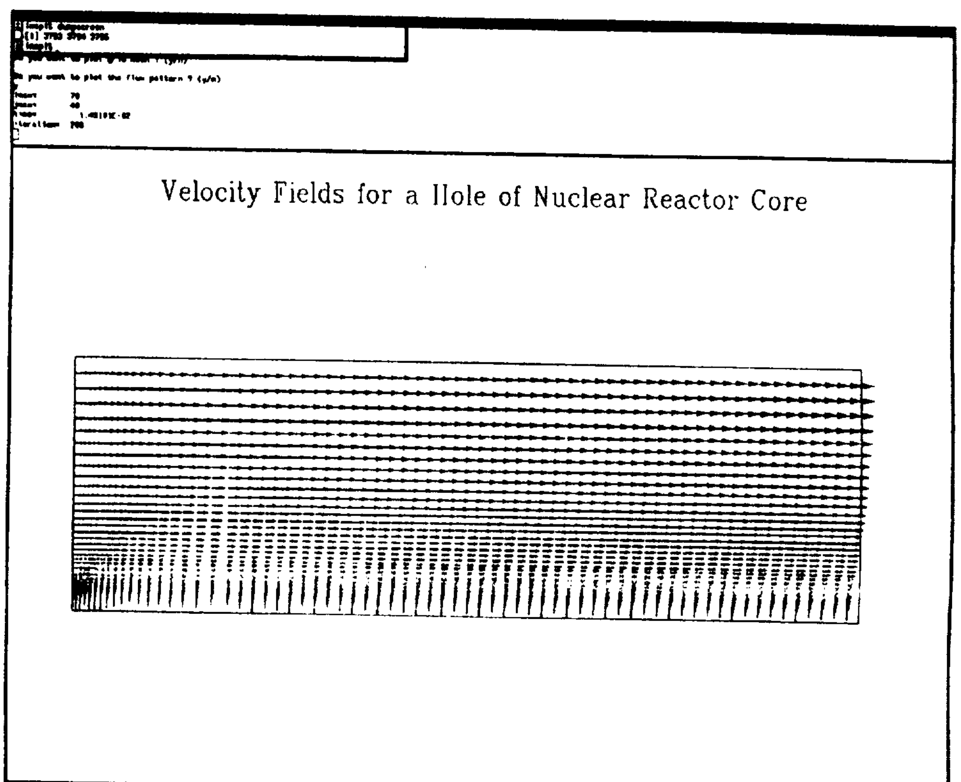

Flow Pattern of A Jet-induced Flow in A Chamber

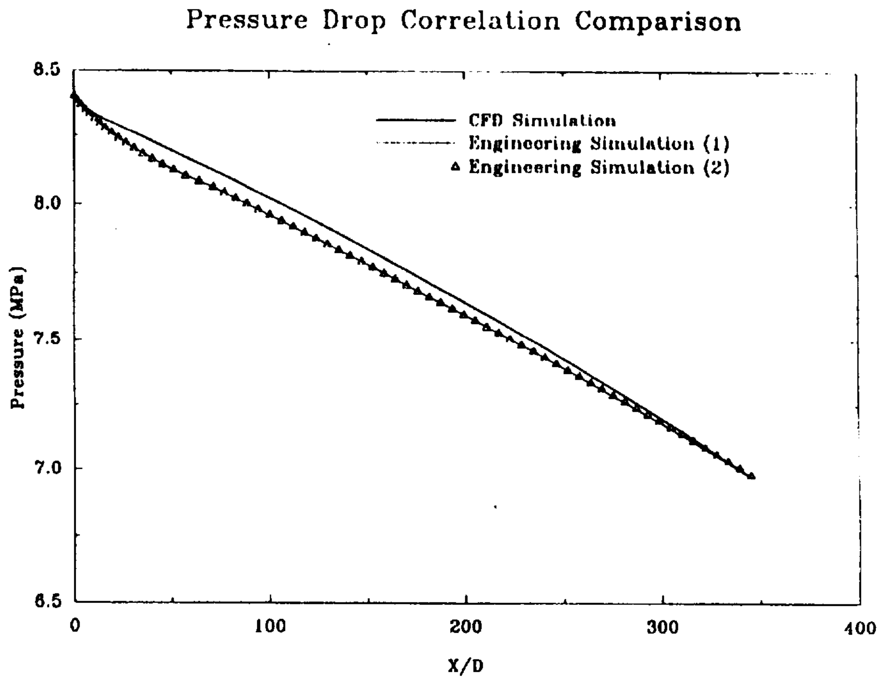

Flow Pattern of A Jet-induced Flow in A Chamber


U-velocity Contour of Jet-induced Flow in A Chamber


V-velocity Contour of Jet induced Flow in A Chamber


Grid System for a Hole of Nuclear Reactor Core


Velocity Fields for a Hole of Nuclear Reactor Core



Nusselt Number (Nu) and Pressure Drop (ΔP) Correlation Comparison with CFD Analysis

(I) CFD Analysis

Energy Equation

$$\frac{\partial}{\partial X}\left((e+\sigma_x)u+\tau_{yx}v-K_c\frac{\partial T}{\partial X}\right)+\frac{\partial}{\partial Y}\left((e+\sigma_y)v+\tau_{xy}u-K_c\frac{\partial T}{\partial Y}\right)=S_e$$

Numerical Algorithm: MacCormack hybrid implicit-explicit, finite volume method

Conductive Heat Flux

$$q_c'' = -K_c\left[\frac{\partial T}{\partial r}\right]_{r=R}$$

Convective Heat Flux

$$q_x'' = h_c(T_w - T_b)$$

$$T_b = \frac{\int_A \rho C_p u T dA}{\int_A \rho C_p u dA}$$

Convective Heat Transfer Coeffient

$$h_c = -\frac{K_c\left[\frac{\partial T}{\partial r}\right]_{r=R}}{T_w - T_b}$$

Nusselt Number

$$Nu = \frac{h_c D}{K_c}$$

☐ DIFFUSION APPROXIMATION

$$q_r'' = -\frac{4}{3a_R}\nabla e_b = -\frac{16\sigma_{SB}T^3}{3a_R}\nabla T = -K_r\nabla T$$

$$K_r = \frac{16\sigma_{SB}T_w^3}{3a_R}$$

Using the perfect gas law,

$$k_r = \frac{16\sigma_{SB}\kappa}{3\sigma_{ph}P}T^4$$

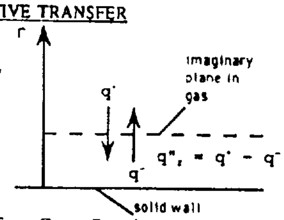
WHERE

a_R : Rosseland Mean Opacity
σ_{SB} : Stefan-Boltzmann Constant
σ_{ph} : Photon Collision Cross Section per Molecule
κ : Boltzmann's Constant
P : Gas Pressure
T : Gas Temperature

☐ APPROXIMATION BY USING 1-D EQUATION OF RADIATIVE TRANSFER

$$i(r) = \int_0^r a\ I(r')\exp(-a(r-r'))\,dr'$$

$$q''_r = q^+ - q^- = \pi\,(i^+(r) - i^-(r))$$



WHERE

$i^+(r)$: Radiation Intensity in the Positive Direction(From Gas to Boundary)
$i^-(r)$: Radiation Intensity in the Negative Direction(From Boundary to Gas)
I(r) : Source Function ($=\sigma T^4/\pi$)

Nusselt number & Prandtl number

$$Re = \frac{\rho u D}{\mu(T_b, T_1)}$$
$$Pr = \frac{\mu(T_b)C_p(T_b)}{K_c(T_b)}$$

(III) Pressure Drop

Compressible Flow

$$\Delta P = \frac{RG^2T_m}{P_m}\left(ln\frac{\rho_1}{\rho_2} + \frac{2f\Delta Z}{D}\right)$$
$$R = \frac{C_P(\gamma - 1)}{\gamma}$$
$$G = \rho_1\left(\frac{V_1 + V_2}{2}\right)$$
$$T_m = \frac{T_1 + T_2}{2}$$
$$P_m = \frac{P_1 + P_2}{2}$$

Incompressible Flow

$$\Delta P = 2f\frac{\Delta Z}{D}\rho_1 V_1^2\left(\frac{T_1 + T_2}{2T_1}\right) + \rho_1 V_1^2\left(\frac{T_2}{T_1} - 1\right)$$
$$f = 0.0014 + \frac{1}{8}Re^{-0.32}$$

(II) Nusselt Number Correlations

(1) Colburn Equation

$$Nu = 0.023 Re^{0.8} Pr^{\frac{1}{3}}$$

(2) Dittus-Boelter Equation

$$Nu = 0.023 Re^{0.8} Pr^{0.3}$$

(3) Sieder-Tate Equation

$$Nu = 0.027 Re^{0.8} Pr^{\frac{1}{3}} \left(\frac{\mu_b}{\mu_w}\right)^{0.14}$$

(4) Petukov Equation

$$Nu = \frac{RePr}{X}\left(\frac{f}{2}\right)$$

$$X = 1.07 + 12.7\left(Pr^{\frac{2}{3}} - 1\right)\left(\frac{f}{2}\right)^{\frac{1}{2}}$$

$$f = 0.0014 + \frac{1}{8} Re^{-0.32}$$

(5) Karmon-Boelter-Martinelli Equation

$$N_u = \frac{RePr\sqrt{\frac{f}{2}}}{0.833\left(5Pr + 5ln(5Pr + 1) + 2.5ln\left(Re\frac{\sqrt{\frac{f}{2}}}{60}\right)\right)}$$

$$f = 0.0014 + \frac{1}{8} Re^{-0.32}$$

Axial Distance Correction

$$Nu(x) = Nu\left(1.957\left(1 + \frac{x}{D}\right)^{-0.35}\right)\sqrt{\frac{T_b}{T_w}}$$

$$Nu(x) = Nu\left(1 + \frac{2ln\frac{Pr}{0.7}}{\frac{x}{D}}\right)$$

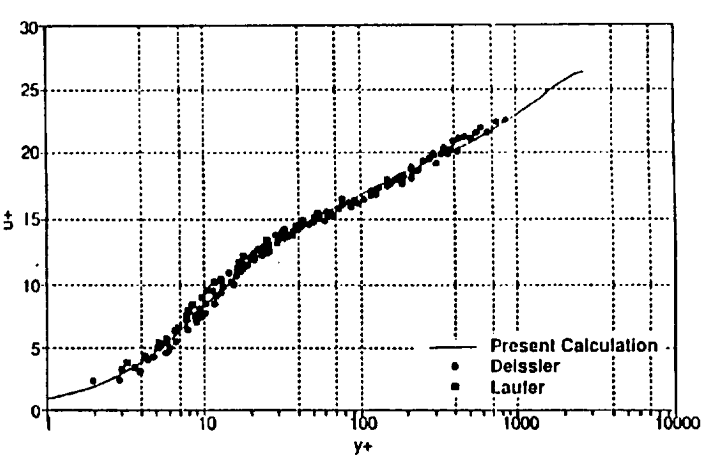


Figure 6.2 Velocity distribution for a fully developed turbulent flow in tube. (Re=1.6 E+4)

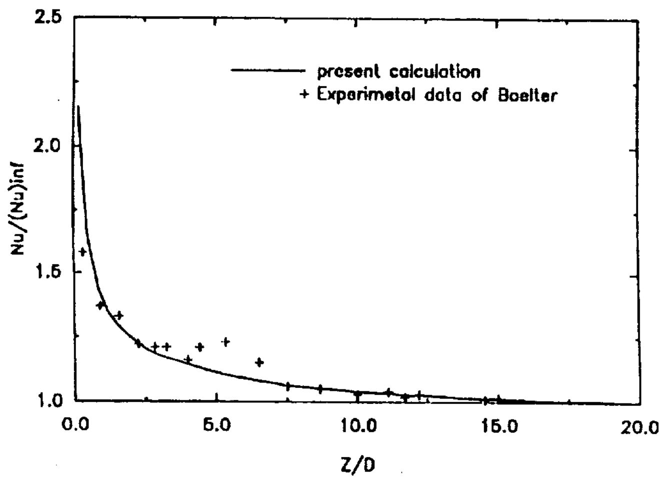


Figure 6.17 Nusselt number vs. axial position for a developing isothermal pipe flow at a Reynolds number of 53000. $(Nu)_{inf}$ is the Nusselt number evaluated at Z/D=20.

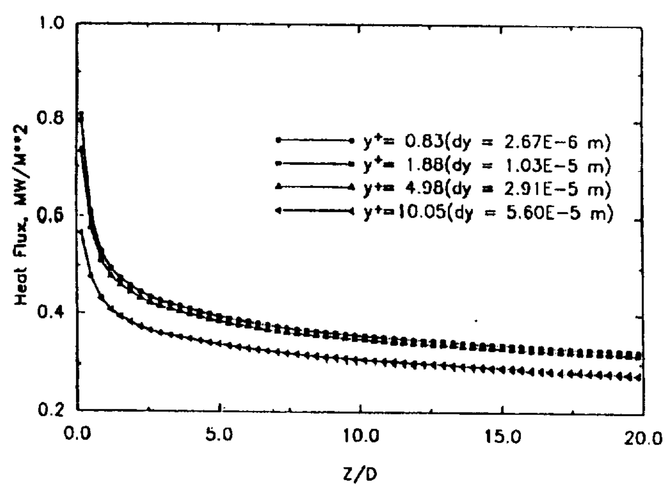


Figure 6.15 Heat transfer rates obtained by Navier-Stokes solver for various boundary cell size. A 60x60 grid is used. (T_{in}=4000 K, T_w=1800 K, P_{in}=1 atm, and P_{out}=0.5 atm)

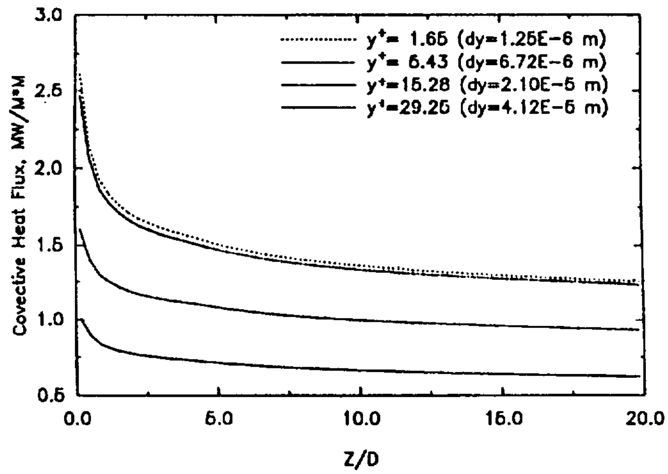


Figure 6.16 Heat transfer rates obtained by Navier-Stokes solver for various boundary cell size. A 60x80 grid is used. (T_{in}=4000 K, T_w=1800 K, P_{in}=10 atm, and P_{out}=9.5 atm)

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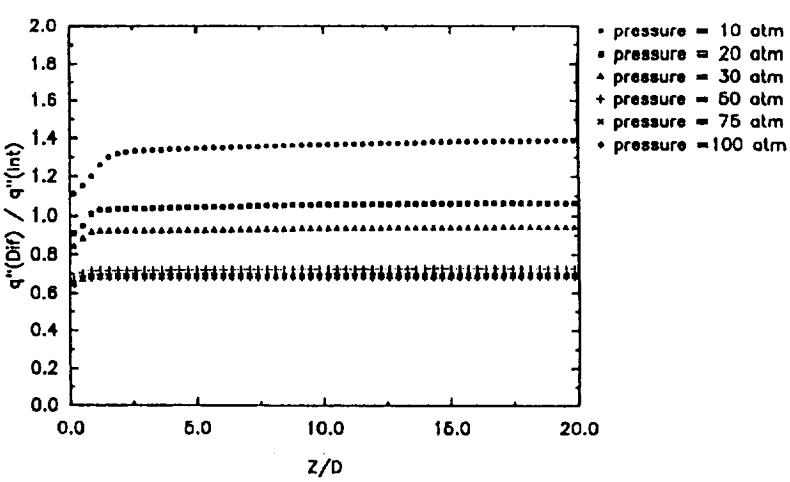

Figure 6.19 Comparative result between diffusion approximation and 1-D integral approximation for varying the gas opacity due to different flow conditions.

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N93-26953

NUCLEAR ENGINE SYSTEM SIMULATION (NESS) VERSION 2.0

- OVERVIEW -

22 JANUARY 1992

PRESENTED BY:
DENNIS G. PELACCIO AND CHRISTINE M. SCHEIL
SCIENCE APPLICATIONS INTERNATIONAL CORPORATION
ALBUQUERQUE, NM 87123
AND
LYMAN J. PETROSKY
WESTINGHOUSE ELECTRIC CORPORATION
MADISON, PA 15663

PRESENTED AT:
1992 NUCLEAR PROPULSION - TECHNICAL INTERCHANGE MEETING
NASA LEWIS RESEARCH CENTER
SANDUSKY, OH

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TOPICS

- BACKGROUND
- FEATURES
- COMPARISONS
- CONCLUDING REMARKS

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BACKGROUND

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NUCLEAR THERMAL PROPULSION (NTP) ENGINE SYSTEM ANALYSIS PROGRAM DEVELOPMENT
- Overall Objective -

- Develop a Stand-alone, Versatile NTP Engine System Preliminary Design Analysis Program (Tool) to Support Ongoing and Future SEI Engine System and Vehicle Design Efforts

  - Perform Meaningful (Accurate), Preliminary Design Analysis - Tank to Nozzle
  - Have Flexibility:
    - -- To Handle a Wide Range of Design Options to Support Preliminary Design Activities
    - -- To Be Easily Upgraded in Terms of Analysis Capability
  - Be Available to the SEI Community, Possibly as an Industry Standard
  - Be Done Promptly and Efficiently
  - Initial Effort:
    - -- Focused on NERVA/NERVA Derivative, Solid-Core NTP Systems
    - -- Based on Upgrading SAIC's NTP ELES Design Code by Incorporating Westinghouse's ENABLER Reactor and Internal Shield Models

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NUCLEAR THERMAL PROPULSION (NTP) ENGINE SYSTEM ANALYSIS PROGRAM DEVELOPMENT

- Observations -

- **No NTP-Specific Code is Commonly Available for Use in SEI Propulsion and Vehicle Design Studies**
  - Versatile, Verified NTP Analysis Design Tool Could Be of Great Use to the Community
- **It Is Envisioned That NESS Is One Key Element in Developing a Robust (Industry Standard Type) Analysis Capability (Design Workstation) to Support NTP Development Into the 21st Century**
  - Enhancements in Terms of Additional Technology/Design Options and/or Analysis Capabilities Possible With the NTP ELES Model

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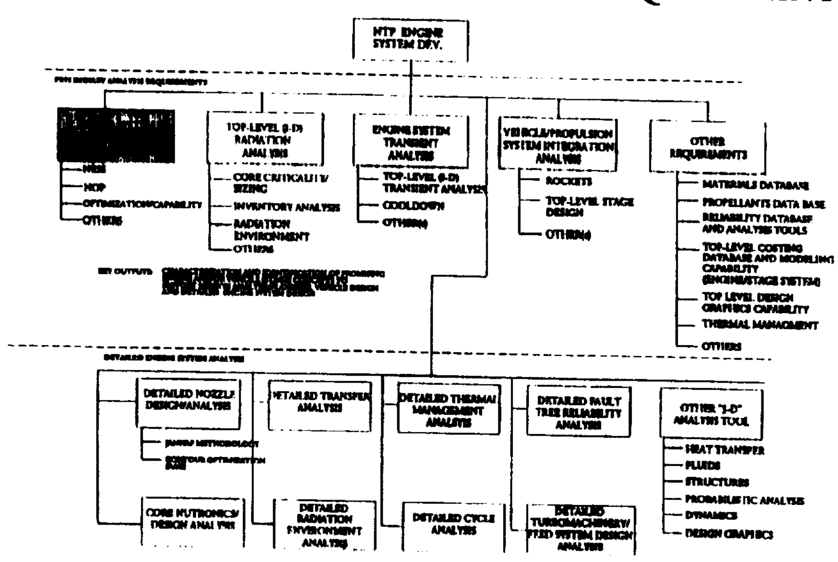
NUCLEAR THERMAL PROPULSION ENGINE DEVELOPMENT ANALYSIS CAPABILITY REQUIREMENTS



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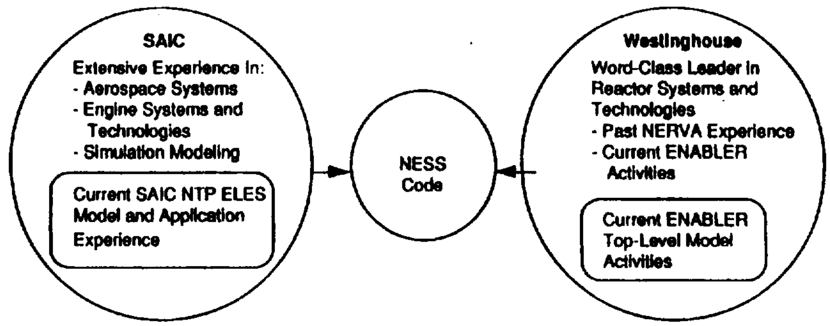
TEAM RESOURCES USED TO SUPPORT NESS DEVELOPMENT



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EXPANDED LIQUID ENGINE SIMULATION (ELES) COMPUTER MODEL
- Background -

- Its Major Objective is to Conduct Preliminary System Design Analysis of Liquid Rocket Systems and Vehicles
- Delivered by Aerojet in the Early 1980's (1981-1984) Under Sponsorship by the Air Force Rocket Propulsion Laboratory (Now Phillips Laboratory)
  - Over $1.2 Million Spent by the Air Force in Its Development
  - Available Through the Air Force
- ELES Has Been Well Distributed and Accepted Within the Propulsion Community for Preliminary Liquid Propulsion System Design Analysis
- ELES Draws on Past Experience and Knowledge From Aerojet and Others
  - Encompasses Aerojet Vast Engineering Base and Expertise in Liquid Propulsion
    - In-house Experience Included in the Model
  - Has Legacy to Experts Active in the Community

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EXPANDED LIQUID ENGINE SIMULATION (ELES) COMPUTER MODEL (Cont.)

- Background -

- ELES Model Uses Mechanistic as Well as Empirical Models of Components/Subsystems
- The Model Is Well Structured, User Friendly, Easily Modified, and Documented
- A High Degree of Verification has Been Done on the ELES Code

> - ELES Is a Comprehensive Industry Type, Standard Code Available to Perform Preliminary Steady-State Liquid Propulsion Design Analysis
>   - A key Starting Point in Initial NTP Engine System Development

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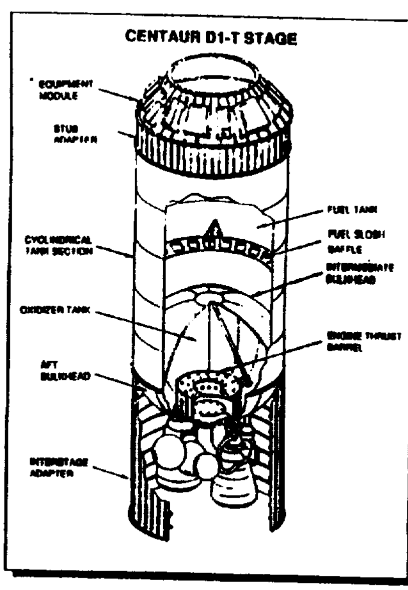
ELES VERIFICATION EXAMPLES

- N-II DELTA (DELTA 2ND STAGE)
- TRANSTAR (TITAN 3RD STAGE)
- CENTAUR/RL-10 DT-1 STAGE
- SPACE SHUTTLE MAIN ENGINE

CENTAUR/RL-10 D1-T VERIFICATION SUMMARY

	ACTUAL	CALC	ACTUAL/CALC
Turbine Pressure Ratio	1.337	1.299	1.029
Regen Jacket ΔT	418	503	0.83
Ox Pump Outlet Pressure	897	904	0.99
Fuel Pump Outlet Pressure	990	954	1.04
Engine System	665	634.9	1.05
TPA Weight	76.1	80.6	0.94
Stage Dry Weight	4048	3952	1.02
Stage Burnout Weight	4602	4384	1.05
Stage Length	360	357.3	1.01
Engine Performance	444	444.6	1.00



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NESS PROGRAM DEVELOPMENT EVOLUTION

FY 1989-90
INITIAL SAIC
NTP-ELES DEVELOPMENT

FY 1991
NESS-VERSION 1.0
- ENABLER I ENGINE
SYSTEM

FY 1992
NESS-VERSION 2.0
- ENABLER II ENGINE
SYSTEM

FY 1993
NESS PUBLIC RELEASE
THROUGH COSMIC
- PC and Vax Versions

- Extensive Anchor/Verification of the Program Performed For Each Development Phase
- BETA Versions of NESS - Versions 1&2 Are Successfully in Operation At NASA Lewis

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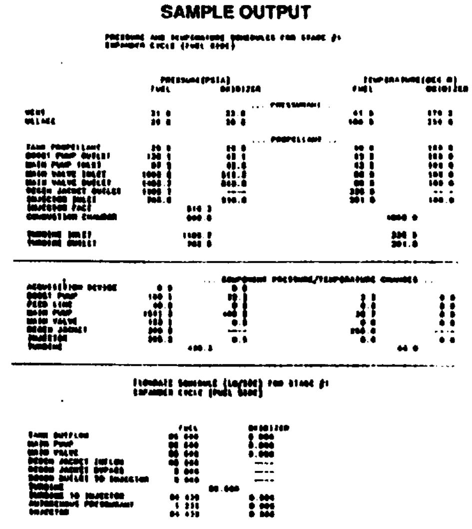
PAST NTP ELES ANALYSIS CODE MODIFICATIONS AND VERIFICATIONS

- ELES-NTP Version Developed and Verified
  - Modifications Performed
    - Incorporation of H_2 and CO Property Tables
    - Monopropellant Turbopump-fed System Modifications
    - Reactor Weight and Dimension Correlations Added
    - Off-Design Engine Operation Capability
  - Verification Conducted
    - Rocketdyne Performance and Weight Data
    - Westinghouse NERVA Data
    - Compared with NASA 90-Day Study Input
  - Much Developed Under SAIC In-House Fund Sponsorship

SAMPLE OUTPUT



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GENERAL NTP ENGINE SYSTEM FEATURES MODELED BY NESS

- Incorporates a Near-Term Solid-Core NERVA/ NERVA-Derivative Reactor Designs
  - Westinghouse ENABLER I&II NTP Reactor Designs
    - Strong Westinghouse R-1 Reactor Design Legacy

- Incorporates State-of-the-Art Propulsion System Technologies and Design Practices

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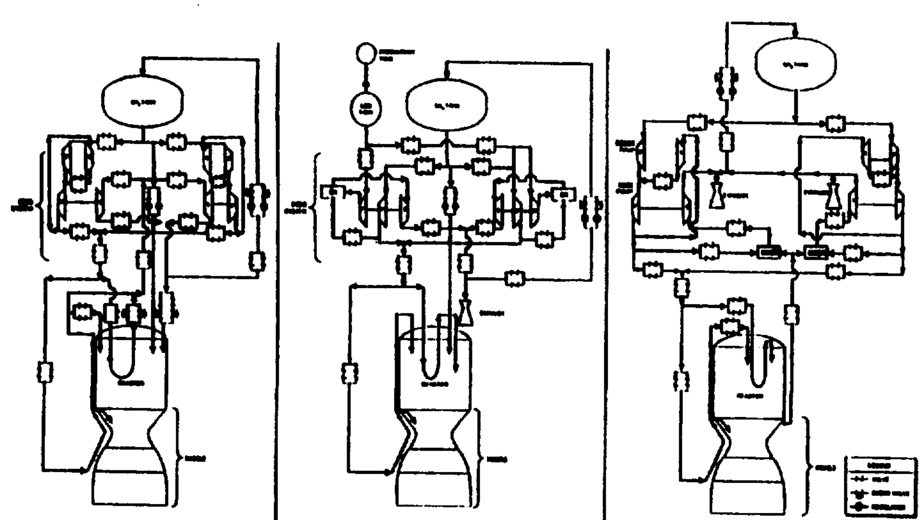
REPRESENTATIVE NTP EXPANDER, GAS GENERATOR, AND BLEED ENGINE SYSTEM CYCLES MODELED BY NESS



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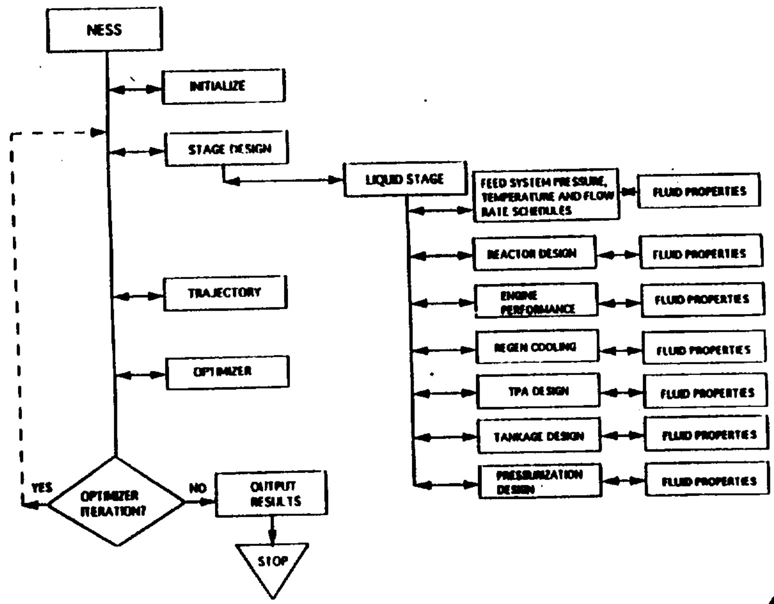
NESS PROGRAM OVERVIEW



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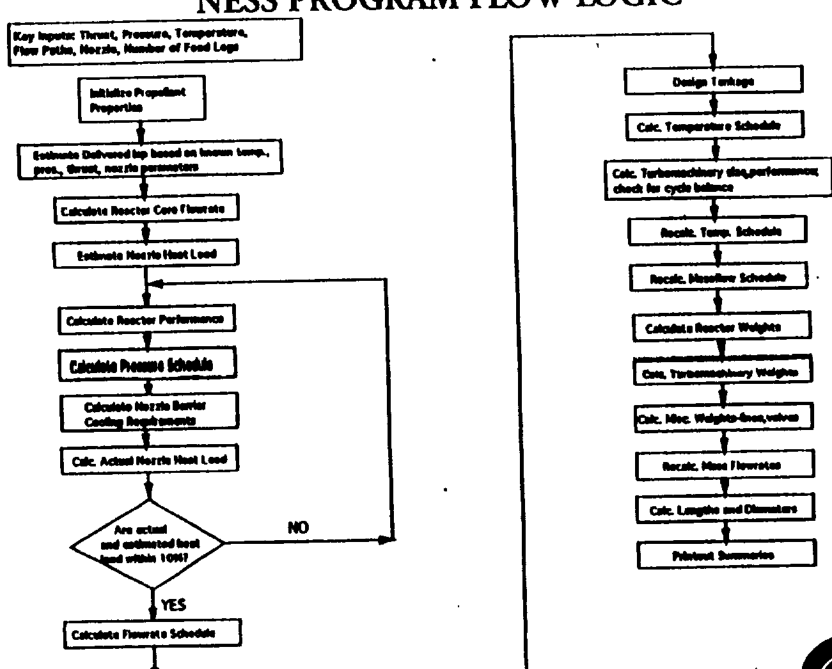
NESS PROGRAM FLOW LOGIC



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TOP-LEVEL KEY NESS FLAGS AND INPUT VARIABLES

Module	Variable	Value	Result
General Input:			
Cycle Type	KCYCLE	=1	Gas Generator Cycle
		=3	Expander Cycle
		=7	Bleed Cycle
Thrust Level	FVAC	–	Set thrust level
Chamber Pressure	PC	–	Set pressure
Chamber Temperature	TCHAMBER	–	Set temperature
Reactor Type	IREACTR	=1	Use ENABLER I
		=2	Use ENABLER II
Choose double-cone?	IDBLEXN	0,1	If =1, double-cone used
Booster pump design?	ITRATE	0,1	If=1, booster pump design
Bleed cycle solver	IXOLVE	=0	Input bleed flow fractions
		=1	Input turbine inlet temp.
Input burn time?	IUSRBRN	0,1	If =1, input burn time
User-defined TPA?	ISTSET	0,1	If =1, input TPA values
Nozzle:			
Exit area ratio	EPS	–	Set exit area ratio
Use extension?	IEXNOZ	0,1	If =1, use extension
Use 3-portion nozzle?	NOZTYP	0,1	If =1, use 3-portion nozzle
Attach area ratio 1	EPSATT	–	Set ext. attach area ratio
Attach area ratio 2	EPSAT2	–	Set 2nd ext attach area ratio
Nozzle cooling	KCOLNZ	=2	Regen cooling of nozzle ext.
		=3	Trans-regen cooling
		=4	Radiation cooling
		=5	Film cooling (GG cycle only)
Regen Cooling:			
Turbine Bypass Ratio	BYPTUR	–	Set turbine bypass flow
		=1.0	Tie tube flow drives turbine
Barrier Temp. Fraction	DIFTBF	–	Set barrier temperature
Reactor:			
Reactor flow paths	CONFIG	=1	Original flow paths
		=2	Tie tube flow drives turbine
Fuel Type	FTYPE	=1	Graphite fuel
		=2	Composite fuel
		=3	Carbide fuel
Support Pattern	SPAT	=2:1	Set support pattern
		=3:1	
		=6:1	
Nozzle Flow Percent	NFP	–	Set nozzle tube flow
Fuel Sealing Factor	FALPHA	–	Set fuel sealing factor
Tankage:			
Tank Type	NCTNK	=0	Tandem tankage
		=1	Non-concentric tankage
Pressurization Method	KOASPL	=0	Cold helium or cold GG
		=1	Autogenous
Turbopump Assembly:			
Pump Configuration	ICNFIG	=1	Gearbox
		=2	Single shaft TPA
		=3	Twin TPA in series
		=4	Twin TPA in parallel
		=5	Multiple feed leg TPA
Turbopump Type	IPTYPE	=0	Centrifugal Pumps
		=1	Axial Pumps
Use Boost Pumps?	IBPFL	0,1	If =1, use fuel boost pump
Number of Feed Legs	NTPA	–	Set number of TPA feed legs

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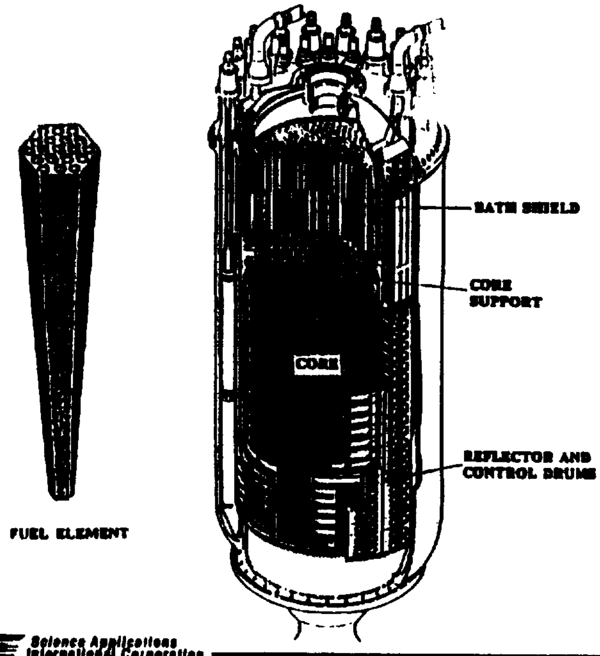
ENABLER (NERVA TYPE) NUCLEAR THERMAL ROCKET ENGINE



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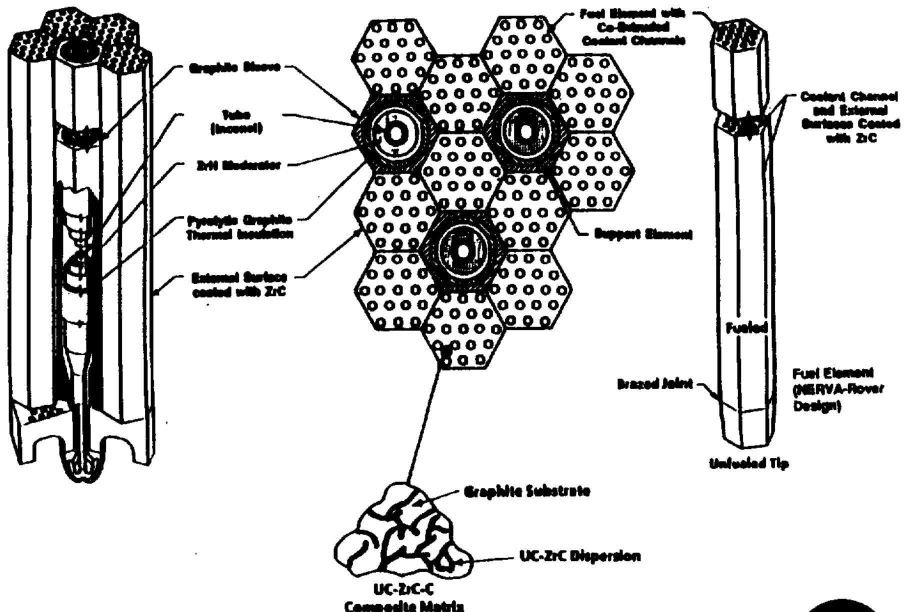
PRISMATIC FUEL ELEMENTS AND SUPPORTS



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REACTOR FUELD AND SUPPORT ELEMENT PARAMETERS

Fuel Element Composition	Graphite	Composite	Carbide
Temperature Range (°K)	2200-2500	2500-2900	2900-3300
Fuel	Coated Particle	UC . ZrC Solid Solution and Carbon	(U,Zr) C Solid Solution
Coating	ZrC	ZrC	—
Unfueled Support Element Composition	Graphite	ZrC-Graphite Composite	ZrC
Unfueled Element Coating	ZrC	ZrC	—

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REACTOR PARAMETERS AS A FUNCTION OF THRUST LEVEL

Thrust (lbf)	15,000	25,000	>50,000
Reactor Power Range	275-400	460-670	920-6700
Fuel and Support Element Length (inch)	35	35	52
Pressure Vessel Length (inch)	82.6	84	101.6
Fuel Element Power (MW)	0.629	0.808	1.20
Relative Fuel Element Power Density	0.778	1.0	1.0
Ratio of Fuel Elements (N) to Support Elements	2:1	3:1	6:1

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INTERNAL SHIELD SIZING

- Sized to Meet Radiation Leakage Requirements Established for the NERVA Program
- Radiation Leakage Limits at a Plane 63 Inches Forward of the Core Center

Type of Radiation	Radiation Leakage Limits Within Pressure Vessel Outside Radius
Gamma Carbon KERMA Rate	1.8×10^7 Rad(c)/hr
Fast Neutron Flux	2.0×10^{12} n/cm^2-sec
Intermediate Neutron Flux	3.0×10^{12} n/cm^2-sec, 0.4 eV $\leq$ En ≤ 1.0 MeV
Thermal Neutron Flux	6.0×10^{11} n/cm^2-sec En < 0.4 ev

- Materials and Thickness
  - For Thrust Level ≥ 50,000 lbf
    - -- 12.6 Inches of Borated Aluminum Titanium Hydrid (BATH)
    - -- 1.3 Inches Lead
  - For Thrust Levels < 50,000 lbf, BATH and Lead Thickness Slightly Reduced Due to Lower Core Power Density

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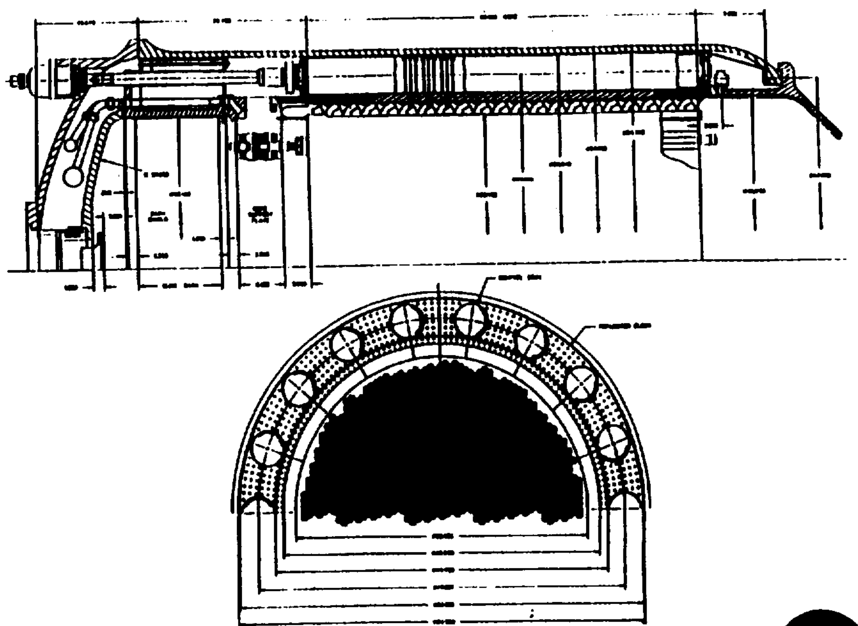
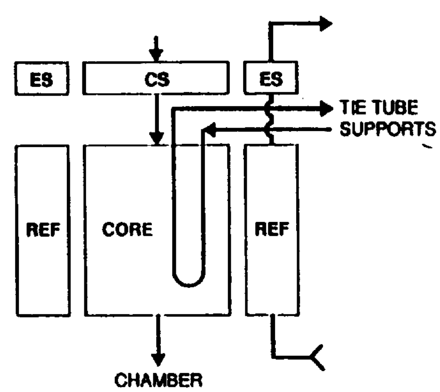
LAYOUT DRAWING OF THE R-1 REACTOR



REACTOR THERMAL MODEL

HEAT GENERATION	
Core	~1,500 MW
Tie Tubes	3-7%
Reflector	1-2%
Central Shield	~0.2%
Ext. Shield	~0.03%

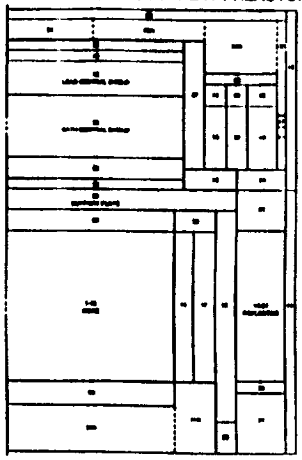
COMPONENT BLOCK DIAGRAM



REACTOR WEIGHT MODEL

- Based On R-1 Engine Design
- 53 Reactor Regions Itemized
- Masses Adjusted With Changes In Core Size

MODELED REGIONS IN THE R-1 REACTOR



REACTOR WEIGHT MODEL REGIONS (EXAMPLE)

REGION NUMBER	REGION DESCRIPTION	MATERIAL
1 - 15	Core	Fueled Element Unfueled Element Pyro Sleeve A-286 SS-304 Hydrogen
16	Core Periphery	Graphite-G Pyrofoil ZrC ([illegible]% Dense) TZM Moly Hydrogen
17	Lateral Support	P03 Graphite Z1A Graphite Pyrofoil Hydrogen
18	Structure	P03 Graphite Al-6061 A-286 Hydrogen

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NON-NUCLEAR AUXILIARY COMPONENT WEIGHTS

- **Updated Weight Correlations Incorporated for the Following Auxiliary Components:**
  - Instrumentation
  - Pneumatic Supply System
  - Reactor Cooldown Assembly
  - Thrust Structure

- **Based on Past Work by TRW (1965) Which Developed Detailed Weight Correlations for Such Components Based on Evolving NERVA Designs**
  - Updated to Take Into Account Advances in Technology and Design Practices

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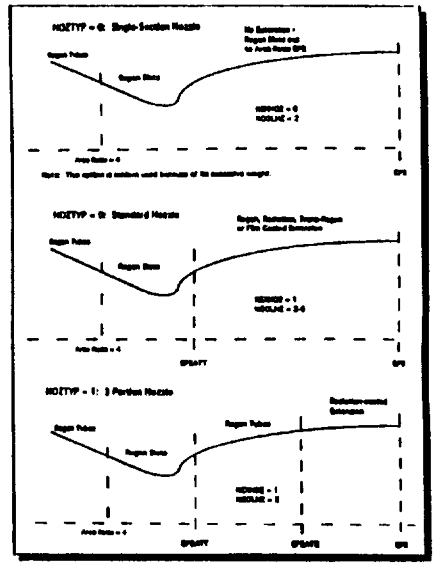
NESS NOZZLE DESIGN OPTIONS

- **STATE-OF-THE-ART NOZZLE DESIGN OPTIONS AVAILABLE**
  - Regenerative Cooled Slotted-Tube Construction, Radiation Cooled Extension
  - Initialized With Up-to-Date Materials
  - Capable of Analyzing Nonconventional Nozzle Designs
  - Translating and/or Gimbaling Nozzles Possible



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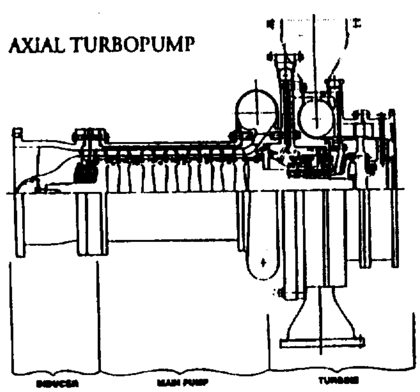
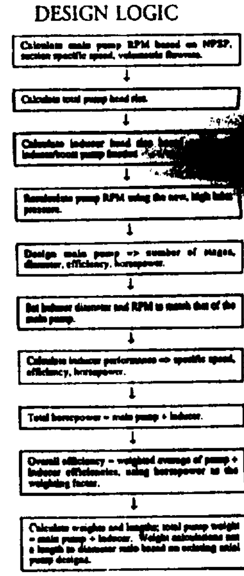
AXIAL TURBOPUMP DESIGN MODULE DEVELOPED AND INTEGRATED INTO NESS VERSION 2.0

AXIAL TURBOPUMP



- Design Correlations Draw on Past Axial Turbopump Decisions and Test
  - Liquid Rocket Engine Axial Flow Turbopumps, NASA SP-8125, April 1978
- Axial Trubopump Weight Model Anchored on:
  - Recent Rocketdyne Design Studies
  - Past Cermet NTP System Design Study

DESIGN LOGIC



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MAJOR NESS ENGINE SYSTEM ENGINEERING DESCRIPTION AREAS

- System Pressure, Temperature and Mass Flow Schedule
- Turbopump Design and Operation
- Nozzle Peformance Losses
- Regenitatively Cooled Nozzle Design
- Reactor Subsystem Design and Operation

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TYPICAL ENGINE SYSTEM DESIGN SUMMARY

ENGINE SUMMARY

EXPANDER CYCLE				
[illegible] II				
CENTRIFUGAL PUMPS				
THRUST LEVEL =	75000.0	lbf	[illegible]	N
CHAMBER PRESSURE =	1000.0	psia	[illegible]	kPa
CHAMBER TEMPERATURE =	4860.0	deg R	2700.0	deg K
NOZZLE EXIT AREA RATIO =	200.0		200.0	
NUMBER OF FEED LEGS =	1		1	
TOTAL PROPELLANT FLOWRATE =	83.1	lbm/s	37.7	kg/s
REACTOR				
COMPOSITE FUEL				
FUEL SCALING FACTOR	0.87		0.87	
REACTOR WEIGHT	[illegible]	lbm	[illegible]	kg
SHIELD WEIGHT	2434.3	lbm	[illegible]	kg
PRESSURE VESSEL DIA.	45.1	in	114.7	cm
PRESSURE VESSEL LENGTH	87.0	in	221.0	cm
CORE PROPELLANT MASS FLOW	80.7	lbm/sec	36.6	kg/sec
NOZZLE				
CONVERGING NOZZLE WEIGHT	[illegible]	lbm	76.0	kg
NOZZLE EXTENSION WEIGHT	[illegible]	lbm	27.6	kg
SECOND NOZZLE EXTENSION WEIGHT	[illegible]	lbm	[illegible]	kg
TOTAL NOZZLE WEIGHT	[illegible]	lbm	[illegible]	kg
AREA RATIO	200.0		200.0	
THROAT DIAMETER	7.5	in	19.0	cm
EXIT DIAMETER	[illegible]	in	[illegible]	cm
NOZZLE LENGTH	163.0	in	414.1	cm
DELIVERED VACUUM ISP	[illegible]	sec	[illegible]	N-sec/kg
DELIVERED THRUST	75000.0	lbf	[illegible]	N
TURBOPUMP ASSEMBLY (TOTAL FOR ALL FEED LEGS)				
MAIN PROP. TURBOPUMP WT	[illegible]	lbm	[illegible]	kg
PROPELLANT BOOST PUMP WT	[illegible]	lbm	29.2	kg
[illegible] PUMP WEIGHT	0.0	lbm	0.0	kg
TPA IGNITION WEIGHT	[illegible]	lbm	[illegible]	kg
BLEED LINE/VALVE WEIGHT	0.0	lbm	0.0	kg

MISC. HARDWARE WEIGHTS				
THRUST MOUNT	[illegible]	lbm	[illegible]	kg
SUPPORT HARDWARE	[illegible]	lbm	[illegible]	kg
ENGINE LINES	[illegible]	lbm	[illegible]	kg
MAIN VALVE	[illegible]	lbm	173.0	kg
GIMBAL + POWER SUPPLY	[illegible]	lbm	[illegible]	kg
MARGIN (2.0%)	83.1	lbm	37.7	kg
TOTAL NONNUCLEAR WEIGHT	[illegible]	lbm	[illegible]	kg
TOTAL ENGINE SYSTEM				
TOTAL ENGINE WEIGHT	[illegible]	lbm	[illegible]	kg
TOTAL ENGINE WEIGHT WITHOUT SHIELD	[illegible]	lbm	[illegible]	kg
THRUST/WEIGHT RATIO WITH SHIELD	[illegible]	lbf/lbm	[illegible]	N/kg
THRUST/WEIGHT RATIO WITHOUT SHIELD	7.2	lbf/lbm	[illegible]	N/kg
REACTOR SAFETY ROD WT.-LAUNCH ONLY	[illegible]	lbm	141.0	kg
TOTAL ENGINE LAUNCH WEIGHT	[illegible]	lbm	[illegible]	kg
TOTAL ENGINE LAUNCH WT. W/O SHIELD	[illegible]	lbm	[illegible]	kg
PUMP-OUT CONDITIONS				
PUMP-OUT THRUST	60000.0	lbf	[illegible]	N
PUMP-OUT CHAMBER PRESSURE	800.0	psia	[illegible]	kPa
PUMP-OUT ISP	[illegible]	sec	[illegible]	N-sec/kg
PUMP-OUT CHAMBER TEMPERATURE	4860.0	deg R	2700.0	deg K
OVERALL DIMENSIONS				
OVERALL ENGINE LENGTH =	[illegible]	in	[illegible]	cm
OVERALL ENGINE DIAMETER =	[illegible]	in	[illegible]	cm

Note: In Addition to Normal Flight Design/Operating Conditions Presented Pump Out Operating and Launch Weight Parameters are Given.

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SAMPLE DESIGN CASE SUMMARY

Case No./ Parameter	1	2	3	4	5	6	7	8
Cycle Type	Expander	Expander	Bleed	Gas Generator	Expander	Bleed	Gas Generator	Expander
Thrust Level (lbf/N)	75,000/ 333,600	75,000/ 333,600	75,000/ 333,600	75,000/ 333,600	75,000/ 333,600	35,000/ 155,700	250,000/ 1,112,000	75,000/ 333,600
Reactor Type	ENABLER I	ENABLER II	ENABLER II	ENABLER II	ENABLER II	ENABLER I	ENABLER I	ENABLER I
Reactor Fuel Type	Composite	Composite	Composite	Composite	Carbide	Composite	Composite	Composite
Chamber Pressure (psia/KPa)	1,000/ 6,895	500/ 3,348	500/ 3,348	500/ 3,348	1,000/ 6,895	500/ 3,348	500/ 3,348	1,000/ 6,895
Chamber Temperature (°R/°K)	4,860/ 2,700	4,860/ 2,700	4,860/ 2,700	4,860/ 2,700	5,580/ 3,100	4,860/ 2,700	4,860/ 2,700	4,860/ 2,700
Nozzle Area Ratio	500:1	200:1	200:1	200:1	500:1	200:1	200:1	500:1
No. of Propellant Feed Legs	2	2	2	2	2	1	3	2
Turbopump Type	Centrifugal	Centrifugal	Centrifugal	Centrifugal	Axial	Centrifugal	Axial	Axial
Reactor Fuel Scaling Factor	1.00	0.67	0.67	0.67	0.67	0.67	1.00	1.00

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NESS VERSION 2.0 OPERATING ENVIRONMENT

- Well Organized Worksheet to Initialize Your Design Are Provided
- Uses Improved Name List Input File
  - Each Input Variable is Defined
- Operates on VMS/VAX System
  - Over 30,000 Lines of Code
- Personal Computer Compatible Version is Available
  - Requirements
    - 486-33 MHz Computer
    - 6 MB RAM
    - 80 MB Hard Drive
    - Leheay Fortran with Extended Memory Required

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NESS PROGRAM USER'S GUIDE

Science Applications International Corporation

Copy No. __

Nuclear Engine System Simulation (NESS)

Volume 1 — Program User's Guide

Contract No. NAS3-25809

December 1991

Prepared for

NASA Lewis Research Center
Nuclear Propulsion Office
21000 Brookpark Road
Cleveland, OH 44135

Prepared by

Science Applications International Corporation
21151 Western Avenue
Torrance, CA 90501

SAIC

CONTENTS

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COMPARISIONS

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CYCLE PARAMETER COMPARISON*
- 75,000 lbf, ENABLER I, Expander Cycle -

Parameter	Rocketdyne	SAIC - ELES NTP	SAIC NESS
Total Flowrate (kg/s)	36.7	36.9	37.27
Pump Discharge Pres. (psia)	1,544	1,538.3	2,298.3
Turbine Flowrate, % Pump	50	50	50
Turbine Inlet Temp. (°K)	555.6	555.3	622.3
Turbine Inlet Pres. (psia)	1,412	1,416.8	1,969.0
Turbine Pressure Ratio	1.25	1.295	1.739
Reactor Inlet Pres. (psia)	1,130	1,255.4	1,132.1
Reactor Power, (MW)	1,645	-	1,587
Reactor Core Flowrate (kg/s)	36.7	36.9	36.2
Nozzle Chamber Temp (°K)	2,700	2,700	2,700
Nozzle Chamber Pres. (psia)	1,000	1,000	1,000
Nozzle Exit Diameter (m)	4.15	4.15	4.22
Nozzle Expansion Ratio	500	500	500
Specific Impulse-Vac (sec)	923	922.8	912.9
Pump Speed (rpm)	37,500	34,913	40,583

* Rocketdyne uses their Mark 25 type axial turbopump (4 stages); SAIC ELES-NTP used a single-stage centrifugal pump; SAIC NESS, Sample Case No. 8, uses a 5-stage axial pump.

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ENGINE SUBSYSTEM WEIGHT COMPARISON*
- 75,000 lbf, ENABLER I, Expander Cycle -

Parameter	Rocketdyne	SAIC ELES-NTP	SAIC NESS
Specific Impulse - Vac (sec)	923	922.8	912.9
Reactor (kg)	5,824	5,823	4,783
Internal Shield (kg)	—	1,523	1,108
Nozzle Assembly (kg)	440	421	535
Turbopump Assembly (kg)	304	104	221
Nonnuclear Support Hardware (kg) - Lines, Valves, Actuators, Instrumentation Thrust Structure	1,815	1,264	1,493

* Rocketdyne uses their Mark 25 type axial turbopump (4 stages); SAIC ELES-NTP used a single-stage centrifugal pump; SAIC NESS, Sample Case No. 8, uses a 5-stage axial pump.

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EFFECT OF WALL TEMPERATURE ON PERFORMANCE*

Wall Temperature (°R)	Barrier Temperature (°R)	Isp (Sec.)	Fuel Film Cooling Fraction
1460	1630	912.9	0.03
1800	2106	915.9	0.03
2000	2429	917.5	0.02
2400	2892	919.4	0.02
2800	3418	921.2	0.02
3000	3651	921.9	0.02
3200	3864	922.4	0.02

* Core Temperature = 4860°R (2700°k)

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DESIGN CASE COMPARISION OBSERVATIONS

- NESS Design Exhibits 1% Lower Peformance Than Other Designs
  - NESS Model More Accurately Predicts Nozzle Cooling Losses-Upstream Film Cooling Required to Meet Maximum Wall Temperature Requirements
- Integrated Reactor/Engine System Design Effects Accounted for in the NESS Design
  - Sized to Take Into Account Heat Captured by the Coolant Before It Enters the Reactor
  - Corresponds to Some Difference in Cycle Pressures, Temperatures, and Turbopump Operating Parameters
- Other Weight Differences From Improvements in NESS Weight Correlations
  - 3-Section Nozzle Design
  - Non-Nuclear Auxiliary Components
  - Update H_2 Properties

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CONCLUDING REMARKS

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CONCLUDING REMARKS

- The NESS Preliminary (ENABLER I&II) Design Analysis Program Characterizes a Complete Near-Term Solid-Core NTP Engine System in Terms of Performance, Weight, Size, and Key Operating Parameters for the Overall System and Its Associated Subsystem
  - Incorporates Numerous State-of-the-Art Engine System Technology Design Options and Design Functions Unique to NTP Systems
  - Extensively Verfied and Documented
- The NESS Program is Deemed Accurate to Support Future Preliminary Engine and Vehicle System Design and Mission Analysis Studies
  - NESS Has Been Successfully Operated and Checked Out at NASA Lewis
- Future Recommendations:
  - Incorporate Other NTP Reactor Types
    - -- Particle Bed
    - -- Pellet Bed
    - -- Low Pressure
    - -- Wire Core
    - -- In situ Propellant Based Reactor Designs
  - Incorporate a Radiative Heating Model
  - Update the Material Library
  - Upgrade the NESS Performance Prediction Module

- NESS Devleopment Is One of Many Key First Steps Required to Support NTP Development
- It Is Envisioned that NESS Will Be One Key Element of an Advanced NTP Engine System Design Workstation

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N93-2695

10/13/92

SAFSIM OVERVIEW

by Dean Dobranich

Sandia National Laboratories
Nuclear Technology Department

October 1992

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Sandia National Laboratories

An overview of the SAFSIM computer program is provided in this presentation.

SAFSIM is being developed at Sandia National Laboratories and is currently funded by the Air Force SNTP program.

Slide 1

SAFSIM
Systems Analysis Flow SIMulator

an "Engineering" computer program to simulate the integrated performance of complex systems involving fluid mechanics, heat transfer, and reactor dynamics

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SAFSIM is a general purpose, FORTRAN computer program to simulate the integrated performance of complex systems involving fluid mechanics, heat transfer, and reactor dynamics. SAFSIM provides sufficient versatility to allow the engineering simulation of almost any system, from a backyard sprinkler system to a clustered nuclear reactor propulsion system. SAFSIM is based on a 1-D finite element model and provides the analyst with approximate solutions to complex problems.

Although SAFSIM can be used to model specific components in detail, its major strength is the ability to couple multiple components together to investigate synergistic effects between components. This is important because, in general, a system of optimized components does not produce an optimum system. Non-lineararities in the physics can produce system performance that might not be expected from analysis of an isolated component.

Slide 2

Desired Program Attributes:

- ✔ versatile
- ✔ fast running
- ✔ robust
- ✔ quality assessed
- ✔ documented
- ✔ benchmarked (when possible)
- ✔ transportable (FORTRAN77)

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SAFSIM is being developed with versatility as its primary attribute. Thus, it can be used to assess the performance of a variety of user-defined systems on a consistent and unbiased basis.

Speed and robustness are also key attributes that are incorporated in the overall development goals of SAFSIM.

SAFSIM documentation, benchmarking, and quality assessment are ongoing activities.

SAFSIM has been run on a VAX8650, a Sun Spark station, and an HP9000 workstation in addition to a 486/25 PC on which it is being developed.

Slide 3

Basic Physics Modules

- Fluid Mechanics
- Structure Heat Transfer
- Reactor Dynamics

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Three basic physics modules are included in the current version of SAFSIM: (1) Fluid Mechanics (solution of the conservation equations governing single-phase fluid flow), (2) Structure Heat Transfer (solution of the heat conduction equation for solid structures), and (3) Reactor Dynamics (solution of the time-dependent equations governing nuclear reactor neutron density, including reactivity feedback and decay heat). These three physics modules are described more fully in the following charts.

Slide 4

Fluid Mechanics

- 1-D Finite Element Model
  - compressible thermal energy equation with advection/conduction/convection
  - compressible mechanical energy equation
- Multiple, user-specified, liquid or gas flow networks
- Single phase with ideal gas, polynomial, or user-supplied equation of state options
- Multiple gases with mixing models

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The fluid mechanics physics module is based on a 1-D finite element model and solves the conservation of mass, momentum, and energy equations for a single-phase fluid. Compressible or incompressible fluids can be simulated. Thermal and mechanical energy equations are solved iteratively to provide the solution to a total energy equation.

The 1-D finite elements can be connected in series or parallel to create any desired flow network. Multiple networks can be included to model, for example, a heat exchanger with gas on one side and liquid on the other.

The user can select the equation of state for the different fluids in all networks. Choices are: ideal gas, polynomial function of temperature (for incompressible fluids), and user-supplied. An interface is in place within SAFSIM to facilitate inclusion of a user-specified equation of state. Thus, an understanding of the internals of SAFSIM is not required to add an equation of state.

Mixing models are provided to allow simulation of multiple gases in a network. Thus, different gases can be tracked throughout a network and fluid properties for the mixture are automatically determined.

Slide 5

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Fluid Mechanics (continued)

- Porous media finite element
- Compressor/Pump element
- Special choked flow boundary element
- Distributed flow manifold element (with options for transpiration flow and tees)
- Super element capability
- Automatic K-factors for expansions and contractions
- Open or closed networks

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Special finite elements allow simulation of flow in porous media, compressors/pumps, and manifolds. Also, a special element allows implementation of a choked flow boundary to model a nozzle. The manifold element includes options to automatically account for transpiration flow (blowing/sucking conditions) and branching flows with respect to friction factors and heat transfer coefficients.

Super elements allow a series of finite elements to be combined into one "super element". This greatly increases computational speed for solution of the mechanical energy equation. Accuracy is also improved because a smaller matrix is produced, resulting in less round-off error.

K-factors are automatically determined for expansions and contractions if desired. Separate K-factors can be included for both forward and reverse flow for each finite element. Also, additional l/d can be added to account for bends, obstructions, etc... A relative wall roughness can also be included.

Both open and closed networks can be modeled.

Slide 6

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Fluid Mechanics (continued)

- Convection based on log-mean delta-T
- Upwinding with automatic determination of upwind factors based on Peclet number
- Pressure, mass flow rate, temperature, zero heat flux, and mass fraction boundary conditions
- Three matrix solvers
  - Gauss-Seidel, iterative
  - Cholesky decomposition, direct
  - Gauss elimination, direct

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Convection heat transfer in the thermal energy equation is based on the log-mean temperature difference which increases accuracy, especially for low flow simulations. To accomplish this, a special technique was developed to allow the linear, 2-noded elements of SAFSIM to provide the accuracy of a higher order element with minimal extra computational expense.

Upwind elements are used for solution of the thermal energy equation. The optimum upwind factor is determined for each element based on the Peclet number, which provides a measure of advective dominance. Thus, problems that are advectively or conductively dominated can be simulated.

Boundary conditions for the fluid mechanics solution can be specified at any node in the network.

Three numerical solvers are provided to add robustness. The user can select a solver or let SAFSIM execute the three solvers in succession until a solution is achieved.

Slide 7

Structure Heat Transfer

- 1-D Finite Element Model
  - automatic timestep control
  - subtimesteps for each structure
- Automatic spherical, cylindrical, or rectangular geometry finite element generator via input if desired
- Temperature-dependent properties
- Automatic implicitness factors

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The structure heat transfer module is based on a 1-D finite element model and solves the heat conduction equation for solid structures (pipe walls, plates, fuel rods or particles, thermocouples,...). Automatic timestep control can be selected for each structure if desired and each structure can have its own subtimestep. Thus, structures with large time constants can run at large timesteps and are not forced to run at the small timesteps required of structures with much smaller time constants.

Although geometry input must be completely specified by the analyst, automatic mesh generation is provided for structures with spherical, cylindrical, or rectangular geometry.

Conductivity and specific heat can be temperature dependent if desired and several options are available for specifying property values, including tables, polynomials, and power laws.

The implicitness factor is automatically determined for all nodes of each structure, at each subtimestep. This ensures that the best accuracy is achieved for any given timestep.

Slide 8

Heat Transfer (continued)

- Multiple exchange surfaces for each structure finite element
- Extensive built-in HTC correlation library
  - laminar and turbulent flows
  - internal and external flow geometries
  - gases, liquids, and liquid metals
- Temperature, heat flux, and convective/radiative boundary conditions

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Each finite element can have multiple exchange surfaces. An exchange surface allows heat transfer between the structure and the coolant (via convection or radiation) or between different structures (via radiation or conduction). For example, a structure finite element representing a pipe wall may have one exchange surface to model forced convection heat transfer to a coolant flowing through the inside of the pipe and another exchange surface to model free convection to another coolant on the outside of the pipe. A third exchange surface could be added to model radiation to the outside coolant, if desired.

SAFSIM allows the analyst to select a HTC correlation for laminar flow conditions and another for turbulent flow conditions for each exchange surface. A built-in library contains over 90 correlations including internal and external flow geometries. Correlations for gases, liquids, and liquid metals are included. Also, an interface is provided to allow the analyst to easily add her own correlations.

Either temperature, heat flux, or convective/radiative boundary conditions can be used for each structure.

Slide 9

Reactor Dynamics

- Point (0-D) Kinetics Model with feedback
  - multiple reactors
  - adaptive timesteps
- Multiple feedback coefficients for fuel, moderator, control rods/drums ...
- User-specified precursor and decay heat groups (automatic concentration initialization if desired for steady state)
- Euler or fifth-order Runge-Kutta solvers

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The reactor dynamics physics module is based on a point (0-D) kinetics model and includes reactivity feedback and decay heat. Multiple reactors can be specified and multiple feedback coefficients are allowed for each reactor to account for all system interactions. The analyst has complete control over how the feedback coefficients are defined. Multiple reactors can be coupled via user defined feedback coefficients if desired. Also, special-purpose "control laws" can be added to the program to simulate reactor startup and shutdown transients. Adaptive timestep control can be employed. A source term also can be included.

Any number of delayed neutron groups and decay heat groups can be specified. Initial precursor concentrations can be input or calculated automatically by SAFSIM based on steady-state conditions.

Two solvers are available for integration of the reactor dynamics equations: (1) Euler, and (2) Runge-Kutta-Fehlberg (RKF). The analyst can switch between solvers during a problem if desired.

Slide 10

Miscellaneous

- Automatic steady-state option
- Function-controlled variables
- User-supplied subroutine interfaces:
  - functions
  - equation of state and fluid properties
  - heat transfer coefficients
  - reactor dynamics control laws
  - special-purpose input and output

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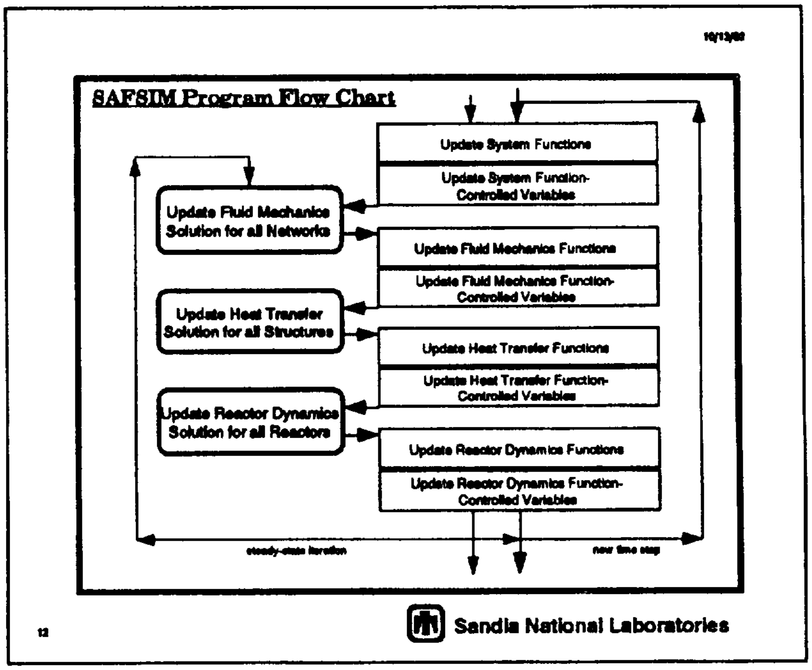
Although SAFSIM is a time-dependent computer program, it can be used to perform steady-state calculations. Two methods are available. The first method is to simply run a transient simulation until the time derivative terms are sufficiently small. SAFSIM offers a second method in which the time-derivative terms are set to zero and wall temperature iterations are performed to obtain consistency between the fluid mechanics and structure heat transfer physics modules. This automatic steady-state method can be combined with the first method if desired.

Function-controlled variables are a unique feature of SAFSIM that allow the analyst to specify most of SAFSIM's input variables as functions of any of it's output variables. An extensive library of mathematical functions is available within SAFSIM or the analyst can add his own. For example, flow lengths and areas can be specified as functions of structure temperature to simulate expansion effects.

SAFSIM provides 5 user-supplied subroutine interfaces to allow the analyst to tailor SAFSIM to problem-specific modeling needs. These interfaces streamline the process of adding special subroutines.

Slide 11



This chart provides a top level flow diagram of SAFSIM and indicates the computational sequence for both steady-state and transient analyses. The three physics modules, along with function-controlled variables and functions, are explicitly coupled to simulate the integrated performance of an entire system. Employing explicit coupling between the different physics modules (which all may have vastly different characteristic time constants) greatly increases program versatility. For very rapid transients the system timestep can be decreased to more tightly couple the different parts of the system.

Slide 12

Program Status

- **All physics modules operational**
- **Cleanup and enhancements in progress**
- **Benchmarking and documentation in progress**

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SAFSIM is a functioning computer program and is currently being used to solve a variety of problems at Sandia National Laboratories. However, SAFSIM is not complete and additional development is anticipated. Benchmarking and documentation are extremely big tasks that are expected to proceed concurrently with development.

Three manuals are planned to document SAFSIM: (1) a theory manual that will contain a description of the governing equations and numerics; (2) an input manual that contains a complete description of all of the input variables required to build an input model; and (3) an application manual that will provide benchmark problems in addition to several example problems. The input manual (Sandia National Laboratories internal report SAND92-0694) is complete and is being distributed as of October, 1992.

Slide 13

Future Enhancements

- Turbine element
- Built-in bandwidth minimizer for mechanical and thermal energy equations
- Blowdown tank option
- Structural Mechanics Physics Module
- LU decomposition with iterative refinement for large networks
- Restart capability

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To expand the class of problems for which SAFSIM is applicable, several enhancements are planned:

- addition of a turbine finite element
- a built-in bandwidth minimizer to increase the speed and accuracy of execution
- a boundary condition option to allow easy and quick simulation of tank blowdown
- a structural mechanics physics module based on a 1-D finite element model to predict the linear and nonlinear stress-strain behavior of solid structures, including plasticity and creep
- addition of an LU decomposition solver with iterative refinement to account for roundoff error when modeling extremely large networks
- restart capability to allow continuation of a problem

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Future Enhancements (continued)

- Kaganove solver for reactor dynamics
- 2-D tables and other special functions
- Pre- and post-processing (graphical)
- Dynamic temperature, mass flow rate, and density terms in fluid mechanics equations
- Upwind elements for the mechanical energy equation
- Liquid metal modeling options

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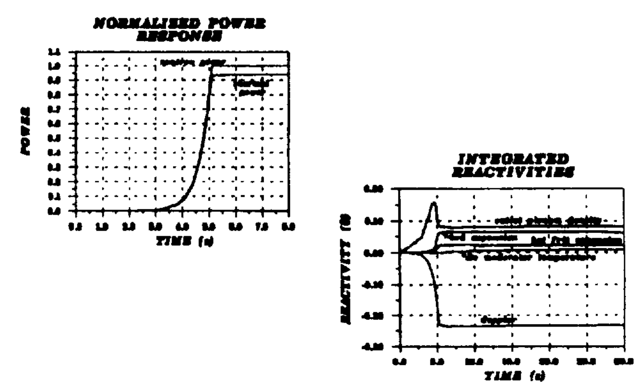
- addition of a Kaganove solver for long-duration reactor dynamics problems
- 2-dimensional table capability for functions along with many other special mathematical functions to enhance modeling capability
- graphical pre- and post-processing routines to facilitate input model building and output interpretation
- addition of all the dynamic terms in the fluid mechanics module
- addition of upwind elements to the mechanical energy equation to allow simulation of supersonic flow
- input options to allow simulations involving liquid metals (such as an accumulator and an electromagnetic pump)

Slide 15

Reactor System Startup Transient



These graphs show results of a SAFSIM application in which a system based on a particle bed reactor is brought to full power in 5 s. In addition to the particle bed fuel element, the moderator, reflectors, vessel, and control drums are modeled. The MIT-SNL control law is used to control the startup of the reactor. Feedback effects due to coolant density, fuel temperature, moderator temperature, bed and hot frit expansion, and control drum rotation are included in the model. The input model includes 64 fluid mechanics finite elements, 145 structure heat transfer finite elements, and 1 nuclear reactor. The problem was run on a 486/25 MH PC and required 4 minutes of CPU time to simulate 30 s of transient time. The average timestep was about 5 ms for the fluid mechanics. The same problem required 30 s of CPU time on an HP9000 workstation.

Slide 16

Current Applications of SAFSIM

- SNL
  - PBR System Startup/Shutdown Transients
  - PBR Element Performance
  - NET Simulation
  - ETS Simulation
- NASA
  - Simulation of NERVA NRX/EST System

17 Sandia National Laboratories

This chart (and the next) lists several applications of SAFSIM that are in progress and demonstrates the versatility of SAFSIM. Simulation of the NERVA NRX/EST system is the only application so far that has experimental data for an entire propulsion system for comparison to SAFSIM calculation. The model is being built at NASA/Lewis and currently contains 240 fluid mechanics finite elements. Agreement with experimental data is excellent.

Slide 17

Current Applications (continued)

- B&W
  - PBR Element Performance
  - Reactor System Performance
- Grumman
  - Propulsion System Control Studies

Sandia National Laboratories

SAFSIM applications in progress. (see preceding chart)

Slide 18

N93-26960

KINETIC - A SYSTEM CODE FOR ANALYZING NUCLEAR THERMAL PROPULSION ROCKET ENGINE TRANSIENTS

ELDON SCHMIDT, OTTO LAZARETH, AND HANS LUDEWIG

BROOKHAVEN NATIONAL LABORATORY
UPTON, NY 11973

PRESENTED AT:

NUCLEAR PROPULSION TECHNICAL INTERCHANGE MEETING

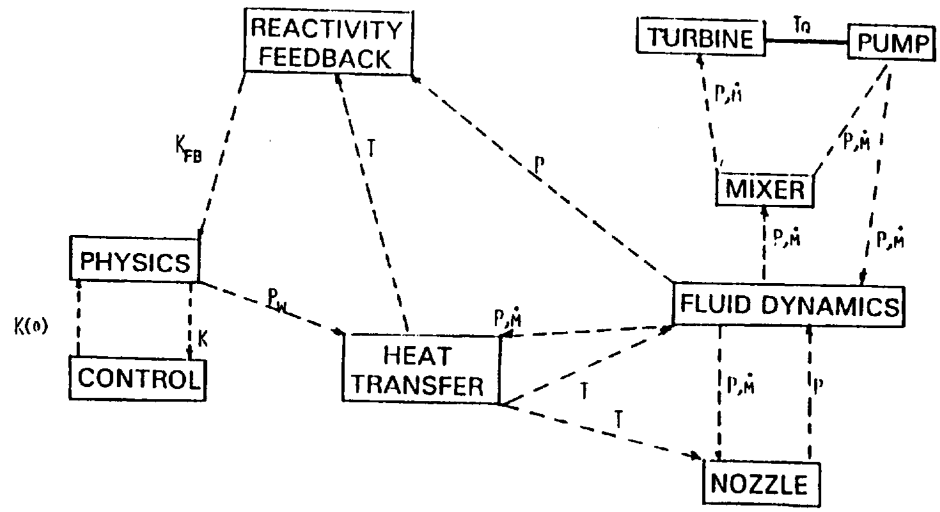
LEWIS RESEARCH CENTER

OCTOBER, 1992

OVERVIEW

- OUTLINE OF KINETIC CODE
- DESCRIPTION OF TEST PROBLEM
- SELECTED RESULTS
- CONCLUSIONS

KINETIC INFORMATION FLOW DIAGRAM



KINETIC NEUTRONIC EQUATIONS

Point kinetic equations

$$\dot{n} = \frac{\kappa(1-\bar{\beta})-1}{\tau} n + \sum_{i=1}^{6} \lambda_i C_i \tag{1}$$

$$\dot{C}_i = \frac{\beta_i \kappa n}{\tau} - \lambda_i C_i \qquad i = 1,\ldots,6 \tag{2}$$

Transformation (n,C) to (ω,Y)

$$\omega = \frac{\dot{n}}{n} \qquad\qquad Y_i = \frac{\lambda_i \tau C_i}{n} \tag{3}$$

Transformed equations

$$\tau\omega = \kappa(1-\bar{\beta})-1 + \sum_{i=1}^{6} Y_i \tag{4}$$

$$\dot{Y}_i = \kappa\beta_i\lambda_i - (\lambda_i + \omega)Y_i \qquad i = 1,\ldots,6 \tag{5}$$

Control equation

$$\tau\dot{\omega} = \dot{\kappa}(1-\bar{\beta}) + \sum_{i=1}^{6} \dot{Y}_i \tag{6}$$

KINETIC NEUTRONIC EQUATIONS
(PERIOD CONTROL ALGORITHM)

Let ω_T be the desired power trace and ω the actual trace. A simple linear restoration function can be written.

$$\dot{\omega} = \gamma(\omega_T - \omega) \tag{7}$$

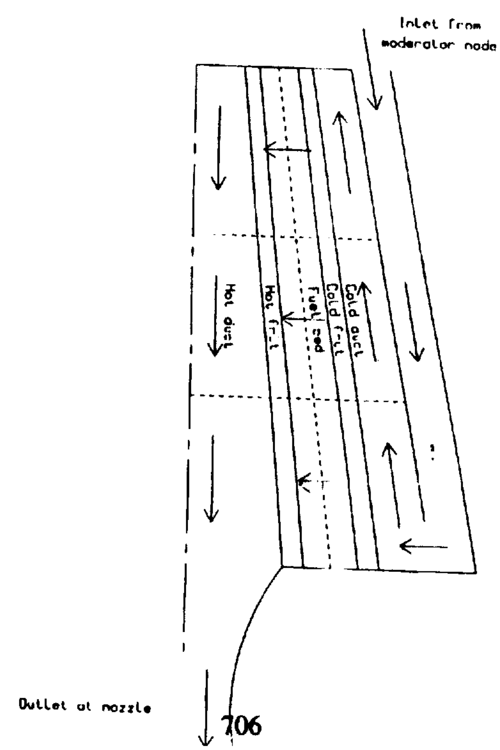
Eliminating κ from equations (6) using equation (4) and letting $\omega = \omega_T$, results in equation (8) (defining G).

$$\sum_{i=1}^{6} \dot{Y}_i = G(\lambda_i, \beta_i, \tau, \omega_T) \tag{8}$$

Equations (6),(7),and (8) result in an equation for κ in the measurable quantity ω and known quantities λ_i, β_i, τ and ω_T.

$$\dot{\kappa}(1-\bar{\beta}) = \tau\gamma(\omega_T - \omega) - G(\lambda_i, \beta_i, \tau, \omega_T) \tag{9}$$

FUEL ELEMENT COOLANT FLOW DIAGRAM



TURBO-PUMP/NOZZLE ALGORITHM

- GIVEN A PUMP ROTATIONAL SPEED DETERMINE PUMP (P,m) FROM PERFORMANCE CURVES.
- GIVEN CHAMBER TEMPERATURE CALCULATE NOZZLE (P,m).
- CALCULATE SYSTEM PRESSURE DROP.
- FROM THESE THREE RELATIONSHIPS (2 PRESSURES AND A FLOW)--OBTAIN TORQUE REQUIRED FOR PUMP FROM PUMP PERFORMANCE CURVES.
- FROM TURBINE PERFORMANCE CURVE AND INERTIAL EQUATION DETERMINE DELTA TORQUE BETWEEN PUMP AND TURBINE AND THUS CHANGE IN TPA SHAFT SPEED.
- REPEAT ABOVE STEPS FOR NEW TIME STEP.

KINETIC HEAT TRANSFER EQUATIONS PER NODE

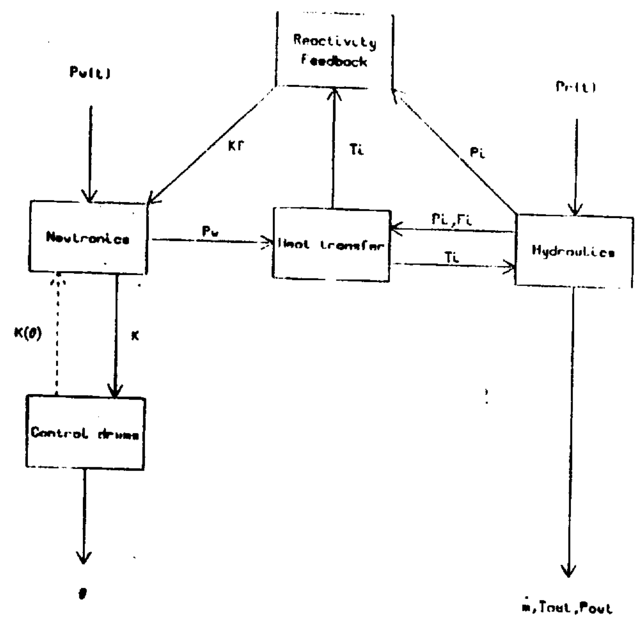
Temperature of solid (S)

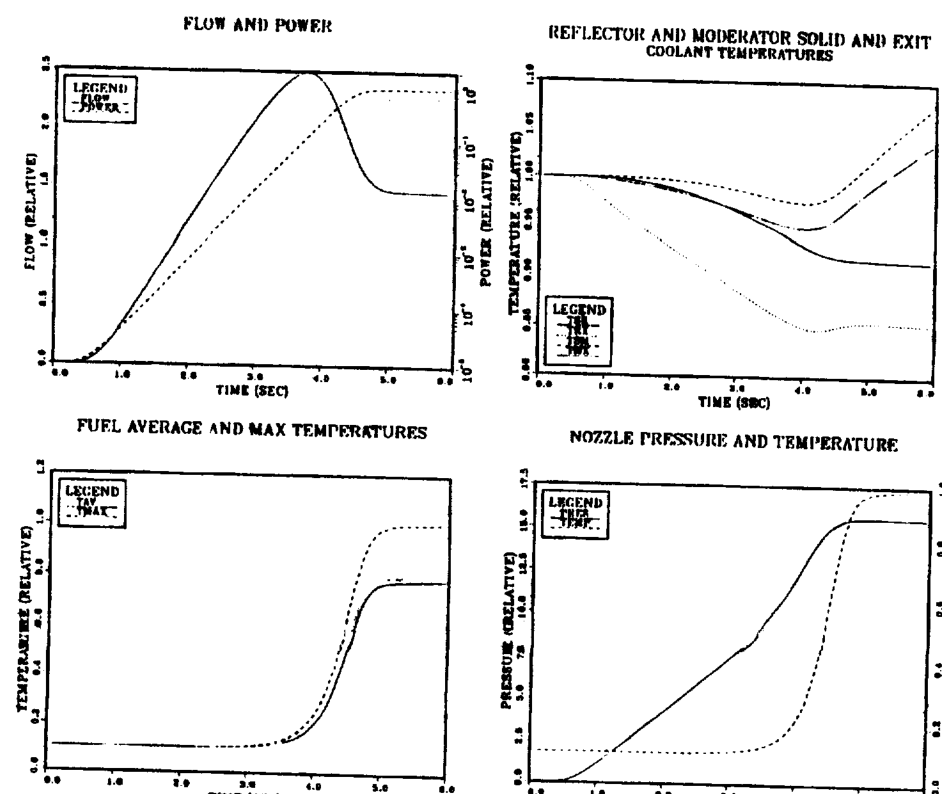
$$M_s C_p^s \dot{T}_s = Q - \dot{m} (H_{OUT} - H_{IN}) \qquad (1)$$

Temperature of coolant as a function of position (C)

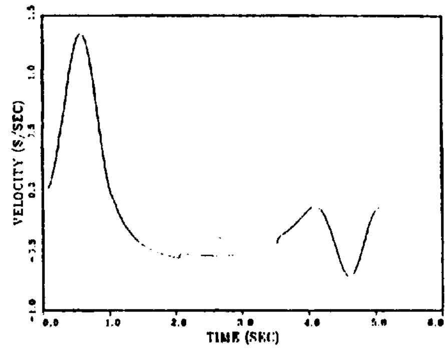
$$hP(T_s - T_c)dx = \dot{m}\, C_p^c dT_c \qquad (2)$$

TEST PROBLEM DIAGRAM

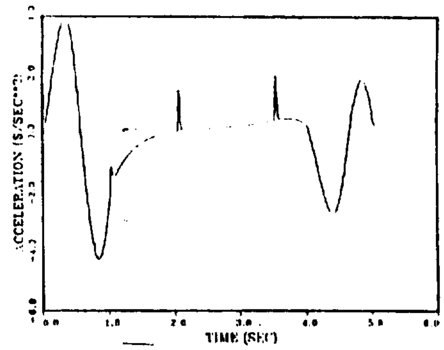




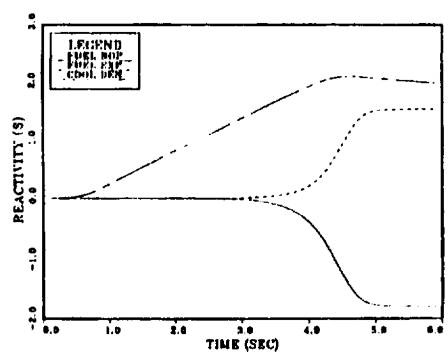
REACTIVITY VELOCITY



REACTIVITY ACCELERATION

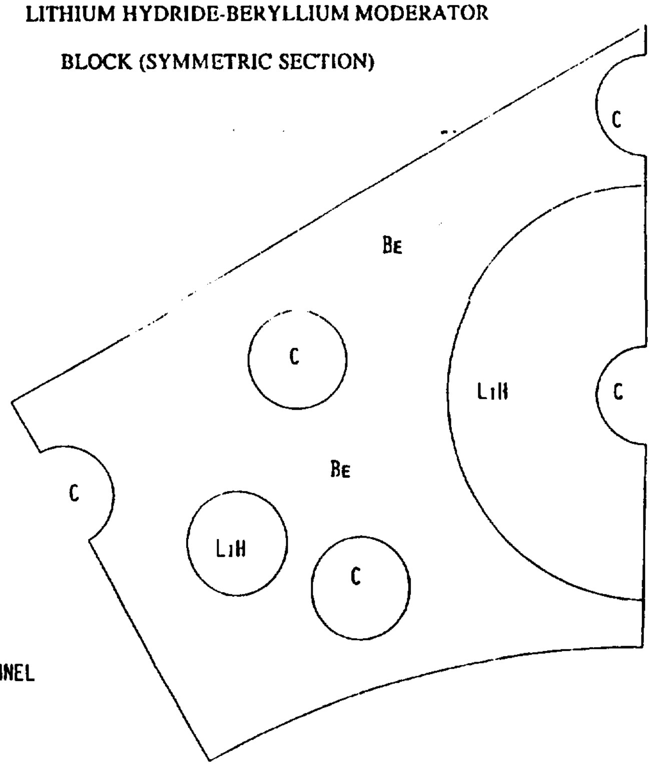


FEEDBACK REACTIVITIES



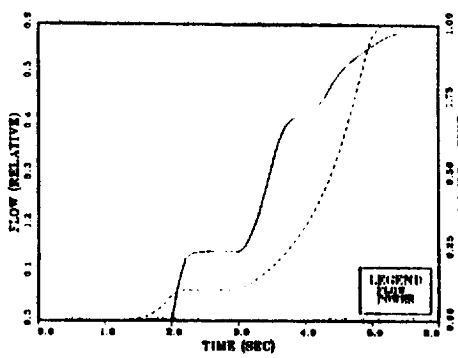
LITHIUM HYDRIDE-BERYLLIUM MODERATOR

BLOCK (SYMMETRIC SECTION)

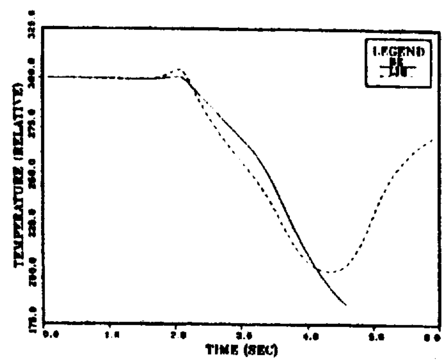


C = COOLANT CHANNEL

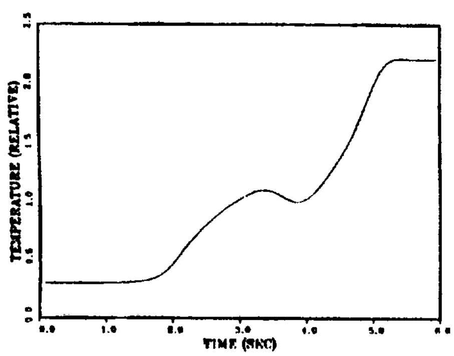
FLOW AND POWER



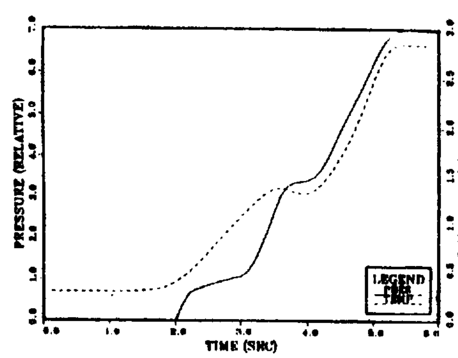
HE AND LH MEAN TEMPERATURES



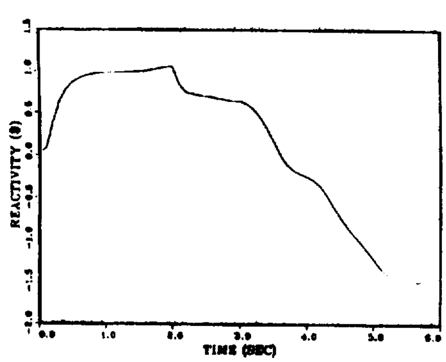
FUEL AVERAGE TEMPERATURE



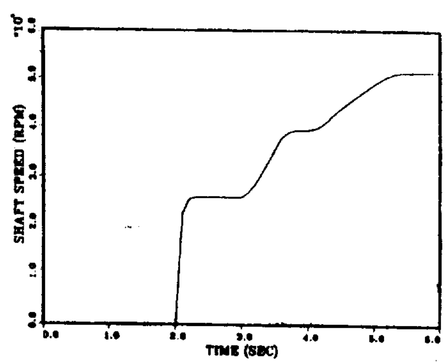
NOZZLE PRESSURE AND TEMPERATURE



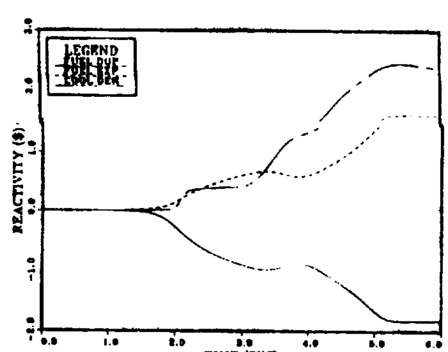
CONTROL REACTIVITY



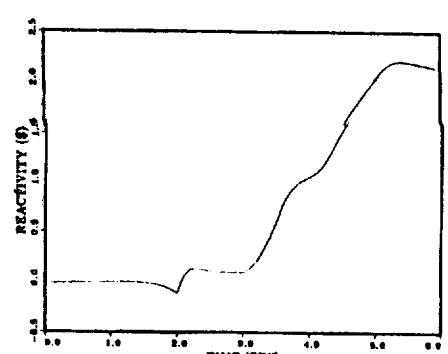
TPA SHAFT SPEED



FEEDBACK REACTIVITIES



FEEDBACK REACTIVITY



CONCLUSIONS

- THE KINETIC CODE SYSTEM IS A VIABLE TRANSIENT ANALYSIS ALGORITHM FOR STUDYING PBR BASED NTP START UP AND SHUTDOWN BEHAVIOR

- THE CODE FLEXIBILITY ALLOWS INVESTIGATION OF
  - TPA START STRATEGIES
  - REACTOR DESIGN VARIATIONS TO MINIMIZE FEEDBACK EFFECTS
  - ENGINE SHUTDOWN STRATEGIES

- TWO-PHASE FLOW AND MULTI-DIMENSIONAL REACTOR KINETICS ARE CURRENTLY NOT MODELED

N93-26961

Next Generation System Modeling of NTR Systems

John J. Buksa and William J. Rider
Los Alamos National Laboratory
October 22, 1992

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Introduction

- NTR Modeling Challenges
- Current Approaches
- Shortcomings of Current Analysis Methods
- Future Needs
- Present Steps Toward These Goals

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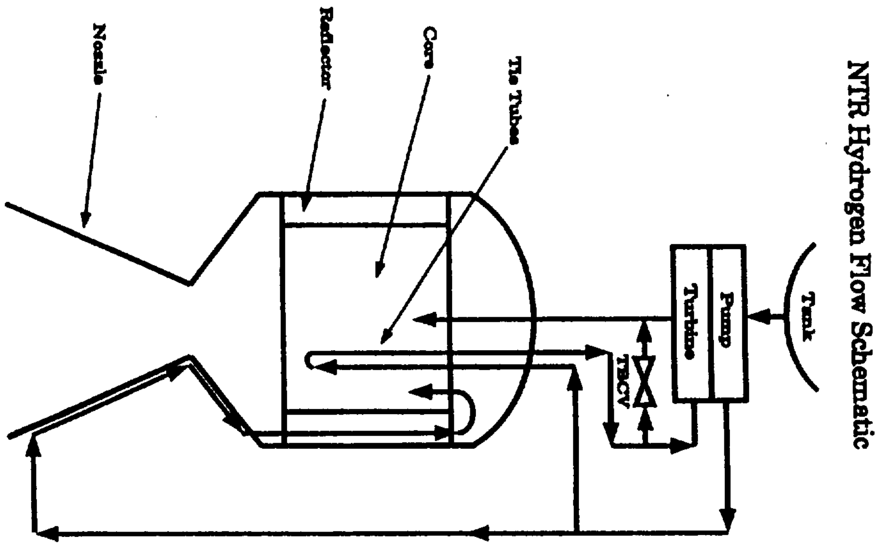

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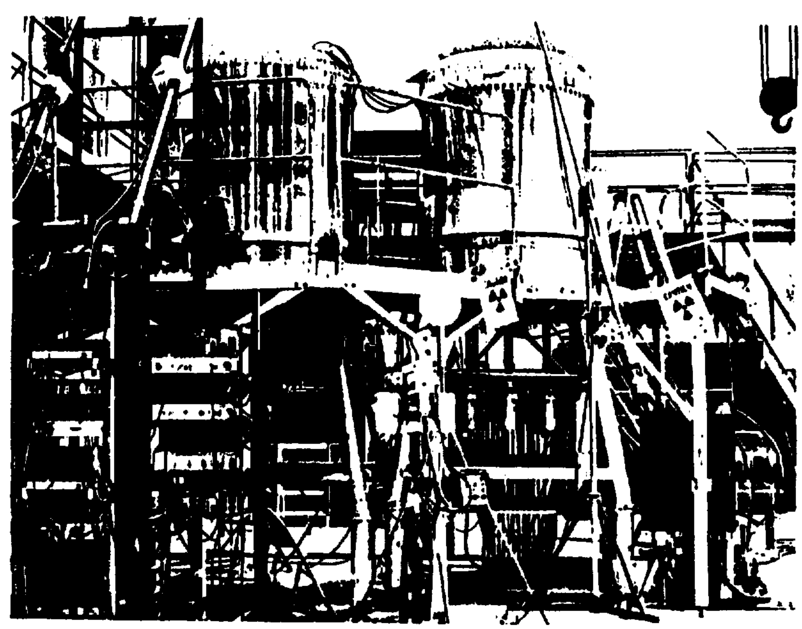
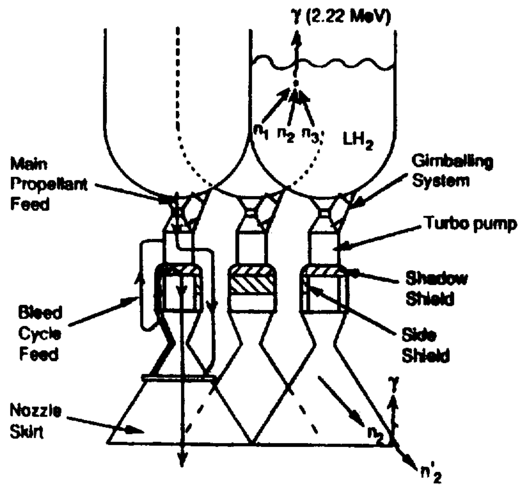
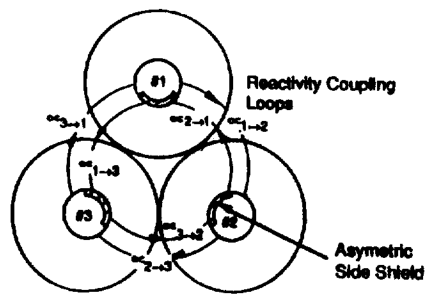


Figure 1. The Coupled Cores in Kiva-3, Pajarito Site. "Test Kiwi" is on the left, and PARKA is on the right.

ENGINE COUPLING PHENOMENA





3/92

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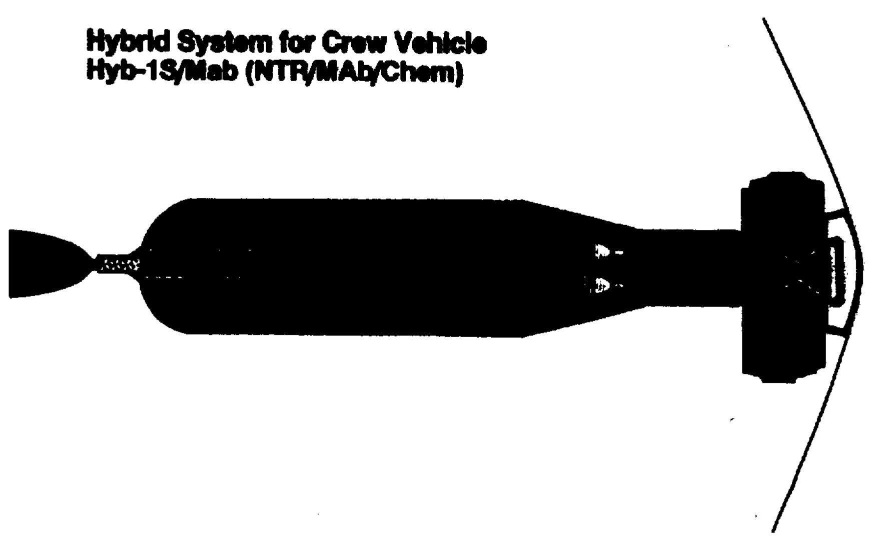

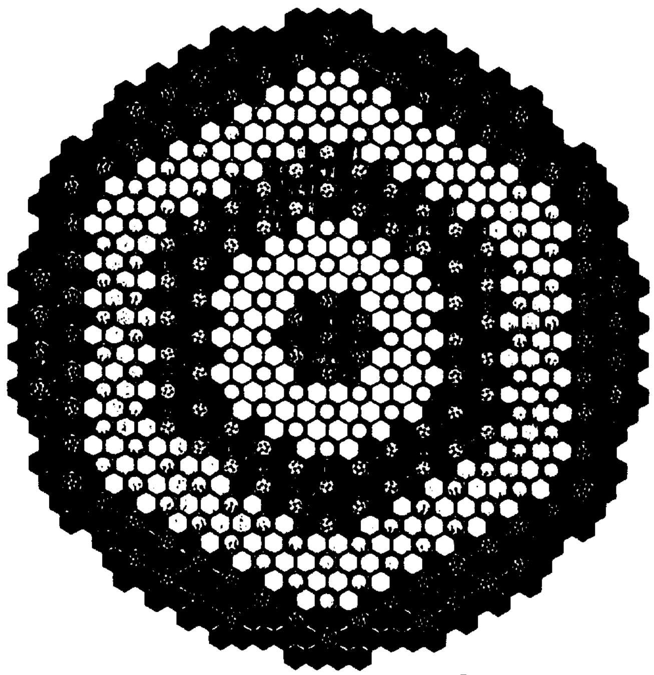

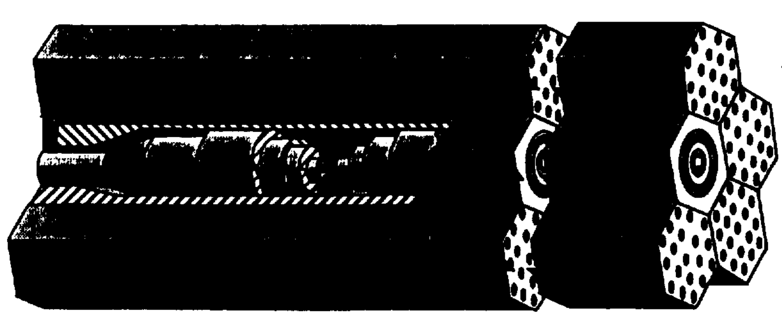

3D NTR Cluster

- Design: performance (SS operation) and lifetime (fuel / criticality)
- Startup and Shutdown
  (two phase T-H, neutronics, kinetics, heat transfer, low strain rate hydro)
- Water Immersion
  (kinetics, neutronics, all hydro)
- Impaction
  (kinetics, neutronics, high strain rate hydro)
- Engine-Out Operations
  (all except high strain rate hydro)

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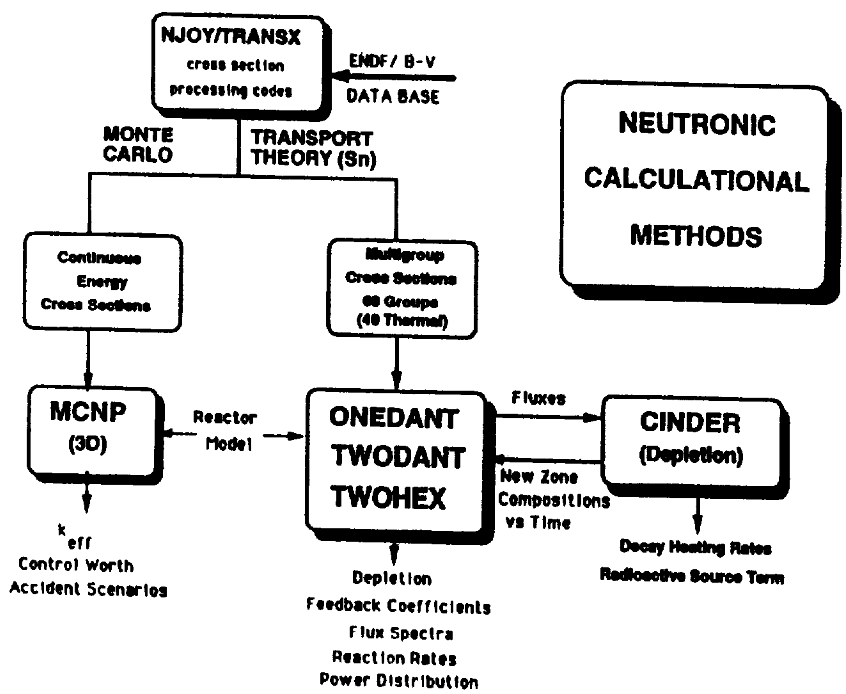

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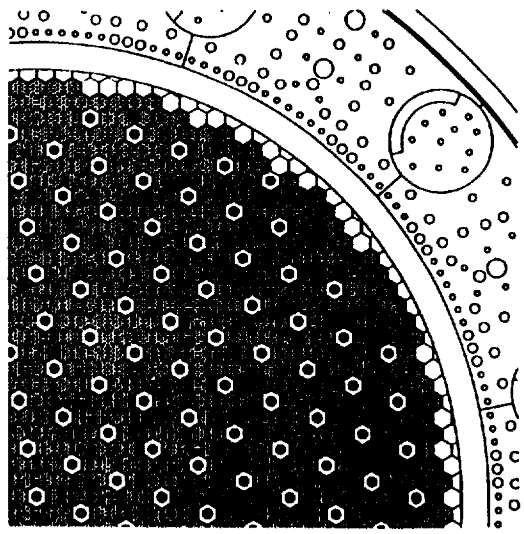
DETAILED MCNP MODELING OF NUCLEAR THERMAL ROCKETS – WESTINGHOUSE NRX-A6 REACTOR



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Thermal-Hydraulic Analysis Methods

- Extensive experience in both space and terrestrial reactors
- TRAC
  - Developed for LOCA analysis of PWRs
  - Highly developed models for two-phase flow
  - Low/zero gravity models are available
  - Useful for facility/more general system analysis
- HERA
  - Developed for solid core terrestrial reactors
  - Useful for the thermal analysis of general systems including space nuclear systems
- KLAXON
  - New thermal hydraulic systems code designed specifically for gas cooled, space reactors
- THROHPUT
  - State-of-the-art heat pipe modeling from startup to shutdown

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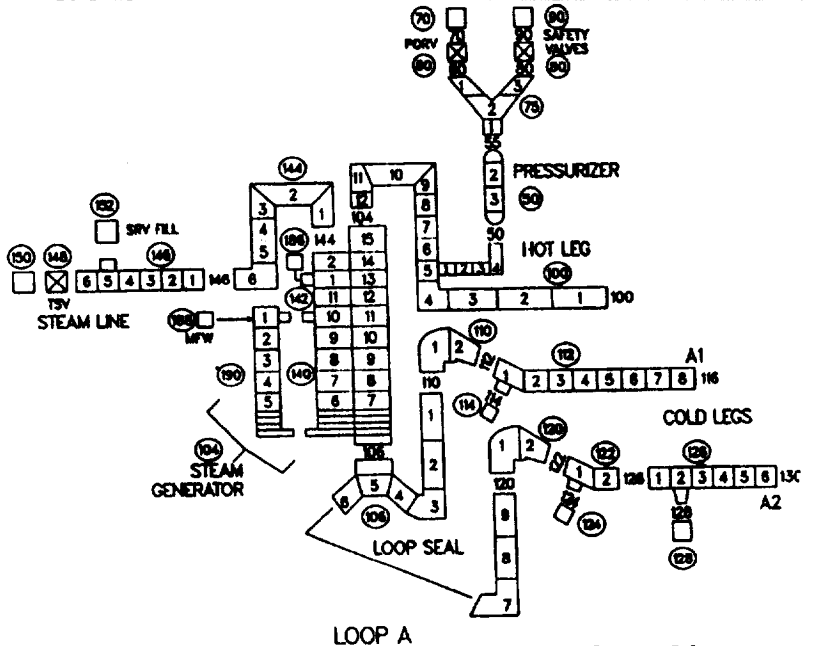
Example TRAC Noding Diagram



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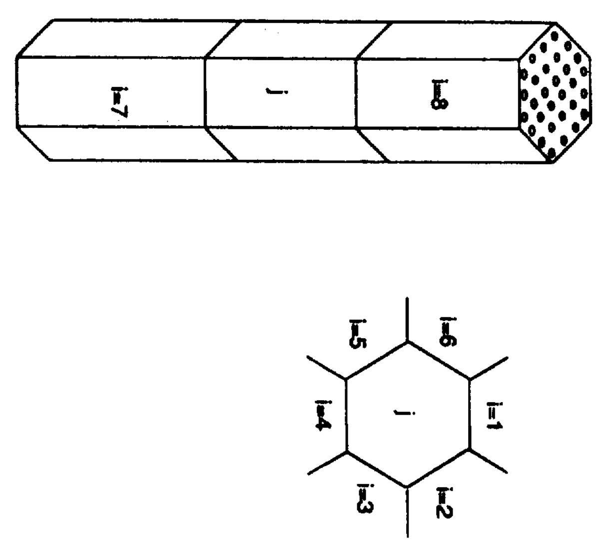
Description of HERA



- HElium Reactor Analysis
- Fully three-dimensional allowing for complex geometries to be accurately represented.
- Flexible input allows a large number of test cases.
- The code computes solution with minimal computational effort.

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Thermal-Hydraulic Modeling: Prismatic Fuel

- HERA: HElium/Hydrogen Reactor Analysis
- Used to model reactor core and core components with axially homogeneous construction
- Three-dimensional, fully transient, arbitrary user defined geometries
- Programmed to be computationally efficient, especially on vector supercomputers
- Currently exists in stand-alone mode and coupled to TRAC. Connection to KLAXON is planned
- PATRAN grid generator and visualization translators currently being written
- Coupling to Storm's corrosion model envisioned → **Core Lifetime**
- Component and core T-H model planned (fuel element, support element, and periphery)

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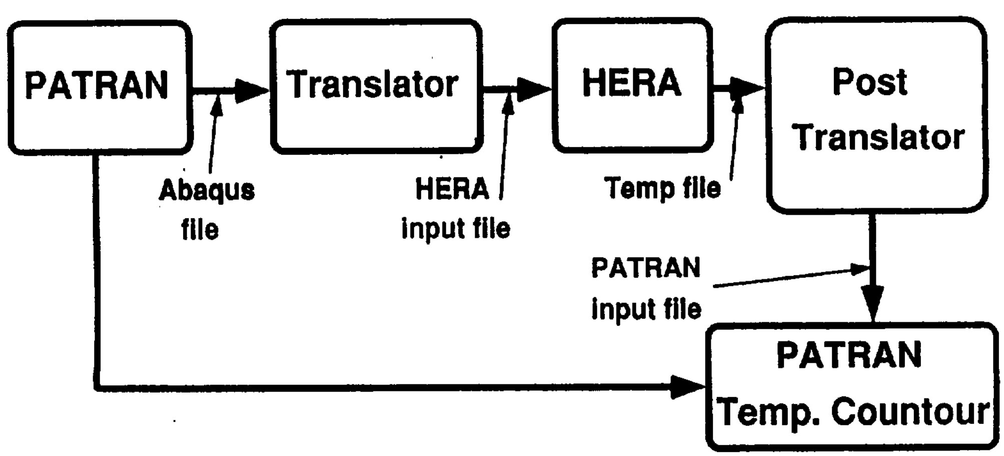
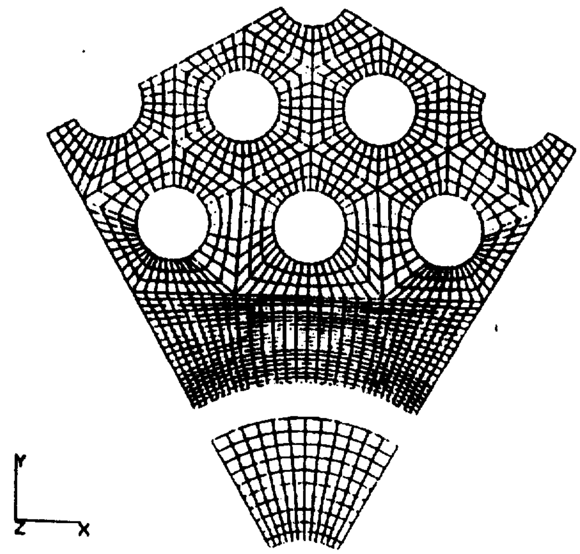
Methodology: New

Specific Outline:

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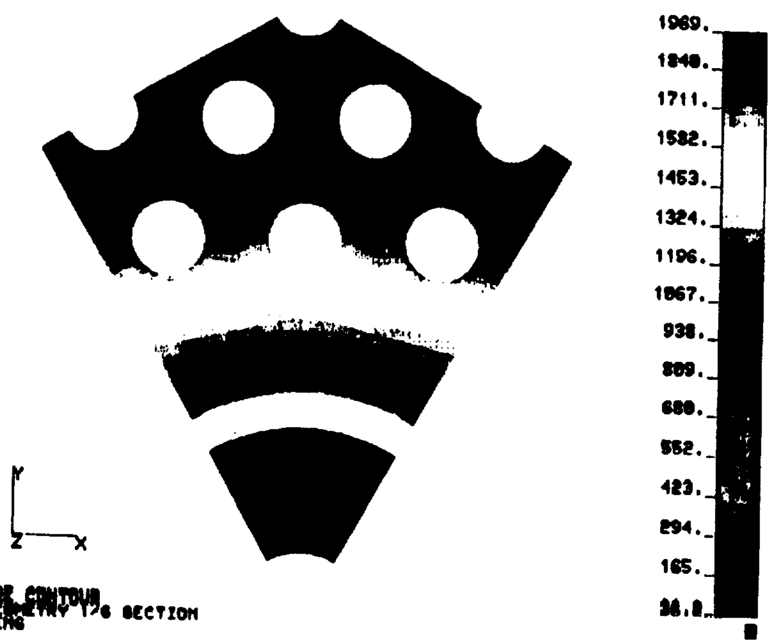
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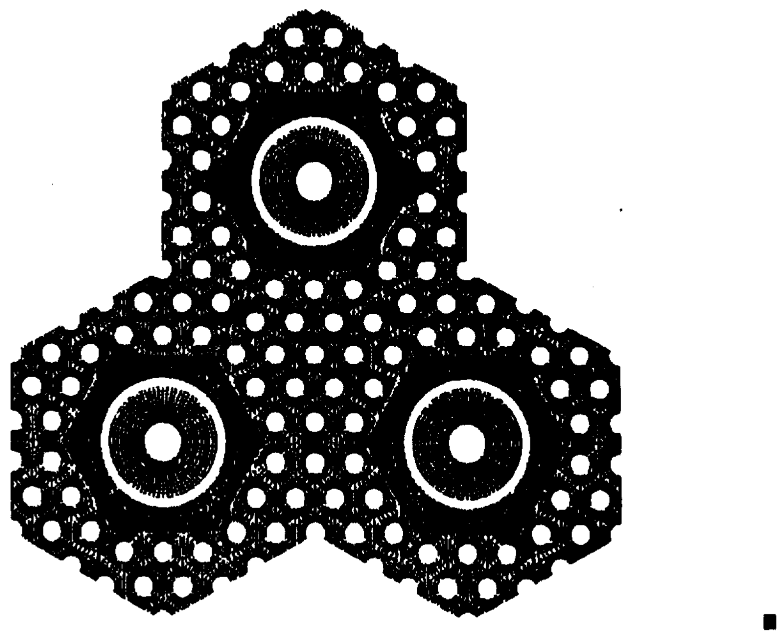
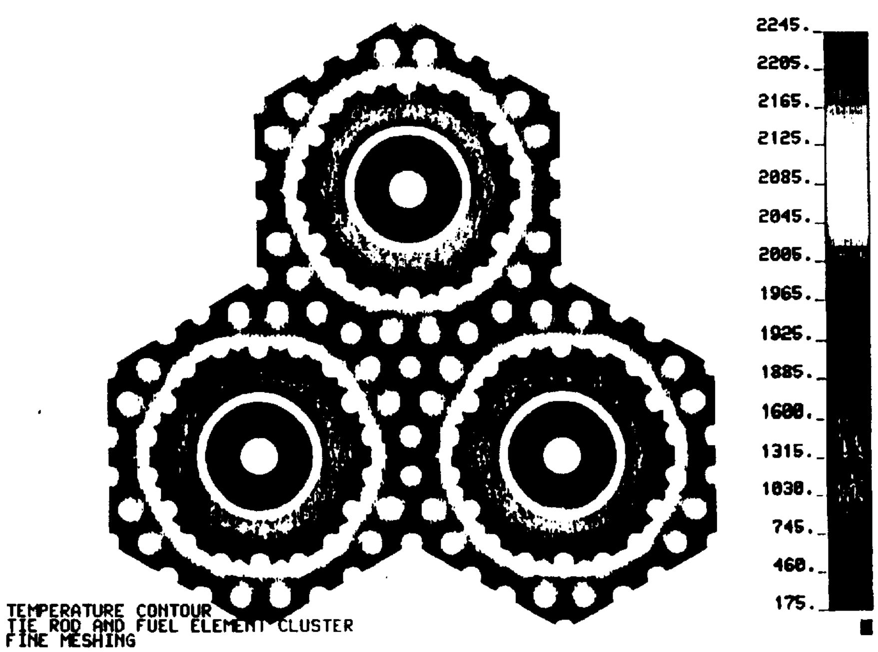
TEMPERATURE CONTOUR
CLUSTER GEOMETRY 1/6 SECTION
FINE MESHING

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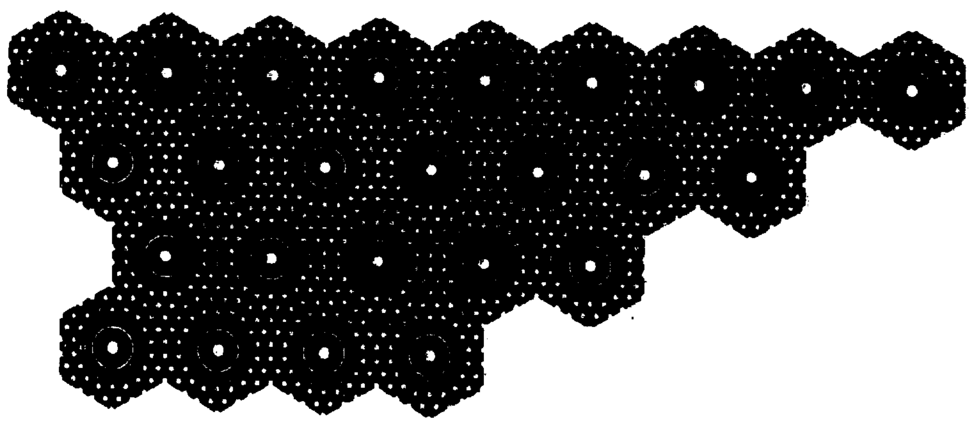
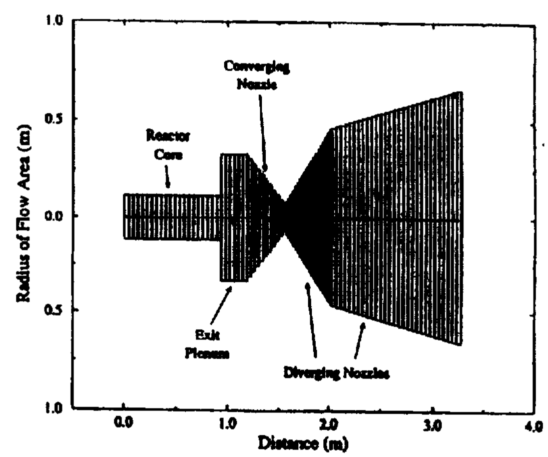
TEMPERATURE CONTOUR
TIE ROD AND FUEL ELEMENT CLUSTER
FINE MESHING



KLAXON
GAS-COOLED REACTOR SYSTEMS MODELING CODE

Time-dependent analysis of systems operating with compressible gas working fluids. TRAC-like pipe, plenum, etc. component models, fill and break capabilities, and advanced flow modeling numerics for shock following in nozzles.



NTR Geometry

Future Development

- **Connection to HERA**
- **Validation with systems data**

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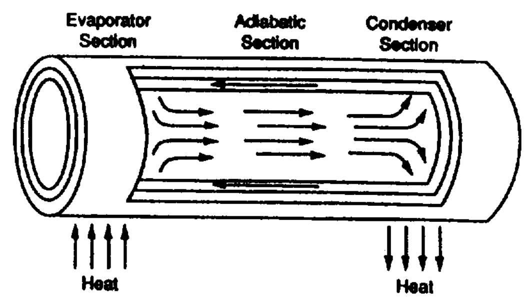
THROHPUT
HEAT PIPE MODELING CODE

Transient thermal-hydraulic heat pipe modeling code with:

- **Multi-region capability (wall, fluid, mixed, gas)**
- **2-D convection and conduction heat transfer**
- **Li melt model**
- **Gravity and non-gravity capillary pressure models**

Future development:
Benchmarking and validation with LANL experiments



Heat Pipe Operation

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Why Level 3/4 Model Development?

- Level 3/4 Improvements
  - Integral versus Ad hoc
  - Physics versus Assumptions
  - Confidence versus Safety Margin
  - Machine versus Human Intervention
- Examples
  - Reactor Compaction/Immersion Accidents
  - Reactor Startup

Future Needs

- ❒ Better All Around Resolution of Problems
- ❒ System Design Optimization Tools
- ❒ Complete Utilization of Modern Technology (Computers and Algorithms)
- ❒ Use of Integrated Physics Codes

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REACTOR DESIGN AND ANALYSIS GROUP

Los Alamos Perspective

- **Emphasis on Simulation Instead of Testing**
  - current ES&H environment dictates reduced testing of nuclear systems
- **Interagency NTP Modeling Team**
  - Role, Impact, Importance, Visibility
- **Effort Should be Commensurate With the SEI**
  - ambitious, high profile, high tech, national importance

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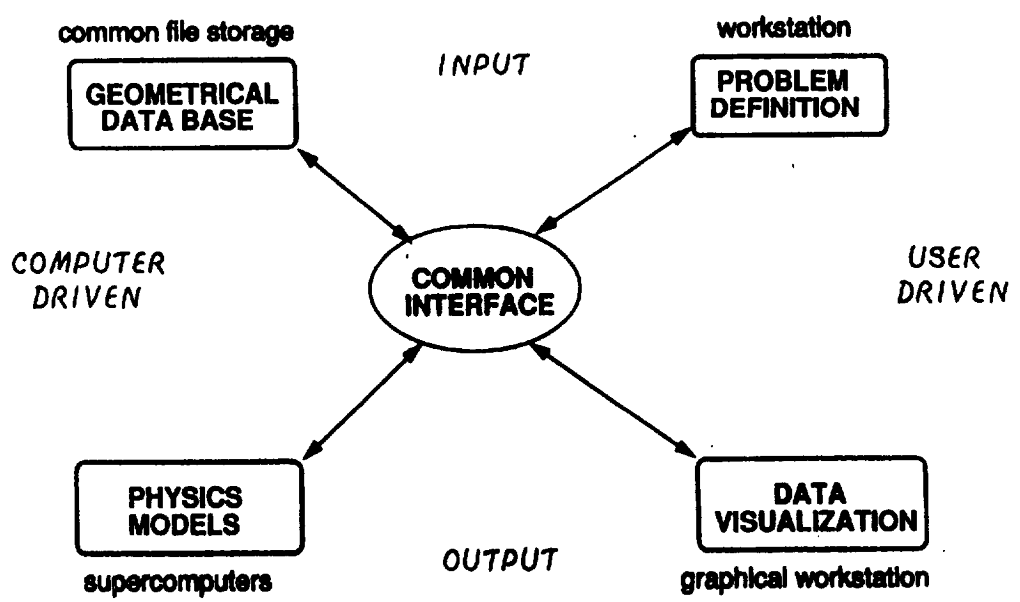
The Need for New Code Development in Level 3/4

- No "Real" Level 3/4 Codes Exist
- Codes will be Heavily Relied on
- Testing will be Restricted by ES&H Requirements
- Current Codes are Designed to Analyze Primarily Terrestrial Reactors
- Current Codes use Outdated Methodologies
- Current Codes are Designed for Older Computer Architectures

2

Advanced Architecture: Description



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Advanced Architecture: Potential Physics Packages

- Neutronics (including cross-sections, dosimetry)
- Spatial Kinetics
- Generation/Depletion
- Thermal-Hydraulics (two phase)
- Low Strain Rate Hydro
- High Strain Rate Hydro (solid and fluid)
- Heat transfer (conduction, radiation)
- Chemistry/Materials

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LANL Current Status

- Outlined Needs and Requirement for Level 3/4 Code Development
- Investigated LANL Capability
- Example LANL Capability
  - NIKE — A time-dependent S_n radiation transport code with arbitrary 3-D meshes on a CM (or Cray)
  - NIKE is coupled to PAGOSA/X3D for high strain rate hydrodynamics on CM (or Cray)
  - Genesis of Level 3/4 code capability for compaction/immersion accident analysis
  - Starting demonstrative NIKE/PAGOSA NTR analysis effort
  - Thermal-hydraulic work continues with work on improving both KLAXON and HERA

3

Other Laboratory Capability

- Fluid dynamics codes
  - Developed for a large range of physical situations varying from incompressible to highly compressible flows
  - Advanced methodologies
- High Strain Rate Solid/Hydrodynamics
  - Applicable to events involving reactor impaction/disassembly
  - Examples: launch accidents, reentry, water immersion
  - Coupled directly to other physical phenomena (neutronics for instance)
  - Advanced methodologies
- High Performance Computing
  - One of two DOE centers of excellence
  - ICN (3 CMs, 7 Cray YMPs)
  - ACL

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ADVANCED COMPUTING LABORATORY

Acting as a university/industrial/laboratory interface for state of the art computations, emphasizing:

- State of the art hardware for massively parallel computation (largest CM-2s and CM-5 in the nation)
- Wide area gigabit network for distributed parallel computing (using ANSI standard: HIPPI)
- Advanced scientific visualization using high speed networking and parallel computational methods
- Software tools/algorithms development for distributed parallel computation (NSF Science & Tech. center: CRPC)
- Emphasizing "real" applications running in parallel environment (Grand Challenges and beyond)

Purposes of the ACL

- To respond to the rapid changes in hardware and software
- To investigate new "Grand Challenge" computing environments
- To provide more "access" to Los Alamos from the outside world
- Provide high performance testbed for networking and visualization
- Stimulate practical algorithm development for massively parallel computing
- Function as one of the Dept of Energy High Performance Computing Research Centers

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Table 1: TODAY

Project	Description	Operations	Memory	Disk/Access
Porous Media	2-d immiscible flow	10^{16}	8 Gbytes	40 GBytes
Novel Materials	2-d molecular dynamics	10^{14}	500 Mbytes	64 GBytes
	3-d multimaterial hydro (200^3 pts)	10^{15}	8 GBytes	100GBytes
Plasma physics	transport scaling	10^{15}	8 GBytes	200 GBytes
Global Ocean	decade, 20 levels, 1/2°	10^{15}	500 MBytes	250 GBytes
Brain Topology	3-d reconstruction	10^{13}	200 MBytes	10 GBytes
QCD	quenched lattice (32x32x32x64)	10^{16}	500 MBytes	500 MBytes

Table 1: TOMORROW

[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
Porous Media	3-d immiscible flow	10^{18}	1 Tbytes	4 TBytes
Novel Materials	3-d molecular dynamics	10^{18}	20 Gbytes	3 TBytes
	3-d multimaterial hydro (1000^3 pts)	10^{18}	1 TBytes	20TBytes
Plasma physics	numerical Tokamak	10^{18}	1 TBytes	100 TBytes
Global Ocean	century, 40 levels, 1/4°	10^{17}	4 GBytes	20 TBytes
Brain Topology	3-d reconstruction	10^{15}	15 GBytes	1 TBytes
QCD	quenced lattice (64x64x64x128)	10^{18}	8 GBytes	8 TBytes

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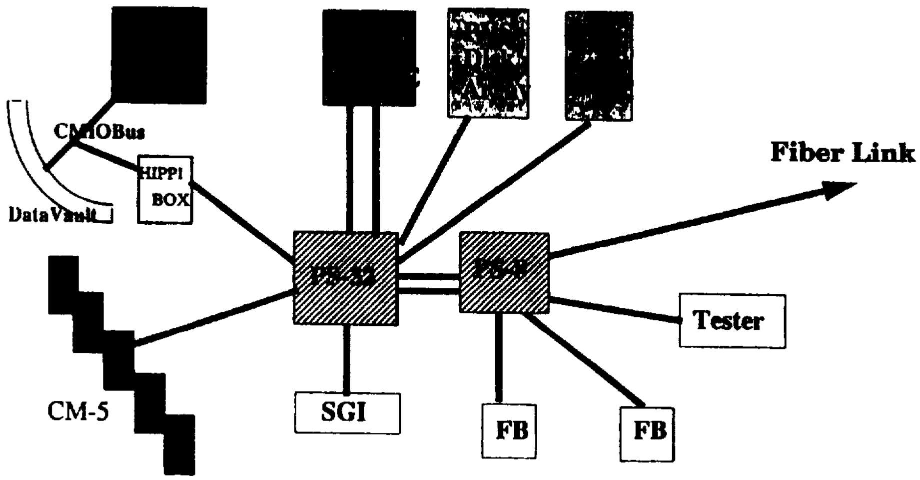
Applications on the CM-2

- QCD
- Condensed Matter Physics
- Free Lagrange Hydrodynamics
- Global Ocean Model
- Lattice Gas (porous media)
- Oil Reservoir: Mobil (11Gflops sustained)
- Tokamak Fluid Turbulence
- Fokker Planck
- Crystal Formation
- Many Body Problem
- Plasma Particle Simulations
- Molecular Dynamics
- Neural Networks

Existing ACL HIPPI Network



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PAGOSA

- ❐ A 3-D Multi-Material Hydrodynamics Code on the Connection Machine
- ❐ High-Speed Hydrodynamics and High-Rate Deformation of Solids
- ❐ Eulerian, Second-Order Predictor Corrector Lagrangian Step with Third-Order High-Resolution Advection
- ❐ High-Resolution Interface Reconstruction Algorithm
- ❐ Highly Efficient for the Connection Machine

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The NIKE Codes

- NIKE-R
  - 3-D Rectangular Mesh
  - Corner Finite-Difference Scheme
- NIKE-T
  - 3-D Arbitrarily-Connected Tetrahedral Mesh
  - Linear-Continuous Finite-Element Discretization
- Common Characteristics
  - Solve Even-Parity S_n Transport Equation with Anisotropic Scattering in Cartesian Geometry
  - Time-Dependent, Steady-State, k or α Eigenvalue Calculations
  - Essentially Positive Solutions - No Flux Fixup
  - Inner and Outer Iteration DSA - Unconditionally Stable and Effective
  - Very Efficient Simplified P_n Option - No Ray Effects

Conclusions

- ❐ **Current Modeling Approaches are Generally Inadequate**
- ❐ **In the Future Modeling will be Relied on Heavily**
- ❐ **Los Alamos has begun to Lay the Groundwork for Future Modeling Capabilities**

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N93-26962

ROCKET ENGINE NUMERICAL SIMULATOR

OVERVIEW PRESENTATION

presented by

Ken Davidian

Space Vehicle Propulsion Branch

Space Propulsion Technology Division

October 22, 1992

1

ROCKET ENGINE NUMERICAL SIMULATOR

CONTENTS

- RENS Definition
- Objectives
- Justification
- Approach
- Potential Applications
- Potential Users
- RENS Work Flowchart
- RENS Prototype
- Conclusions

ROCKET ENGINE NUMERICAL SIMULATOR

RENS DEFINITION

- Rocket Engine Numerical Simulator (RENS) Performs Liquid Rocket Engine Propulsion System Analyses and Design
- RENS Gives Engineer a 3-D Transient Tool for Analyzing Engine Systems (Tanks - Feed System - Thrust Chamber)
- RENS Will Surpass/Encompass Capabilities of Current System Codes (ROCETS & Generic Power Balance)

3

ROCKET ENGINE NUMERICAL SIMULATOR

RENS DEFINITION

- RENS is Long Term and Large Scope
- RENS Features Include:

- System Executive
- Data Management
- Graphical User Interface
- Incorporation of Users' Technical Codes
- Easy to Use
- Industry/University/ Gov't Advisory Group
- Public Domain
- Evolution of Capabilities

4

ROCKET ENGINE NUMERICAL SIMULATOR

OBJECTIVES

- Enable spontaneous and adaptive rocket definition, generation, performance evaluation, and failure analysis.
- Develop capability to simulate component and system level performance of rocket propulsion systems.
- Provide rapid and accurate assessment of rocket to increase design efficiency.
- Incorporate and integrate validated computational simulation codes/technologies.

5

ROCKET ENGINE NUMERICAL SIMULATOR

JUSTIFICATION

- Following capabilities required by NASA to do our job: independent verification of proposed rocket performance, new rocket designs, assess impact of new rocket technologies.
- Standardized industry design/analysis tool (industry-university-government participation).
- Streamline, enhance, and alter research & analysis process to reduce time and cost.

6

ROCKET ENGINE NUMERICAL SIMULATOR

APPROACH

- The RENS program will be patterned after, and will leverage from, the Numerical Propulsion System Simulator (NPSS), currently under development at NASA LeRC for aircraft propulsion systems.
- RENS will incorporate component level descriptions to predict performance and reliability.

7

ROCKET ENGINE NUMERICAL SIMULATOR

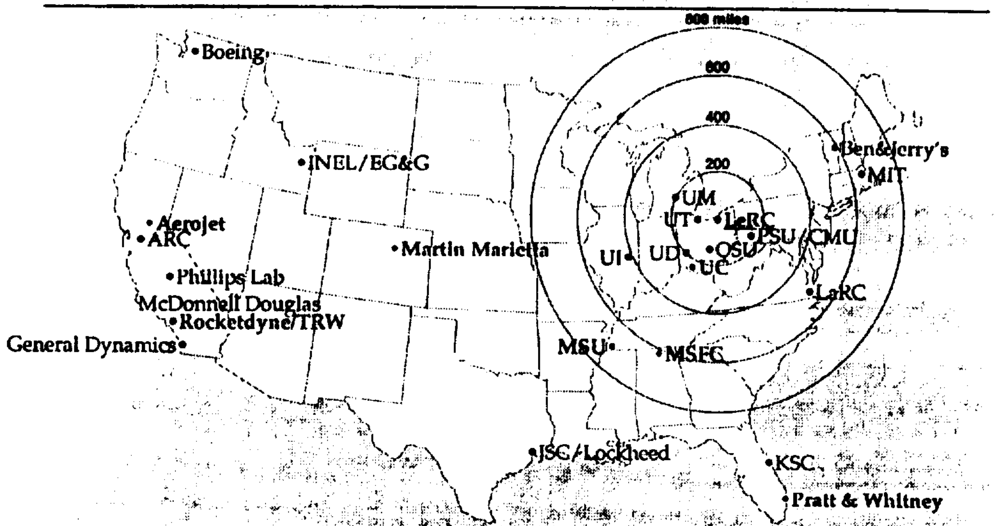
POTENTIAL APPLICATIONS

- Chemical Propulsion Systems
- Nuclear Thermal Propulsion Systems
- Propulsion System Test Facilities
- Nuclear Electric Propulsion Systems
- Space Power Systems

8

ROCKET ENGINE NUMERICAL SIMULATOR

POTENTIAL USERS

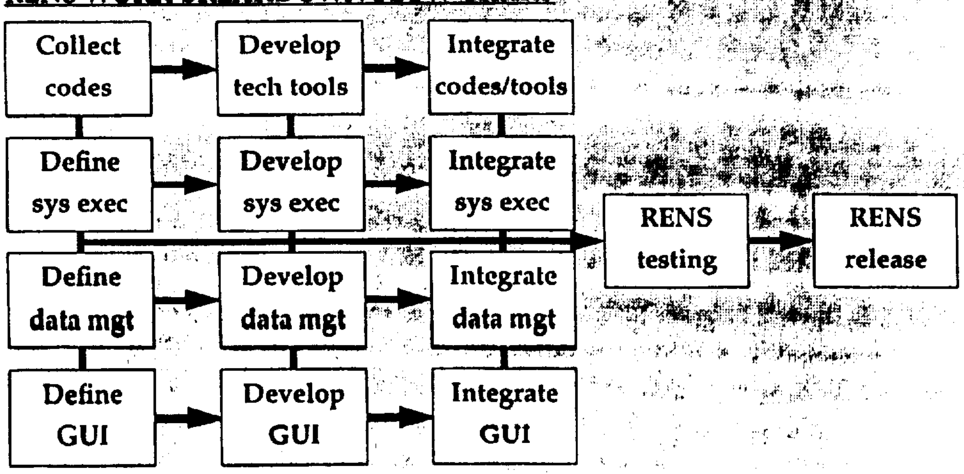


9

ROCKET ENGINE NUMERICAL SIMULATOR

RENS WORK STRUCTURE

RENS WORK BREAKDOWN FLOW CHART



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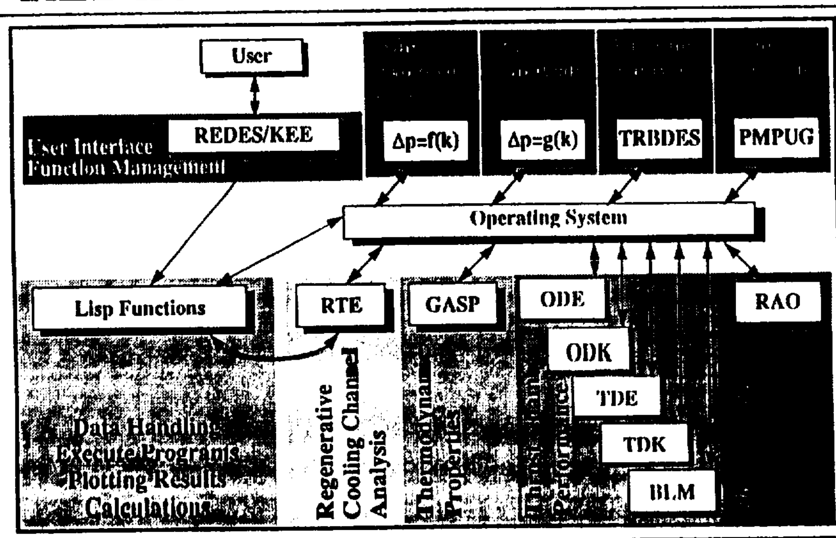
ROCKET ENGINE
NUMERICAL SIMULATOR
RENS PROTOTYPE - REDES

- Prototype Capability Initiated in 1989 with Rocket Engine Design Expert System (REDES).
- REDES Used to Conduct Various Studies and Model Various Engines:
- Nozzle Performance Parametrics (SSME, RL10)
- Nozzle Design (NTR)
- Rocket Engine Test Facility Capability Assessment (NASA LeRC Rocket Engine Test Facility Ejectors)

11

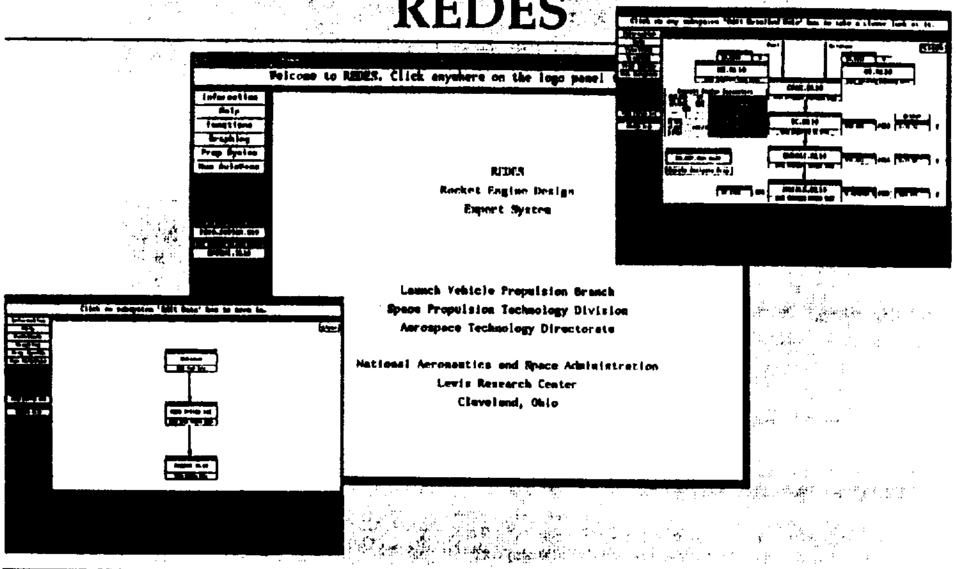
ROCKET ENGINE
NUMERICAL SIMULATOR
REDES ANALYTICAL DOMAIN



C-3

ROCKET ENGINE NUMERICAL SIMULATOR

REDES



13

ROCKET ENGINE NUMERICAL SIMULATOR

CONCLUSIONS

- RENS Capabilities Required For Simulation Development.
- Simulation Capability Required By Gov't, Industry, and University in Many Technical Disciplines.
- RENS Prototype Exists at LeRC.

14

ORIGINAL PAGE IS OF POOR QUALITY

ROCKET ENGINE NUMERICAL SIMULATOR

RENS USER SURVEY (part 1 of 2)

Q: How Would You Use RENS?

Q: What Would You Add To the Current RENS Description? What Would You Delete?

Q: What Do You Like About the Current RENS Description? What Do You Dislike?

Q: What Would Be the Impact of Using RENS On Your Organization? Technology Benefit? Cost Benefit?

5

ROCKET ENGINE NUMERICAL SIMULATOR

RENS USER SURVEY (part 2 of 2)

Q: Would You Be Interested In Developing Some Portion of RENS? What Portion?

Q: How Would You Justify Expending Resources In the Use of RENS to Your Management?

Q: May We Cite Your Responses In Our Advocacy Presentations to NASA Headquarters?

6

FACILITIES

N93-26963

Facilities
Chairman - Darrell Baldwin

NASA		
1:15	LeRC Facilities	Darrell Baldwin
1:30	Plum Brook Facility Overview (LeRC-PB)	Robert Kozar
2:00	NEP Facilities (LeRC)	Bob Vetrone
DOE		
2:15	LANL Studies (LANL)	Mike Hynes
2:45	Break	
3:00	INEL Studies (INEL)	Thomas Hill
DOD		
3:15	Air Force Facility (Sandia)	Dave Beck
3:30	Effluent Treatment System (Sandia)	Larry Shipers
TOUR		
3:45	Logistics (LeRC-PB)	Henry Pfanner
4:00	Tours B-2 High Temperature Facility Space Power Facility	
6:00	Adjourn	

Nuclear Propulsion Facility Requirements

Nuclear Facilities

Thermal Propulsion	Electric Propulsion
Fuel Development	Fuel Development
Reactor Development	Reactor Development
Materials Radiation Testing	Materials Radiation Testing
Integrated System Testing	Integrated System Testing

Non-Nuclear Facilities

Thermal Propulsion	Electric Propulsion
Nozzle Development	Power Conversion System Development
Turbopump Development	PMAD System Development
Propellant Tank Development	Thruster System Development
Control System Development	Control System Development
Valve and Mechanism Testing	Valve and Mechanism Testing
Material Compatability Testing	Material Compatability Testing
System Structural Testing	System Structural Testing
Cold Flow Verification Testing	Integrated System

NASA LEWIS CANDIDATE FACILITIES

CLEVELAND

- ELECTRIC PROPULSION LABORATORY (TANK 5)
- ELECTRIC PROPULSION LABORATORY (TANK 6)
- ROCKET ENGINE TEST FACILITY
- MATERIALS AND STRUCTURES LABORATORY
- ZERO GRAVITY FACILITY
- HYDROGEN ENVIRONMENT MATERIALS LABORATORY
- HOT HYDROGEN TEST BED
- SIMULATION AND CONTROL FACILITY

PLUM BROOK STATION

- SPACECRAFT PROPULSION RESEARCH FACILITY
- HIGH TEMPERATURE FACILITY
- SPACE POWER FACILITY
- CRYOGENIC PROPELLANT TANK RESEARCH FACILITY
- ROCKET DYNAMICS AND CONTROL FACILITY
- PLUM BROOK REACTOR FACILITY

INTERAGENCY FACILITY PANEL (NASA, DOE, DOD)

- DURING FY91, THE FACILITY PANEL IDENTIFIED APPROXIMATELY 220 EXISTING GOVERNMENT, UNIVERSITY, AND INDUSTRY FACILITIES WHICH COULD BE MADE AVAILABLE TO SUPPORT NTP AND NEP RESEARCH AND DEVELOPMENT PROGRAMS (REF: NASA TM - 105710)
- WITH APPROPRIATE UPGRADES AND MODIFICATIONS, AND DEPENDING ON THE PROPULSION CONCEPTS SELECTED, VIRTUALLY ALL DEVELOPMENT AND TEST WORK CAN BE ACCOMPLISHED IN EXISTING FACILITIES
- SINCE MOST OF THESE CANDIDATE FACILITIES WERE DESIGNED AND OPERATED UNDER SAFETY AND ENVIRONMENTAL REGULATIONS THAT ARE NOW OBSOLETE, MANY WILL REQUIRE MAJOR RENOVATIONS AND / OR ADDITIONS IN ORDER TO COMPLY WITH CURRENT REGULATIONS
- LEAD TIMES FOR PARTICULAR FACILITIES WILL VARY IN THE RANGE OF 2-4 YEARS FOR NON-NUCLEAR FACILITIES AND FROM 4-8 YEARS FOR NUCLEAR FACILITIES. ESTIMATED CONSTRUCTION COSTS RANGE FROM $400M TO $800M DEPENDING ON SELECTED PROPULSION SYSTEM CONCEPTS AND ASSOCIATED TEST OPTIONS

N93-26964

ROBERT KOZAR
10-21-92

Plum Brook Facilities

Spacecraft Propulsion Research Facility (B-2)

The facility was designed to test space vehicles and upper stage rocket engines in a simulated space environment. The vacuum test chamber can accommodate space vehicles up to 22' diameter by 50' long.

This facility is to be restored as part of the advanced cryogenic engine program. Additional facility upgrades will be made which will allow the use of this facility to perform integrated engine non-nuclear testing.

- Cold flow distribution verification and thermal investigations
- Solar irradiation / cold soak thermal cycling verification
- Verification of structural static loading

Hydrogen Heat Transfer Facility (HHTF)
(Currently the Hypersonic Tunnel Facility)

When restored to its original capability of handling large flows of hot hydrogen, this facility will be used as a testbed to perform NTR nozzle performance verification using hot hydrogen at altitude.

- Verification of simulation model results
- Verification of thermal and vibration performance
- Verification of nozzle erosion / corrosion characteristics performance

Plum Brook Facilities

Rocket Dynamics and Control Facility (B-3)

This facility was designed for altitude tests on various components for large rocket engines such as would be needed for interplanetary travel. It was used to test the structural integrity of the Centaur-Viking vehicle and its protective shroud. The existing facility presently includes a 200,000 gallon liquid hydrogen storage tank. NPO intends to use this facility for propulsion system vibration testing with altitude simulation.

- Verification of structural dynamic loading
- Cold Flow stability in vibration environment

Cryogenic Propellant Tank Site (K-Site)

This facility has been used as a research test chamber where liquid hydrogen rocket fuel tanks up to 18' in diameter were tested in a 25' diameter spherical thermal vacuum chamber. This facility is currently operational and has been used for recent slush hydrogen work associated with the NASP program

It will provide a facility for NTP and NEP propellant tank testing.

- Verification of tank insulation performance
- Functional leak testing of filler plumbing
- Verification of structural and vibration performance
- Acent / decent profile testing
- Slush hydrogen investigations

Plum Brook Facilities

Space Power Facility (SPF)

This facility is a very large vacuum chamber (100' diameter, 120' height) for testing spacecraft and / or their subsystems and components in a simulated space environment. It was specifically designed for testing space nuclear electric power systems in a hard vacuum, cold wall environment. It is intended to use this facility for nuclear electric propulsion component and integrated system tests.

- Non-nuclear system tests
- Functional testing of NEP components
- Heat source, radiators, power conversion, PMAD, thrusters
- Functional testing of integrated NEP systems
- Functional testing of the NEP stage

N93-26965

SPACE PROPULSION TECHNOLOGY DIVISION

NASA Lewis Research Center

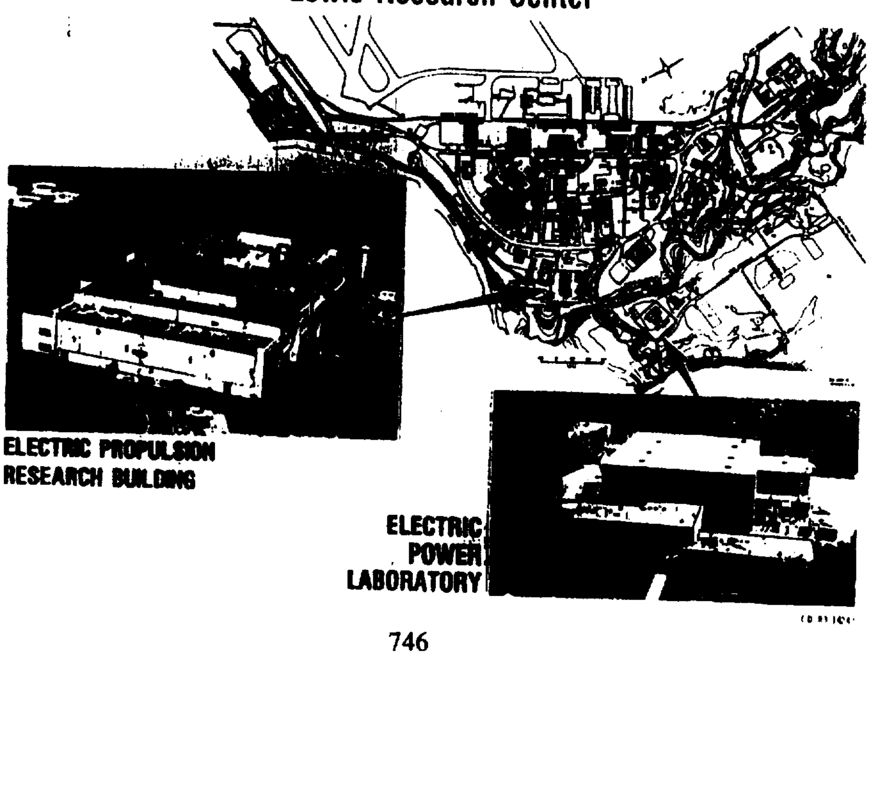
NEP FACILITIES (LERC)

Nuclear Propulsion Technical Interchange
October 21, 1992

R. H. Vetrone
Facility Manager/EPL, EPRB, Stirling

NASA
C-84-03073

SPACE SIMULATION FACILITIES
Lewis Research Center



SPACE PROPULSION TECHNOLOGY DIVISION

NASA Lewis Research Center

EPRB
ELECTRIC PROPULSION RESEARCH BUILDING(#16)

FACILITIES

VACUUM CHAMBERS (9): RANGE FROM 3FT. TO 10FT. DIA.

BELL JAR SYSTEMS (6)

CAPABILITIES

EXTREMELY HIGH (~ 1000 STD L/M - H_2 @ 10^{-1} TORR) PUMPING SPEEDS

HIGH VACUUM LEVELS (10^{-7} TORR)

CRYOPUMPED CHAMBERS

ACTIVITIES

COMPONENT DEVELOPMENT

THRUSTER TESTING

POWER CONDITIONING INTEGRATION

SPACE PROPULSION TECHNOLOGY DIVISION

NASA Lewis Research Center

EPL
ELECTRIC POWER LABORATORY (BLDG.301)

FACILITIES:

VACUUM CHAMBERS(3): 5FT. X 15FT.; 15FT. X 63FT; 25FT. DIA. X 82FT. LONG
BELL JAR SYSTEMS(7)

MAJOR FEATURES:

CLOSED LOOP REFRIG. SYSTEM TO ODP TRAPS
FULLY AUTOMATED
<<< UTILIZATION - >>> LOW OPERATING COST & MANPOWER REQUIREMENTS

TANK 6:

- * 20 OD PUMPS; 4 FORELINE BLOWERS; 3 MECHANICAL PUMPS
- * > 240 KW THERMAL REJECTION LN_2 COOLED SHROUD
- o SOLAR SIMULATOR

TANK 5:

2000 PUMPS; 4 FORELINE BLOWERS; 4 MECHANICAL PUMPS
$41M^2$ CRYOPANEL - GHe/LHe REFRIGERATOR/LIQUIFIER CRYO-SYSTEM

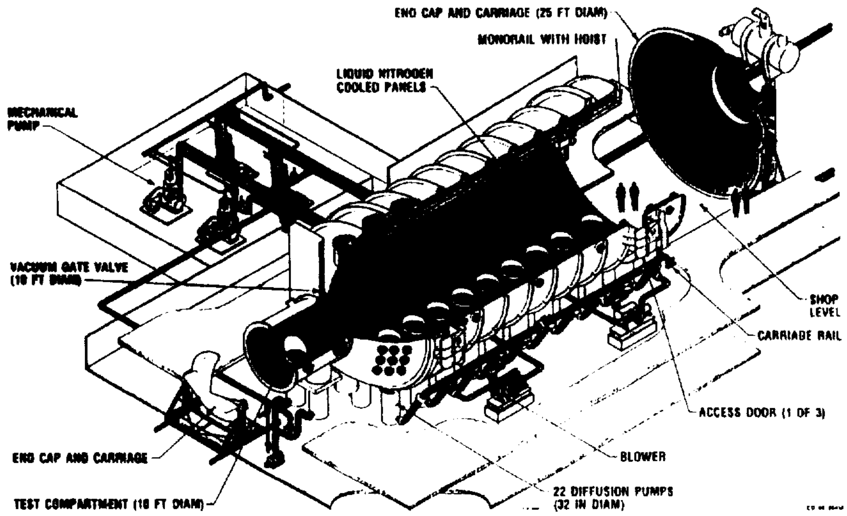
* EXPECTED IN POST 1991 COF PROJECT
o ADVOCATE: 5400; INSTALL & OP 1994/1995

NASA
C-86-7959

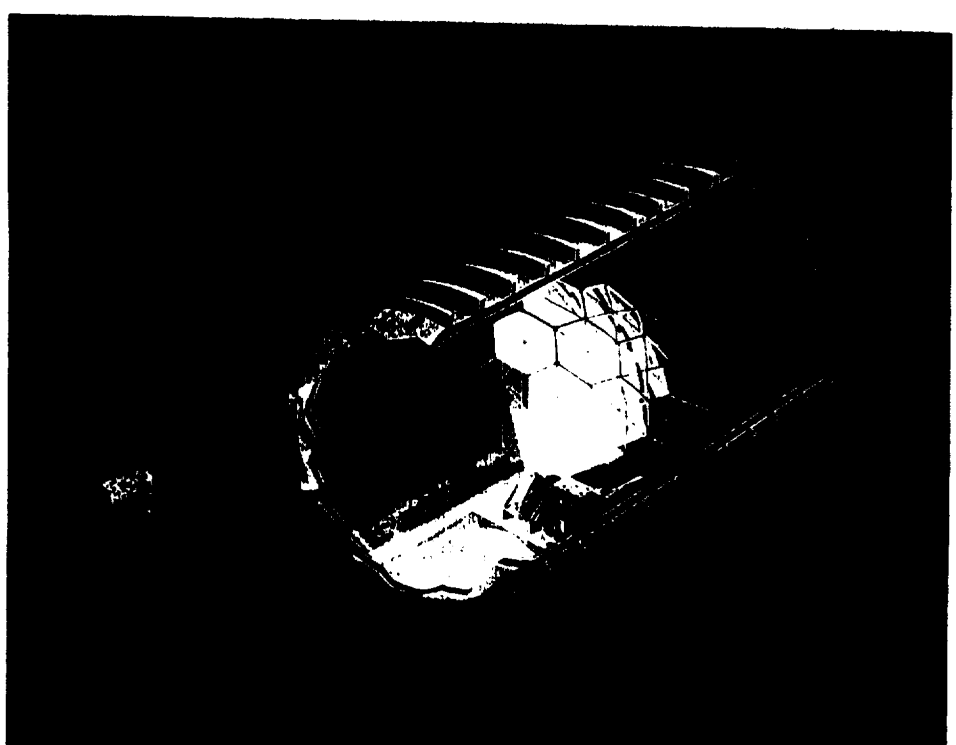
Lewis Research Center

TANK 6 VACUUM FACILITY

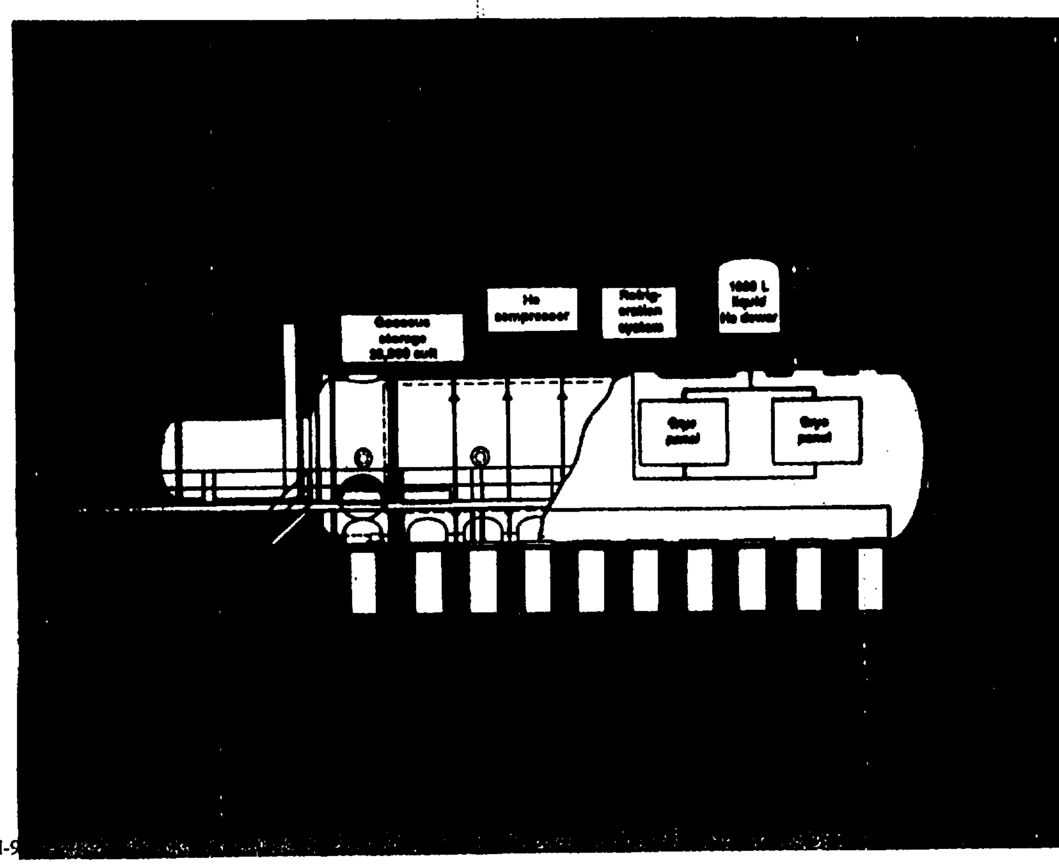
(25 FT DIAM X 82 FT OVERALL)



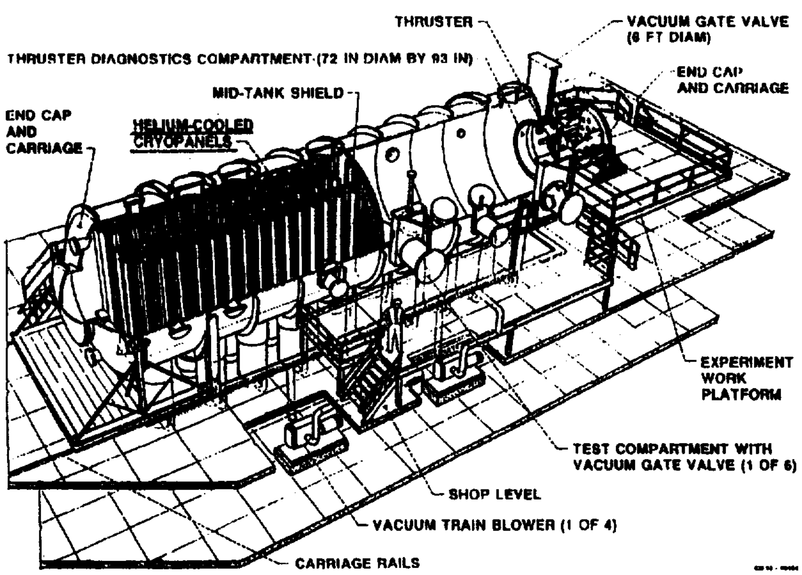
NASA
C-92-05795



NASA


Gaseous
storage
He
compressor
Refrig-
eration
system
liquid
He dewar
Cryo
panel
Cryo
panel

NP-TIM-9



SPACE PROPULSION TECHNOLOGY DIVISION

NASA Lewis Research Center

NUCLEAR ELECTRIC PROPULSION

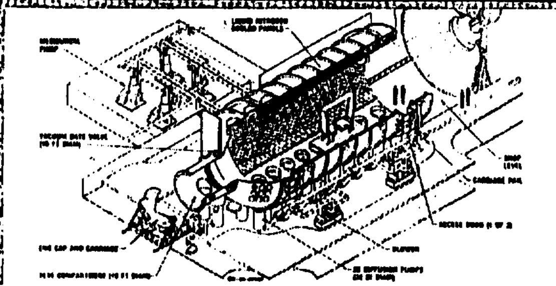
LOW THRUST, ELECTRIC	ION		MPD	
	5KW (Xe)	25KW (Xe,Kr)	100KW (H_2)	200KW (Ar)
$\dot{M}$(Mg/s)	5.3	27	40	320
REQ'D.PRESS.(TORR)	$<1.0X10^{-5}$	$<1.0X10^{-5}$	$<3.0X10^{-4}$	$<3.0X10^{-4}$
TANK 5 FACILITY				
(20)ODP/$\dot{M}$(Mg/S)	5.3	22	25.5	100
ACTUAL PRESS(TORR)	$1.3X10^{-5}$	$3.7X10^{-5}$	$4.8X10^{-4}$	$2.3X10^{-4}$
CRYOPANEL/$\dot{M}$(Mg/S)	8.0	TBD	TBD	155
ACTUAL PRESS (TORR)	$1.2X10^{-5}$	TBD	TBD	$1.0X10^{-4}$
FOCUS		*	*	

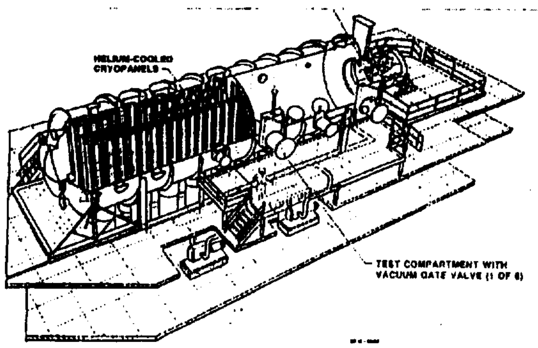
[USING FOUR(4) FORELINE BLOWERS & MECH. PUMPS = 300 Mg/SEC.@ $6X10^{-1}$ TORR - H_2]

SPACE PROPULSION TECHNOLOGY DIVISION

NASA Lewis Research Center

Test Facilities For Electric Propulsion At NASA LeRC





- Tank 6 has been cleaned of mercury contamination
- 20 oil diffusion pumps (ODP's) provide 90 000 l/s xenon pumping capability
- Tank 6's 41 m² cyropump system operational
- Tank 5's pumping capability is unique to NASA with 20 ODP's and cryopump yielding 140 000 l/s for xenon

CD-92-61505

N93-26966

Los Alamos Studies of Nevada Test Site Facilities for the Testing of Nuclear Rockets

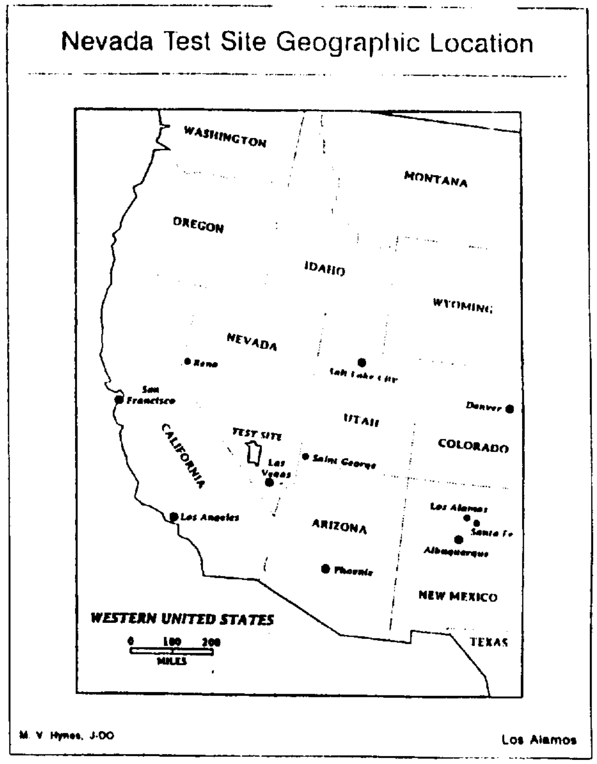
Nuclear Propulsion
Technical Interchange Meeting

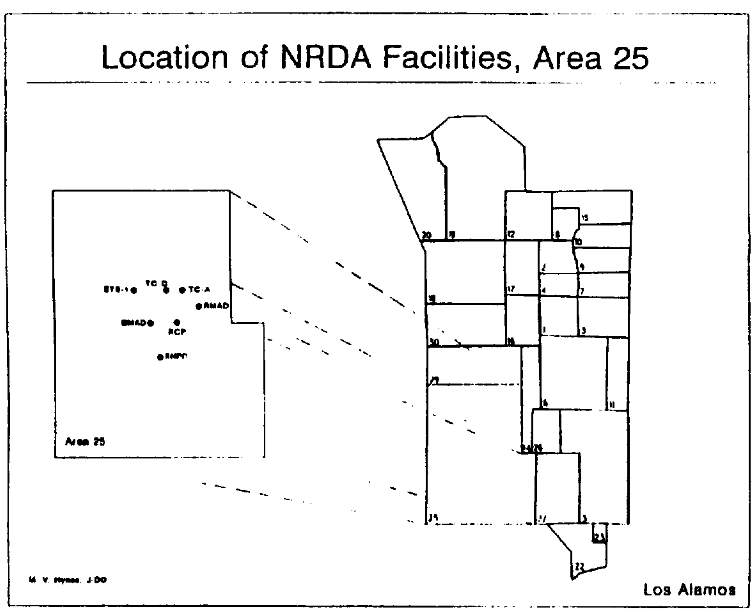
October 20-23, 1992
NASA-Lewis Research Center
Plum Brook Station

Michael V. Hynes
Field Test Division
Los Alamos National Laboratory

Los Alamos

Recent NASA/DOE studies for the Space Exploration Initiative have demonstrated a critical need for the ground-based testing of nuclear rocket engines. Experience in the ROVER/NERVA Program, experience in the Nuclear Weapons Testing Program, and involvement in the new nuclear rocket program has motivated our detailed assessment of the facilities used for the ROVER/NERVA Program and other facilities located at the Nevada Test Site (NTS). The ROVER/NERVA facilities are located in the Nevada Research & Development Area (NRDA) on Jackass Flats at NTS, approximately 85 miles northwest of Las Vegas. To guide our assessment of facilities for an engine testing program we have defined a program goal, scope, and process. In particular we have assumed that the program goal will be to certify a full engine system design as flight test ready. All nuclear and non-nuclear components will be individually certified as ready for such a test at sites remote from the NRDA facilities, the components transported to NRDA, and the engine assembled. We also assume that engines of 25,000-100,000 lb thrust levels will be tested with burn times of 1 hour or longer. After a test, the engine will be disassembled, time critical inspections will be executed, and a selection of components will be transported to remote inspection sites. The majority of the components will be stored for future inspection at Jackass Flats. To execute this program scope and process will require ten facilities. We considered the use of all relevant facilities at NTS including existing and new tunnels as well as the facilities at NRDA. Aside from the facilities located at remote sites and the inter-site transportation system, all of the required facilities are available at NRDA. In particular we have studied the refurbishment of E-MAD, ETS-1, R-MAD, and the interconnecting railroad. The total cost for such a refurbishment we estimate to be about $253M which includes additional contractor fees related to indirect, construction management, profit, contingency, and management reserves. This figure also includes the cost of the required NEPA, safety, and security documentation.


Nevada Test Site Geographic Location
WASHINGTON
MONTANA
OREGON
IDAHO
WYOMING
NEVADA
Reno
Salt Lake City
San Francisco
Denver
UTAH
CALIFORNIA
TEST SITE
Las Vegas
Saint George
COLORADO
Los Alamos
Santa Fe
Albuquerque
Los Angeles
ARIZONA
Phoenix
NEW MEXICO
TEXAS
WESTERN UNITED STATES
0 100 200
MILES
M. V. Hynes, J-DO
Los Alamos


Location of NRDA Facilities, Area 25
TC-A
RMAD
RCP
Area 25
M. V. Hynes, J-DO
Los Alamos

Assessment Program Plan

- Phase 0: Preliminaries
  - Formal charter from Jay Norman, Field Test Division Leader
  - Notification of N. Aquilina, NVOO
  - Notification of J. Stewart, NTSO
- Phase 1: Testing Program Design
  - Define testing program goal, scope, and process
  - Determine facilities required to execute testing program
- Phase 2: Facilities Overview
  - Survey of all relevant facilities at NTS
  - Existing and new tunnels
  - Vertical bore holes
  - ROVER/NERVA facilities on Jackass Flats
- Phase 3: Facilities Assessment
  - Determination of most cost effective facilities
  - Detailed functional assessment
  - Detailed cost estimating
- Phase 4: Operational Considerations
  - Infrastructure and support facilities
  - Impact on other users of NTS and Area 25
  - NEPA, safety, and security issues

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Program Goal, Scope, and Process
The New Nuclear Rocket Program

- Program Goal:
  - Flight Test Certify Design of Full Nuclear Rocket Engine System
- Program Scope:
  - Test fire up to 100,000 LbF Thrust engines for up to 1 hour
  - Testing capability for up to 6 tests annually
- Program Process:
  - Mission profile and flight systems specifications determined.
  - Develop engine system design
  - Develop and certify non-nuclear components at sites remote from Engine Test Stand
  - Develop and certify nuclear components at sites remote from Engine Test Stand
  - Transport all components for full engine system test to Engine Assembly/Disassembly Facility
  - Assemble engine
  - Transport engine to Engine Test Stand Facility
  - Conduct all needed tests
  - Transport engine to Engine Assembly/Disassembly Facility
  - Disassemble engine
  - Conduct time critical inspections
  - Package and ship components to remote inspection sites.
  - Analyze results and determine engine performance.
  - Store engine components for future reference near Assembly/Disassembly Facility

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Nuclear Rocket Engine Test Facilities
Program Goal: Flight Test Certify Full Engine System

1.	Transportation facilities for components	DOT Casks
2.	Non-Nuclear assembly facility	EMAD
3.	Nuclear assembly/disassembly facility	EMAD
4.	Rocket engine test stand facility	ETS-1
5.	LH_2/LN_2 & HP gas storage facility/tank farm	ETS-1
6.	Transportation facilities between NTS sites	NRDA RR
7.	Time-critical inspection facilities	EMAD
8.	Storage facility for reference components	RMAD
9.	Storage facility for SNM components	EMAD
10.	Transportation facilities between remote inspection sites	DOT Casks

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EMAD Facility
Engine Maintenance, Assembly, and Disassembly Building

- General Description:
  - Built in 1964 for the assembly and preperation of NERVA engines for testing, refurbishment of radioactively hot engines for additional testing, and disassembly and detailed post mortem inspection of tested engines and components
  - T-Plan multi-storied structure, 260 ft by 350 ft.
  - Divided into 7 separate sections according to specific functions and material traffic flow.
    - Cold assembly area; Hot maintenance and disassembly area; Post mortem cells; High and low level cells; Operating galleries; Shop and service area; Office area
- Functional Capabilities:
  - Cold and hot assembly and disassembly of major engine components and full size engines
  - Assembly line techniques applied due to heavy work load.
  - Special remote operated equipment installed to enable rapid disassembly.
- Cold Assembly Area:
  - Used for receipt and assembly of engines
  - Three major sections all 43 ft high:
    - Core receiving area --- 84 ft by 72 ft
    - Engine receiving area --- 72 ft by 36 ft
    - Cold engine assembly area --- 72 ft by 144 ft
- Hot Maintenance and Disassembly Area:
  - Five major sections all equipped with rectilinear and master-slave manipulators, overhead cranes, specially shielded viewing windows, etc.
    - Main hot bay --- 66 ft by 144 ft by 77 ft high
      5-8 ft thick concrete walls for shielding, rectilinear and master slave manipulators,
    - Core disassembly and examination cell --- 46 ft by 28 ft
    - Engine disassembly and examination cell --- 46 ft by 28 ft
    - Crane maintenance balcony
    - Hot and cold transfer tunnel
- Post Mortem Area.
  - Twelve independently shielded cells with shielded door openings to a common cell service area
  - Each cell equipped with special viewing windows, master-slave manipulators, transfer carts, and specialized inspection equipment

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Los Alamos

Summary of Final Assessment Results
J-Division Review of Nuclear Rocket Facilities at NTS
NRDA, Jackass Flats, Nevada

- Determined general program goals, scope, and process for full engine system test.
- Surveyed all possible facilities at NTS for application to program requirements.
  - Tunnels, existing and new
  - Existing ROVER/NERVA facilities
- Determined that existing facilities on Jackass Flats have the most potential for meeting program requirements in a cost driven assessment.
- Cost estimated upgrade of existing facilities for New Nuclear Rocket Program to be about $253M.
  - Richardson and Means Formalism
  - All additional fees included
- Recommend pursuing upgrade of existing facilities out of operating budget with NEPA and Safety Analysis concurrent.
- Estimated time to completion = 3 years.
- Recommend feasibility study of scrubber design alternatives and optimization in FY93.
  - Estimated cost = $350K
- Recommend full conceptual design study in FY93.
  - Estimated cost = $1M

M. V. Hynes, J-DO

Los Alamos

ETS-1 Facility
Engine Test Stand Number 1

- General Description:
  - Built in 1966 for the ground development testing of a downward firing NERVA-type engine in a flight simulated environment.
  - Originally designed for the test of a 50,000 LbF, 1 GW engine with a 300 s run time.
  - Upgrade to 75,000 LbF engine not completed.
- Physical Description of ETS-1 Complex:
  - Test stand connected to an underground control point building by a 1150 ft tunnel.
  - Cryogenic dewar and High Pressure gas vessel tank farm
  - Interconnecting process piping
  - Engine compartment radiation shield
  - Diffuser/Ejector exhaust duct
  - 2.5 Mgal demineralized deluge and cooling water storage tank.
  - Cooling water drainage ditch
  - Instrumentation and Controls, general utilities and support systems.
- The Test Stand consists of:
  - 160 ft, 100 ton aluminum structure supporting a 77,000 gal 50 psig LH2 vacuum jacketed run tank, Instrumentation and Controls terminations, and an elevator.
  - Below grade pipe chase
  - Exhaust gas duct vault
  - Mechanical and electrical equipment room
  - 3 ft wide by 40 ft high by 100 ft long concrete shadow shield
  - Process piping and distribution system
- The Control Point Building consists of:
  - Underground structure partitioned for control and recording data reduction
  - 2000 channels of data available
  - Above ground equipment room
  - HV & AC capability for all of ETS-1
  - I & C cabling, steam lines, and AC ducts in shielded tunnel

M. V. Hynes, J DO

Los Alamos

Facilities Cost Summary ($M)

Cost Item	E-MAD	ETS-1	R-MAD	Railroad	Subtotal
Basic Facility	17.574	50.930	2.473	0.624	71.601
Indirect	8.435	25.000	1.187	0.299	34.921
Home Office	6.502	22.500	0.915	0.231	30.148
NEPA Documentation	1.500	1.000	0.250	0.250	3.000
Safety Analysis	2.000	4.200	0.085	0.500	6.785
Security Plan	0.500	0.000	0.000	0.000	0.500
Construction Management	3.576	9.800	0.503	0.127	14.006
Inspection	0.000	3.800	0.000	0.000	3.800
Profit	3.251	9.800	0.458	0.115	13.624
Contingency	5.364	51.000	1.258	0.190	57.812
Management Reserve	3.576	13.000	0.503	0.127	17.206
Subtotal	52.278	191.030	7.632	2.463	253.403

M. V. Hynes, J DO

Los Alamos

N93-26967

Idaho National Engineering Laboratory

Space Nuclear Thermal Propulsion

Evaluation of PIPET at the INEL's CTF

T. J. Hill

October 21, 1992

PRESENTATION OUTLINE

- Study Scope
- Existing CTF Status & Infrastructure
- Assumptions
- Results
- Other Studies

SCOPE FOR FEASIBILITY REPORT

- Evaluate the Feasibility and Provide an ROM Estimate of Cost and Schedule for Testing the PIPET Reactors in the Contained Test Facility (CTF)

STUDY EVOLUTION

- Task was Identified at Meeting on June 11-12, 1992
- Task was Authorized to Start August 12, 1992
- Supported Three Meetings With Sandia
- Supported LANL Study for ETS-1

PIPET FACILITY REQUIREMENTS

Building	Size
• Receiving & Support Building	10,000 ft sq
• I & C Building	2,900 ft sq
• Reactor Systems Support (Test Building &Area)	Undefined
• Fuel Storage Support (Handling, Storage, & Shipping of Irradiated Material	Undefined
• Disassembly Building	7,500 ft sq
• Test Evaluation Center	6,400 ft sq

EXISTING CTF FACILITIES

- TAN 650 - Containment Building - 70 ft Dia by 129 ft High
- TAN 630 - Control & Data Acquisition Building - 18,000 ft sq
- TAN 624 - Containment Vessel Entry Building - 3,600 ft sq
- TAN 607 - Warm Shop - 4,080 ft sq
- TAN 604 - Maintenance Shop - 11,000 ft sq
- TAN 601/602 - Administration Building - 58,000 ft sq
- TAN THS - Hot Shop - 8,160 ft sq
- TAN THC - Hot Cell - 350 ft sq
- TAN 668 - Heavy Equipment Cleaning Facility - 2,800 ft sq

CTF BACKGROUND

- Contained Test Facility (CTF) was Loss-of-Fluid Test Facility (LOFT)
- LOFT was designed to study safety issues in a PWR
- CTF & associated facilities consist of a containment vessel, control and data rooms, maintenance shops, administrative buildings, hot shop, hot cells, warm shop, utilities, ES&H infrastructure
- CTF containment vessel is 70 ft. in dia. by 129 ft high, is an ASME Sect. III, Class B vessel rated at 40 psi, 360,000 cu ft volume with a 24 by 33 ft high door. 60 ft under 50 T Polar Crane

CTF REPORT ASSUMPTIONS

- PIPET/CTF test series will consist of testing five reactor cores and one technology demonstration engine.
- PIPET cores up to 550 Mw and run times up to 1,000 sec. Demonstration engine 1,000 Mw, Max. run time of 500 sec.
- Use of mechanical and electrical components and systems developed for SNTP.
- Determine feasible SNTP components and systems lay out for CTF.
- No design optimization of equipment and components.
- Existing INEL facilities and infrastructure will be used.
- No other programs or projects are assumed to restrict CTF use.
- Facilities will be upgraded to meet current codes and standards.
- Costs are based on SNTP Program.

ETS SIZE INFORMATION

PIPET COMPONENT SIZES

Component	Qty	Diameter (ft)	Length (ft)	Nozzle sizes (IPS)
Debris tank	1	15'-6" ID	30'-0" Tan.-Tan. ~38' Overall	24" ID inlet 60" OD outlet
Hot Gas Cooler	1	11' - 0" OD	60'	60" OD inlet 42" OD outlet
Process gas filter	4	9'-0" OD	30'-0" Tan.-Tan.	24" OD
Cryogenic mixer	1	4'-0" OD	5'-0"	
Noble gas adsorber	8	8'-0" OD	8'-0" Tan.-Tan.	20" OD

ETS COMPONENT ARRANGEMENT EVALUATION

Arrangement Option	Ramification
No Confinement	(1) Maximum radiological release
Reactor Only	(1) Maximum radiological release (2) Difficult materials problems
Reactor and Debris Trap	(1) Confinement of majority of particulate (2) Adequate access for maintenance (3) Single Large Containment Vessel Penetration Reqd.
Rx, Debris Trap, Heat Exchanger	(1) Confinement of majority of particulate (2) Adequate access for maintenance (3) Redesign of hx required (4) Several Large Containment Vessel Penetrations Reqd
RX, DT, Hx, Process Filters	(1) Confinement of all particulate (2) Reduced access for maintenance (3) Redesign of hx required (4) Several Large Containment Vessel Penetrations Reqd
RX, DT, Hx, Process Filters, Gas Adsorbers	(1) Confinement of all particulate (2) Very limited access for maintenance (3) Redesign of hx required (4) Several Large Containment Vessel Penetrations Reqd

PROPOSED ETS CONFIGURATION

- Size and Number of ETS Components Favored Locating Part of System Outside of Containment Vessel
- ETS Inside Containment Vessel Negated Flexibility for Other Test Reactor Programs
- Higher Temperature Components Located in Containment Vessel

The Cost Evaluation Results

- A potential savings is possible from the use of existing facilities.

CTF SCHEDULE

- Current Preliminary Project Schedule for PIPET starts In-pile Testing in 1st Quarter of 1997.
- INEL experience indicates that the design and procurement of large high-pressure storage tanks will be critical path.
- The use of existing CTF facilities will allow an earlier start of facility equipment installation.
- Significant reactor testing infrastructure exists to support the PIPET activities.

The PIPET schedule is not impacted at INEL.

The Bottom Line

The existing facilities are robust and provide ample space for the planned operations with the potential for both cost and schedule improvement.

N93-26968

SPACE NUCLEAR THERMAL PROPULSION

SNTP

US AIR FORCE
Phillips Laboratory
Grumman, Babcock & Wilcox
Brookhaven & Sandia National Labs

Air Force Facility

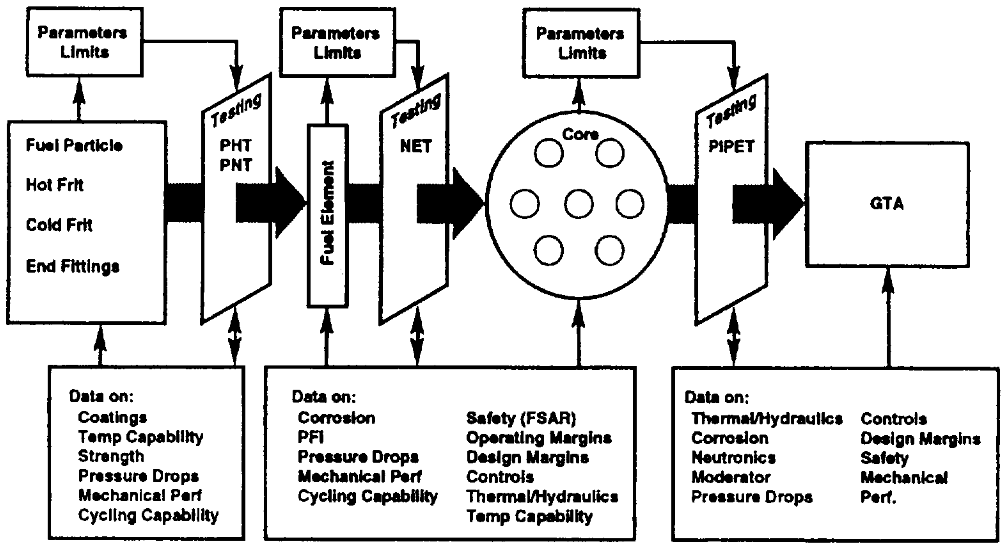
David F. Beck
PIPET Project Manager
Sandia National Laboratories

The Space Nuclear Thermal Propulsion (SNTP) program is an initiative within the U.S. Air Force to acquire and validate advanced technologies that could be used to sustain superior capabilities in the area of space nuclear propulsion. The SNTP program has a specific objective of demonstrating the feasibility of the particle bed reactor (PBR) concept.

SNTP

The term PIPET refers to a project within the SNTP program responsible for the design, development, construction and operation of a test reactor facility, including all support systems, that is intended to resolve program technology issues and test goals.

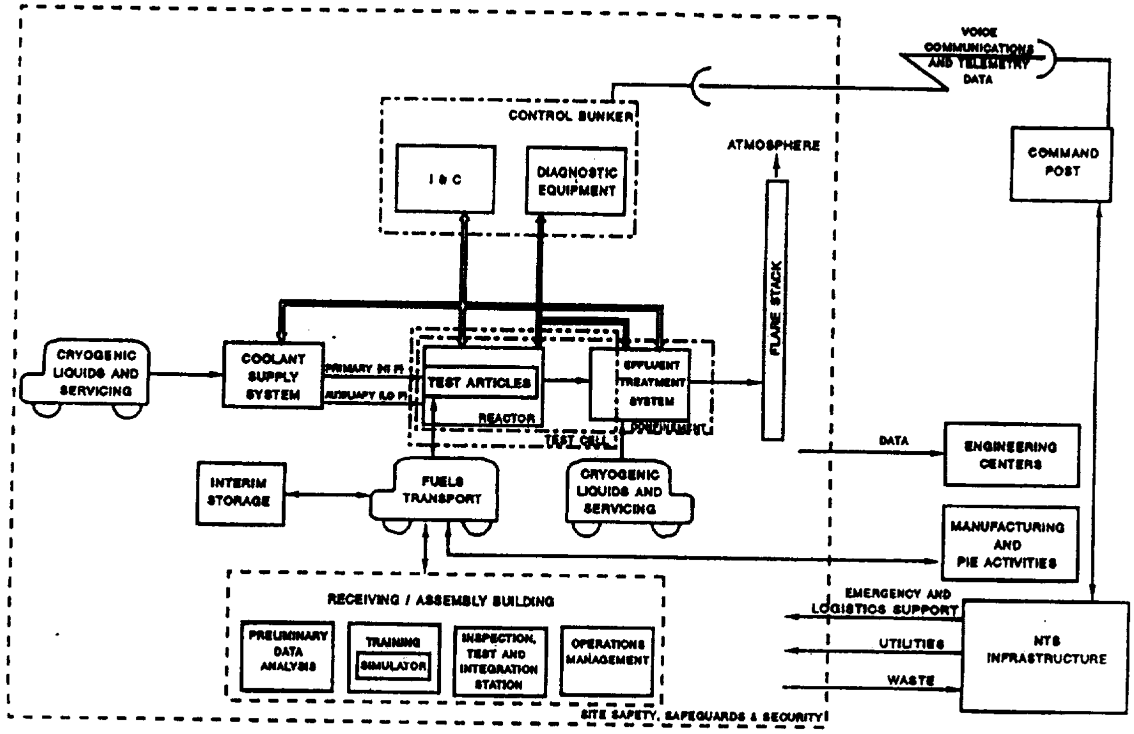
Experiment Data Flow



The PIPET project will provide the necessary capability to complete the final steps in the SNTP program nuclear test plan.

No known reactor facility in the world is capable of providing prototypical test conditions for SNTP PBR fuel or fuel elements. Although certain nuclear tests (pre-PIPET) within the current SNTP program may probe the design envelope of the fuel and fuel element, the best that can be accomplished is very short run times and very low flow conditions for sub-sized or nonstandard fuel element designs (e.g., PNT and NET). The high-power densities that make the PBR so attractive will never be tested to prototypical design conditions until the PIPET element-test reactor is built.

No operational reactor facility in the U.S. is capable of testing a flight-like NTP reactor core or engine under power (some limited capability exists in the CIS, but even this does not include any cryogenic hydrogen support and is not currently configured for propulsion type testing). No facility in the world is capable of providing nuclear test support for NTP reactors or engines under the current and rightful concern for protecting the environment and public health. The investment in building a high power density fuel element test reactor can be leveraged into a facility that can also provide test support in meeting certain NTP ground test requirements.



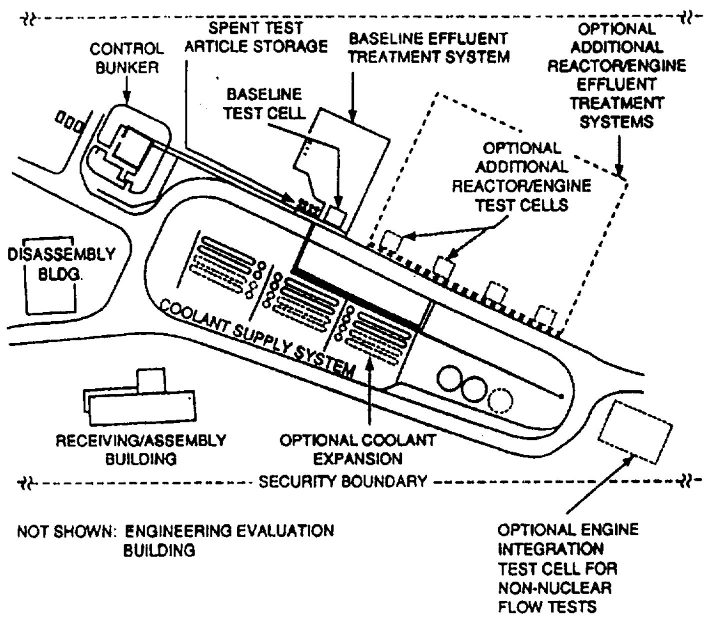
PIPET TEST REACTOR SYSTEM

The PIPET system includes:

1) Major interfaces with the host site for utilities & logistics support.

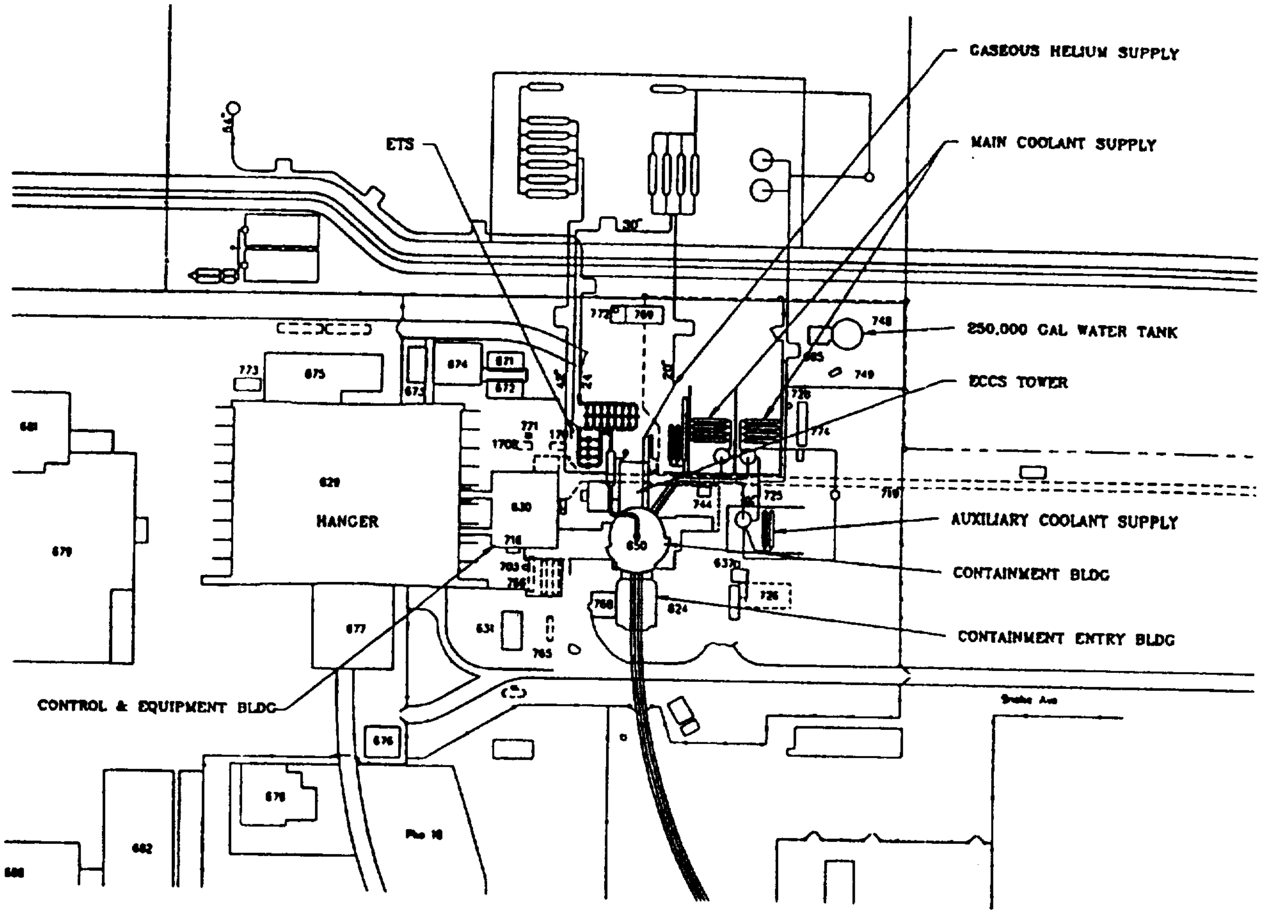
2) Facilities including a control bunker, a receiving and assembly building, temporary dry storage areas for irradiated materials, a disassembly building, and test cell(s).

3) A reactor coolant supply system consisting of a cryogenic hydrogen supply and hydrogen effluent treatment system.

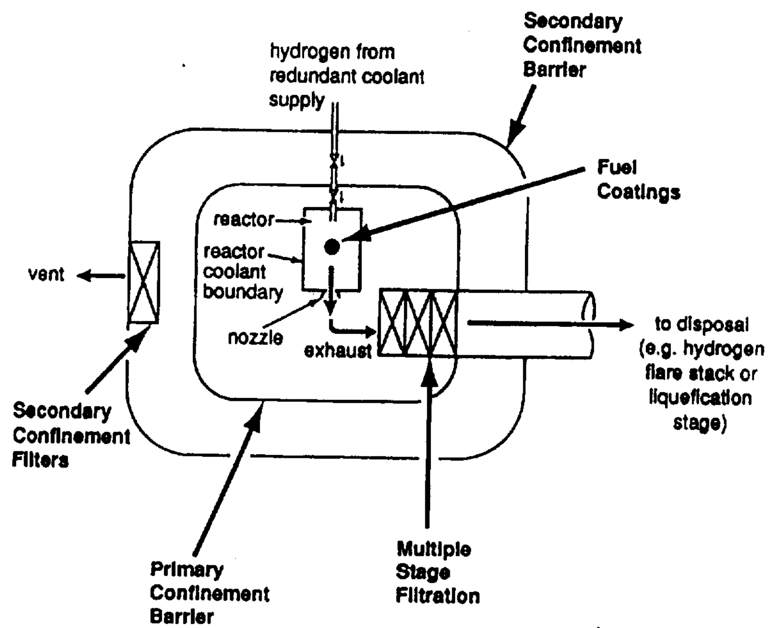


SNTP Baseline Facility Conceptual Site Plan Illustrating Laydown Space for Optional Expansions

One location for the PIPET test station supported by the SNTP program Environmental Impact Statement is a "green-field" location on the Nevada Test Site (NTS). This would involve essentially all new construction, with designs developed to meet program requirements.

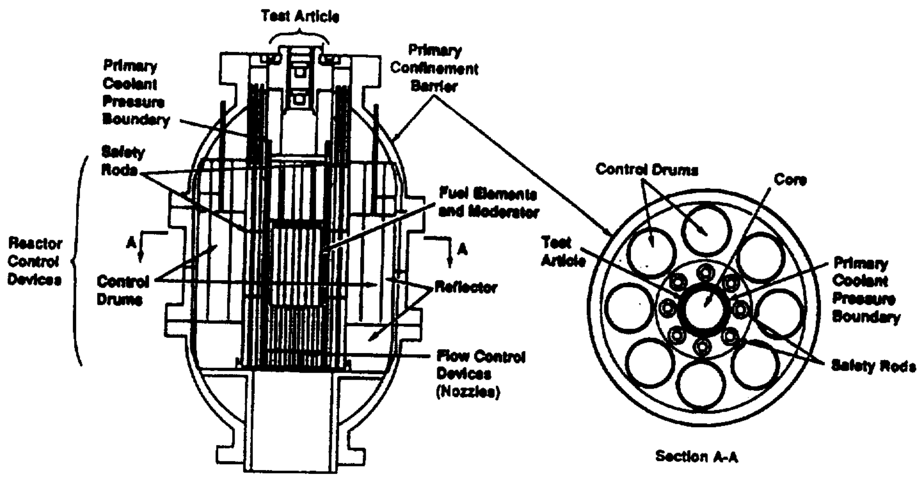


The second alternative site for the PIPET facility is a location within Test Area North (TAN) of the Idaho National Engineering Laboratory (INEL). This would involve renovation, adaptation and use of existing structures such as the Contained Test Facility (CTF) and TAN 607 Hot Shop Complex.



Depiction of the Use of Multiple Physical Barriers and Single-Failure Criteria in Preventing the Unmitigated Release of Fission Products

The Space Nuclear Thermal Propulsion program is committed to achieving the highest practicable levels of safety both in program activities and in the ultimate safety both in program activities and in the ultimate product of the program. Safety considerations will include: protection of the health and safety of the public; protection of the health and safety of all employees where program activities are done; protection of the environment and lands from contamination or damage as a result of program activities; and protection of the property and facilities used in the program. Unmitigated release of fission products is prevented by use of concepts such as 'defense in depth.' This includes administrative, physical, and operational controls and measures. Physical controls for ground testing on NTP concepts involve multiple barriers including fuel coatings, primary confinement systems, and secondary confinement systems. Physical barriers to be employed that will prevent the unmitigated release of fission products are diagrammed above. As implemented for the SNTP program, the primary confinement barrier around the reactor looks much like a reactor vessel in a conventional power plant design, but is functionally much different. The mechanical structure used to support and direct flow through the multiple stage filtration system also serves as the balance of the primary confinement barrier. The secondary barrier includes the test cell structures, which may serve multiple functional needs (for example, weather protection and shielding).



Simplified Drawing of the Test Bed Preliminary Design

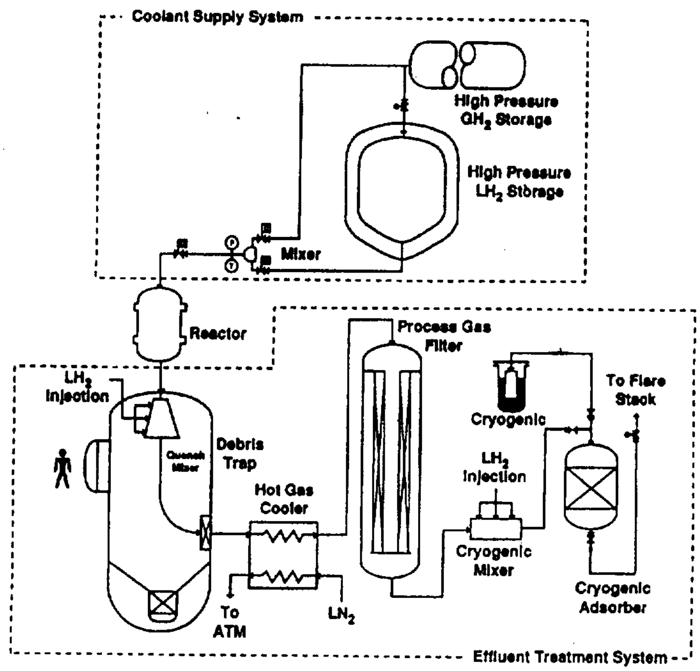
The test reactor by design contains two major subsystems — a test bed and a test article.

The test bed nominally provides:

1. A primary fission product confinement barrier.
2. Interfaces between the test article and other programmatic equipment (for example, coolant supply, effluent treatment, and instrumentation and controls).
3. An experiment volume in which the test article (fueled portion of the reactor) is tested.
4. Independent reactivity systems to bring the overall reactor system to the desired preoperational reactivity state; control startup, shutdown, and operational transients; and provide scram capability.

Test articles are designed for ease of removal to enable rapid test turnaround, ease of reconfiguration, and minimal worker exposures. Reactivity controls within the test bed are designed for ease of removal, so that test articles containing their own reactivity control mechanisms can take advantage of the confinement and programmatic equipment interfaces without having to relay on other design features. Test article design options can thus be seen to include:

1. A hybrid core design where a previously qualified test article design has a single fuel element replaced with a new design.
2. A new test article that makes use of all the inherent features found in the test bed.
3. A new test article with integral reactivity control systems, only making use of the confinement barrier and subsystem interfaces of the test bed.
4. Replacement of the entire test bed/test article assembly with a new reactor design.



Simplified Line Diagram Of The Baseline Reactor Coolant System

A primary coolant system has been designed that meets the safety and performance requirements of the SNTP program for use in the development, demonstration, and qualification of NTP fuel elements, reactors, and engines. (Integrated stage qualification, including high-altitude simulation, is not a requirement for the current program.) The functional requirements of the reactor coolant system design includes:

1. Provide an adequate, redundant, highly reliable supply of cryogenic hydrogen at required pressures, temperatures, and flow rates (hydrogen supply - coolant supply system).

2. Interface with the primary heat source (test reactor or engine).

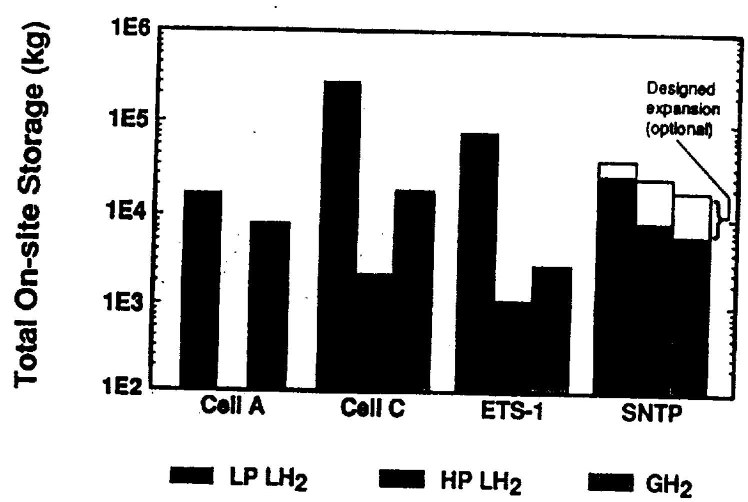
3. Cool the hot primary flow to temperatures compatible with structural and heat exchanger materials. Catch any core debris material resulting from failures (planned or unplanned) and maintain it in a coolable, subcritical configuration. Allow access for remote/robotic retrieval of core debris. Provide initial, coarse-filtering to prevent downstream heat exchanger plugging and act as a getter for plate out of fission products with boiling points above the cooldown temperature (debris trap).

4. Provide additional cooling of exhaust flow to temperatures compatible with downstream particulate filters (hot gas cooler).

5. Filter out particulates entrained in the exhaust flow (process gas filter).

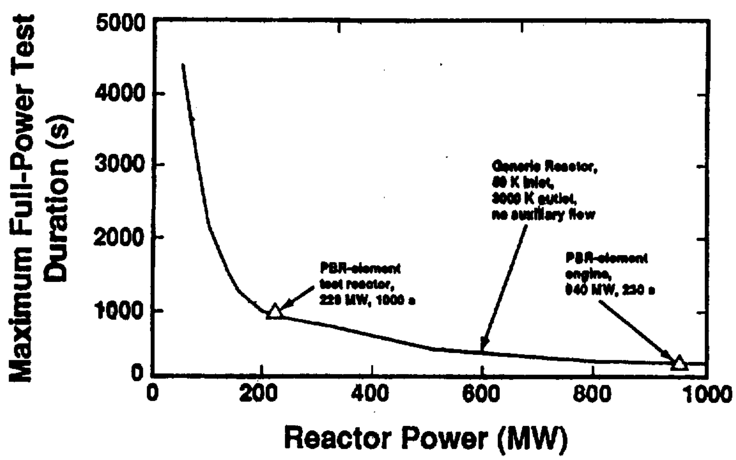
6. Retain any fission products still in volatile form (for example, krypton and xenon) for a sufficient time to allow for decay (cryogenic mixer/adsorber stage).

7. Dispose of cleaned effluent (flare stack).



NTP Facility Baseline Hydrogen Storage Capacities

The PIPET facility includes an initial, baseline coolant supply capacity designed to envelope the minimum test duration requirements of the SNTP program. Optional supply system expansions are planned that will provide capability to meet maximum test duration requirements. The figure above provides a comparison between the planned SNTP program PIPET test facility on-site hydrogen storage capacities against the test-cell hydrogen installations of the ROVER/NERVA Program.



Baseline Facility Test Durations

The planned baseline reactor coolant supply system, although designed to meet several operating point requirements, is best represented by an extensive set of operating envelopes that are a function of mass flow rates, temperatures and pressures. However, to illustrate the system performance, a generic NTP reactor was used to generate a test duration envelope as a function of reactor power. This curve is, roughly speaking, a line of constant energy. Also shown are operating points for two conceptual PBR test article designs.

SUMMARY

- A nuclear test facility has been designed that meets SNTP facility requirements including:
  - safety and environmental policies
  - minimum impact on waste streams
  - provisions for appropriate safeguards and security
  - meets minimum SNTP performance levels
  - supports expansion to maximum SNTP performance levels
- The design approach taken to meet SNTP requirements has resulted in a nuclear test facility that should encompass a wide range of NTP test requirements that may be generated within other programs. The SNTP PIPET project is actively working with DOE and NASA to assess this possibility.

Additional information concerning these facilities can be found in:

Allen, G.C. et al. (1992), "Ground Test Facilities for Evaluating Nuclear Thermal Propulsion Engines and Fuel Elements," in Proceedings of the 1992 Nuclear Technologies for Space Exploration, Jackson, WY, 16-19 August 1992, pp 514-523.

Beck, D.F. et al (1993), "Test Facilities for Evaluating Nuclear Thermal Propulsion Systems," to be presented at the Tenth Symposium on Space Nuclear Power and Propulsion, Albuquerque, NM, January 1993.

Shipers, L.R., and Allen, G.C. (1992), "Handling Effluent From Nuclear Thermal Propulsion System Ground Tests," presented at the Third Specialist Conference on Nuclear Power Engineering in Space Nuclear Rocket Engines, Semipalatinsk-21, Republic Kazakhstan, September.

Shipers, L.R., and Brockmann, J.E. (1993), "Effluent Treatment Options for Nuclear Thermal Propulsion System Ground Tests," to be presented at the Tenth Symposium on Space Nuclear Power and Propulsion, Albuquerque, NM, January 1993.

Nuclear Technology Department

EFFLUENT TREATMENT FOR NUCLEAR THERMAL PROPULSION GROUND TESTING

Larry R. Shipers

NUCLEAR PROPULSION
TECHNICAL INTERCHANGE MEETING - 1992
NP-TIM-92

NASA-Lewis Research Center
Plum Brook Station

Sandia National Laboratories

Ground testing of fuel, fuel elements, and engine assemblies at a suitable facility is required to support the development of nuclear thermal propulsion (NTP) systems. Given the current Environmental Safety and Health (ES&H) regulations, policies, and guidelines in the USA, it is not planned today to vent the potentially contaminated hydrogen that these tests will generate directly to the environment. In order to minimize the potential safety and environmental impacts of NTP ground tests, the gaseous reactor effluent needs to be confined, treated, and/or scrubbed of radioactive fission products prior to its unrestricted release.

Nuclear Technology Department

Objectives

Define Treatment Functions
Review Concept Options
Discuss PIPET ETS Concept
Outline Future Activities

Sandia National Laboratories

Over the years, several different options have been evaluated by Sandia National Laboratories to either process the hot hydrogen effluent simultaneously with the test being conducted or configure the test facility in a manner that real time processing is not required. The evaluation effort was initiated by identification and formulation of a wide range of concept options to treat NTP test article exhaust. The concept options considered ranged from closed cycle (venting the exhaust to a closed volume or recirculating the hydrogen in a closed loop) to open cycle (real time processing and venting of the effluent). A number of variations of these general concepts are still under consideration. This paper defines the functions any effluent treatment system must perform, reviews the various concept options to handle effluent from nuclear thermal propulsion system ground tests, presents the current lead effluent treatment concept for the PIPET project, and outlines future effluent treatment studies to be performed.

Nuclear Technology Department

Reactor Exhaust

Hydrogen Flow at 1 - 40 kg/s
Temperatures in Excess of 3000 K
Trace Concentrations of Particulate, Volatile Species, Halogens, and Noble Gases
Entrained Core Material and Debris

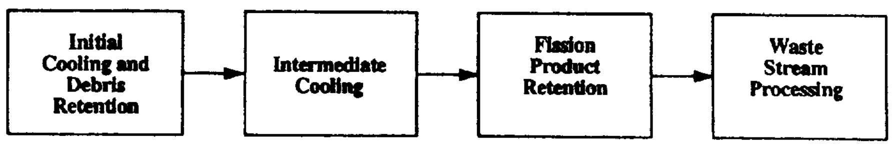
Sandia National Laboratories

Prismatic (NERVA Derived), particle (PBR and Pellet bed), and refractory (Cermet, Wire Core) fuel forms are all candidates for ground testing as a part of a NTP development program. Consideration of these varied concepts leads to a consistent set of functional requirements for any system designed to treat the reactor exhaust during ground testing. In all cases, fuel operating temperatures in the range of 2700 - 3400 K are planned. Significant quantities of cryogenic hydrogen will be required to cool NTP reactors tested under prototypic conditions. Small fuel element test reactors with powers on the order of 50 MW would require 1 kg/s coolant flows while large ground test of reactors with powers as high as 2000 MW would require coolant flows in the range of 40 kg/s.

As the hydrogen coolant flows through a fuel element and is heated by direct contact with the nuclear fuel, it can be expected to become contaminated with fission products and/or fuel particulate. The potential for the generation of other hazardous compounds within the hydrogen also exists. The risk of significant contamination is especially high early in the development process when new and advanced fuel forms are expected to be tested. The reactor exhaust can also be expected to contain significant quantities of core material and debris. The effluent treatment system design must allow for the potential of significant core failure and relocation that may occur during the development of any NTP concept.

Nuclear Technology Department

Effluent Treatment Functions



Sandia National Laboratories

Any system designed to treat the exhaust from a solid core NTP ground test reactor must perform four basic functions:

1. Initial cooling of the hot reactor exhaust to temperatures compatible with normal engineering materials. In addition, any debris and large particulate ejected from the core must be retained and maintained in a subcritical configuration.

2. Intermediate cooling to temperatures at or below atmospheric. While this cooling stage is not necessary, its inclusion in the system enhances the performance of many concepts.

3. Fission product retention to prevent uncontrolled release of contaminants to the environment. This stage must be designed to retain small particulate, halogens, noble gases, and other volatile species.

4. Waste stream processing to properly handle retained fission products, cleaned or processed hydrogen effluent, and any other potentially contaminated fluids introduced in or generated by the system.

The collection of components that performs these functions is normally referred to as an effluent treatment system (ETS).

Nuclear Technology Department

Effluent Treatment Categories

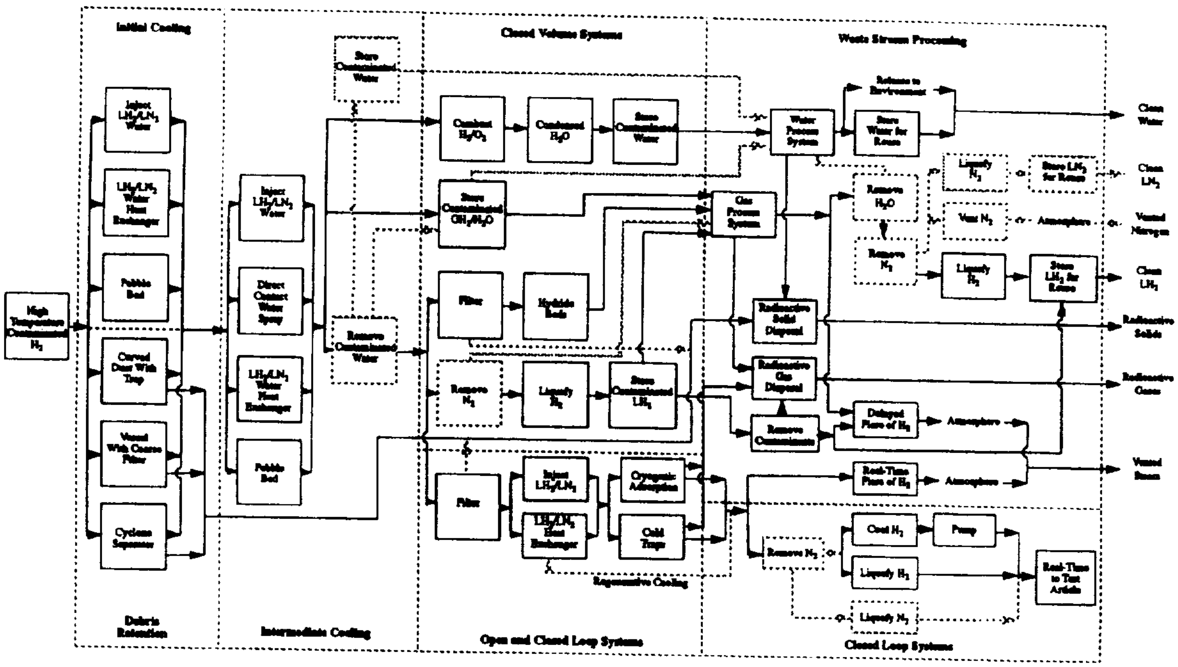
Closed Volume Systems
Delay and Accumulate Effluent

Open Systems
Real-Time Effluent Processing

Closed Loop Systems
Recirculate Effluent as Coolant

Sandia National Laboratories

ETS concepts can be grouped into three broad categories: closed volume systems, open systems, and closed loop systems. Closed volume systems delay and accumulate the effluent generated during reactor power operations and then process the effluent at much reduced flow rates at some time after power operations. Closed volume systems include concept options such as venting the effluent to storage vessels or metal hydrides. In an open system, the effluent is processed and vented to the atmosphere as it is produced during reactor power operations. Open systems are characterized by large capacity filtration and adsorption equipment. A closed loop system performs real time processing of the effluent and then recirculates the hydrogen to the reactor inlet to be reused as coolant. Care must be used when comparing a closed loop system to other types of ETS concepts. The closed loop system both treats the reactor exhaust and performs the additional function of supplying coolant to the reactor inlet. The appropriate functional relationship is maintained when a closed loop system is compared to another ETS concept in combination with the concept and components used to supply coolant to the test reactor.



Effluent Treatment Options

Sandia National Laboratories

A map of effluent treatment options is shown. The high-temperature contaminated hydrogen effluent is shown entering on the left. Waste products resulting from the treatment process are should on the right. The major functional divisions of initial cooling, debris retention, closed volume systems, open and closed loop systems, and waste stream processing are labeled and outlined in dashed lines. Tracing a path through this figure (with appropriate consideration of branching) will define a complete functional effluent treatment system.

The commonalities of ETS component options and the impacts of component choices are illustrated. Each of the three categories (closed volume, open, closed loop) of effluent treatment concepts have the same options for components to perform the initial cooling, debris retention, and intermediate cooling functions. The concepts differ in the components used for fission product retention and waste stream processing. The choice of the method used for initial cooling can also influence the components that must included in the intermediate cooling, fission product retention, and waste processing stages. Optional downstream functions which may be required (dependent upon upstream component choice) are shown with dotted lines.

Concept Evaluation

Total System Approach
Reliability and Redundancy
Passive Systems
Avoid Exotic Materials and Concepts
Maintenance, Inspection, and Testing
Support and Posttest Processing Systems
Expansion Potential
Capital and Life Cycle Costs
Decontamination and Decommissioning

Sandia National Laboratories

Evaluation of effluent treatment concepts should be performed from a total system approach considering potential environmental impacts, safety, operations, potential future activities, and total cost. Any system designed must have a high degree of reliability and redundancy. Passive systems, such as blowdown rather than pumping, should be employed whenever practical. Exotic materials and concepts should be avoided. Steps should be taken to minimize occupational exposure during required in-service maintenance, inspection, and testing. Performance of the maintenance and inspection using remote or robotic means should be considered. The ETS support systems (coolant storage, water removal, etc.) and post test processing systems (decay heat, pebble bed heat, waste processing, etc.) can have significant impacts on overall system complexity and cost. The potential for future expansion should be considered. Any ETS concept is, to a first approximation, a power limited system. If it is desired to significantly increase reactor power (and thus flow) it would be necessary to significantly increase the size of the velocity limited components or to use process trains in parallel. A total energy limit, defined by the system storage capacity (coolants, heat sinks, closed volume fission product retention, etc.), also exists for an ETS. Both the first and the life cycle costs of system options should be evaluated. Evaluation to date has shown that the use of large complex equipment and systems should be minimized for a limited testing program since a large number of tests are required to offset the increased capital cost with decreased operating costs. The system end of life decontamination and decommissioning costs should also be considered.

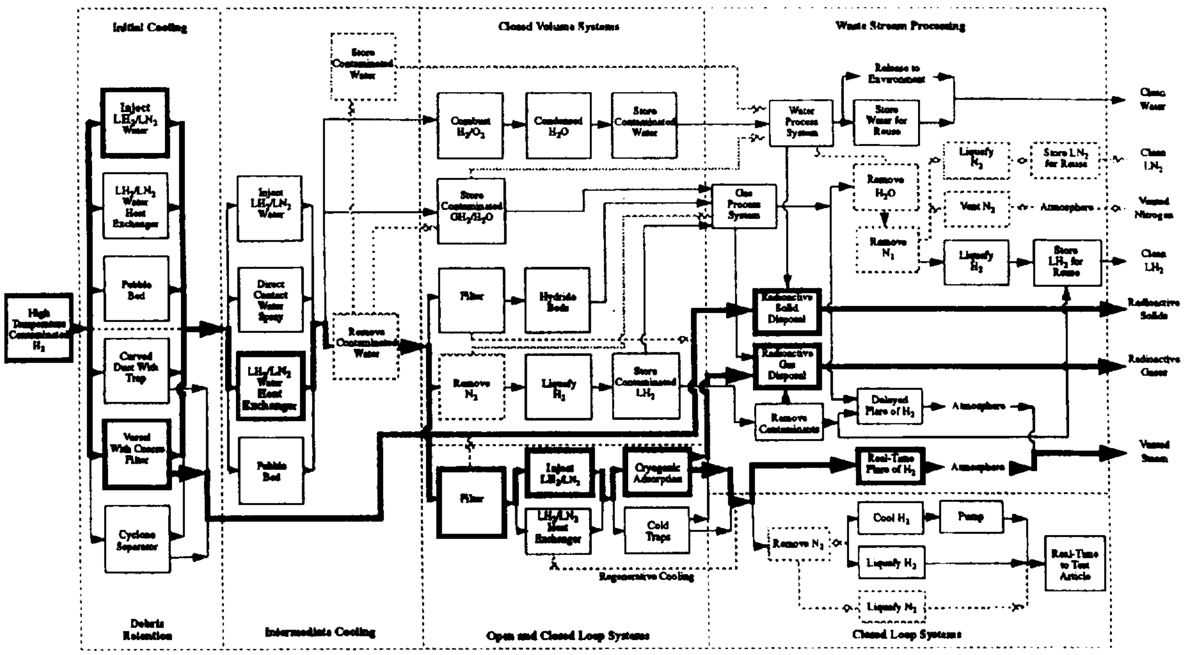
Nuclear Technology Department

PIPET ETS Envelope

Maximum Reactor Power		1 GW
Duration at Maximum Power		240 sec
Duration at 40 MW Power	$\geq$	1 hr
Maximum Flow at 3000 K		20.4 kg/s
Maximum Flow at 1100 K		66.4 kg/s
Inlet Pressure at Maximum Power		1.4 MPa
Inlet Pressure at 120 MW		0.4 MPa

Sandia National Laboratories

The current PIPET effluent treatment system is designed to support operation of ground test reactors at power levels up to 1 GW. The maximum duration of continuous full power operation is limited by the available coolant storage. The current design will support operation of 1 GW test reactors with a 3000 K exhaust temperature for a duration of 240 sec. Duration is increased if the reactor is operated at either a lower power level or a lower mixed mean inlet temperature. Durations well in excess of 1 hour may be obtained by the current ETS design for reactor powers in the range of 40 MW. The system volumetric flow rate is limited by the interstitial velocity in the system filtration and adsorption components. This leads to an inlet mass flow rate limitation that is a function of the effluent mixed mean temperature. The maximum inlet flow rate is 20.4 kg/s at a 3000 K inlet temperature. As the effluent temperature is reduced, the maximum allowable inlet mass flow rate increases. At a mixed mean effluent temperature of 1100 K, the allowable inlet mass flow rate is 66.4 kg/s. The volumetric flow constraint also establishes the system operating pressure limits. In order to reduce the size of the system components, the ETS was designed to operate at an inlet pressure of 1.4 MPa for the maximum flow and power conditions. This design pressure is sufficiently below the reactor design operating pressures (6.9 MPa chamber and 3.4 MPa throat) to insure decoupling the test article pressure response from that of the ETS. As the reactor power (and inlet flow) are reduced the system operating pressure may be reduced while a constant volumetric flow rate is maintained. At a reactor power of 120 MW the current ETS could be operated at an inlet pressure as low as 0.4 MPa.



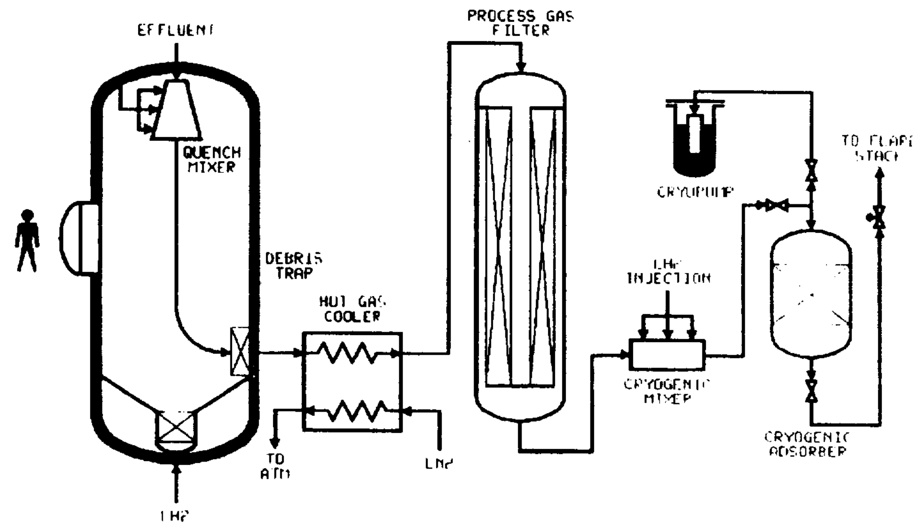
PIPET Effluent Treatment Concept

Sandia National Laboratories

The effluent treatment concepts illustrated were evaluated during the development of the PIPET concept (shown in heavy lines). Concepts in addition to the lead concept (including water injection, gasholder, hydride, heat exchanger, pebble bed, and closed loop systems) have been developed to high levels and are still under consideration. The lead PIPET effluent treatment concept is an open system that uses liquid hydrogen injection for initial cooling, a liquid nitrogen heat exchanger for intermediate cooling, granular filters to remove particulate, liquid hydrogen injection to cool to cryogenic temperatures, and cryogenic charcoal adsorbers to remove halogens, noble gases, and other volatile species. A flare stack combusts the treated hydrogen effluent prior to venting to the environment.

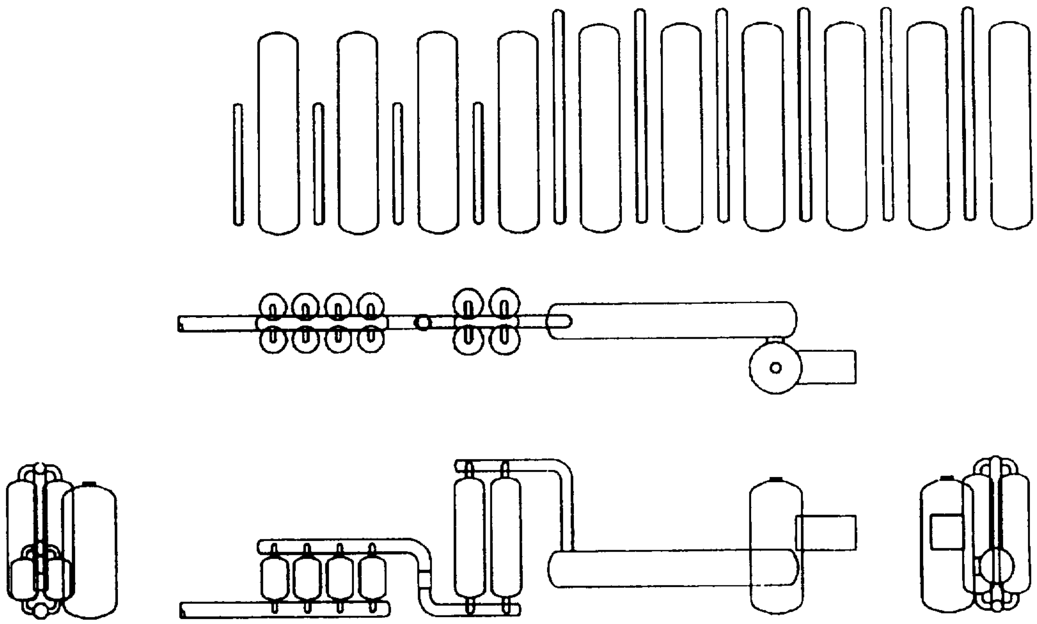
Provisions are included to handle both the solid contaminants retained in the debris trap and gaseous contaminants retained in the cryogenic adsorbers. Access is provided to remove debris retained in the trap between operations. The filters and adsorbers are designed to retain the trapped particulate and halogens for the life of the facility. However, the noble gases are only retained in the adsorbers when cryogenic temperatures are maintained. When the adsorbers warm, the xenon and krypton will off-gas. Provisions for two procedures for the long-term disposal of the noble gases are incorporated into the design. The adsorbers may be isolated (valves included in the design) (1) to allow the noble gases to decay prior to releasing to the environment in a controlled manner or (2) to allow the noble gases to diffuse to a cryopump (included in the current design) to collect and concentrate the contaminants for appropriate disposal.

PIPET ETS CONCEPT



Sandia National Laboratories

The lead PIPET ETS concept is shown. The initial quench mixer (located in the debris trap) cools the effluent to 1100 K (a reasonable material upper limit temperature for stainless steel). The debris trap is a large jacketed liquid hydrogen cooled pressure vessel (~9.1 m x 5.5 m ID). A coarse filter is located at the exit of the debris trap to serve two functions: (1) to retain large particulate (on the order of 300-500 micron) in the debris trap and (2) to provide a large surface area and thermal mass for the plate out of any high temperature aerosols prior to leaving the debris trap. Access to the debris trap interior for inspection and debris removal is provided through an airlock. A large (~21 m x 3.4 m ID) liquid nitrogen to hydrogen tube in shell heat exchanger cools the effluent to ambient temperature. The heat exchanger cold side is operated at a pressure above that of the effluent stream so that leaks will not bypass the process train. Large (~9.1 m x 2.7 m OD) radial flow granular filters remove small particulate. The effluent enters by the inner annulus, flows radially outward and is collected in the outer annulus. A second liquid hydrogen injection quench mixer is used to cool the effluent to the 160 K cryogenic adsorber operating temperature. Large (~3.0 m x 2.4 m OD) axial flow cryogenic activated impregnated charcoal adsorbers remove halogens, noble gases, and other volatiles. A pressure regulating valve is located downstream of the cryogenic adsorbers to control the system operating pressure. Active pressure control during startup and shutdown may allow system operating pressure to be maintained sufficiently below the reactor operating pressure for decoupling of the test article pressure response from that of the ETS.



Sandia National Laboratories

A potential layout of the lead PIPET effluent treatment system concept has been developed. Top, front, left side, and right side views are shown. The liquid hydrogen and liquid nitrogen storage vessels (with their associated gas pressurization storage) are shown in the top view. Piping sizes range from 0.5 to 1.5 m diameter. Four granular filters manifolded in parallel are required by the current design. The eight required cryogenic adsorbers (manifolded in parallel) are also shown.

Future Activities

SEI Requirement Impacts

- **Increased Reactor Power**
- **Extended Duration**
- **Altitude Simulation**

Single Failure Evaluation

The impacts of SEI requirements on effluent treatment system design will be evaluated. These requirements include operation at increased reactor power, extended periods of continuous full power operation, and decreased system backpressure for altitude simulation. All of these design requirements may have significant impacts on ETS concept selection, design, and cost. Operating at increased reactor power (and flow) requires increased storage capacity for closed volume systems and either increased component size or parallel process train for open and closed loop systems. Increased duration requires large storage capacities for both open and closed volume systems. The need for low ETS operating pressures to support altitude simulation requires sufficient pressure recovery from the high-speed flow to overcome the system backpressure. Since many of the system components will be sized based upon flow velocity, the overall system size can be expected to increase as operating pressure decreases. The potential exists to incorporate a diffuser into the debris retention component design. Injectors or ejectors could be used to lower the system inlet pressure and cool the effluent stream.

Critical system functions (initial cooling, fission product retention, etc.) should be performed in a manner such that a single failure will not lead to loss of ETS function and fission product releases to the environment. The impacts to the public and the environment of ETS single component failures will be assessed. Appropriate features will be incorporated into the system design to mitigate any negative impacts.

NUCLEAR ELECTRIC PROPULSION

SYSTEM CONCEPTS

N93-26970

NASA LEWIS RESEARCH CENTER

Nuclear Electric Propulsion Systems Overview

Michael P. Doherty
NASA Lewis Research Center
Nuclear Propulsion Office

Presented at
NP-TIM-92
NASA LeRC Plum Brook Station
October 20, 1992

NUCLEAR PROPULSION OFFICE

NASA LEWIS RESEARCH CENTER

Nuclear Propulsion Background
Customer Technology Needs - NEP
Code SL Top-Level Requirements

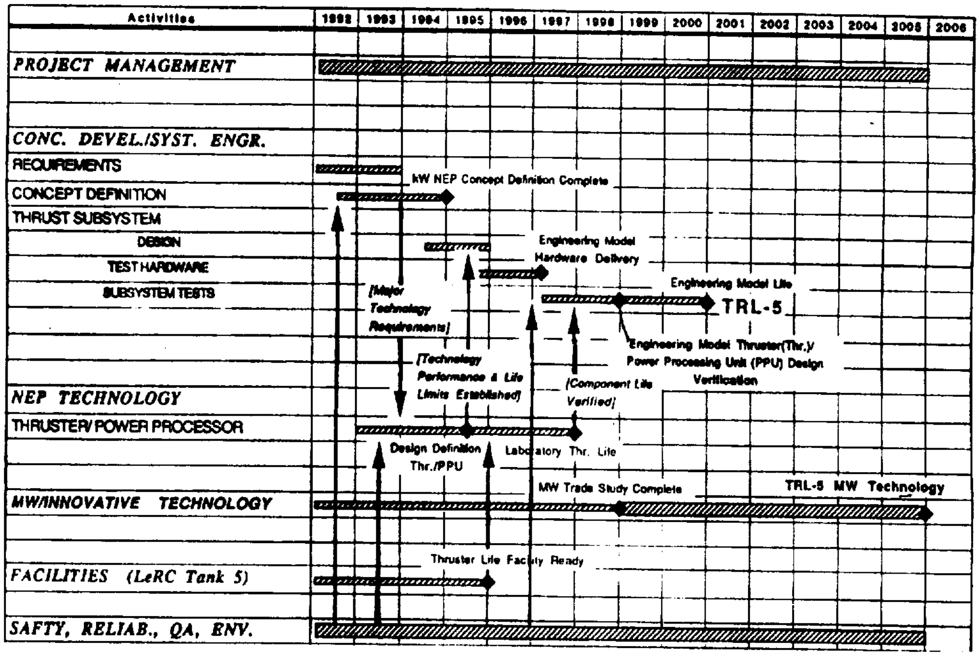
- **Time Frame:Long Term (> 10 years)**

- **Missions of Interest:**
  - Pluto Orbiter
  - Neptune Orbiter
  - Jupiter Grand Tour
  - Multiple Mainbelt Asteroid Rendezvous
  - Comet Nuclear Sample Return
  - Mercury Orbiter
  - Uranus Orbiter/Probe

- **Requirements:**
  - Generally, the Division foresees a need for low-thrust propulsion, in particular, nuclear electric propulsion (NEP). NEP would provide a large reduction in propellant mass, provide commonality from mission to mission, allow for launch date flexibility, and reduce trip times over conventional ballistic approaches. NEP would significantly enhance the mission feasibility/performance and science return and, in at least two instances, enable the mission (Jupiter Grand Tour and Pluto Orbiter).
  - The Division has need for a propulsion system with high reliability, longevity, autonomy, compactness, and safety. Specific requirements include:
    - Power Level of 50 - 100 kWe
    - Operate at Full Power for 4 - 8 years
    - Life Time of 8 - 15 years

NUCLEAR PROPULSION OFFICE

The primary customer for Nuclear Electric Propulsion, Code SL, the Solar System Exploration Division of the Office of Space Science and Applications (OSSA), foresees their need for NEP based upon its being the most viable means to provide for desirable science missions to a number of planetary, asteroidal, and cometary destinations early in the 21st century. NEP enables a number of the proposed missions and allows for orbiter missions to the major satellites of Jupiter, Uranus, Neptune, and Pluto, and yields more frequent launch opportunities. Analyses to date imply that successful and timely performance of the desired planetary missions will require a space nuclear electric power source rated nominally at 4 to 8 years full power life, 50-100 kilowatts-electric (kWe) power, and 25 watts per kilogram (W/kg) and ion electric engines having a specific impulse of 5000 to 10,000 seconds and 10,000 hours of individual thruster life.



Schedule for the Nuclear Electric Propulsion Project.

The Nuclear Electric Propulsion Project includes six elements: project management, concept development/ systems engineering, NEP technology, megawatt/ innovative technology, facilities, and safety/ reliability/ quality assurance/ environment.

The concept development/ systems engineering element will serve to document OSSA customer system requirements for NEP, define NEP systems which meet OSSA customer requirements, and design, fabricate, and test the required 100 kWe electric propulsion thrust system. The NEP technology element will serve to design, verify, and validate the performance and life of component technologies for electric thruster and power processor, and their required thermal subsystems. The MW/ innovative technology element will serve to identify technologies having benefit for higher power Moon and Mars NEP applications and to perform fundamental MW technology demonstration tests. The facilities element will serve to identify and advocate the facility infrastructure that is necessary for testing of kilowatt-rated non-nuclear technologies for NEP. The safety/ reliability/ quality assurance/ environment element will serve to perform studies and assessments to establish requirements upon the safe, environmentally acceptable design, development, test, deployment, and operations of space nuclear electric propulsion.

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NEP for the Space Exploration Initiative

- **Office of Exploration Requirements (PROJECTED)**
  - Mission: Mars Cargo and Piloted, with potential early use for Lunar Cargo Application
  - Reduced trip time for piloted missions
  - Reduced IMLEO for cargo, piloted missions
  - Provides launch date, stay time flexibility
  - Reduced resupply mass
- **Technology Readiness Level 5 by approximately 2005**
- **Critical Technical Performance Parameters**

- Electric Power to Thrusters:	5-10 MWe
- Specific Mass:	<10 kg/kWe
- Full Power Lifetime:	2-10 years
- Operation and Control	Autonomous
- Thruster Lifetime	10000 hours
- Restart Capability	Multiple

NUCLEAR PROPULSION OFFICE

Although not currently the baseline propulsion system for Moon/ Mars human exploration missions, NEP is being considered as a possible means to meet the Office of Exploration (OEXP) requirements for transportation of cargo and crew to Mars. The OEXP requirements are shown in the chart.

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NEP On-Going Systems Tasks

- **Power Conversion, Heat Rejection, and PMAD Modeling (MW)**
  - Create Models for Government Use
    - Power Conversion: K-Rankine and Brayton
    - Heat Rejection: Heat Pipe
    - PMAD: includes high temperature
- **Reactor Modeling (MW)**
  - Create Reactor Models for Government Use
    - High Temp Pin-Type (Liquid Metal Cooled)
    - Cermet (Liquid Metal Cooled)
    - High Temp Gas Cooled (UC/C matrix)
- **Concept Definition of System for Planetary Science (100 kWe)**

  Define and Baseline a System Which Has Multimission Capability

  Power Level Baselined

  System Configuration Established

  Implications upon ELVs Stated

NUCLEAR PROPULSION OFFICE

Key technical issues associated with megawatt NEP have been addressed by FY92 tasks in NEP flight processing, operations and disposal, and NEP operational reliability.

NEP concept development/ system engineering activities have also included modeling of NEP subsystems, specifically reactor, power conversion, heat rejection, and power management/distribution for megawatt applications.

Additionally, a conceptual definition study for 100 kWe NEP has recently been initiated. The objective of the study is to assess the applicability of a common NEP flight system to meet the specific propulsion requirements of the OSSA missions, accounting for differences in mission-specific payload and delivery requirements.

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NEP On-Going Systems Tasks (Continued)

- **Flight Processing, Operations, Disposal (MW)**
  - Assess the NEP Piloted Mission System and Profile, Identify Issues, Propose Resolutions
    - Launch Sequencing, LEO Basing, Assembly
    - Crew Rendezvous
    - On-orbit Refurbishment
    - Disposal
- **NEP Operational Reliability Assessment (MW)**
  - Reliability Assessment of Piloted Mission/ System to Identify Technologies Where There is a High Reliability Payoff

NUCLEAR PROPULSION OFFICE

Key technical issues associated with megawatt NEP have been addressed by FY92 tasks in NEP flight processing, operations and disposal, and NEP operational reliability.

NEP concept development/ system engineering activities have also included modeling of NEP subsystems, specifically reactor, power conversion, heat rejection, and power management/distribution for megawatt applications.

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20 kWe Mission/System Study

- **In response to HQ directive:**
  - provide a "good" set of 20-50 kWe NEP missions
  - delineate a flight system development program
- **Approach:**
  - conduct science and mission analysis activities (JPL lead)
  - conduct NEP system studies consistent with mission requirements (LeRC lead)
- **Products:**
  - 20-50 kWe mission set defined
  - flight system development plan, schedule, cost documented
- **Schedule: Late November**

NUCLEAR PROPULSION OFFICE

A joint JPL/LeRC mission/ system study for 20-50 kWe NEP has recently been initiated. The objectives of the study are to develop a good set of low power, near term "mission from planet Earth" NEP missions and to delineate a development program for 20-50 kWe class NEP, which lays the groundwork for the development of 100 kWe (greater than 10 year lifetime and reduced mass) class NEP necessary for outer planetary space science applications.

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Agenda

• 20 kWe System Studies (LeRC)	Jeff George
• 100 kWe Concept Definition (SAIC)	Alan Friedlander
• Reactor Subsystems (ORNL)	Felix Difilippo
• PC, HR, PMAD Subsystems (R/D)	Dick Harty
• MW Flight Processing (SAIC)	Mike Stancati
• MW Operational Reliability (SAIC)	Jim Karns

NUCLEAR PROPULSION OFFICE

The speakers to follow will provide further detail, analysis, results and conclusions of the systems concepts/ systems engineering tasks performed in FY92.

N93-26971

"20 kWe" NEP SYSTEM STUDIES

Nuclear Propulsion Technical Interchange Meeting
LeRC Plum Brook Station
October 20, 1992

Jeff George
Advanced Space Analysis Office

NASA Lewis Research Center
Advanced Space Analysis Office

Introduction

- Investigate low power options for nuclear electric propulsion (NEP) demonstration missions

- Use technologies which are applicable to later NASA missions through growth and scalability

- What is desirable in a "demonstration" system/mission?
  - Applicable to "production" systems and missions
    - Technologies
    - Power levels
    - Temperatures
  - Applicable to NASA mission needs

- LeRC inhouse power systems analysis:
  - Advanced Space Analysis Office
  - Power Technology Division

NASA Lewis Research Center
Advanced Space Analysis Office

Initial Study Groundrules

- Mission
  - 1998 - 2000 Launch
  - Launch to escape - No earth orbital spirals
  - Meaningful scientific return
  - Smallest feasible launch vehicle
- System
  - Near term technology
  - 2 - 3 year system lifetime
  - Scaled SP-100 reactor
  - Technology evolable to 100 kWe needed for outer planet exploration missions
- Groundrules will evolve as study progresses

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Advanced Space Analysis Office

Power System Groundrules/Assumptions

- 10 - 50 kWe
- 3 year life
- 2000 V to load
- 15 m reactor-to-payload separation distance
- 1.0×10^{12} n/cm^2
- 5×10^4 rad gamma
- 17 degree half-angle
- 10 % excess heat rejection capacity

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Power System Technologies Assessed

Reactor

- "Customized" SP-100
  - Scaled to meet thermal power requirements
  - Reactor redesign required
- Prototypical 2.4 MWt SP-100
  - Current design
  - Thermal power "rich" for 10-50 kWe

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Power System Technologies Assessed (cont.)

Power Conversion

- Thermoelectrics
  - Current SP-100 program choice
  - Static
  - Power limited to approx. few 100's kWe
  - $z = 0.67 \times 10^{-3}$ 1/K multicouple (Aug. 92 projected)
- Brayton
  - Dynamic
  - Scalable to multimegawatts
  - 1144 K demonstrated technology
  - 0.9 recuperator effectiveness
  - 1 + 1 redundancy (100%)
- Stirling
  - Dynamic
  - Power limited to approx. 1 MWe
  - 1050 K demonstrated technology
  - 1 + 1 redundancy (100%)

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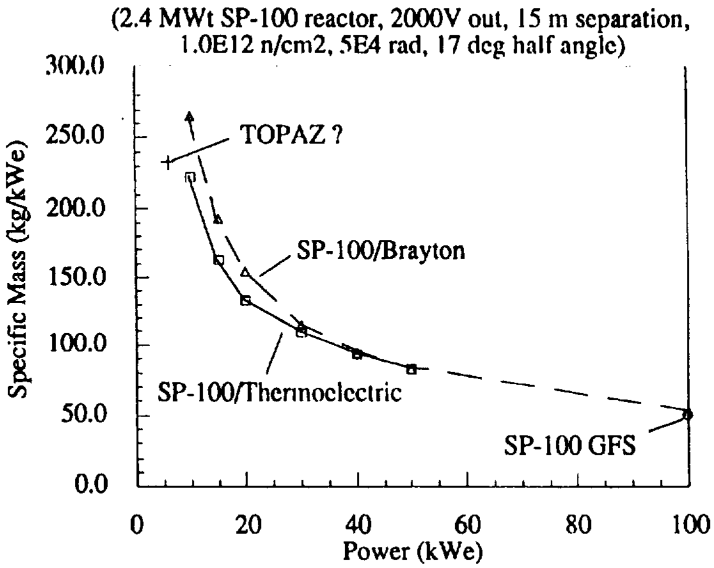
"Prototype" SP-100 System Specific Mass



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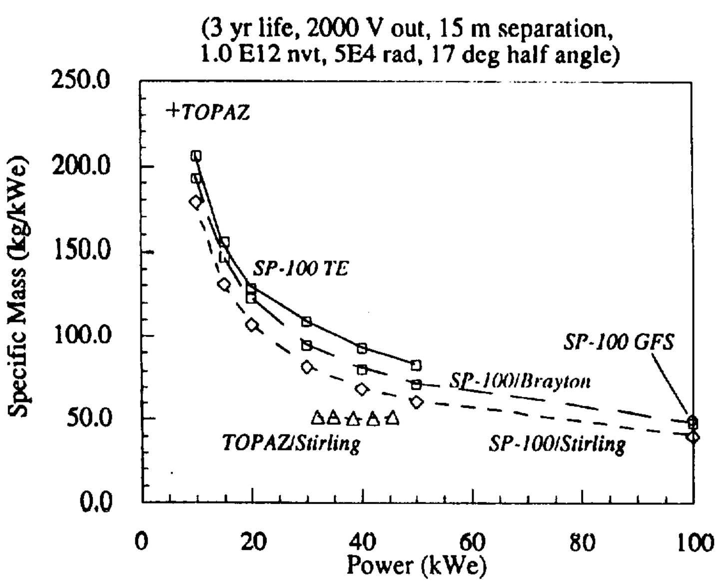
"Custom" SP-100 System Specific Mass



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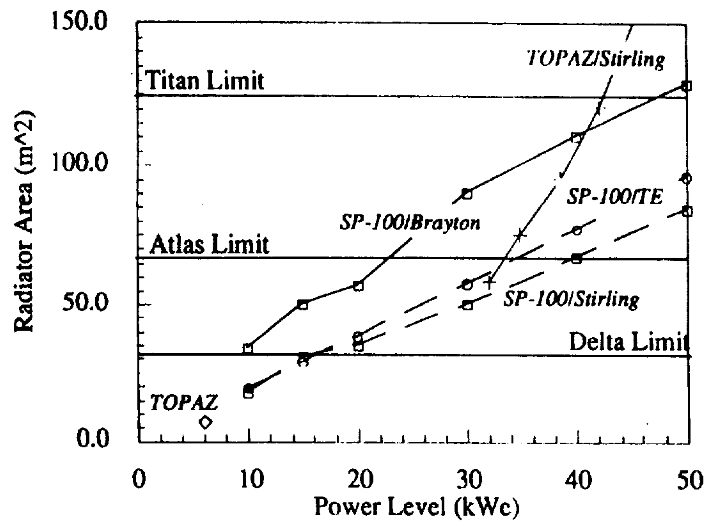
Radiator Packaging Limits

(No Deployment)



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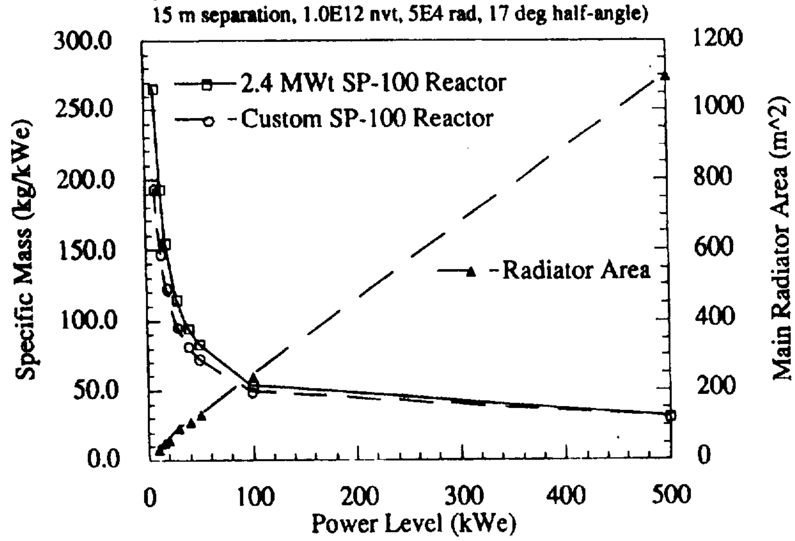
Brayton System Specific Mass and Radiator Area

(3yr life, 1144 K turbine inlet, 1+1 redundancy, 2000 V out, 15 m separation, 1.0E12 nvt, 5E4 rad, 17 deg half-angle)



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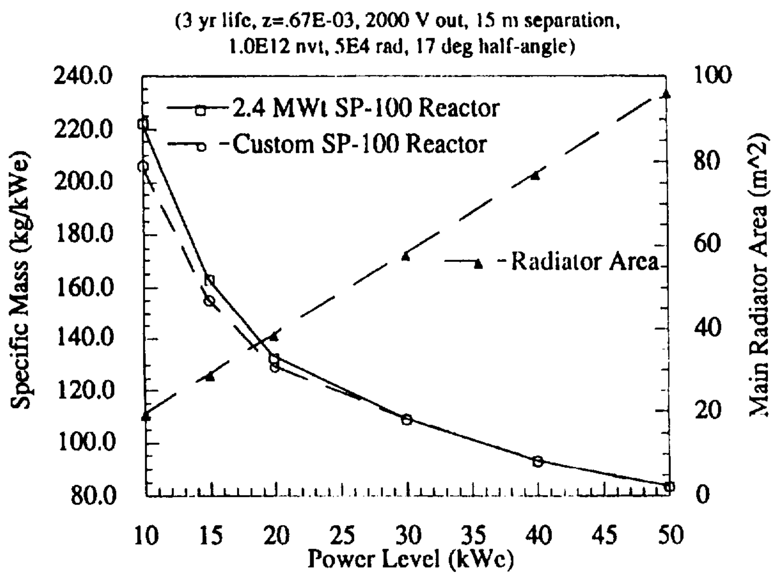
Advanced Space Analysis Office

Thermoelectric Specific Mass and Radiator Area



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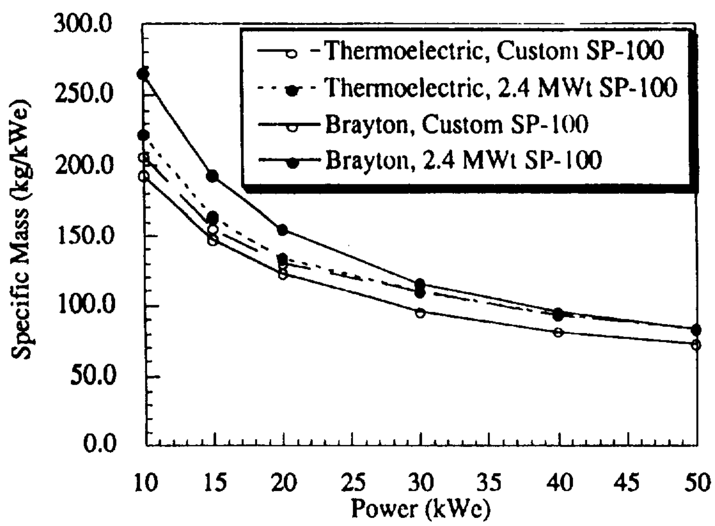
Specific Mass for "Prototype" vs. "Custom" SP-100-based Systems



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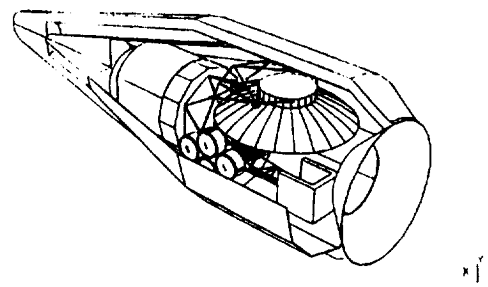
Advanced Space Analysis Office
NEP: System Concepts

System Packaging Limits on Power Level (kWe)

ELV	TE	Stirling	Brayton
Delta	15	15	10
Atlas	35	40	20
Titan	>50	>50	50



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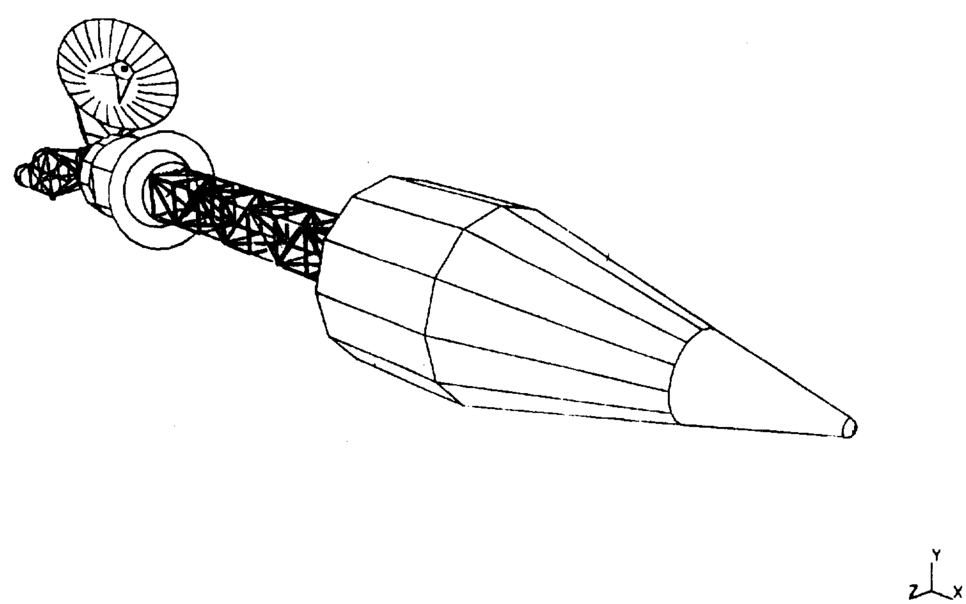
Conceptual NEP Science Mission Spacecraft Design



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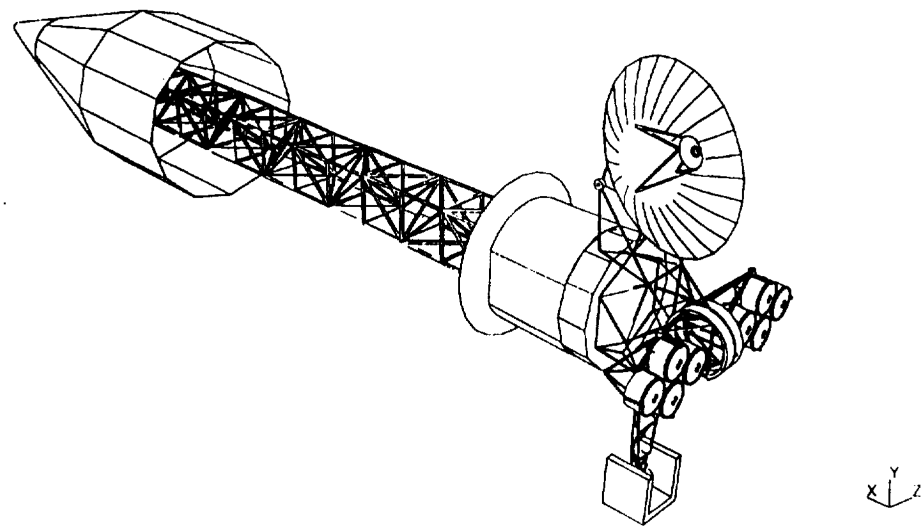
Conceptual NEP Science Mission Spacecraft Design



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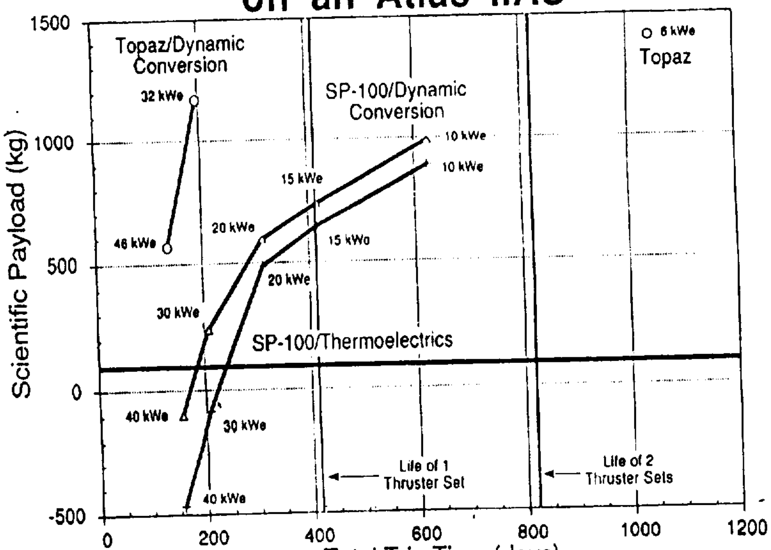
Van Allen Payload Delivery off an Atlas IIAS



Notes:	Xenon Ion Thruster Efficiencies 3500 Sec. Minimum Isp 5000 Sec. Maximum Isp	Initial NEP Spacecraft Orbit 1000 km Circular 30% Dry Mass Contingency NASA/LeRC/ASAO/SRO

NASA Lewis Research Center
Advanced Space Analysis Office

Summary

- Power system options for low power NEP demonstration missions investigated
  - 10-50 kWe
  - 2.4 MWt versus "Custom" SP-100
  - Brayton, Stirling, Thermoelectric

- Van Allen Mapper Mission identified as candidate 15 - 20 kWe demo.

- Investigation of other candidate missions continues

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Advanced Space Analysis Office

N93-26972

CONCEPTUAL DEFINITION of a 50-100 kWe NEP SYSTEM for PLANETARY SCIENCE MISSIONS

by

Alan Friedlander
Science Applications International Corp.
Schaumburg, Illinois

at

Nuclear Propulsion Technical Interchange Meeting
NASA-LeRC Plum Brook Station

October 20-23, 1992

SAIC

STUDY OBJECTIVES and SCOPE

- OVERALL TASK OBJECTIVE

  SAIC's Task Order 23, under Contract No. NAS3-25809 for NASA LeRC (NPO), has the Phase I objective of assessing the applicability of a common NEP flight system of the 50-100 kWe power class to meet the advanced transportation requirements of a suite of planetary science (robotic) missions, accounting for differences in mission-specific payloads and delivery requirements.

- CANDIDATE MISSIONS (post-2005 Launch Dates)

  (1) Comet Nucleus Sample Return
  (2) Multiple Mainbelt Asteroid Rendezvous
  (3) Jupiter Grand Tour (Galilean satellites and magnetosphere)
  (4) Uranus Orbiter/Probe (atmospheric entry and landers)
  (5) Neptune Orbiter/Probe (atmospheric entry and landers)
  (6) Pluto-Charon Orbiter/Lander

- CONCEPTUAL DESIGN TRADES

  — Moderate and Major Levels of Exploration Capability (i.e. payloads)
  — Flight Time vs Power Level and Specific Impulse of NEP Operation
  — Launch Vehicle Capability (injection to Earth escape - no spiral escape) in Mass Performance and Packaging: Titan IV/Centaur vs HLV/Centaur
  — NEP Flight System Configuration (e.g. subsystem functions and location)

STUDY ORGANIZATION and SCHEDULE

- **SUBTASK ACTIVITIES**
  - **(1) Mission Model Definition**
  - **(2) System Model Definition**
  - **(3) Analysis of Mission Performance and System Commonality**
  - **(4) Assessment of System Capability and Recommendations**
  - **(5) Task Reporting**

- **LEVEL-OF-EFFORT**
  - **— 632 Direct Labor Hours**

- **SCHEDULE**
  - **— 4 Calendar Months (October 1992 - January 1993)**
  - **— Subtask 1 Completed on October 16**
  - **— Subtask 2 in Progress, Subtask 3 Start on October 26)**
  - **— Final Report Briefing end of January (annotated vu-graphs)**

SAIC

NEP MISSION MODEL - SCIENCE PAYLOAD DEFINITION

MISSION: PLUTO-CHARON ORBITER/LANDER

SCIENCE INSTRUMENTS	MASS (kg)	
EXPLORATION CLASS:	MODERATE	MAJOR
• Attached Mission Module		
Imaging Subsystem	57	57
UV Imaging Spectrometer	13	13
Visual-IR Mapping Spectrometer	33	33
Composite IR Spectrometer	30	30
Cosmic Dust Analyzer	- -	8
Magnetometer	7	7
Radio Science Subsystems	11	11
Cassini Plasma Spectrometer	14	14
Radio Plasma Wave Spectrometer	- -	21
Ion & Neutral Mass Spectrometer	- -	9
Microwave/Thermal IR Radiometer	- -	15
Total	165	218
• Pluto and Charon Landers	Pluto Only	Pluto and Charon
Tenuous Atmosphere Science (Separated)		
Neutral Mass Spectrometer	4.0	4.0
Ion Mass Spectrometer	3.0	3.0
Retarding Potential Analyzer	3.0	3.0
Electron Temperature Probe	2.0	2.0
Surface Sampler	- -	13.0
Multi-Spectral Imager	[illegible]	[illegible]
Magnetometer	0.4	0.4
Alpha-Proton/X-Ray Spectrometer	2.0	2.0
Scanning Electron Microscope	- -	12.0
X-Ray Diffractometer	- -	6.0
Petrographic Microscope	- -	6.0
Seismometer	2.2	2.2
Temperature Sensors	0.1	0.1
Total	21.7	56.7

Table 7. Pluto Orbiter/P(optional lander) Performance Summary
Requirements: $M_{PL} \geq 1410$ kg

PLO/P with (Titan IV/Centaur +NEP) (From Yen and Sauer, 1991)

FT (yr)	FTI (yr)	VHL (km/s)	P0 (kw)	ISP (sec)	P_R (kw)	P_e (kw)	T_L (yr)	T_P (yr)	N_e	N_t	N_{PPU}	M_0 (kg)	M_P (kg)	M_{PP} (kg)	M_{TH} (kg)	M_{NEP} (kg)	M_{PL} (kg)	V_{AC} (km/s)
13.5	13.5	2.4	58	8095	13	12	1.32	7.8	5	40	2	8315	3134	2844	1162	4006	1175	37.6
14.0	14.0	2.4	57	8238	14	11	1.37	7.9	5	40	2	8303	3009	2829	1143	3972	1322	36.4
14.5	14.5	2.4	56	8358	14	11	1.41	8.0	5	40	2	8301	2905	2815	1127	3942	1454	35.3
15.0	15.0	2.4	56	8461	14	14	1.15	8.0	4	36	2	8314	2822	2804	1079	3883	1609	34.4
15.5	15.5	2.3	55	8556	14	14	1.18	8.1	4	36	2	8351	2763	2800	1070	3870	1718	33.7
16.0	16.0	1.0	58	9390	16	15	1.22	10.3	4	44	2	8967	3075	2989	1192	4181	1711	38.7
16.5	16.5	1.0	57	9617	16	14	1.28	10.6	4	44	2	8952	2964	2980	1172	4152	1836	37.9
17.0	17.0	1.1	56	9812	16	14	1.33	10.9	4	44	2	8931	2856	2968	1152	4120	1955	37.1
17.5	17.5	1.2	55	9970	17	14	1.38	11.1	4	44	2	8909	2755	2953	1134	4087	2067	36.2
18.0	18.0	1.2	54	10121	17	13	1.43	11.2	4	40	2	8887	2662	2937	1083	4020	2205	35.3

- Orbiter is a NEP enabled mission mode.
- Minimum flight time=14.5 years, total mission time ~16.5 years.
- Feasibility indicated but margin may not be sufficient.
- Nominal P0 ~ 55 kW, ISP~ 8400 sec.
- May be a viable and attractive option if mass growth in all components can be controlled.

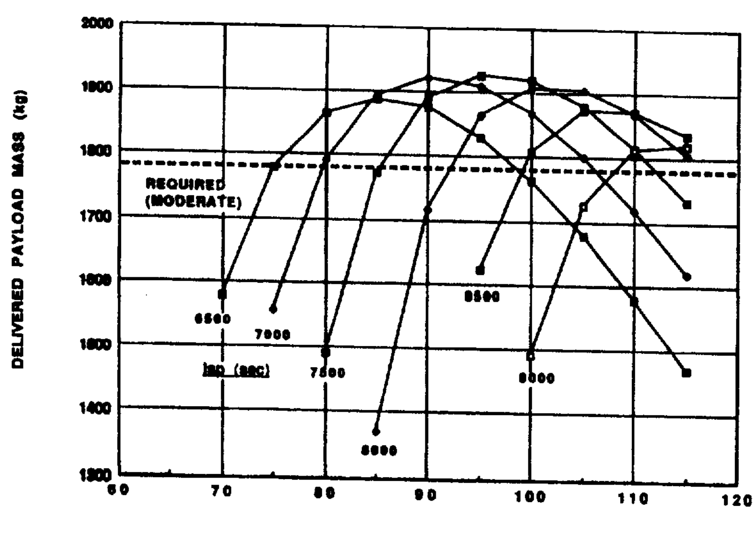
NEP-TRANSPORTED MISSION ELEMENT MASSES (kg)

MISSION EXPLORATION CLASS	CNSR		MMBAR		JGT		UO/P		NO/P		PCO/L	
	MOD.	MAJ.	MOD.	MAJ.	MOD.	MAJ.	MOD.	MAJ.	MOD.	MAJ.	MOD.	MAJ.
• Attached Mission Module Subsystems												
Telecommunications	52	52										
Antennas	86	86										
Command & Data	53	53										
Attitude Control	92	92	SAME		SAME		SAME		SAME		SAME	
Power Cabling & Control	160	160										
Thermal Control	50	50										
Mechanical Devices	58	58										
Structure	275	275										
Science Payload	121	180	116	138	160	200	174	238	174	238	165	218
Contingency (20%)	189	201	188	193	197	205	200	213	200	213	198	209
Subtotal	1136	1207	1130	1157	1183	1231	1200	1277	1200	1277	1189	1253
• Deployed Elements (Propulsion and Contingency Incl'd)												
Separated Orbiter	--	--	--	--	--	979	--	--	--	--	--	--
Atmospheric Entry Probe	--	--	--	--	--	--	234	337	234	337	--	--
Tenuous Atmosphere Probe	--	--	--	--	--	--	--	--	62	--	--	--
Landers	233	466	--	454	--	917	--	--	--	656	564	1114
Penetrators	--	--	272	272	304	--	--	308	--	--	--	--
Sample Return Capsule	120	120	--	--	--	--	--	--	--	--	--	--
Support Structure (5%)	18	29	14	36	15	95	12	32	15	50	28	56
Subtotal	371	615	286	762	319	1991	246	677	311	1043	592	1170
Total Element Mass	1507	1822	1416	1919	1502	3222	1446	1954	1511	2320	1781	2423

Table 11. Summary of NEP System Design Parameters
(From Yen and Sauer, 1991)

Mission	UO/P	NEO/P	PLO/P	PLO/P	JGT
LV	HLV	HLV	Titan IV	HLV	Titan IV
FT (yr)	10.5 - 14.	12 -15	14.5	11.5 -14	5 - 7
P0 (kW)	98 - 92	101 - 100	56	103 - 99	58 - 48
ISP (sec.)	8400 - 1 0000	7800 - 9500	8400	7200 - 8100	8700 - 10000
N_t	70 - 78	72 - 7 7	40	72 - 64	40 - 36
Tp (yr)	8.3 - 12.3	7.9 - 1 0.7	8.0	7.0 - 7.7	8.2 - 11.5
Mission Time (yr)	14 - 19	14.5- 18	16.5	13 - 16	12 - 15

Mission	JGT	MMBAR	MMBAR	CNSR
LV	HLV	Titan IV	HLV	HLV
FT (yr)	5 - 6.5	13.5	11	6.7-7.6
P0 (kW)	97 - 97	40	93	92-96
ISP sec.)	8500 - 9800	5300	6000	~ 5000
N_t	63 - 60	25	70	50-60
Tp (yr)	7.9 - 10.	5	6.3	4.0
Mission Time (yr)	11 - 14	13.5	11	8



Pluto Orbiter/Lander Mission, Mpl - Po - Isp Trades
TF = 12 years, C3 = 3.2, HLV/Centaur (Mo = 13,700 kg)

N93-26973

Scoping Calculations of Power Sources for NEP*

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Oak Ridge, Tennessee 37831-6363 USA

Viewgraphs to be presented at the *Nuclear Propulsion Technical Interchange Meeting*, October 20-23, 1992, NASA Lewis Research Center.

* Managed by Martin Marietta Energy Systems, Inc., under contract DE-AC05-84OR21400, U.S. Department of Energy.

Definition of the Problem (From NASA-LERC)

Power Levels (P): 10-50 Mw
Core Life (D): 2-10y

Which Implies:

Energy Released: 7305-182,625 Mwd; or
the burnup of ~: 9.1-228 Kg of ^{235}U

Types of Reactors to be Analyzed:

1. **High Temperature Gas-Cooled Reactors of the NERVA derivative type.**
2. **Lithium-Cooled Advanced Fuel Pin. One-phase flow.**
3. **Lithium-Cooled Cermet. One-phase flow.**

For an input *P* and *D*, it is required to calculate:

(a) Composition and Masses of the core.

(b) Mass of the Reflector.

(c) Mass of the Shielding.

(d) Temperature and Pressure Distributions.

Elements to Build the Reactors

1. Gas Cooled, NERVA Type

Core

(a) Fuel Element, hexagonal 1.913 cm flat to flat, dispersion of UC-ZrC in a graphite matrix, 19 coolant holes (d = 2.8mm), ZrC clad.

(b) Support Element: ZrH_2 on inconel tube, central and lateral coolant around the ZrH_2, pyrolitic graphite and graphite as thermal shield.

Coolant: He (for direct Brayton cycle)

Reflector: Be, radial

Control: B_4C sheet on drums that rotate in reflector

Safety Rods in Core

Pressure Vessel: Outside the reflector

Elements to Build the Reactors (continued)

2. Advanced Fuel Pin

Core: Rods, 6.35mm diameter (may vary); UN pellets; clad, tantalum alloy (Astar-811C or T-111) 0.635mm thick; tungsten liner 0.122mm thick; He gas gap 0.025mm thick.

Coolant: Liquid Lithium

Reflector: OBe

Control: B_4C sheets on drums in reflector.

Pressure Vessel: Between Core and Reflector

Elements to Build the Reactors (continued)

3. Cermet (ceramic-metal)

Core: Hexagonal Fuel Element; UO_2 (or UN) in a matrix of W (with some Re); clad, W-Re-Mo alloy.

Coolant: Liquid Lithium

Reflector: Be

Control: B_4C sheets on drums in Reflector.

Pressure Vessel: Between Core and Reflector.

Shielding

(Common to the three designs)

LiH or B_4C for neutrons, W-Mo alloy for gammas.

Geometry: shadow shield.

Estimation based on

(a) source term,

(b) first collision shielding,

(c) removal cross section, and

(d) buildup factors.

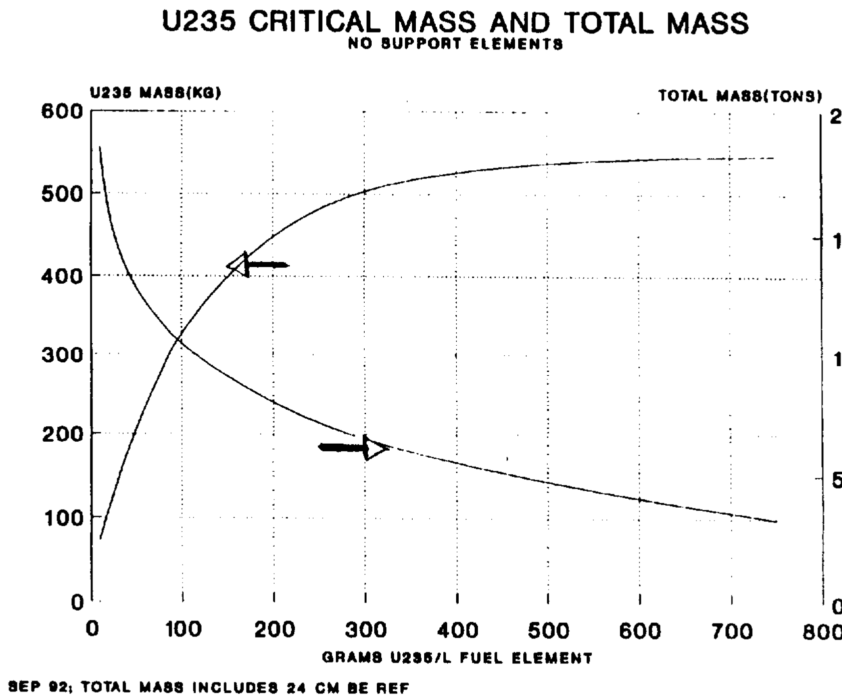
Results for the Gas-Cooled Reactor

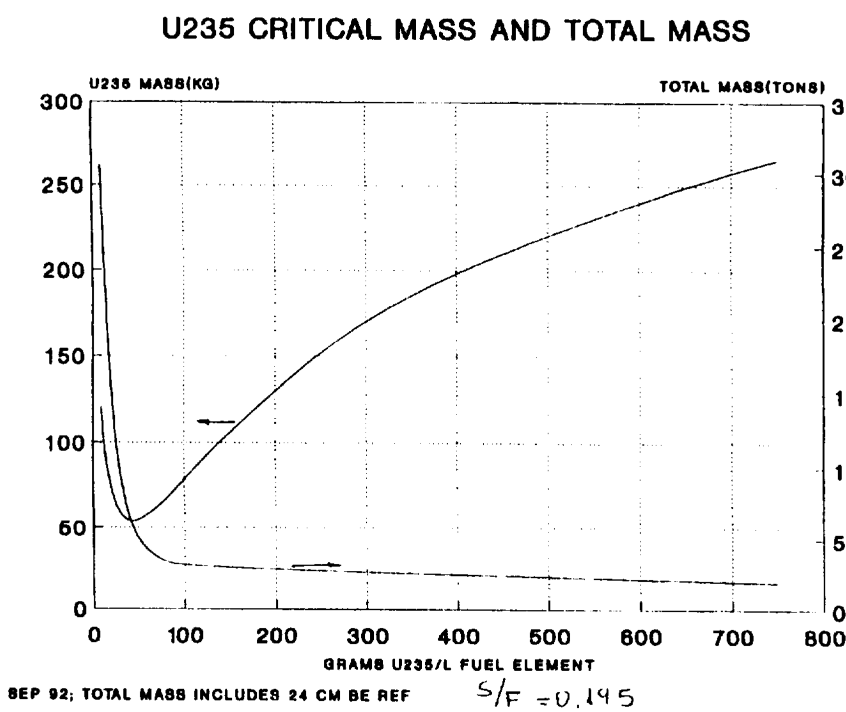
Variables to choose in order to meet demand:

(1) ^{235}U density in fuel element

(2) Ratio S/F of the number of support over fuel elements

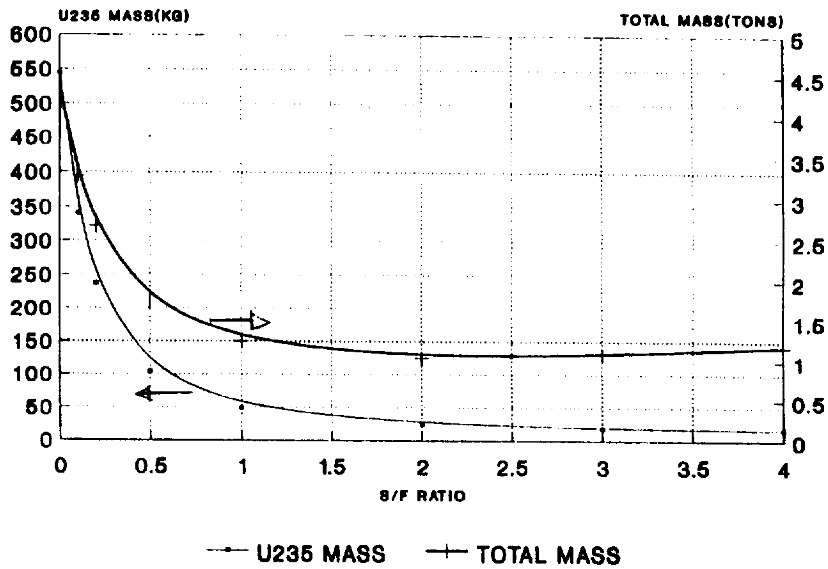
Given conditions at channel inlet (flow, p and T) compute pressure, temperatures and velocities considering single phase 1D steady flow. Use usual correlations from ANS handbook about gas-cooled reactors.


U235 CRITICAL MASS AND TOTAL MASS
NO SUPPORT ELEMENTS
U235 MASS(KG)
TOTAL MASS(TONS)
600
500
400
300
200
100
0
20
15
10
5
0
0
100
200
300
400
500
600
700
800
GRAMS U235/L FUEL ELEMENT
SEP 92; TOTAL MASS INCLUDES 24 CM BE REF


U235 CRITICAL MASS AND TOTAL MASS
U235 MASS(KG)
TOTAL MASS(TONS)
300
250
200
150
100
50
0
35
30
25
20
15
10
5
0
0
100
200
300
400
500
600
700
800
GRAMS U235/L FUEL ELEMENT
SEP 92; TOTAL MASS INCLUDES 24 CM BE REF
S/F =0.145

U235 CRITICAL MASS AND TOTAL MASS

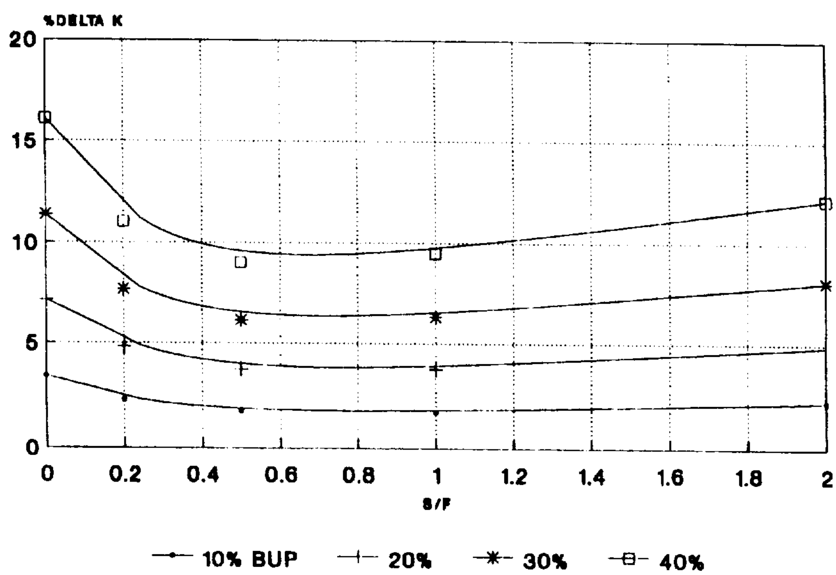
AS FUNCTION OF SUPPORT/FUEL RATIO



9-92;0.5KG U6/L INCLUDES 24 CM BE REF

REACTIVITY WORTH OF BUP AS F(S/F)

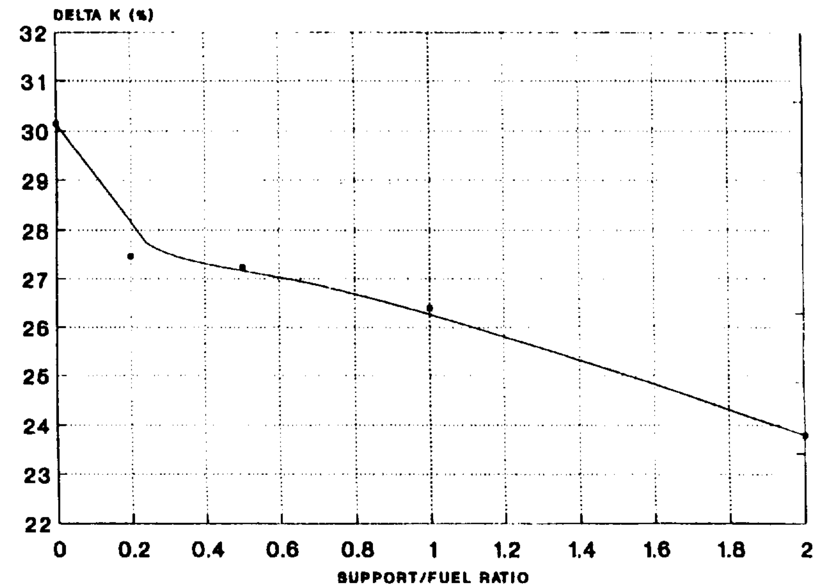
0D+DB2 CALCULATION



SEP 92; 500 G U235/L FUEL

REACTIVITY WORTH 30CM RADIAL BE

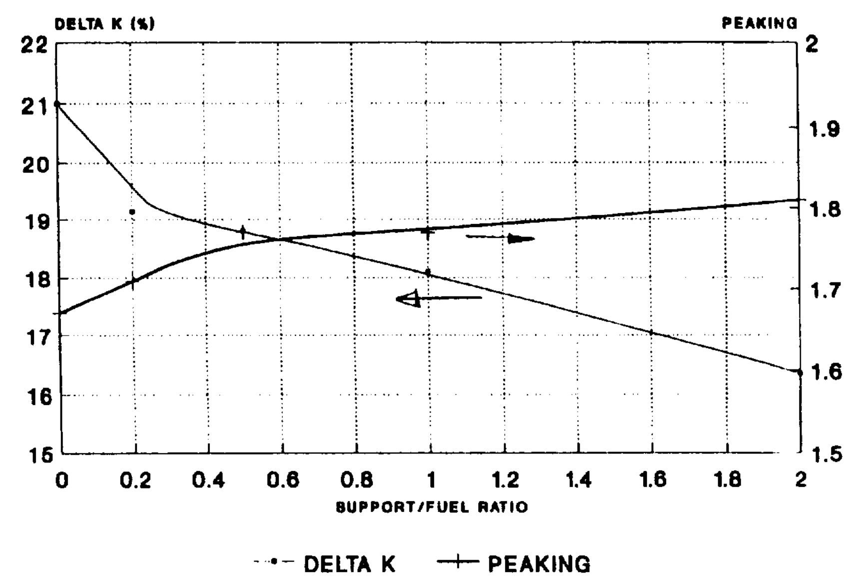
FOR CORE 500. G U5/L FUEL



9-92

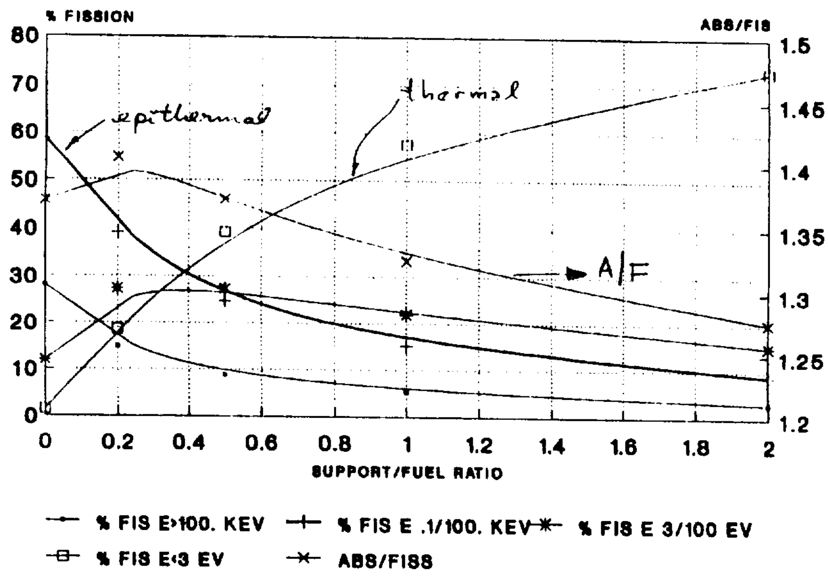
MAX DELTA K FOR B4C DRUM IN BE REF

AND PEAKING FACTOR AT BOL



600 GU5/L;30.CM BE; 2MM B4C

SPECTRAL INDICES AND ABS/FIS IN U235



500 GU5/L;30.CM BE; 2MM B4C

Initial Approach for Use of this Model

Fuel density of 500g ^{235}U/L fuel is a reasonable compromise between good heat transfer and low total mass for the reactor.

Then, the parameter S/F is chosen to meet the demand:
P (Power), D (core Life), BU (% at burnup)

(1) With P, D, and BU estimate ^{235}U mass at BOL for slightly subcritical bare reactor. This then define S/F.

(2) With S/F and BU define $\Delta k)_{BU}$ due to burnup.

(3) Add (a) estimated Δk due to steady Xe and Sm (~3% max), (b) Δk Xe for buildup after trip, (c) 2% Δk for EOL operation and (d) 2% (estimated) due to structural material.

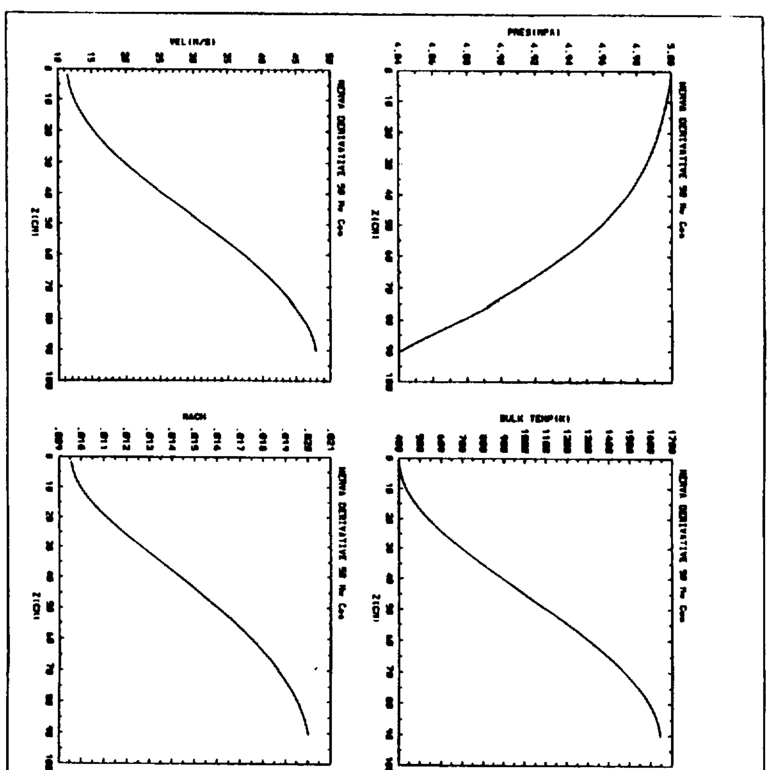
Initial Approach for Use of this Model (continued)

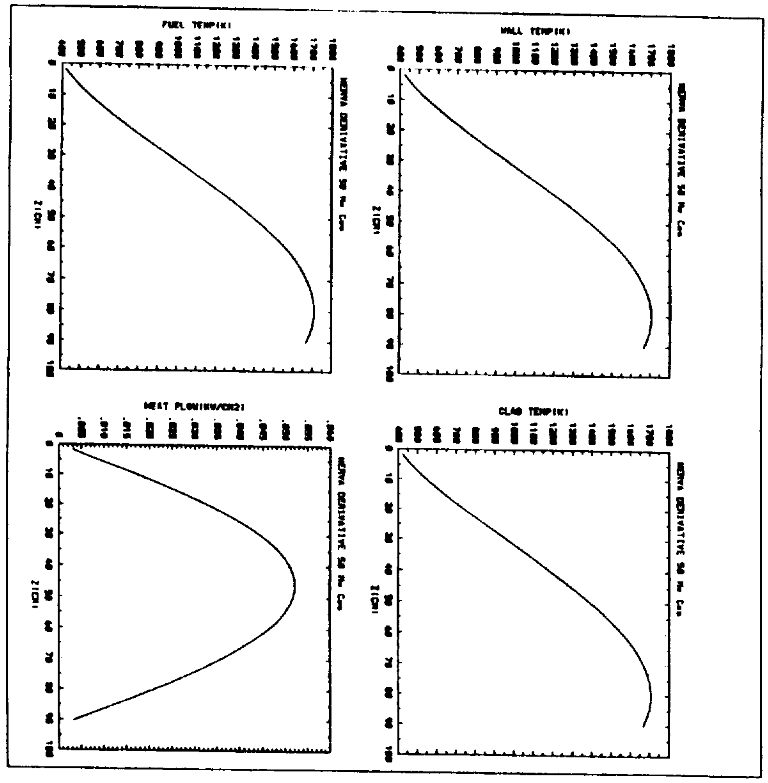
(4) With S/F find Δk of 30cm Be reflector.

(5) If 30cm of Be does not match the required Δk go to (1) change the ^{235}U mass.

(6) Check if control rods in reflector are sufficient to control the reactor.

(7) Check consistency of the A/F assumed.





Results for Initial Use of the Model

- **A model has been generated to allow initial scoping calculations of gas-cooled reactor power sources for NEP.**

- **High power, long mission would require control mechanism in the core or burnable poison.**

- **The algorithm to use the model is going to be attached to the thermalhydraulic and shielding calculations in order to have a PC program useful for mission analysis. Work in progress.**

- **The previous criteria is going to be applied to the other two designs.**

N93-26974

NEP POWER SUBSYSTEM MODELING

Nuclear Propulsion Technical Interchange Meeting

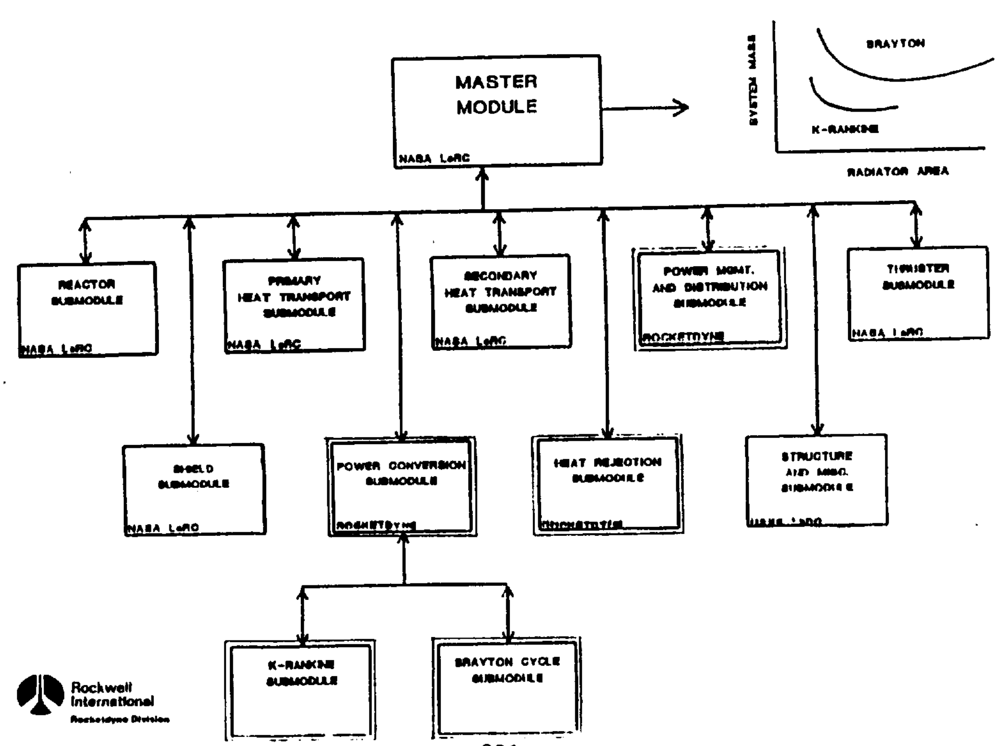
October 20-23, 1992

NASA-Lewis Research Center
Plum Brook Station

Rockwell International
Rocketdyne Division

The Nuclear Electric Propulsion (NEP) system optimization code consists of a master module and various submodules. Each of the submodules represents a subsystem within the total NEP power system. The master module sends commands and input data to each of the submodules and receives output data back. Rocketdyne was responsible for preparing submodules for the power conversion (both K-Rankine and Brayton), heat rejection, and power management and distribution.

NEP SYSTEM OPTIMIZATION CODE



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The basic objective of each task was to perform detail performance modeling for selected subsystems of an NEP system. The output of each task is software (computer disk) and a users manual providing a detailed model description, limitations, assumptions, and inputs and outputs.

TASK ORDER OBJECTIVES AND OUTPUT

TASK OBJECTIVES

- CHARACTERIZE AND PERFORM DETAILED MODELING OF SELECT SUBSYSTEMS FOR A NUCLEAR ELECTRIC PROPULSION SYSTEM
  - POWER CONVERSION
    - LIQUID METAL RANKINE
    - GAS COOLED BRAYTON
  - HEAT REJECTION
  - POWER PROCESSING AND DISTRIBUTION

TASK OUTPUT

- SOFTWARE AND USERS MANUAL DESCRIBING DETAILED MODELS USED
- SUFFICIENT DETAIL TO PROVIDE THE FOLLOWING ON THE COMPONENT AND SUBSYSTEM LEVEL
  - MASS
  - PERFORMANCE
  - DIMENSIONS
  - PHYSICAL OPERATING CONDITIONS
  - RELIABILITY

C-4

GROUND RULES AND REQUIREMENTS

GENERAL
- POWER LEVEL RANGE - 100 kWe TO 10 MWe
- OPERATING LIFETIME - 2 TO 10 YEARS
- OPERATING ENVIRONMENT - LOW EARTH ORBIT TO INTERPLANETARY SPACE
- TECHNOLOGY TIME FRAME - 2005 TO 2020

K-RANKINE
- TURBINE INLET TEMPERATURE - 800 TO 1500 K
- TEMPERATURE RATIO - 1.25 TO 1.6
- TURBINE TYPE - AXIAL FLOW
- WORKING FLUID - POTASSIUM

BRAYTON
- TURBINE INLET TEMPERATURE - 1200 TO 1500 K
- TEMPERATURE - 2.5 TO 4.0
- TURBINE TYPE - AXIAL AND RADIAL FLOW
- WORKING FLUID - He AND HeXe

HEAT REJECTION
- TEMPERATURE RANGE - 750 TO 1250 K (K-RANKINE), 300 TO 1000 K (BRAYTON)
- RADIATOR TYPE - HEAT PIPE
- HEAT PIPE WORKING FLUIDS - NH_3, H_2O, Hg, K, Na, Li
- GEOMETRY - FLAT, CYLINDRICAL, CONICAL

POWER PROCESSING AND TRANSMISSION
- TRANSMISSION LENGTHS - 25 TO 300M
- VOLTAGE LEVEL - 200 TO 10,000 VOLTS
- AC FREQUENCY RANGE - 100 Hz TO 20 kHz
- COLD PLATE TEMPERATURE - 60 TO 200°C

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The facing page lists the key ground rules and requirements for each task. The values were agreed to with NASA. The values represent the applicable range of interest and range of the current data base.

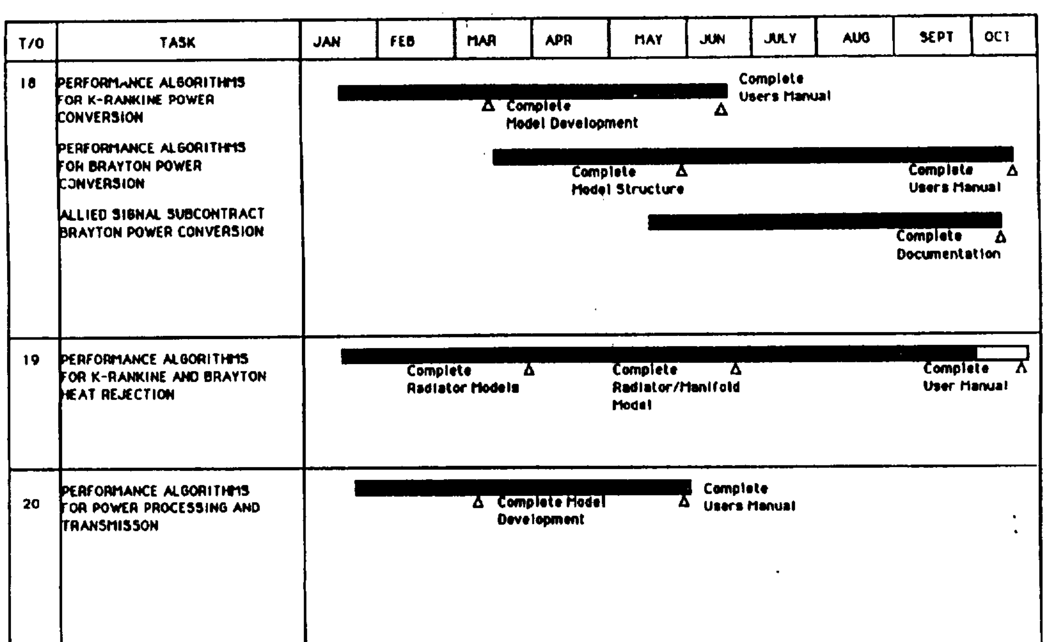
The models being developed are based on first principles. Where this is not possible such as heat transfer coefficients and aerodynamic efficiencies, algorithms are used to describe these parameters. Using first principals provides a great deal of flexibility for the user. The user, however, must be knowledgeable in the particular component being modeled. Default values are provided to aid the user in establishing realistic initial values.

MODULE ARCHITECTURE CHARACTERISTICS

- BASED ON FIRST PRINCIPLES WITH SOME EMPERICAL CORRELATIONS
- STEADY-STATE DESIGN CODE
- DEFAULT VALUES USED AS A STARTING POINT TO AID USER
- USER MUST HAVE SOME KNOWLEDGE OF BASIC PRINCIPLES

The schedule for developing the models is presented on the facing page. All activities have been completed with the exception of the Heat Rejection Task Order. The software for this Task Order has been completed and the users manual is in preparation. The task orders also includes user support to aid NASA in integration with the master module.

SCHEDULE AND MILESTONES FOR NEP SUBSYSTEM MODEL DEVELOPMENT TASK ORDERS 18, 19, 20



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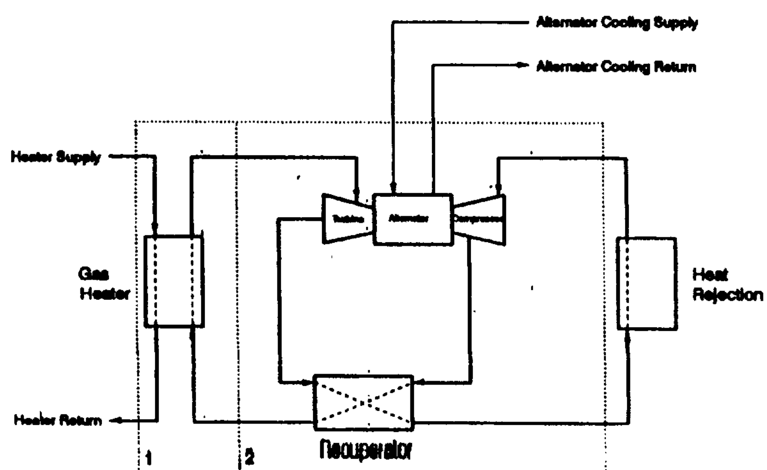
Brayton Power Conversion Module Flow Diagram

The facing viewgraph shows a typical flow diagram for a closed Brayton cycle (CBC) system. The Power Conversion Module computer code provides for two heat source configurations; (1) liquid metal-to-gas primary heat exchanger, or (2) a gas cooled reactor configured into the CBC loop. The scope of the power conversion module for those two cases is indicated on the facing page.

The Brayton power conversion module provides for the cycle state point calculations, component performance projections, and component sizing. The components include the turbine, compressor, alternator, recuperator, and ducting. A primary heat exchanger performance and sizing routine is provided for the gas heater option.

Power Conversion Module Flow Diagram



1. Full system module boundary
2. Gas reactor system option module boundary

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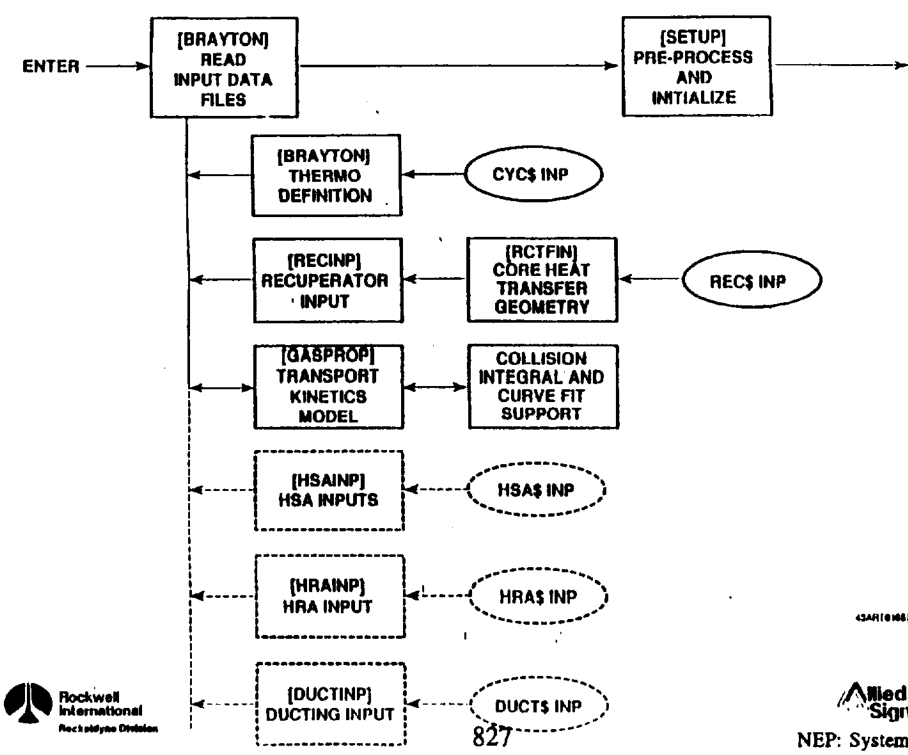
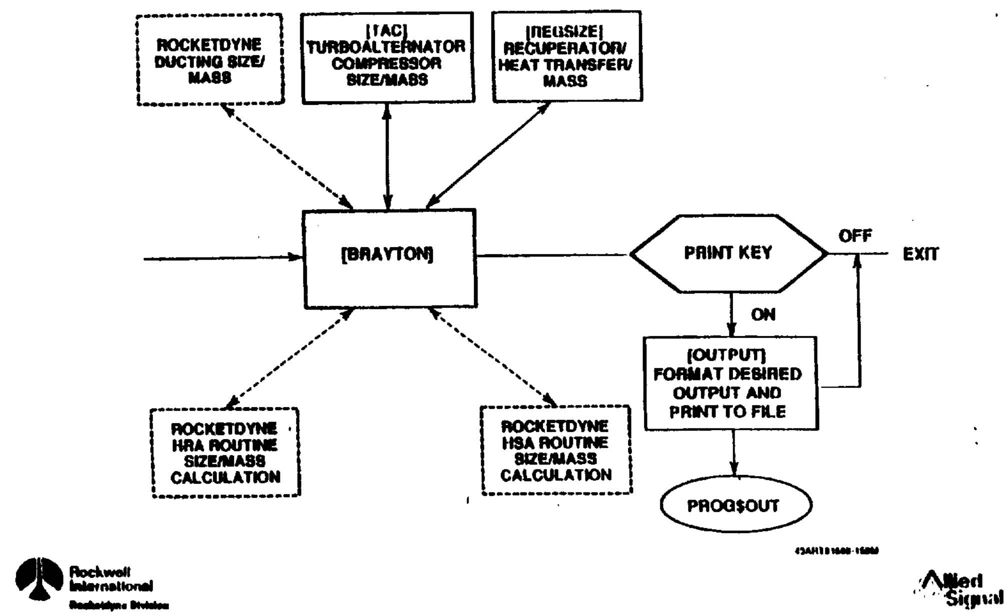
Power Conversion Module Computer Program Block Diagram

The next three viewgraphs give the computer program structure for the Brayton power conversion module. The first chart shows the input file structure for the program. Once the data files have been read and the appropriate preprocessing completed, the code moves on to the cycle state point definition routines including component performance computations. The second chart gives the layout of the subroutines used in the cycle statepoint definition portion of the code. Following the statepoint definition, the code moves into the detailed component sizing. The third chart gives the layout of the subroutines used in the component sizing portion of the code. Output options for the code are also provided.

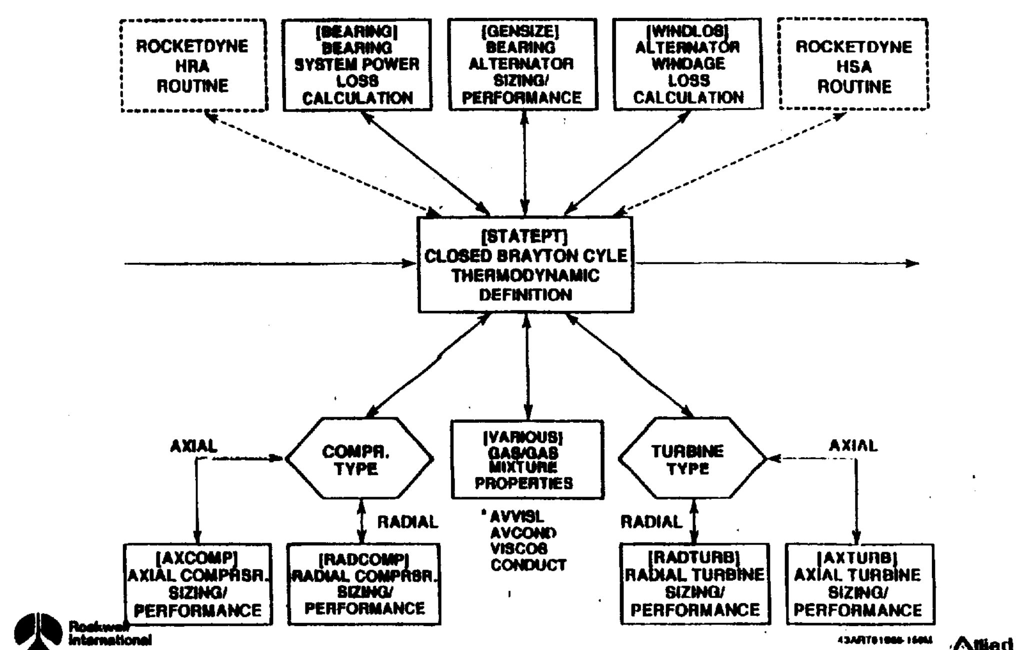
DATA INPUT/SETUP MODULE



POST PROCESSOR/OUTPUT MODULE



THERMODYNAMIC MODEL



The facing page is a table illustrating the input variables the heat rejection submodule receives and directs to the various routines, and the output variables generated by the routines that the heat rejection submodule directs to the master module. Since there are numerous variables, only a partial listing of some of the key variables where included in the table.

Brayton Power Conversion Module

Key Inputs

- Axial or radial
- Gross electrical power
- Turbine inlet temperature
- Pressure ratio
- Cycle beta
- Specify 2 of 3
  - RPM
  - Specific Speed
  - Compressor inlet temperature
- Recuperator effectiveness
- Pressure drop allocations
- Molecular weight options

- plus more than 30 others

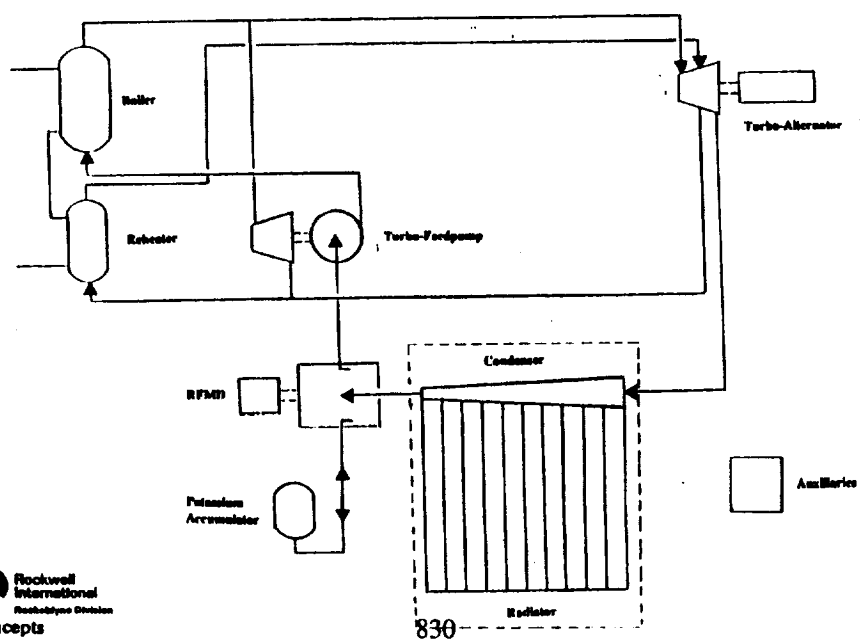
Key Outputs

- TAC mass
- Recuperator mass
- Turbine efficiency
- Compressor efficiency
- Alternator mass
- Cycle statepoints
  - Temperatures
  - Pressures
  - Flows
- 1 of 3
  - RPM
  - Specific speed
  - Compressor inlet pressure

- dozens of performance and geometry related parameters are available

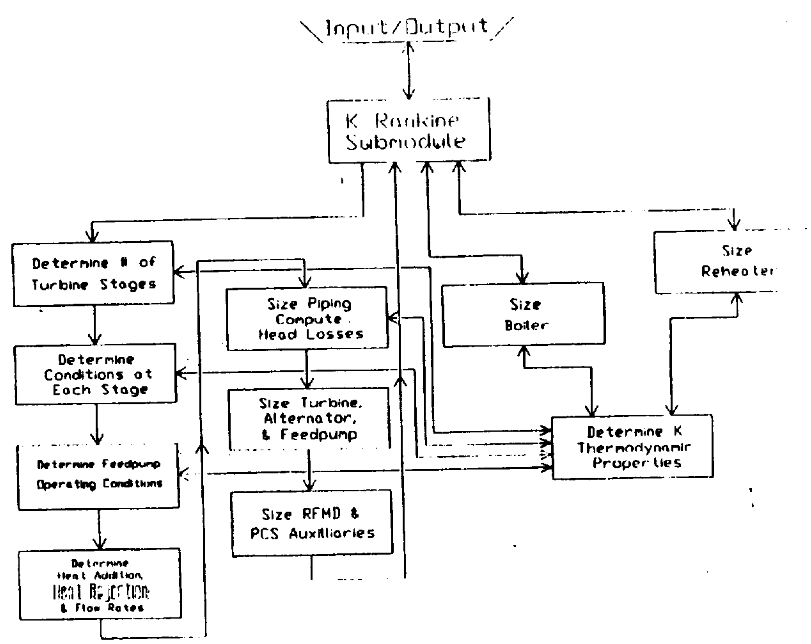
In the potassium-Rankine power conversion subsystem, shown on facing page, the principal flow of potassium vapor leaving the boiler is to the main turbine. A relatively small stream is diverted to the turbine of the turbo feed pump. The main turbine is divided into high-pressure stages and low-pressure stages. Upon exhausting the high-pressure stages, the wet potassium vapor is routed through a reheater to revaporize entrained moisture and re-superheat the vapor stream, upon which the vapor stream leaving the reheater is routed to the low-pressure turbine. Upon exhausting from the low-pressure turbine stages, the vapor is condensed in a shear flow controlled condenser. Latent heat of vaporization is rejected by the condenser to the heat rejection subsystem. Condensate leaving the condenser is directed to a Rotary Fluid Management Device (RFMD). The RFMD provides two phase fluid management and pressurizes the condensate to ensure that sufficient net positive suction head (NPSH) is provided to the main turbo-feedpump. The turbo-feedpump repressurizes the liquid potassium received from the RFMD and directs it to the boiler.

POTASSIUM-RANKINE POWER CONVERSION SYSTEM SCHEMATIC



The potassium-Rankine program structure and interfaces are illustrated on the facing page. The K-Rankine submodule is designed to interface with the master module by receiving input and directing output generated form the K-Rankine routines to the master module. Additionally, the K-Rankine submodule directs the flow of computations and data through the various K-Rankine routines.

NEP K-RANKINE TOP LEVEL FLOW DIAGRAM



The facing page is a table illustrating the input variables the K-Rankine submodule receives and directs to the various routines, and the output variables generated by the routines that the K-Rankine submodule directs to the master module. Since there are numerous variables, only a partial listing of some of the key variables where included in the table. The K-Rankine code requires in the neighborhood of 60 input variables and generates over 500 output variables.

K-RANKINE INPUT/OUTPUT VARIABLE

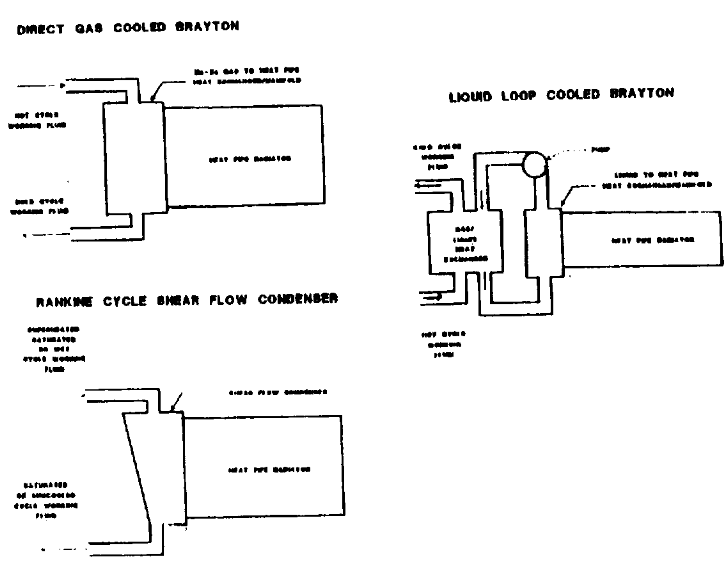
MAJOR INPUT VARIABLES	MAJOR OUTPUT VARIABLES
- Electric Power Out	- System Mass
- Turbine Inlet Temperature	- Heat Input Requirements
- System Life	- Heat Rejection Requirements
- Condenser Temperature	- Electrical Frequency
- Voltage	- + Over 500 Other Output Variables
- + 50 Other Input Variables	

HEAT REJECTION

The heat rejection subsystem design code provides the capability of analyzing three distinct configuration options; namely, direct gas cooled Braytons, liquid loop cooled Braytons and Rankine cycle shear flow condenser units. Algorithms to calculate the mass and performance expected for each component in each of the three subsystems are included. Normally, a relatively complete description of the dimensions and flows involved with the particular component is required to be supplied to the code. An option is offered that permits the code to run with relatively little information (namely; inlet and outlet conditions and system type). The output from this option can then be used as a baseline for other optimization studies.

Note: Flow input to the Rankine condenser manifold must be either saturated or wet. The code cannot accommodate superheated vapor.

RADIATOR FLOW SCHEMATIC OPTIONS



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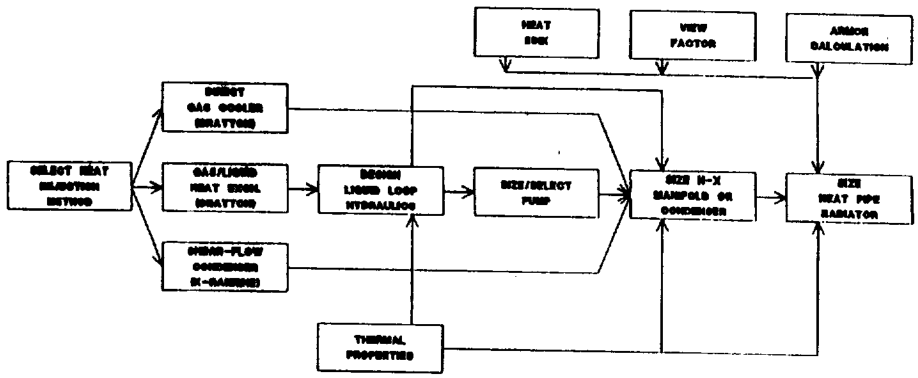
NEP HEAT REJECTION TOP LEVEL FLOW DIAGRAM



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The top level flow diagram for the heat rejection subsystem is shown. The driver code must, as a minimum, supply the subroutine with thermodynamic inlet and outlet conditions and with a heat rejection method selection. The code will then proceed to perform a detailed computation of the performance and mass of the system specified. The computation sequence for these estimates proceeds from first principles and follows the blocks as shown. The code contains all properties and orbit environmental information needed to analyze most operational situations.

HEAT REJECTION INPUT/OUTPUT DESCRIPTION

KEY INPUTS	KEY OUTPUTS
- Inlet Flowrate	- Radiator Area
- Inlet Temperature	- Heat Rejection Subsystem Mass
- Inlet Pressure	- Component Masses
- Amount of Heat to be Rejected (Duty)	- Component Pressure Drops
	- Component Temperature Drops
- Detail Component Dimensions (Optional)	- Detail Component Dimensions, If Not Given

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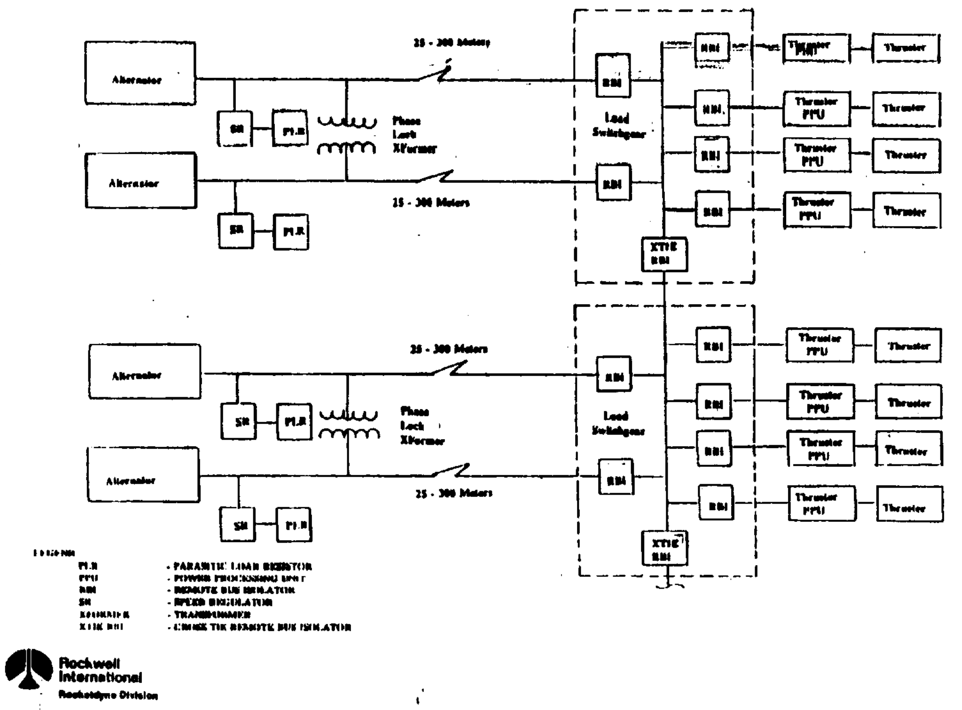
The facing page is a table illustrating the input variables the heat rejection submodule receives and directs to the various routines, and the output variables generated by the routines that the heat rejection submodule directs to the master module. Since there are numerous variables, only a partial listing of some of the key variables where included in the table.

LOW FREQUENCY PMAD ARCHITECTURE



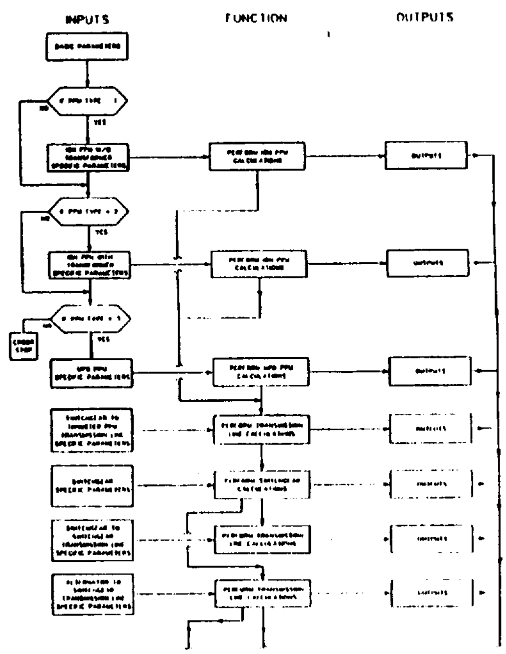
Low Frequency PMAD Architecture

The PMAD model is based on a low frequency PMAD architecture that transmits power to either ion or magnetoplasmadynamic (MPD) thrusters at the alternator voltage and frequency. It does not utilize a rectifier or inverter to change the alternator output power characteristics. This low frequency transmission approach was compared with dc and high frequency ac designs, and determined to have the lowest mass, highest efficiency, and on the basis of complexity judged to have the highest reliability and lowest development costs. Although its power quality is not as good as that provided by a high frequency system, it is adequate for both ion and MPD thruster applications.

This architecture has six main elements: thruster power processing units (PPUs), switchgear units, phase lock transformers, shunt regulators, parasitic load radiators, and transmission lines. The thruster PPUs convert the high voltage ac employed for power transmission into lower voltage dc feeds for the respective thruster elements. The switchgear units perform power switching operations and provide fault protection for the thruster PPUs. The phase lock transformer is only included if counter rotating alternators are employed. It synchronizes the alternator outputs and prevents a torque moment from being applied to the NEP vehicle due to unequal or unbalanced changes in alternator speed. The speed regulator controls the alternator and turbine speed by adjusting the connected load. The objective is to maintain the total connected load, thrusters and parasitic load, at a fairly constant level and prevent the reactor from experiencing power fluctuations. Finally, the transmission lines carry power from the alternators to the switchgear units and distribute it to thrusters.

TASK20VG OCT

NEP PMAD TOP LEVEL FLOW DIAGRAM



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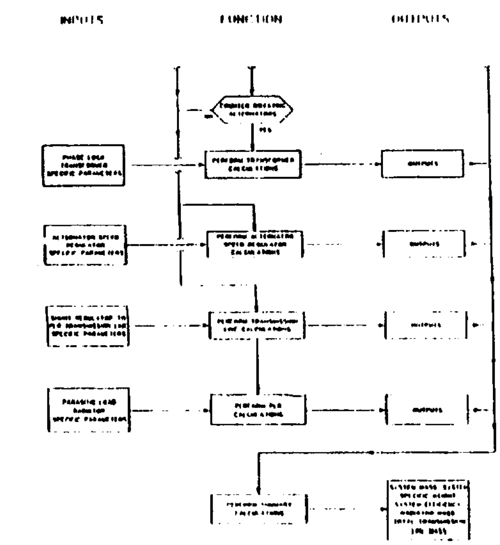

NEP PMAD Top Level Flow Diagram

The model operator largely defines the PMAD architecture by selecting the number of operating and standby PMAD channels, and the number of alternators and thrusters per channel. Then, depending on whether ion or MPD thrusters are being studied, the user selects the appropriate PPU type. The frequency used for power transmission is established by the alternator, and the thruster PPU input voltage selected by the user determines the transmission voltage. The final system level parameter selected ~~by the model operator is the power conditioning component coldplate temperature.~~ Many other component specific parameters can also be changed; however, the default values that are provided are appropriate for most applications. Based on the operator selected inputs, the PMAD model outputs such figures of merit as total PMAD system mass and specific weight, and the end-to-end PMAD system efficiency.

PMAD Model Input and Output Parameters

Key Inputs

Total Output Power Level

Alternator Frequency

Number of PMAD Channels

Number of Alternators per Channel

Number of Thrusters per Channel

Power Processing Unit Type

Component Coldplate Temperature

Numerous Other Inputs such as Transmission Voltage; Transmission Line Lengths; and Power Conditioning Component Configurations, Voltages, Filtering Levels, and Power Processing Element Efficiencies

Key Outputs

Total PMAD System Mass

PMAD System Specific Weight

PMAD System End-to-End Efficiency

Total PMAD Component Mass

Total Transmission Line Mass

Total Electronics Radiator Mass

Numerous Other Outputs such as Transmission Line Temperatures and Efficiencies; and Individual Power Conditioning Component Masses, Efficiencies, and Volumes

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The facing page is a table illustrating the input variables the PMAD submodule receives and directs to the various routines, and the output variables generated by the routines that the PMAD submodule directs to the master module. Since there are numerous variables, only a partial listing of some of the key variables where included in the table.

N93-26975

NEP PROCESSING, OPERATIONS, AND DISPOSAL

FINAL REPORT AND PRESENTATION

Task Order 20

Contract NAS3-25809

by

Science Applications International Corporation

and

Martin Marietta Astronautics Group

for

NASA Lewis Research Center

Nuclear Propulsion Office

October 20, 1992

MARTIN MARIETTA

SAIC
Science Applications International Corporation

Study Purpose

Several recent studies by ASAO/NPO staff members at LeRC and by other organizations have highlighted the potential benefits of using Nuclear Electric Propulsion (NEP) as the primary transportation means for some of the proposed missions of the Space Exploration Initiative. These include potential to reduce initial mass in orbit and Mars transit time. Modular NEP configurations also introduce fully redundant main propulsion to Mars flight systems, adding several abort or fall-back options not otherwise available. Recent studies have also identified mission operations, such as on-orbit assembly, refurbishment, and reactor disposal, as important discriminators for propulsion system evaluation. This study is intended to identify and assess "end-to-end" operational issues associate with using NEP for transporting crews and cargo between Earth and Mars. We also include some consideration of lunar cargo transfer as well.

The study was performed by SAIC and Martin Marietta under direction of Michael Doherty of the NASA/LeRC Nuclear Propulsion Office. Mike Stancati (Study Leader) and Jim McAdams of SAIC performed the rendezvous and disposal modes analysis. Tal Sulmeisters and Dr. Robert Zubrin of Martin Marietta prepared the launch, assembly, and refurbishment sequences. The study team wishes to acknowledge the guidance and valuable comments by Mike Doherty, Jim Gilland of Sverdrup Technology, and Len Dudzinski and Jeff George of NASA/LeRC.

Study Purpose

Identify and assess operational issues associated with using Nuclear Electric Propulsion for SEI missions, including Mars cargo and piloted, and lunar cargo transfer:

- Launch and assembly
- Spiral operations and crew rendezvous
- On-Orbit Refurbishment and maintenance of a reusable NEP transfer vehicle
- NEP disposal

Ground Rules

This study concentrates on operational issues, rather than performance assessment of alternative technologies against some set of user requirements. For this reason, certain items are specified as given. The NEP system is a modular concept, which was identified and studied in several recent activities by LeRC. Changes or enhancements to this basic system are proposed only for operational reasons; beyond very basic calculations, we have not optimized specifications or sizing. Payloads are consistent with many earlier studies to support a crew of four round-trip to Mars.

Commonality of design and operations is preferred throughout. This means, for example, that a single Earth orbit will be selected for both initial assembly and refurbishment between missions. Similarly, common procedures will be used for operation of both piloted and cargo transfer vehicles.

Simplicity of in-space operation is also a ground rule. The processing sequences proposed and evaluated are selected to minimize the complexity of on-orbit operations. Infrastructure and resources are minimized, consistent with safe, effective operation.

Finally, we address reactor disposal using conservative approaches in all cases.

Ground Rules

- Specified NEP reference systems for cargo and piloted transfer vehicles, based upon propulsion module concept studied previously at LeRC
- Payload sizing generally consistent with earlier studies for a crew of 6
  - Mars transit habitat = 40 t
  - Earth Crew Capture Vehicle = 7 t, for Apollo-type reentry with $V_{\infty} \leq 9.4$ km/s
- Prefer common NEP vehicle configurations and processing sequences for piloted and cargo missions
- Minimize on-orbit operations and infrastructure
- Safe reactor disposal for all cases, from normal end of life to propulsion system failure
- Split mission profile
  - cargo MTV carries surface payload and MEV; crew MTV carries return propellant
  - use 2012 cargo/2014 piloted opportunity for calculations

Assumptions for NEP System Scaling

Each module includes a complete propulsion system, from energy source to thrusters, and the necessary structural support. The reactor is designed to deliver 5 MWe at full power, with an efficiency of about 20%. Design life for the reactor is two years at full power. The module mass estimate is just under 37 t, including all subsystems, so the target specific mass is 7.3 kg/kWe. Studies by LeRC and GE indicate that, while this represents an advance in state-of-the-art, it is a reasonable projection for attainable capability in the near term.

Cargo flight to the Moon or Mars would use a transfer vehicle configuration with a single propulsion module. Piloted flights to Mars would include system-level redundancy with two fully configured propulsion modules delivering a total of 10 MWe. In addition to improving nominal performance, the piloted Mars Transfer Vehicle (MTV) features several abort modes for degraded propulsion systems, including loss of an entire module. A parallel study by SAIC (Task Order 19 of this contract) reports a preliminary risk/reliability assessment of the two-module "Hydra."

Assumptions for NEP System Scaling

Each propulsion module - "relatively near-term" technology

- Complete, self-contained propulsion system with: growth SP-100 reactor, K-Rankine power conversion, PMAD, thrusters, heat rejection, and supporting truss structure
- Reactor delivers 5 MWe full power over 2 year life
- Argon ion thrusters, Isp = 5000 s, 10,000 hour life
- Module specific mass (includes all subsystems) = 7.3 kg/kWe

Transfer Vehicle Configurations

- One 5 MWe module for cargo flights
- Two 5 MWe modules for piloted flights

NEP/MCV Concept

- To fit within a [illegible] m fairing presently [illegible] on-orbit assembly, a recommended [illegible] consists of [illegible] segments. The forward [illegible] a short width of 4.5 m and a large width of [illegible] m resulting [illegible] side area [illegible] two segments [illegible] in an area [illegible] sq. m, slightly larger than the [illegible] configuration [illegible] (supplied [illegible]).

- The reactor is mounted on the short width end of the forward segment and can be packaged within the conical region of the shroud.

- The deployment sequence is automated and does NOT require on-orbit assembly. The automated extension of the boom is also possible (a design of such nature was analyzed for the Thermionic Space Nuclear Power system proposal).

- The remaining key items, i.e. two solar pannels ([illegible] kw each), CTV docking port, FTS and an engine pod are launched with each vehicle. Cargo, CTV and the propellant module are launched as lift and packaging capabilities allow. Specific subsystem design concepts would be required to specifically manifest and package a given mission.

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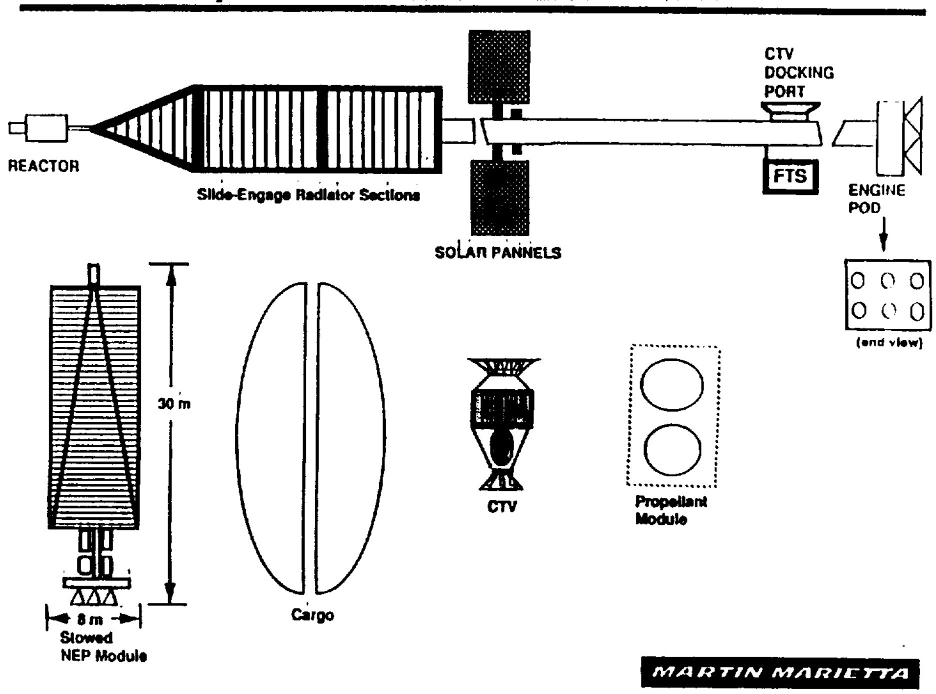
NEP Concept - MCV



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TS920812.7

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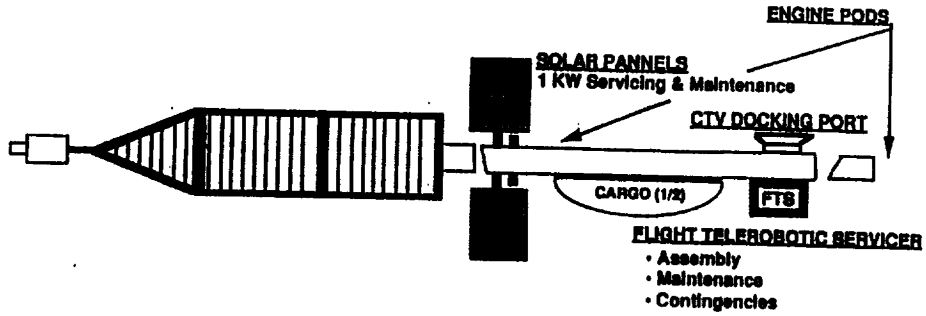
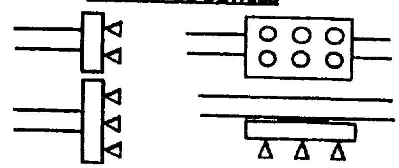
NEP Concept - Key Items



GROUND RULES
- NO Planned EVA for Assembly
- NO Planned Contingency EVA
- Docking Operations ROBOTIC/ Automated

ENGINE POD DETAILS



- Engine Sets Can be Mounted on Side or End of the Boom
- Propellant Tank Pods Can be Mounted on Side or End of the Boom

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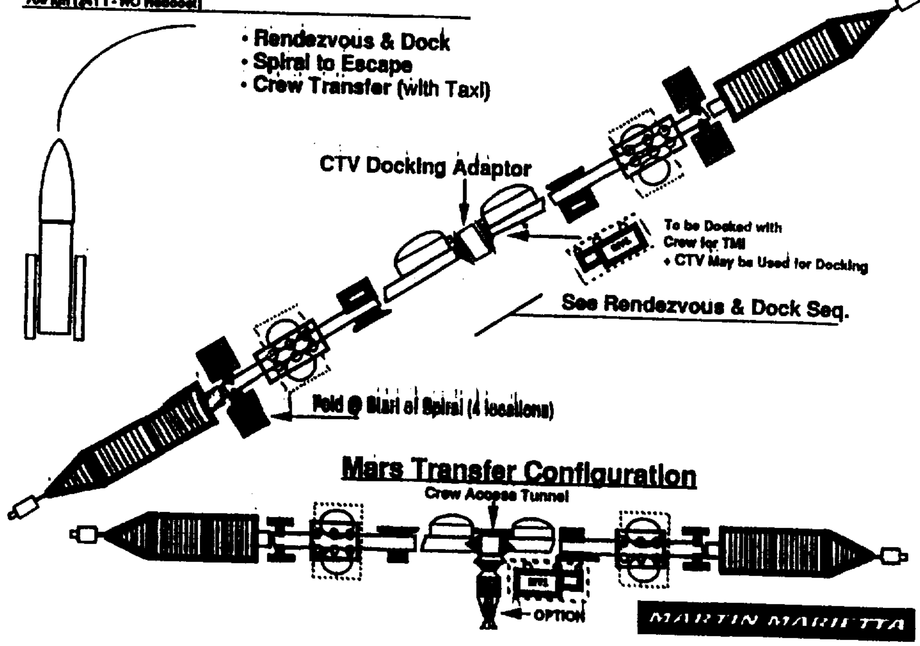
TS920811.1

MPV Orbital Ops

300 km (269 t - Reboost), or
500 km (250 t - Reboost ?), or
700 km (241 t - NO Reboost)



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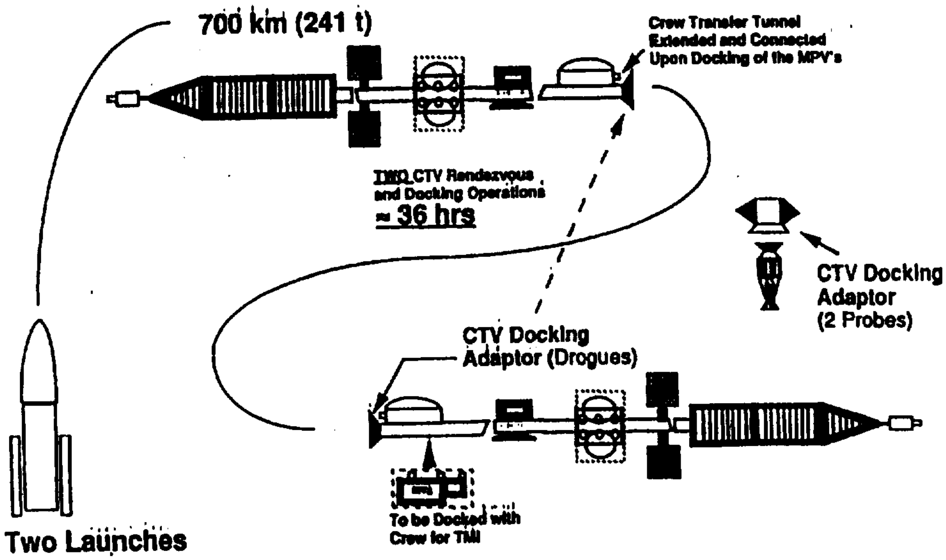
MPV Orb Ops - RENDEZVOUS & DOCK



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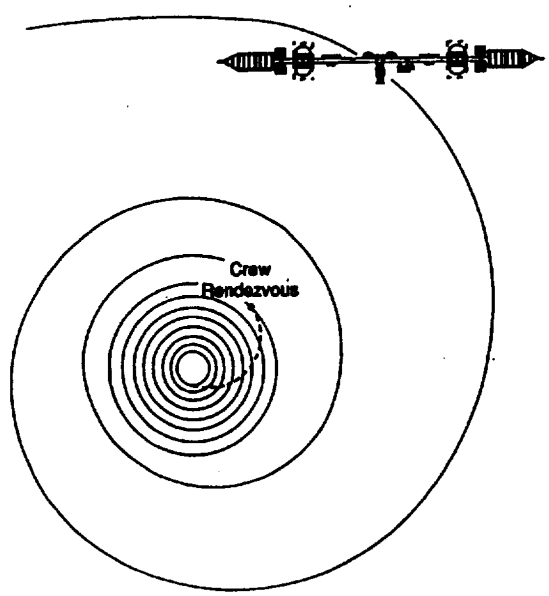
Crew Rendezvous Summary



Earth Departure Spiral

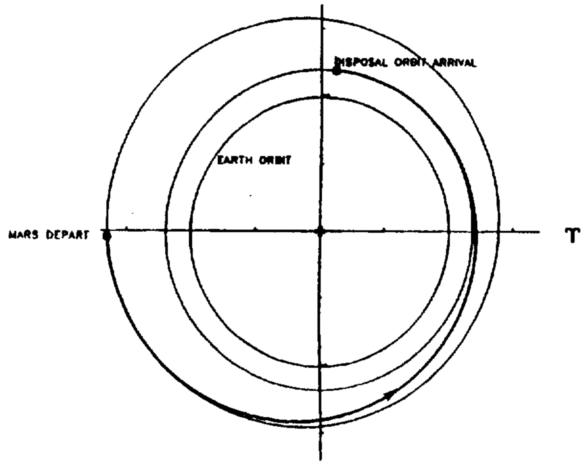
- Crew rendezvous in high Earth orbit (> 20,000 km) prior to escape
- Use co-elliptic approach and terminal closing strategy of Gemini/Apollo
- Applies to all spiral thrusting programs and Earth-Mars trajectories
- Requires a Crew Taxi vehicle
- Option: co-elliptic rendezvous in lunar orbit

Mars Orbit Operations

- A sequence of co-elliptic approaches
- Piloted chase vehicle in each case
- Avoid docking 2 large structures

SAIC

NEP Disposal - Summary



Vehicle and Infrastructure Implications

- Include auxiliary propulsion in 5 MWe module design for orbit raising (150 m/s)
- Separate disabled reactor from rest of module - optional capability
- OTV for assured removal from Earth orbit

- **Nominal End of Life** - use stable heliocentric orbit
  - modest propellant requirements
  - conservative risk management

- **Disabled Vehicle** - use interplanetary path
  - orbit life of $\geq 10^7$ years
  - collision risk similar to asteroids
  - no ΔV

- **What About Earth Orbit?**
  - temporary storage only
  - avoid long-term storage perceived risk

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Ground Rules & Assumptions

GROUND RULES:

- NO Planned EVA's for Basic Assembly or Contingency Operations
- Docking Operations are Automated
- Robotics (i.e. FTS) Used for Maintenance and Refurbishment Ops
- 700 km Orbit is the Point of Departure for Assembly and Return Ops
- Maximize Common NEP Configurations for Cargo and Piloted Missions
- Minimize On-orbit Assembly and Required Supporting Infrastructure

ASSUMPTIONS:

- Use of a Cargo Transfer Vehicle (CTV) is Available
- Flight Telerobotic Servicer (FTS) is Available
- CTV Docking Port is Available on Each Vehicle
- ≈250 t Launch Vehicle with Supporting Facilities is Available

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TS-920911.4

Mass of NEP Vehicle Missions

The NEP vehicles addressed in this study had three missions, Lunar cargo, Mars cargo, and Mars piloted with the mass breakdown as shown on the facing page.For the manned mission, there is an additional cryogenic chemical Crew Taxi with an initial mass in LEO of 57 tonnes. It is used to transport the crew from LEO to the point of rendezvous prior to Trans Mars Injection.

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TS-NEP-1FP

Mass of NEP Vehicle Missions

	Lunar Cargo	Mars Cargo	Mars Piloted
NEP Spacecraft	40	40	80
Habitation & ECCV	0	0	50
Propellant	48	91	177
Tanks	5	9	18
Cargo	140	160	0
Total	233	300	325

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TS-NEP-1

Saturn V Derived Orbital Delivery Capability

The performance calculations shown were based on a Saturn V derived Heavy Lift Vehicle (HLV) under consideration for use in the First Lunar Outpost (FLO) transportation system. FLYIT code (Martin Marietta proprietary launch vehicle simulation) was used. The HLV has a cryogenic 2nd stage. Since performance loss to 700 km is very modest and orbital decay from 700 km is about 30 times greater than from 400 km, this altitude was BASELINED for this study.

Examination of the launch mass requirements with the capabilities indicates the need for TWO launches to support each of the Mars missions, however, considerable excess capability exists. To improve the manifesting efficiency, it is suggested that a "banking" approach be considered where the extra capability is filled with additional propellant, spare components, etc. for use on other missions. These could be stored on orbit, possibly on a platform.

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TS-NEP-2FP

Saturn V Derived Orbital Delivery Capability

Orbital Altitude (km)	Payload (tonnes)
300	259
500	250
700	241

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TS-NEP-2

"Gut Feel" Baseline Mission for NEP

The basic steps to accomplish a cargo or piloted mission using NEP vehicles are summarized. Individual mission sequences along with options are described in following charts. Some of the options, i.e. return to earth of a NEP cargo vehicle are also identified.

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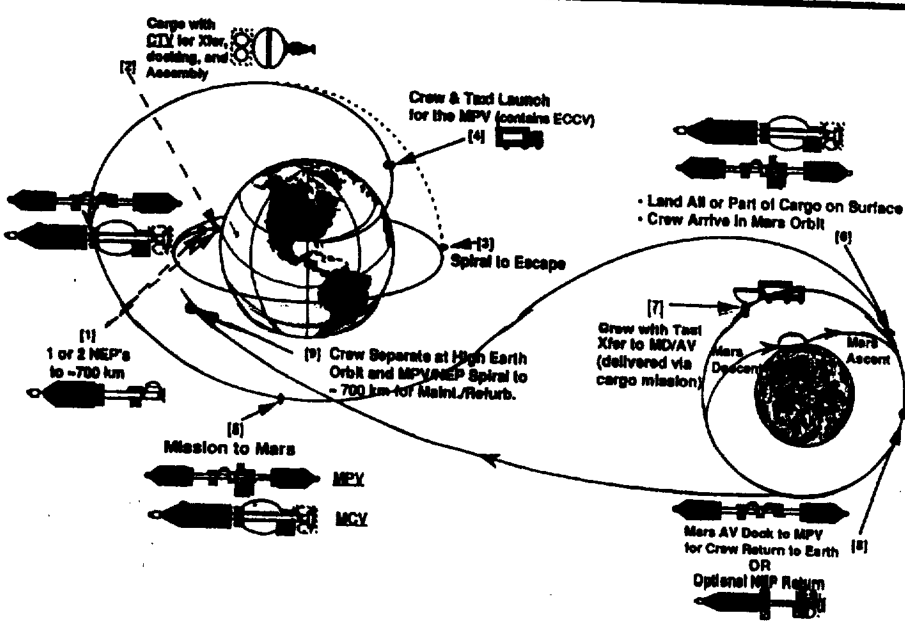
TS-812.2-FP

"Gut-feel" Baseline Mission for NEP



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Mission Sequence - MARS/LUNAR CARGO

The numbers indicate the sequence of functions. Some options are desirable at certain times in the mission as follows:

1. **Take CTV to Mars -**
2. **All cargo left in Mars orbit or some landed on Mars**
3. **NEP from Mars/Lunar flight returned and circularized in ~ 700 km earth orbit.**

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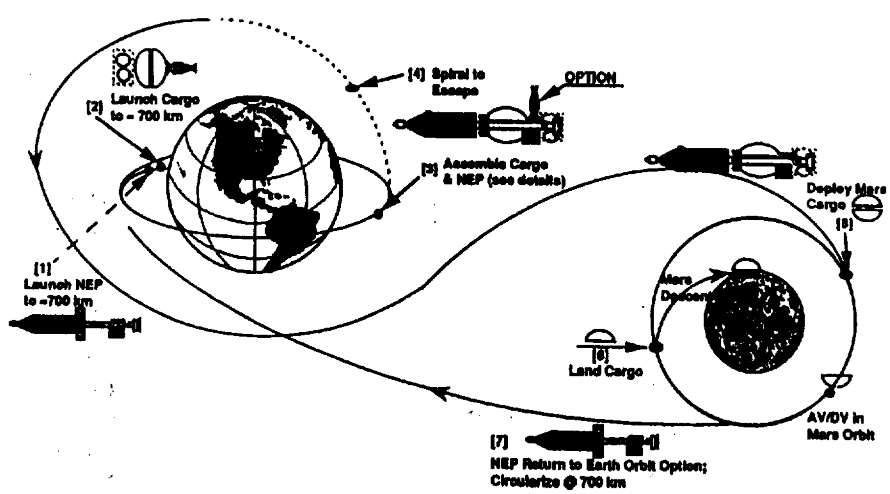
TS- 909.3-FP

Mission Sequence - MARS/LUNAR CARGO



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TS- 920909.3

Mission Sequence - MARS PILOTED, LAUNCH

Two NEP's are launched in separate launches. It may be possible to launch two NEP's with the crew habitats and one ECCV in one launch (this requires some additional conceptual work for the vehicle and habitat design definition). If the NEP's are launched separately, a CTV is used to assemble the two vehicles using a CTV adaptor. This would provide some backup since the CTV can maneuver and it would not require initial designation of each NEP as to which is the target and which is the chase vehicle. It is envisioned though that a stabilization system of some sort will be required on each NEP vehicle. Sizing of these systems and the CTV should be traded and worked in an iterative manner.

Use of the CTV and the adaptor, could provide further redundancy by implementing multiple docking probes.

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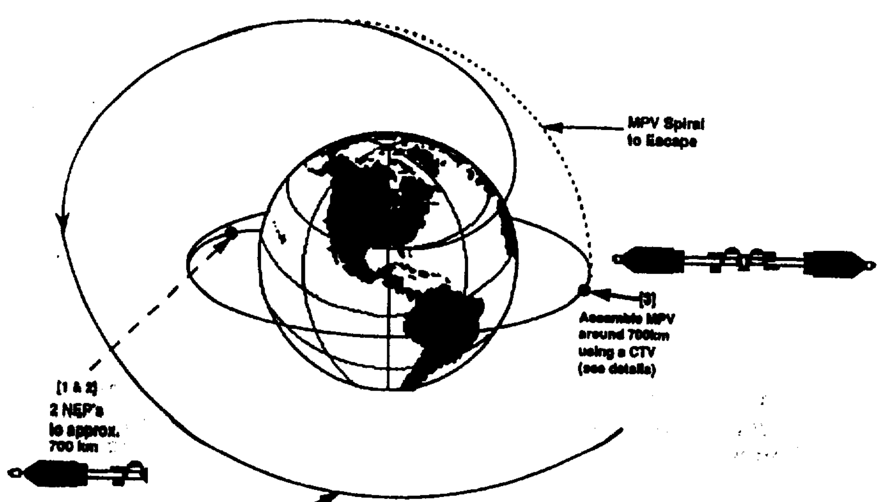
TS-909.1-FP

Mission Sequence - MARS PILOTED, LAUNCH



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TS-920909.1

Mission Sequence - MARS PILOTED, CONT'D

Upon MPV completion of spiraling to escape, the Mars crew is launched in a taxi that has an ECCV capability. The taxi rendezvous with the MPV assembly and continues to Mars. Once the vehicle is circularized in Mars orbit, the crew, using the taxi, transfers to the Mars Descent (MD)/Ascent Vehicle (AV), previously delivered to Mars orbit by the cargo mission. Subsequently the crew lands on Mars and after the requisite stay time, returns to the MPV for return to earth. When high earth orbit is attained, before the spiral down to 700 km, the crew separates in the ECCV for return to LEO or earth direct.

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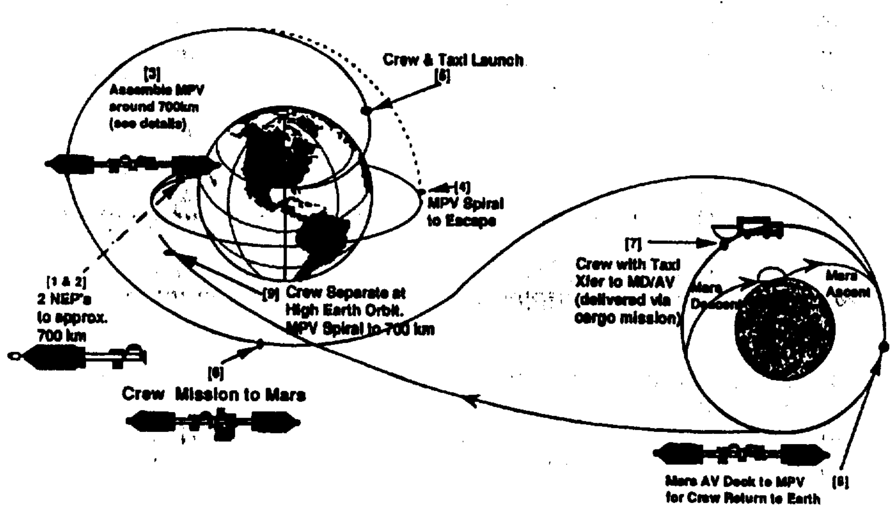
TS-908.5-FP

Mission Sequence - MARS PILOTED, CONT'D.



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TS-920908.5

NEP/MCV - Concept

To fit within a 10X30 m fairing, presently planned for HLV's, and to avoid on-orbit assembly, a recommended radiator design, used in this study, consists of 3 segments. The forward trapezoidal segment, 11 m long has a short width of 4.5 m and a large width of 8 m resulting in a 69 sq. m per side area. The remaining two segments are rectangular, 8X18 m resulting in an area of 144 sq.m per side. Thus the total radiator has an area of 357 sq. m, slightly larger than the baselineconfiguration of 347 sq. m (supplied design).

The reactor is mounted on the short width end of the forward segment and can be packaged within the conic region of the shroud.

The deployment sequence is automated and does NOT require on-orbit assembly. The automated extension of the boom is also possiL (a design of such nature was analyzed for the Thermionic Space Nuclear Power system proposal).

The remaining key items,i.e. two solar pannels (1kw each), CTV docking port, FTS and an engine pod are launched with each vehicle. Cargo, CTV and the propellant module are launched as lift and packaging capabilities allow. Specific subsystem design concepts would be required to specifically manifest and package a given mission.

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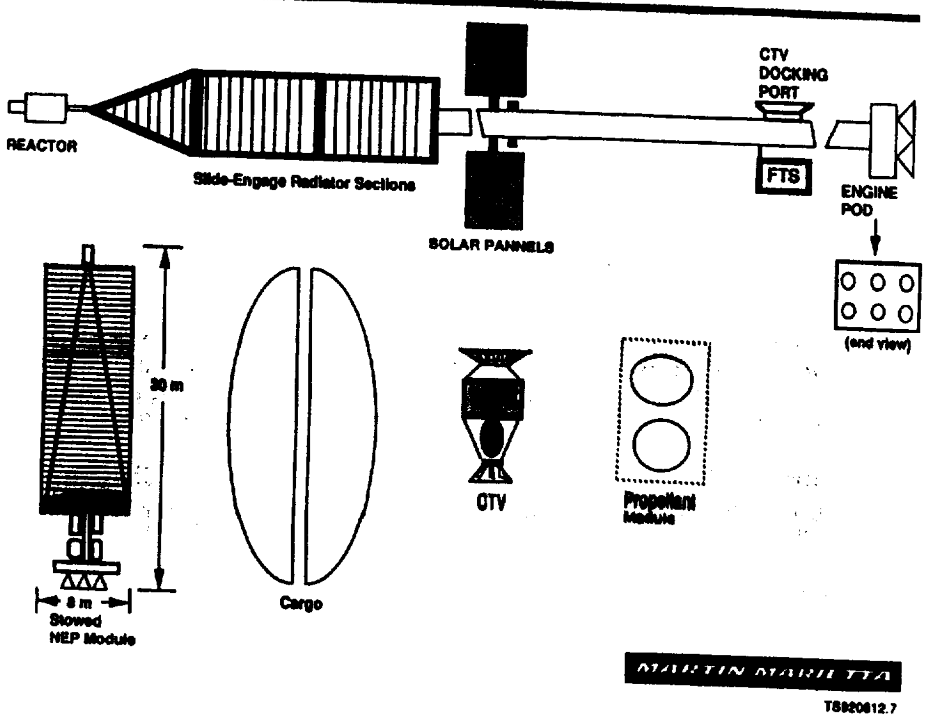
TS- NEP/MCV Conc-FP

NEP Concept - MCV



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TS920812.7

NEP Key Items

The NEP vehicle has a reactor assembly, a boom assembly, an FTS to assist in contingency, repair and on-orbit maintenance operations, an engine pod, located at the end or along the boom, depending on the use of a given vehicle, i.e. cargo/end or piloted/side, a CTV docking port, and two solar pannels (1kw each) to provide communications, control functions (RCS subsystem may be desirable) and FTS operations.

Cargo attachments (docking ports ?) for major cargo items and onboard spares will be provided and require a conceptual design to afford timeline development for maintenance or repair operations (what parameters and to what degree of finesse they must be specified is addressed under the FTS operations part of this study).

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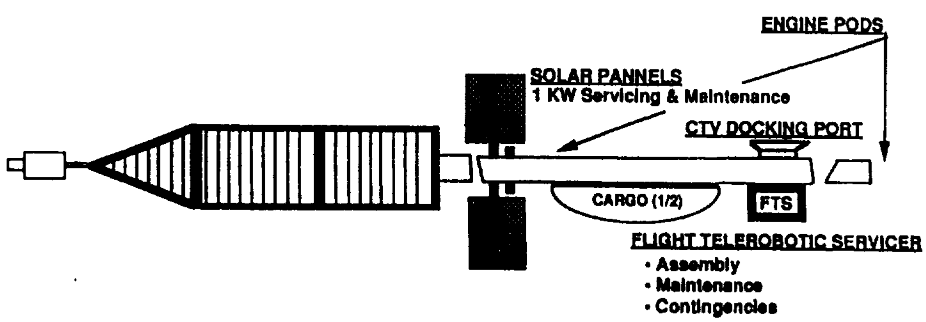
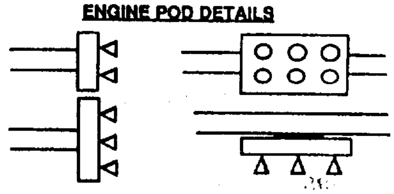
TS-811.1-FP

NEP Concept - Key Items



GROUND RULES
- NO Planned EVA for Assembly
- NO Planned Contingency EVA
- Docking Operations ROBOTIC/ Automated

ENGINE POD DETAILS



- Engine Sets Can be Mounted on Side or End of the Boom
- Propellant Tank Pods Can be Mounted on Side or End of the Boom

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TS920811.1

NEP Ground Ops Flow

The NEP processing cells can handle the basic or cargo as required. Upon completion of packaging and required amount of encapsulation, the basic vehicle or the cargo set is moved to the Vertical Assembly Building for stacking with the launch vehicle.

The only on-pad operations planned would be associated with cryogenic systems and their handling.

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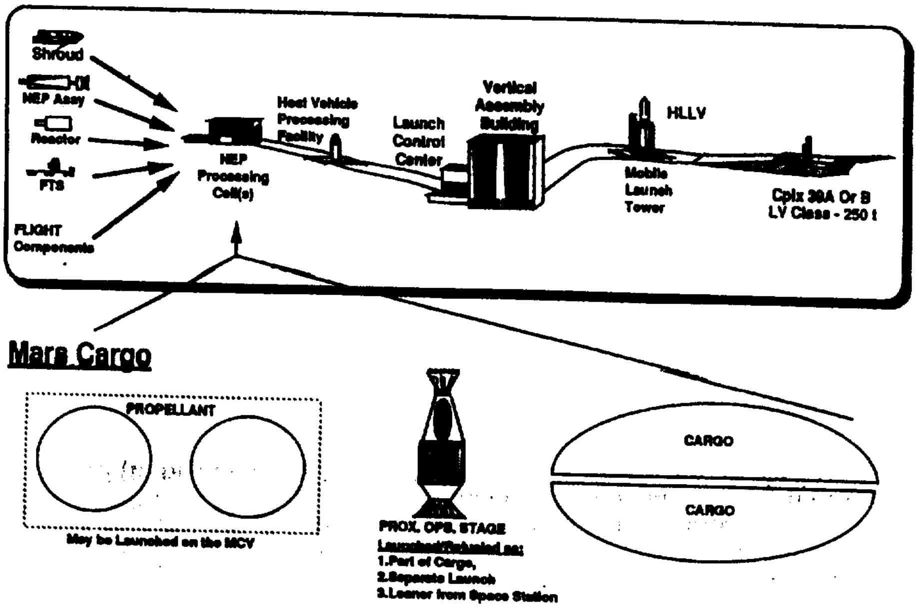
TS- 820.4-FP

NEP - Ground Ops Flow

MCV



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TS920820.4 Rev 2 Sep

NEP Processing

The items to be assembled and stowed (radiator, boom, etc.) are handled in the horizontal processing cell. The sizing of the cell should be based on a 5:1 area ratio of the stowed cargo area, plus the cargo area itself, using the shroud diameter, and adjusted for the maximum length of the unstowed (to be collapsed) items.

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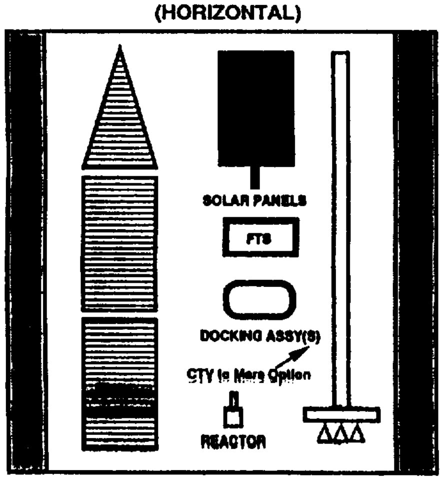
TS-920.5-FP

NEP Processing

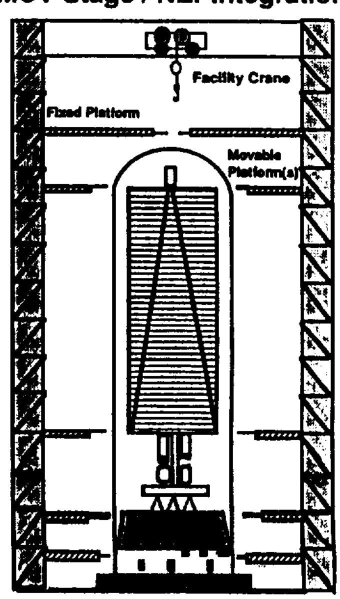
Top View

Radiator Boom and Attachments Processing

(HORIZONTAL)



MCV Stage / NEPIntegration



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TS920820.5

Mars Cargo Processing

As shown earlier in the ground ops flow, the Mars cargo will be transported from the 700 km altitude to Mars orbit using the NEP vehicle. The cargo is planned to be launched using the same HLV and thus the same ground processing facilities are envisioned.

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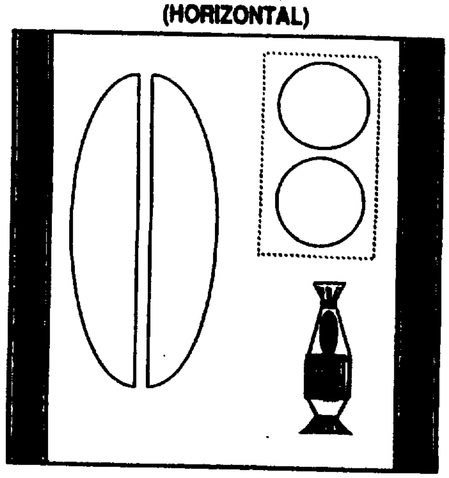
TS-3

Mars Cargo Processing

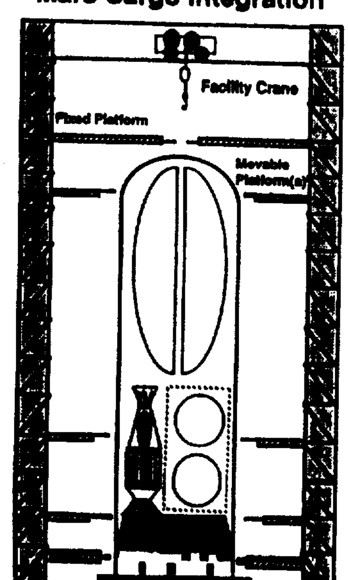
Top View

Cargo, Propellant and Cargo Transfer Vehicle

(HORIZONTAL)



Mars Cargo Integration



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TS920731.2

NEP Orbital Ops Summary - INITIAL LAUNCH

The mission planners can select which item set (NEP or cargo) is the target and which is the chase vehicle. The two will be placed at some altitude apart. They both should be located at the same inclination, thus no mention is made of orbital plane change.

It is envisioned that after the NEP vehicle launch (probably the first launched vehicle to allow confirmation that all systems are operational before committing to launch of the cargo) the stowed systems will automatically deploy and activate the prime subsystems required to communicate with and control the vehicle. The activation and checkout sequence duration will depend on the success of the automated sequences and availability of support resources (TDRSS, etc.). The subsequent cargo launch time will depend on the pad turnaround time or GO for second launch, based on the above described timeline, if a second pad is available.

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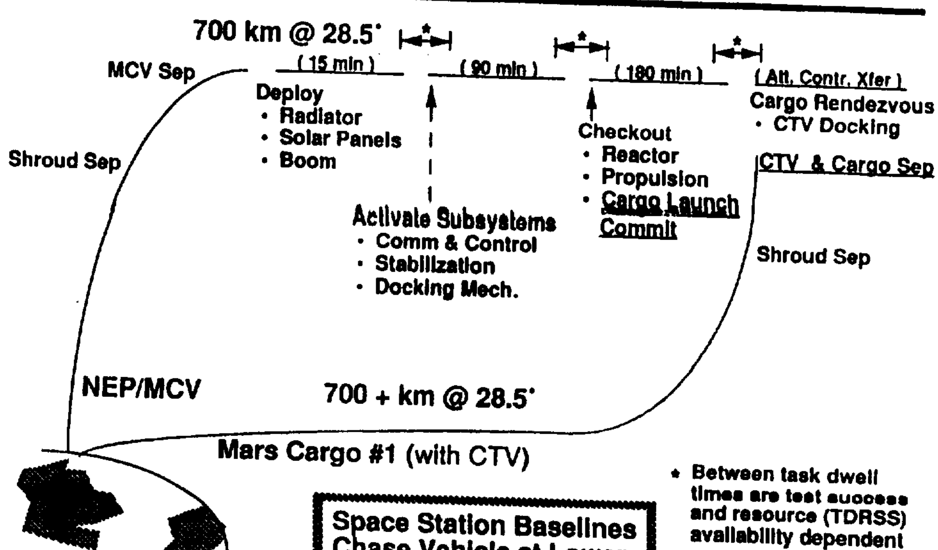
TS-820.1-FP

NEP Orbital Ops Summary - INITIAL LAUNCH



Space Station Baselines Chase Vehicle at Lower Altitude and Behind

* Between task dwell times are test success and resource (TDRSS) availability dependent

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TS920820.1

NEP Orbital Ops Summary - RENDEZVOUS/DOCK

The Mars cargo is transfered from the cargo launch location to the NEP vehicle via the CTV. Upon completion of the rendezvous and docking sequence, i.e. cargo transfer, the CTV can be retained with the vehicle as a resource and eventually taken to Mars, or deployed and returned for storage somewhere in the earth orbit realm (some options are suggested in the "Deploy CTV" sequence.

As shown, the cargo transfer can take from a few hours to a few (could be many in cases of failure or available CTV propellant limitations) days depending on the separation altitude, the desired length for a launch window, available ΔV, and the phasing angle between the two vehicles. A set of parametrics over a desired range should be developed.

There are basically two options to how the cargo is transferred; the CTV *gathers* all cargo pieces at the cargo location and takes the total mass to the NEP, or it can go back and forth to pick up individual or grouped pieces. Though it appears obvious to take the first choice, a trade study is recommended once a CTV is sized (propellant, control authority, docking mechanizm, etc.)

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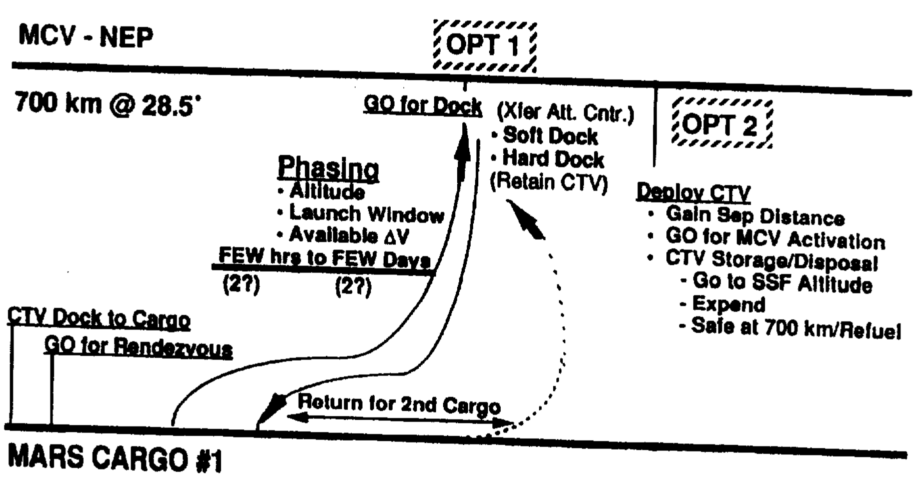
TS-720 2-FP

NEP Orbital Ops Summary - RENDEZVOUS/DOCK



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TS920720.2

NEP Orbital Ops - RENDEZVOUS/DOCK Details

The choice for the 700 km orbit that was baselined (agreed upon in a joint telecon) is referenced, and as one can see, no reboost is required at the 700 km altitude. Additional consideration of radioactive decay is discussed separately.

The times shown for cargo piece capture by the CTV along with the transfer times from cargo location to the NEP vehicle are *ball park* figures estimated from similar activities calculated for specific Space Transfer Vehicle (STV) configuration studies (see referenced sources).

It is recommended that each NEP have an FTS and a CTV docking and retention capability.

One can see that using this cargo transfer approach, a minimum of 32.5 hrs, not counting validation and verification times required by the ground crews, would be required for on-orbit assembly.

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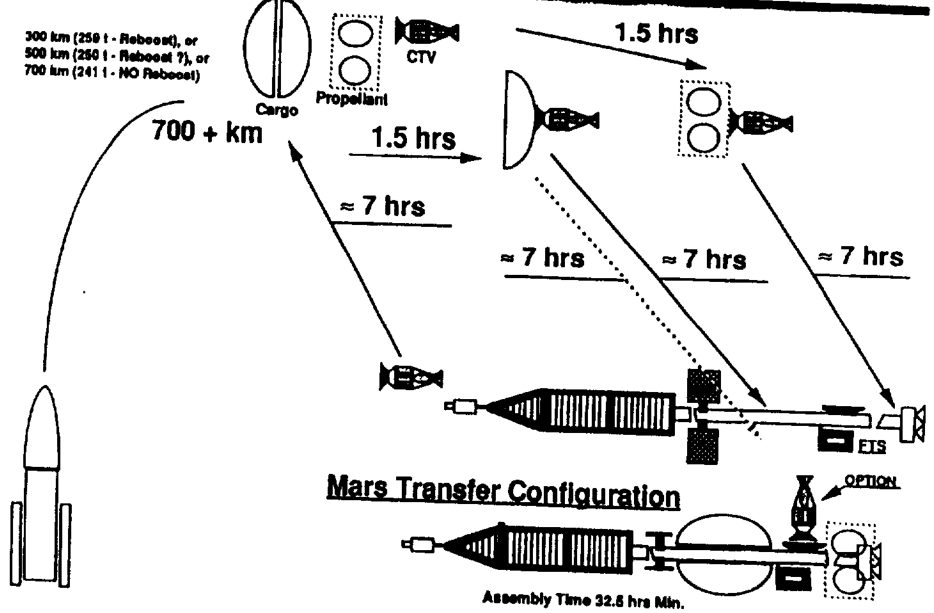
TS-731.1-FP

NEP Orbital Ops - RENDEZVOUS/DOCK Details



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TS920731.1

Orbital Ops Option - 1

When the cargo pieces are assembled before transfer to the NEP and then sequentially attached to the NEP vehicle, it appears that some time and propellant can be saved; assembly time of 22 hrs. However, no validation and verification time has been allocated for the ground crew support/control operations or potential ground resource availability constraints.

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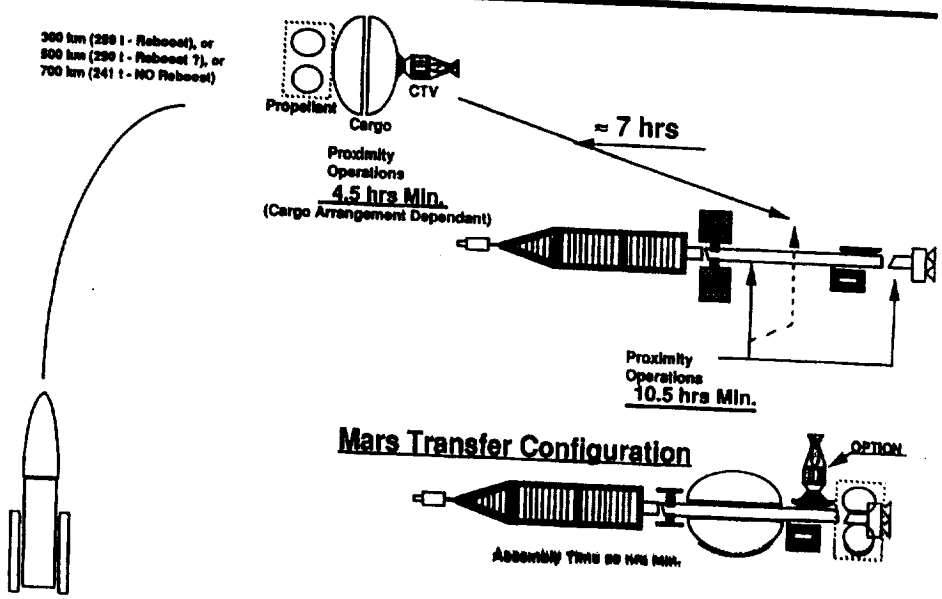
TS-803.2-FP

Orbital Ops Option - 1 (MCV)



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TS920803.2

Orbital Ops Timeline Summary - CARGO ASSEMBLY

The times, based on the STV calculated point design for a Lunar cargo transfer vehicle study #NAS8-37856, as shown would result from the number of individual cargo pieces that must be assembled. In this study we assumed the shown three major pieces.

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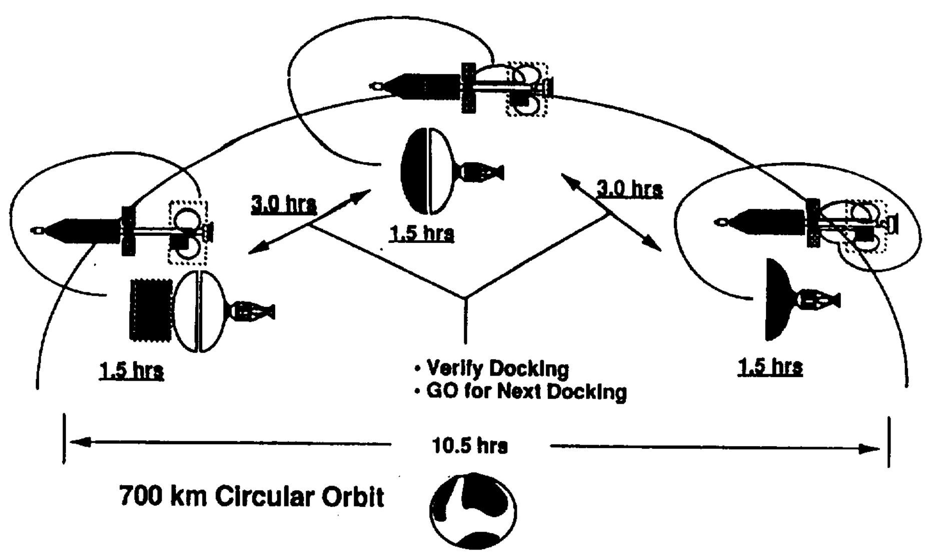
TS-810.1-FP

Orbital Ops Timeline Summary - CARGO ASSEMBLY



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TS920810.1

NEP Concept - MPV

The key differences between a NEP for Mars cargo versus the one for piloted use are:

1. The engine pod is located on the side of the boom so that adjustment for CG is possible and balanced thrust between the two assemblies during Mars transfer and return to Earth can be configured.
2. A crw habitat is provided for on each NEP to balance the CG between the two NEP modules after assembly. They are connected with a tunnel after docking. One of the habitats has an attached Earth Capture Crew Vehicle (ECCV) for contingencies. The second ECCV is carried with the taxi that is brought up as part of the crew launch.
3. A drogue assembly to interface with a CTV docking adaptor using multiple probes so that either NEP can be designated as the target vehicle and also provide backup for docking operations.

It is recommendee that each NEP for the Mars Piloted Vehicle (MPV) also be equipped with an FTS and a CTV docking port (2nd level backup).

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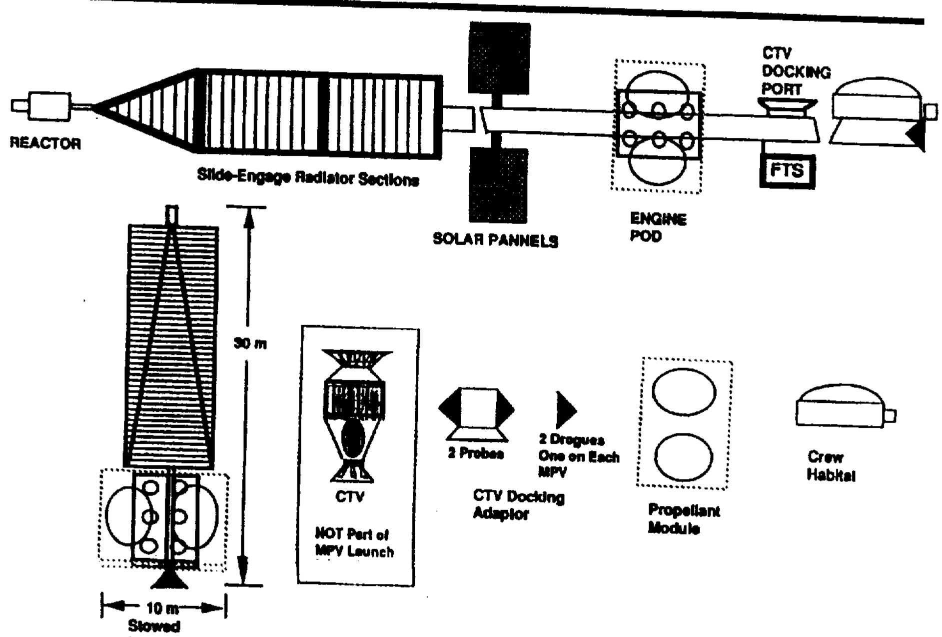
TS-814.1-FP

NEP Concept - MPV



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TS920814.1

MPV Ground Flow

The MPV ground flow is essentially the same as that for the NEP cargo vehicle except for the specific components involved. It takes two launches to get the two NEP vehicles in orbit. The crew with the crew taxi, which also contains an ECCV, is launched as a 3rd flight.

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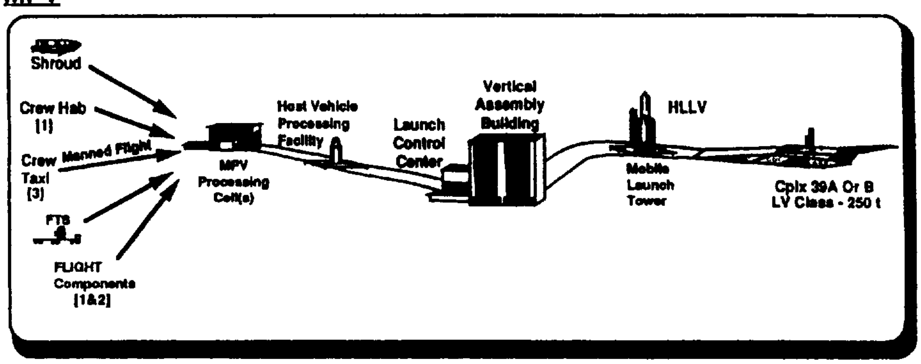
TS-811.3-FP

MPV Ground Flow

MPV



- TWO Launches with NEP Vehicles
  - One Crew Hab (includes ECCV)
  - Crew Taxi (includes ECCV) Launched with Manned Flight
- For GROUND Ops See NEP Processing
- CTV Assumed to be:
  - On-orbit from Cargo Launch
  - On-orbit from Space Station
  - Launched with One of the NEP's for the MPV

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TS920811.3Rev.

MPV Ground Processing

The same ground facilities, using the same sizing estimations as for the NEP cargo vehicle, are used to support the NEP's for the MPV.

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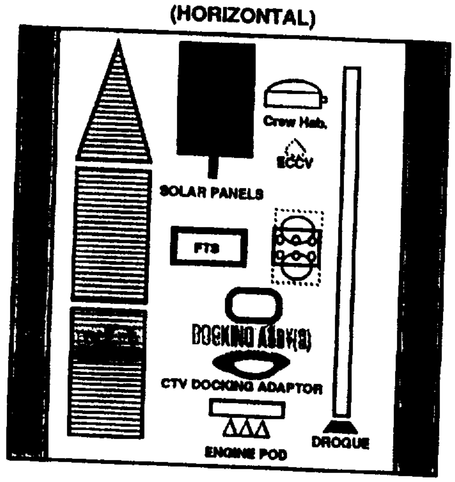
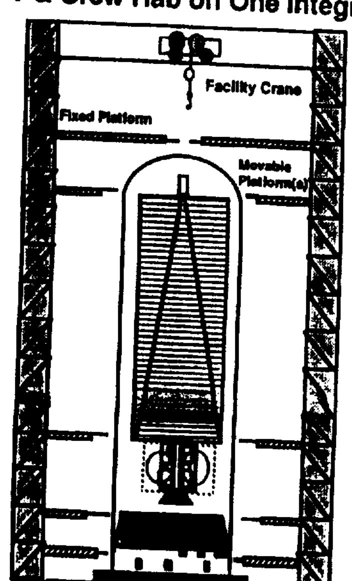
TS-812.2-FP

MPV Ground Processing



NOTE: Taxi has ECCV Capability



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TS920812.2

MPV Orb Ops - RENDEZVOUS & DOCK

Using a CTV, after each vehicle has been checked out, it is estimated based on the earlier detailed task timelines, that the rendezvous and docking operation will require a minimum of 36 hrs.

Once docked, the crew transfer tunnel will be extended connecting both MPV/NEP modules.

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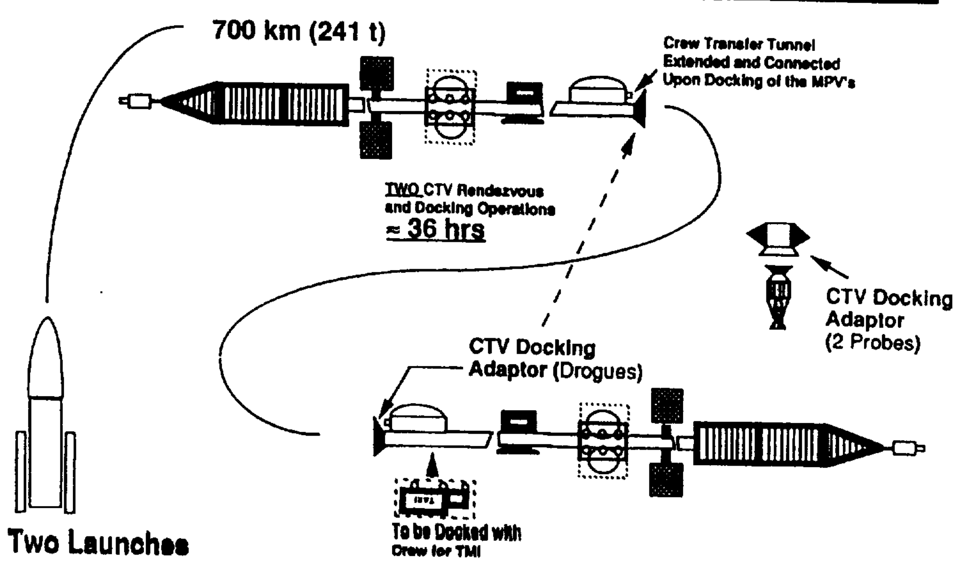
TS-812.2-FP

MPV Orb Ops - RENDEZVOUS & DOCK



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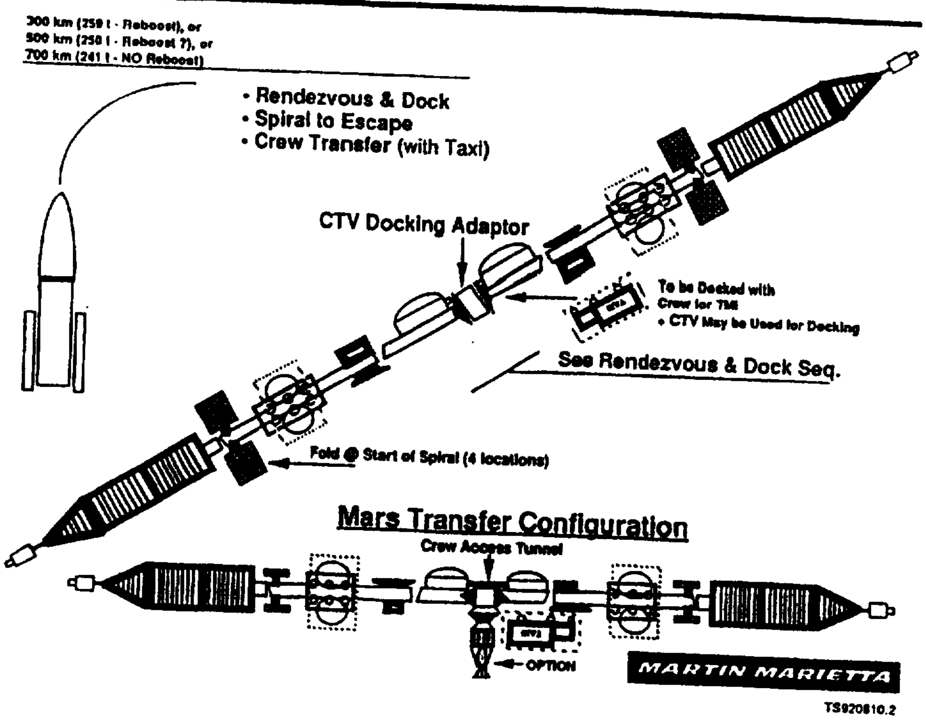
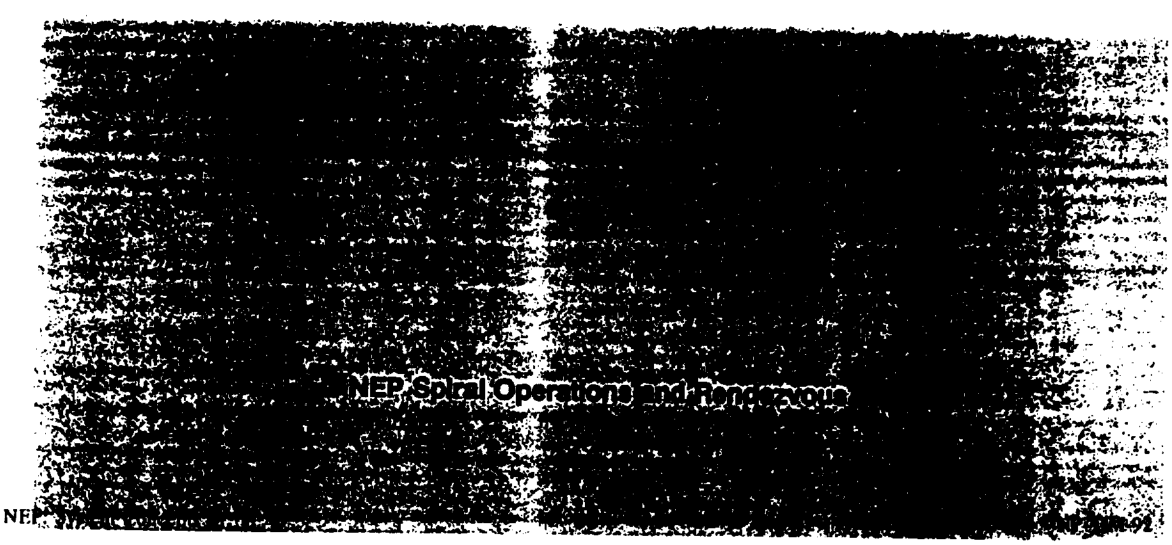
MPV Orbital Ops

For the spiral out or the final Mars transfer configuration, the CTV may be taken along or left behind. The crew taxi is brought up with the crew launch, however, the docking operation may utilize the CTV. As can be seen, sizing of the CTV in terms of control system, available propellant and ground control interfaces is desirable before more detailed task assessments are undertaken.

MPV Orbital Ops





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The Rendezvous Profile

The typical [illegible] sequence divides into [illegible] to achieve [illegible] relative positioning between [illegible] conditions of one phase [illegible] starting conditions for the [illegible] narrow dispersion [illegible] nominal conditions.

The approach phase results in approximate matching of the orbits of the target vehicle and the chase vehicle. Approach usually calls for a sequence of maneuvers by the CV [illegible] orbit parameter at a time: inclination, line of nodes, eccentricity and altitude. After the orbits are nearly matched, or relative differences fall within a predetermined range, the terminal closing phase reduces the separation and closing rate to nearly zero. Final checkout, if required, is followed by the physical docking sequence.

This study examines alternate approach/terminal closing techniques for use at Earth departure and in Mars orbit.

The Rendezvous Profile

Designate a passive Target Vehicle (TV) and an active Chase Vehicle (CV)

- **Approach** Impulse sequence establishes nominal starting conditions for the terminal closing phase

  Example: CV moves to concentric circular orbit just below TV altitude (say 20 km) by adjusting one orbit parameter at a time

- **Terminal Close** Impulse sequence reduces range and range rate for final docking

  Example: CV uses line-of-sight thrusting to raise altitude and close to within a few meters of TV

- **Station-keeping** final (optional) checkout prior to docking

- **Docking** Combination of small impulses and physical grappling devices

Of the several rendezvous schemes considered for Gemini and Apollo, the circular, coplanar method was selected. First, the target vehicle's orbit was established at a selected altitude. Then, the chase vehicle launched and began the approach phase, modifying its orbit with a preplanned impulse sequence. Since these flights involved human crews, time to rendezvous was minimized at the expense of some additional propellant. Autonomous rendezvous could follow the same general procedure, using a maneuver sequence designed to minimize propellant over a longer time interval.

The chase vehicle approach phase ended in a circular, coplanar orbit at slightly lower altitude, with the chaser lagging the target by a few tens of kilometers. For Gemini, the altitude difference was 15 nautical miles, or about 28 km. The range was 30 - 40 N.Mi., since predicted visibility would give a clear line of sight to the Agena target at that range.

The Apollo rendezvous followed a similar sequence. Just after the CSM passed overhead, the LM launched from the surface to a transfer orbit of 60,000 feet by 45 N.Mi. Circularization at 45 N.Mi. gave the starting conditions for terminal closing phase. The entire sequence was completed 3.5 hours after the LM liftoff.

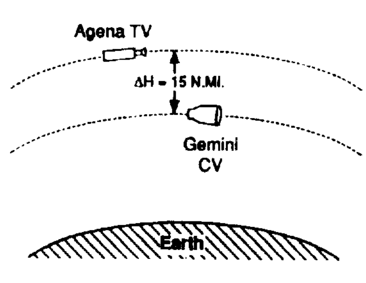
The terminal closing phase for Gemini and Apollo was flown manually, using line-of-sight thrusting by the chase vehicle. The entire approach phase design was intended to produce standard conditions (lighting, direction, range, range rate, and required ΔV) to begin the terminal closing phase. For Apollo, a faster rendezvous approach would have used direct ascent from the surface to standard terminal closing conditions; but the expected dispersion range in starting conditions would have been too large. The concentric orbit approach reduced this dispersion to acceptable values.

Note that the orbits need not be circular: the same control can be achieved with co-elliptic orbits.

Orbit Rendezvous Experience Base

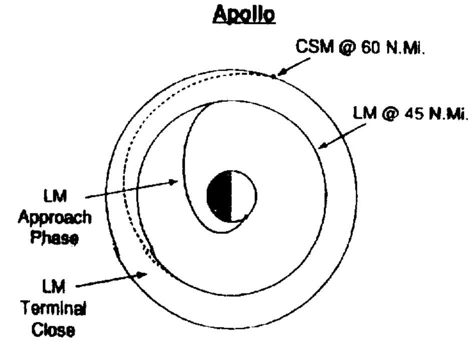
- Approach phase puts target and chase vehicles in circular, coplanar orbits with specified altitude separation, ΔH (can also be co-elliptic)
- Terminal closing phase performed manually, so standard initial conditions are very desirable:
  - approach direction
  - lighting conditions
  - line-of-sight rates
  - nominal ΔV budget

Gemini



- Chase Vehicle below and behind Target to commence Terminal Closing: Range ≅ 30 - 40 N.Mi.

Apollo



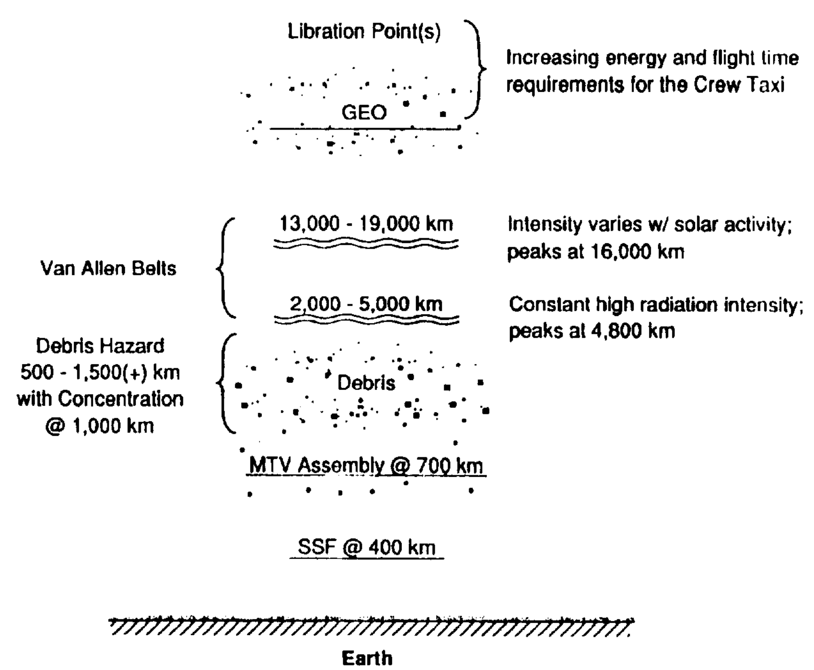
- LM ascends, injects to 60,000 ft x 45 N.Mi., then circularizes at 45 N.Mi. to start Terminal Closing
- 3.5 hours lift-off to docking

Rendezvous Selection Considerations

Crew rendezvous with a spiralling NEP transfer vehicle is complicated by hazard avoidance and timing considerations. Minimizing crew time traversing the radiation belts suggests a location above 19,000 km altitude. But higher orbits mean higher energy requirements for the crew taxi and, more importantly, longer phasing periods for the rendezvous sequence.

The list of operational constraints on the following chart suggests that considerable work will be needed to define near-optimal rendezvous strategies for an NEP transfer vehicle departing Earth. We consider four basic alternatives as a preliminary evaluation.

Rendezvous Selection Considerations



Crew Taxi Rendezvous with NEP Transfer Vehicle

Problem: Pick an Earth orbit location and an approach/rendezvous sequence that:

- minimizes crew exposure to natural and on-board radiation
- minimizes risk of orbital debris impact
- minimizes crew time on board the MTV
- minimizes vehicle design and propulsion requirements for the crew taxi and for the Mars Transfer Vehicle
- minimizes complexity of operational sequences for nominal and fallback modes
- minimizes crew time spent in rendezvous

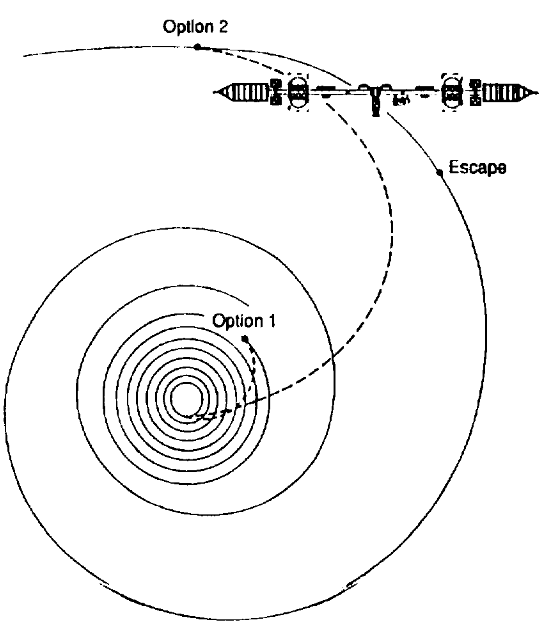
Three of the options proposed for rendezvous are shown opposite. The first is to select a high Earth orbit altitude, above the van Allen belts and free of debris collections. A controlled co-elliptic rendezvous sequence would build on our experience base from early manned programs.

The second option is to rendezvous post-escape, somewhat analogous to the direct ascent approach considered and rejected for Apollo. NEP thrusting would be suspended long enough (exact interval to be determined, but probably a few days) to reduce the radiation hazard and permit the crew taxi to chase a target with relatively stable orbit conditions. Since approach and terminal closing phases are combined, there is one less measure of control over the close approach conditions. Off-nominal burns from LEO departure create a broader range of possible approach conditions than the co-elliptic strategy. Moreover, there is only one chance to "catch the bus."

The third option, not diagrammed on the chart, is to deliver both the MTV and crew taxi to one of the Earth-Moon stable libration points, and rendezvous there. Previous studies (post-Apollo) suggested some advantages for the trans-lunar L2 point as a node, over the L1 point. However, the selection is moot in the case of the reference trajectory and spiral, because the MTV reaches escape conditions well before reaching lunar distance! To use either libration point would require modifying the spiral to use a non-optimal thrust program; this can be done, but at the expense of additional time and propellant for the spiral. This also adds thrust-on time to count against thruster lifetime limits.

The final option is to rendezvous in low lunar orbit. The crew would be sent out on a Lunar Transfer Vehicle, possibly as "hitchhikers" on a regular lunar mission, to board their MTV waiting in orbit. Feasibility of this approach depends on the lunar exploration manifest and infrastructure to support it. A ΔV of about 2-3 km/s would be needed for NEP orbit capture/departure, but this is likely to produce only a small increase in propellant loading. Of course, this approach adds some operations complexity in scheduling concurrent lunar and Mars flights.

Rendezvous Location Options



Option 1: High Earth Orbit

- Suspend NEP thrusting program anytime before reaching escape
  - establish target vehicle orbit
  - power output decay (10- day delay, per MMAG)
- Crew taxi departs LEO to co-elliptic orbit position below and trailing the target NEP vehicle
- Perform co-elliptic terminal rendezvous sequence and dock with NEP
- Continue NEP spiral to escape

Option 2: Post-Escape

- Suspend NEP thrusting program only as long as required for crew safety
- "Direct ascent" trajectory to rendezvous
- Combined approach and terminal closing phases

Option 3: Libration Point Rendezvous

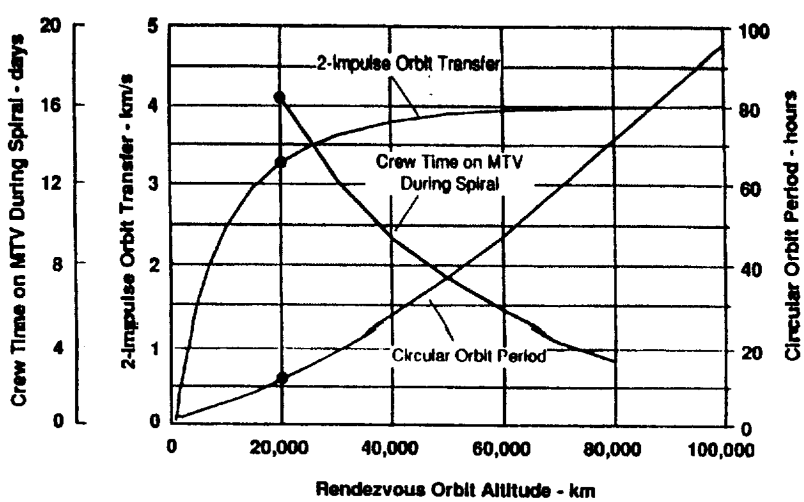
- Both vehicles transfer to L1 (or L2)
- Not shown opposite because this optimal thrust program reaches escape conditions well before lunar distance

Considering the high orbit (Option 1 on the previous page), there are performance impacts of selecting an altitude. A two-impulse transfer from LEO would use the first burn to raise the orbit apogee to the selected altitude, and the second burn to circularize there. Assuming this burn sequence, the ΔV requirement increases rapidly with altitude, but flattens out above geosynchronous altitude (35,786 km). However, the radiation hazard of the van Allen belts forces a selection higher than 19,000 km, so the crew taxi must be able to handle in excess of 3 km/s impulse from the main engines.

At the same time, orbit period is increasing from a few hours at lower altitudes to significant fractions of a day at higher orbits. A longer period implies a longer rendezvous and docking sequence, especially for fall-back options that require more than one or two revolutions. Therefore, even though there is a limited energy savings to be gained from using the lowest possible orbit above the radiation belts, there is an operational advantage. We propose an altitude of 20,000 km, assuming a roughly circular orbit for crew transfer to the departing MTV.

The third curve on the opposite page shows the additional time the crew will spend aboard the MTV if this co-elliptic approach is used. The suggested altitude requires an extra 17 days on board the MTV in addition to the Earth-Mars transfer time.

Mission Performance Impacts of Rendezvous Orbit Selection



- Crew Taxi impulse increases rapidly with altitude; hits a "knee" at ~20,000 km
- Orbit period (circular) increases linearly with altitude. The longer the period, the longer the terminal rendezvous sequence for a co-elliptic rendezvous.

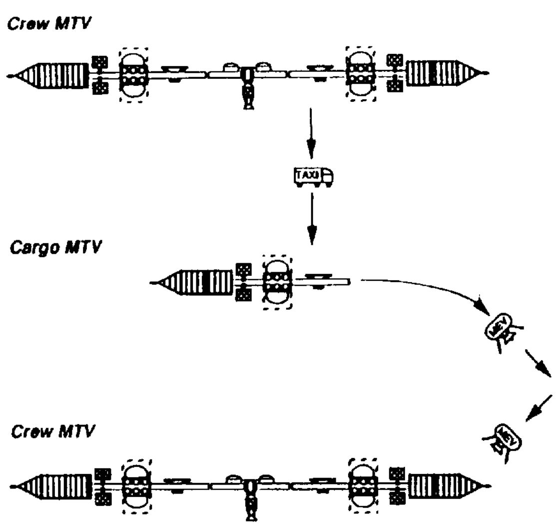
Mars Orbit Operations: MEV Deployment & Return

Several rendezvous an docking operations in Mars orbit are required to support the surface mission and return trip. The cartoon opposite illustrates one approach that may minimize the complexity of each step, but at the expense of adding at least one step to the process.

To begin, the crew MTV spirals to capture at Mars in an orbit that approaches the cargo MTV which has arrived earlier and has already deployed part of the surface payload. From this rough matching of orbit parameters, the crew taxi or another element designed for this purpose completes the terminal closing phase to transfer the crew to the MEV brought out by the cargo vehicle.

After conducting the surface mission, the crew returns directly to the crew transfer vehicle in the MEV, completes a co-elliptic rendezvous, and readies for departure.

Mars Orbit Operations: MEV Deployment & Return



- Crew MTV spirals to rendezvous orbit
- Allow delay of several (≤ 10) days after reactor shutdown before crew movement begins
- Crew Taxi shuttles crew to Cargo MTV for transfer to MEV
- MEV separates and begins descent sequence
- Surface mission
- MEV ascends to co-elliptic rendezvous with Crew MTV
- Crew MTV spirals to escape on Earth return trajectory

The advantage to this approach is eliminating the need to dock the crew and cargo MTVs. The only transfer requirement for the baseline mission profile is to move the crew from transfer element to excursion element and back again; no propellant transfer is required for the crew's return.

Mars Orbit Operations

Several independent rendezvous operations with different active partners

- Crew MTV must perform the gross maneuvers of approach to match orbit parameters with the cargo MTV, already in orbit
- Crew Taxi (or similar element) must perform terminal close and docking to transfer the crew to the MEV.
- MEV must perform complete rendezvous and docking sequence upon return from Mars surface.

Alternative: Crew MTV and Cargo MTV rendezvous

- Requires close maneuvering of two large structures, and appropriate scarring for all operational sequences at Earth and Mars.
- Complicates crew safety on approach: must avoid 3 radiation sources

NEP Rendezvous Approach and Design Implications

Earth Escape

- Rendezvous at Earth-Moon L2 may be incompatible with the optimal thrusting program for spiral escape; spiral time could be extended, but at the cost of extra thrust time.
- Select a high Earth orbit altitude (20,000 km) for co-elliptic approach/rendezvous
  - standard, controlled rendezvous sequence
  - permits delay for power decay after shutdown, before crew approaches
- Crew taxi must have ECCV capability and be able to handle ΔV of 3.5 km/s
- Increases crew time on board MTV by a few days (17 in this case)

Mars MEV Separation/Approach

- Use crew taxi to ferry crew from their MTV to the MEV
- Eliminates the need to rendezvous and dock two large structures

On-orbit Support Requirements

- **PLATFORM in a 720 km Orbit** [*Study Indicates Operational Advantages*]
  - **Reboost**
  - **Attitude Control**
  - **Ops Power**
  - **CTV Storage/Dock**
- **CTV**
  - **Cargo Transfer**
  - **NEP Repositioning/Reboost Backup**
  - **MPV Rendezvous & Dock**
- **Mission Control**
  - **Deployment Verification**
  - **Next Function GO**
  - **Rendezvous/Docking Calculations**
  - **Auto Sequence(s) Overrides**
- **Space Station Interface** (contingencies, backup, CTV?)

TS920825.1

NEP Weight Statement

To assess the ability of the FTS as presently designed to handle specific items, the weight statement as shown was used. Each item was viewed from a mass aspect to see if it is a contender for handling by the FTS. The FTS task column indicates the results. In the case of the power distribution system, the 10000 kg are probably devided between various components, each of which could be handled adequately. However, to finalize such an assessment, the design to at least a conceptual level, for each subsystem component, must be defined. It is the location of each item that will determine how long it takes for the FTS to get to it, what motion is required to twist/pull/push/lift etc. for handling each item, and thus establish requirements on the FTS and the subsystem components. Obviously this is a very interactive and iterative process.

The same discussion as above applies to the Taxi and Crew Habitat handling since they will consist of components.

Repair operations where pull and push functions by the FTS are probably desired, will impact the design requirements placed on these components. Particularly in this group would fall the solar pannel mechanisms, the thrusters, and propellant/electrical connectors.

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TS-812.4-FP

NEP Weight Statement

MCV/LCV	Mass kg	FTS Task
• Reactor/Radiator Assembly	23285	N/A
• Solar Pannel Assembly	163 each	√
• Flight Telerobotic Servicer	700	N/A
• Engine Pod	3000	√
• Propellant Module	10000 dry	√
• Power Distribution	10000	?
• Miscellaneous Structure	4xxx	
+		
• 2 x MD/AV (Cargo)	75000 x 2	
MPV		
• Taxi(with ECCV capability)	57000	?
• CTV Docking Assembly	2000	√
• Crew Habitat Module (with ECCV)	50000	?
MCV/MPV OPTIONS		
• CTV Docking Port	500	N/A
• CTV Docking Adaptor	2000	√
• CTV (Wet)	6000	√

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TS920812.4

Rendezvous, Prox Ops, FTS & Other References

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TS 819.1-FP

Rendezvous, Prox Ops, FTS & Other References

RENDEZVOUS & PROX OPS: (Bill Jackson / JSC [713]483-8303)

- Space Transfer Vehicle, Lunar Transportation Study NAS8-37856, ΔV Allocations, Timelines, and Earth/Lunar Orbit Rendezvous
- NLS Cargo Transfer Vehicle Guidance and Targeting Strategies, Wayne Deaton NASA-MSFC, 8 April 92
- CTV Briefing #3 to MSFC (Martin Marietta Proprietary)

FTS:

- Max Load Carrying Capability Final Report; MMAG Memo FTS-SYS-90-473
- An Analytic Solution for Robotic Trajectory Generation, MMAG Memo FTS-SYS-90-452
- Contract # NAS5-30689

OTHER

- 1 KW SUPER Design for the P91-1 Program

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TS920819.1

FTS - Timeline Considerations

The referenced FTS documents were used for showing a boundary of of how item mass relates to maneuver time including general considerations as listed. This only addresses the motion of lift/move itself. To develop total task timelines, the design (at least at a concept level) is needed.

Note that denser objects can be moved faster since they will be smaller and their CG closer to the attach point, therefore a shorter lever arm.

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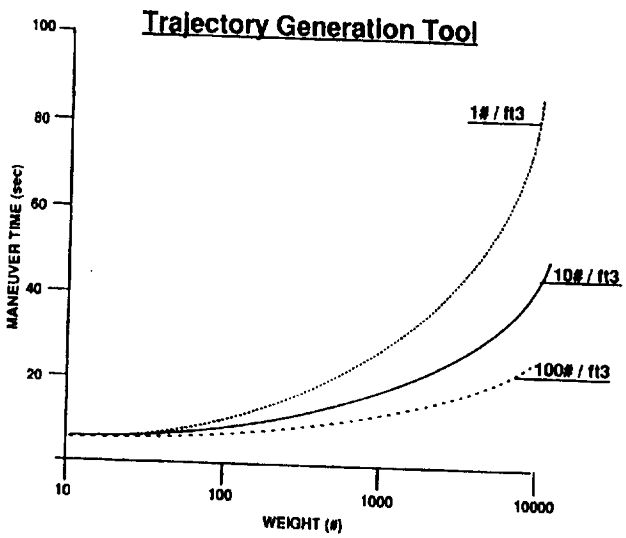
TS-729.1-FP

FTS - Timeline Considerations



CONSIDERATIONS INCLUDED:

1. Joint Torque Limits
2. Joint Velocity Limits
3. Mass Properties
4. Maneuver
5. Position Loop Bandwidth
6. Simulation Model
7. Safe Velocities

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TS920729.1

NEP Orbital Ops Summary - FTS

The tasks listed is a beginning of a long list that needs to evolve as the vehicle conceptual design evolves. The specific item single maneuver time needs to be connected with the task timeline, which requires the knowledge of location, reach distance, etc. and thus leads to the recommendation that a conceptual design for the subsystems an' therefore the total vehicle be undertaken.

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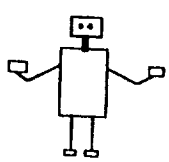
TS-820.6-FP

NEP Orbital Ops Summary - FTS

CONTINGENCIES

- Cargo Secure
- Power Deploy



SINGLE MANEUVER TIME

ITEM	lbs/ft3	sec
• Engines	9.4	15
• Engine Pods	9.4	30
• Power Cond.		
• Solar Panel	3.3	12

MAINTENANCE

- Engines @750kg/5m3
- Engine Pods (4 engines) @3000kg
- Power Conditioner 10000kg/?
- Solar Panels @ 111 kg each

NOTE: 35.32 ft3/m3
2.21 lbs/kg

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TS920820.8

Maintenance & Refurbishment Scenarios

The NEP vehicle is basic for the Mars cargo, Lunar cargo, and the Mars piloted flights. Variations in vehicle configurations depend on the specific mission. As was seen from previous discussions on cargo rendezvous and docking sequences and their relationship to manifests, it appears that a unmanned, passive platform could be of operational advantage. The platform could also have a dedicated FTS to perform such tasks as thruster replacement where the remainder of the pod is operational (failures that have occured before expected end of life).

The numbers under each type of equipment indicate the total number recommended for use in accomplishing a given Mars mission.

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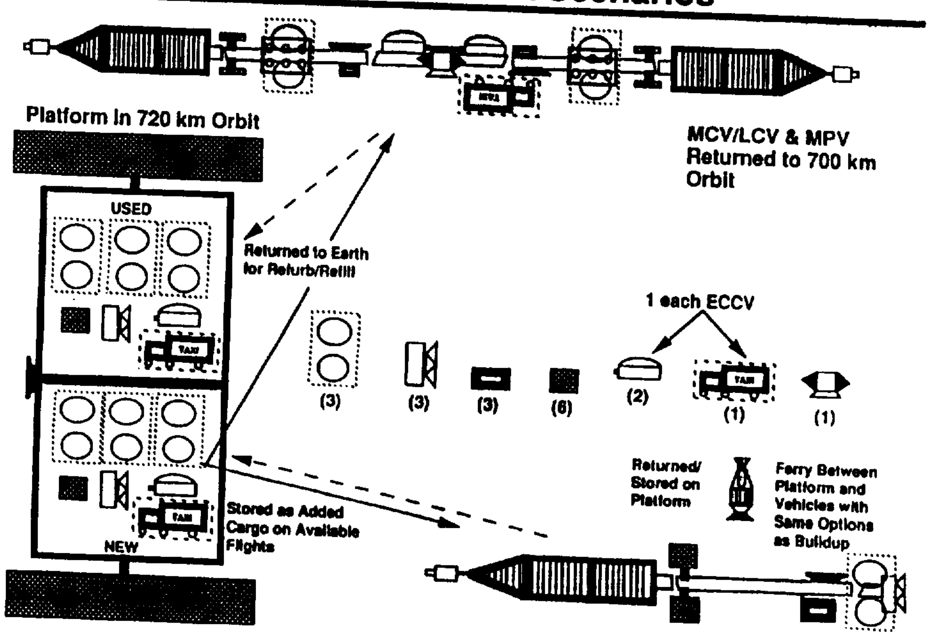
TS-819.3FP

Maintenance & Refurbishment Scenarios



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Vehicle Refueling

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TS-819.4-FP

Vehicle Refueling

- Fluid Transfer NEP Veh. (trade study required - does NOT look favorable)
  - Propellant in Module Form for Initial Vehicle Configuration
  - Maintain Propellant Module Synergism
- Fluid Transfer CTV Appears Favorable

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TS920819.4

Thruster Replacement

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IS-819.5FP

Thruster Replacement

- Thruster OR Engine Pod Replacement Is Feasible with FTS Design
  - Mass drives maneuver time
  - Component design will drive:
    - Accuracy Req.
    - Force Req.
    - Dexterity Req.
    - Reach Req.

These and Moving Distance Determine Total Task Timelines

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TS920819.5

Non-nuclear System Repairs

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TS-819.5-FP(1)

Non-nuclear System Repairs

- In General Possible and Desirable (specific dynamics have been analyzed)
  - Specific Design Dependent
  - Mass Density Dependent
- FTS May be Usable in Conjunction with the CTV

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TS920819.5

Refurb & Maintenance Schedule

Some of the possible candidates for refurbishment and maintenance are identified and their potential schedule suggested. Again, until at least a conceptual level of subsystem design is performed, specific component replacements, their projected reliabilitiy and buildup of that particular function, as shown in this list, can not be accomplished.

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TS-818.3-FP

Refurbishment and Maintenance Schedules

REFURBISHMENT ITEMS	SCHEDULE
• Solar Power - Replace Panel Assembly (2/vehicle)	Each Mission
- Replace Battery Assembly (2/vehicle)	As Req.
• Crew Habitat	Each Mission
• Engine Pods	Each Mission *
• Propellant Module	Each Mission
• Taxi	Each Mission
• CTV Docking Adaptor	Upon Failure
• FTS	10 yrs/Failure
• CTV	As Req.

MAINTENANCE ITEMS	
• Solar Power - Drive Mechanism Inspect/Replace	As Req.
• Crew Habitat - Selective Items	As Req.
• CTV - Selective Items	As Req.

NOTE: * An option of taking extra pods to Mars for scheduled replacement should be considered

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TS920818.3

Decay Power of a 5 MWe NEP

Upon return and subsequent to shutdown of each 5 MWe module, the decay time and power were tabulated. On the basis of these results it is recommended that a minimum of 10 days be allowed before any cargo or propellant loading is initiated. One can see that a further wait to 100 days would only further reduce the doses by a factor of 0.4.

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908.2-FP

Decay Power of a 5 MWe NEP - AFTER SHUTDOWN

Time (days)	Fraction of P rated	Decay Power (kWt)
0.1	0.01	244
1.0	0.005	122
10.0	0.0015	37
100.0	0.0006	15
1000.0	0.0003	7

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920908.2

10 MWe NEP Radiological Inventory if Re-entering

The worst case scenario for a Mars piloted vehicle failing in all aspects upon return to a 700 km LEO orbit would have a radiological inventory as shown. The vehicle has two 5 MWe modules for a total power of 50 MWt. The Mars mission is assumed to last for three full power burn years for a total reactor usage of 150 MWt-years. Since re-entry from a 700 km orbit for this type of vehicle (ballistic coefficient of 200 kg/m2) is expected to be around 54 years, the radiological hazard would be ≈100,000 Ci.

The probable health consequences are ZERO, since odds are 75% that the system will land in the ocean and sink through the bottom immersing 50 to 100 m below the sub-sea bed, thus safe disposal.
If the reactor were to re-enter over prime farm land, breaking up and dispersing, the prime hazard will come from the bone seeking isotopes Sr90 and Cs137, both with half-lives of ≈30 years. Typical crop condemnation level is ≈1 Ci/km2. Thus under the worst smooth scattering possible, about 100,000 km2 could conceivably be contaminated. If the crop were wheat, assuming $2.50 per bushel at 40 bushels to an acre, economic losses would be $2.5 B/yr. Clearly this would not be acceptable and an infrastructure to assure prevention of this type of an accident is recommended.

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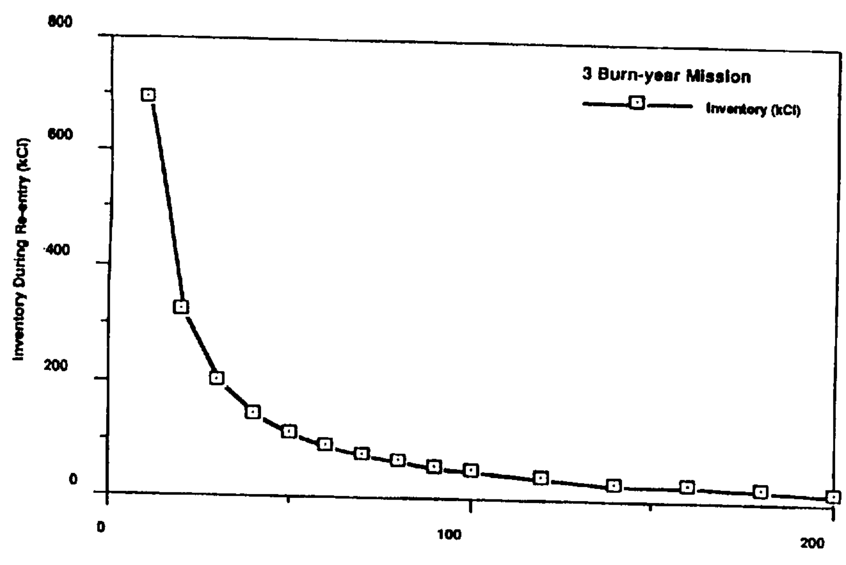
911.1-FP

10 MWe NEP Radiological Inventory if Re-entering



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Further Study Recommendations

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TS-918.2-FP

Further Study Recommendations

- SIZE CARGO TRANSFER VEHICLE (Opt.1=take allong; Opt.2=leave in EO)
  - Control System
  - Propellant (Cryo, Space Storable Cryo, Storables TRADES)
  - Communications
- SIZE FLIGHT TELEROBOTICS SERVICER
  - Cargo Assist
  - Routine Maintenance
  - Potential Contingencies
- POWER SUBSYSTEM DESIGN/TECHNOLOGY REQUIREMENTS
  - Component Performance
  - Component Simulation Models (Transfer Functions)
  - System Design Requirements Based on Simulations
- TRADE CTV vs ATTITUDE CONTROL ON THE MPV
  - Type of Attitude Control
  - Location & Size of Attitude Control (Soft and Hard Dock)
- TOP CUT AT GROUND PROCESSING COSTS
- POTENTIAL FTS ACTIVITY DETAILS (Push, Pull, Twist, etc.)

NOTE:May Establish Synergistic Requirements with Other Systems (BENEFIT)

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TS920918.2

Ground Processing Cost Estimate

Studies performed and on-going in the areas of STV and HLV have generated data for facility sizing, task planning, ground support test and simulation equipment identification, and the associated projected costs. There are cost and task trade and sensitivity models at KSC and MSFC. These could be exercised to gain a feel for the cost bounds associated with processing a NEP vehicle.

The chart shows a sample of the kind of information that can be made available and could be worked in conjunction with a vehicle concept design task.

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TS-812.6-FP

Ground Processing Cost Estimate

TASK DESCRIPTION	LOCATION	DURATION (hrs)	MANPOWER (#)	COST-$
MCV				
Assemble Slider Radiator Sections	HVPF	8	5	xxxx
Install Reactor Assembly	?			
Install CTV Docking Port	HVPF			
Install FTS	HVPF			
Install Engine Pod	HVPF			
Assemble Cargo Modules				
Install CTV				
MPV				

OPTIONS

Standard Tasks:

Mating 2 items ———— 4hrs ———— mech, fluid, electr, sys, qual.

MARTIN MARIETTA

TS920812.6

[illegible]

Disposal Scenarios - Status and Location of Transfer Vehicle

Normal End of Life

- Piloted MTV: on Earth approach/flyby after ECCV separates
- Piloted or cargo MTV: in Earth orbit, after return and capture (option)
- Cargo MTV: in Mars orbit

After Propulsion System Failure

- In Earth orbit
  - during initial system start-up; limited fission product inventory on board
  - during spiral in/out operation, between designated Earth orbit and escape conditions
  - after return from Mars
- During trans-Mars cruise
- In Mars orbit
- During trans-Earth cruise

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Disposal Options - Where to Put It?

Two planetary orbit classes and two heliocentric orbit classes are considered for temporary storage and permanent disposal locations. Each has advantages for certain disposal scenarios, but each also has limitations. This study evaluates all four, and proposes a basic disposal strategy that considers safety, feasibility, and ease of operation.

Planning a solar system ejection or "crashing" into the Sun as a nominal disposal mode demands too much energy, and too much autonomous operations time to be practical. It is possible that the last use of an NEP module could be to power a robotic planetary explorer or a high-energy execliptic mission. However, this introduces further operational complexity and timing issues that are not relevant for preliminary propulsion technology planning.

Disposal Options - Where to put it?

- Earth orbit
  - Orbit lifetime is a function of altitude and the ballistic coefficient of the vehicle or system configuration
  - "Nuclear-safe" must be defined relative to the nature of the risk for each case; altitude of 700 km selected for this case based on lifetime and risk
- Mars Orbit - presumably no closer than Deimos
- Heliocentric transfer flight path
  - Leaves the reactor or vehicle in some interplanetary flight path
  - Most will cross both Earth and Mars, but still have very long life times
- Stable heliocentric orbit
  - Starts out at 1.19 x 1.19 AU - between Earth and Mars
  - Predicted not to be perturbed into a planet crossing path for a <u>very</u> long time; after that, same characteristics as previous case

Earth Orbit Lifetime Versus Orbit Altitude

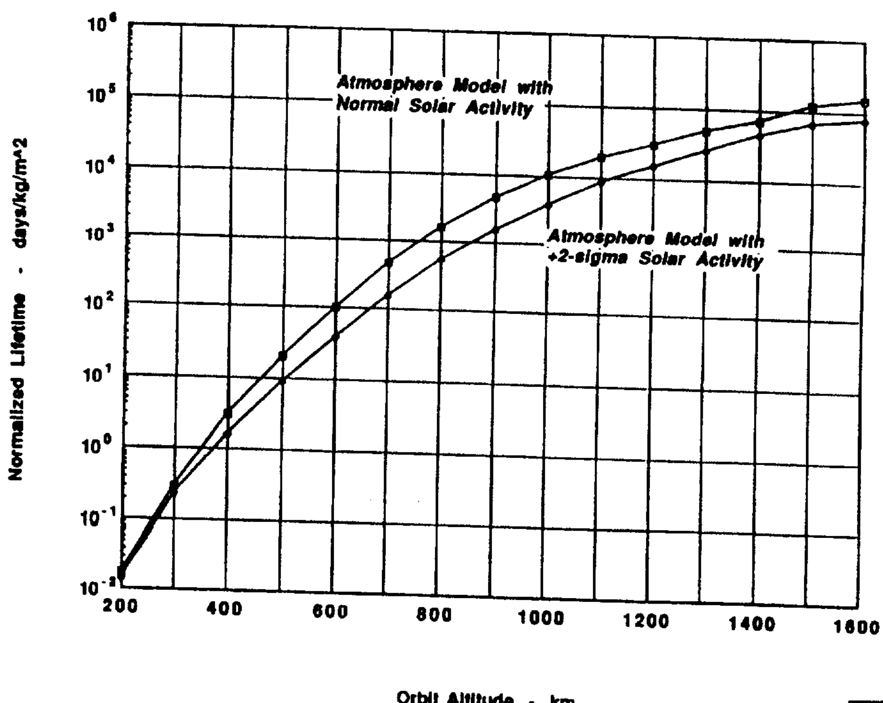
The first, and most critical disposal option is an Earth orbit. This option is included de facto for initial reactor startup and for any reuse scenarios, so the question is how to pick an orbit altitude that matches the risk factors and that is within Earth-to-orbit capability.

Analysis by Martin Marietta in another section of this report indicates that a 700 km altitude is well within the reach of anticipated heavy lift launch vehicles for SEI. In fact, ETO capability degrades only slightly from 400 km to 700 km. Maximum orbit lifetime favors a higher altitude, as the graph opposite will show.

Orbit lifetime is plotted versus orbit altitude for circular orbits from 200 km up to 1600 km. The lifetime is normalized with respect to the ballistic coefficient of the vehicle in orbit. The two curves represent different atmospheric density models: the upper curve assumes normal levels of solar activity, while the lower curve factors in most of the observed high solar activity periods. Both curves will be used to estimate a lifetime range, with the normal activity showing a longer lifetime, and the high activity showing a more conservative shorter lifetime.

To use the curves, the mass and physical dimensions of the orbiting vehicle must be known, and a drag coefficient must be supplied. The table on the next page shows calculated lifetime ranges for some cases of interest for the NEP vehicle.

Earth Orbit Lifetime vs. Altitude



Selected Orbit Lifetimes

Four possible disposal configurations have been evaluated, from a fully loaded MTV to a single propulsion module. Masses for each are shown, as is the area presented if we assume that the largest possible plane area is perpendicular to the direction of motion. Areas are approximate, and the assumption that the largest area will always be presented to produce drag will produce conservative results. Drag coefficients shown are for rough shape equivalents; a complete calculation for this situation is beyond the scope of this study. These quantities are used to calculate a ballistic coefficient for each disposal configuration, which is then multiplied by the normalized lifetime (read off the preceding graph), and converted to years.

The results in the table opposite show the value of higher altitudes for extended life in orbit without reboost procedures. Based on this preliminary analysis, we select a 700 km circular Earth orbit for all operations. This location is also suitable for temporary storage, but probably not for permanent disposal of a spent nuclear reactor.

Selected Orbit Lifetimes

Area based on longest 2 dimensions

Disposal Configuration	Mass kg	C_D	Area m^2	β kg/m^2	Predicted Orbit Lifetime (Yrs) for the Specified Altitude		
					400 km	700 km	1000 km
Mars Transfer Vehicle Fully Loaded	325,000	2	1,525	107	0.5 - 0.9	40 - 140	1110 - 2950
Mars Transfer Vehicle w/o Payload, Propellant	90,000	2	1,425	32	0.1 - 0.3	10 - 40	350 - 880
1 5 MWe Module	36,285	2	710	26	0.1 - 0.2	10 - 30	280 - 720
1 Reactor only	3,500	1.3	10	269	1.2 - 2.2	110 - 350	2800 - 7400

Notes: 1. Estimated area assumes largest plane area is perpendicular to the velocity vector

2. Drag coefficients are only rough approximations by shape

3. Lifetime range determined by using both atmospheric density models

Disposal On an Interplanetary Flight Path

Another disposal possibility, especially suited to a transfer vehicle already in interplanetary flight, is to simply leave the vehicle in some interplanetary flight path. The path selected might be the current one, or it might be specifically designed to minimize the possibility of a future reencounter. This option could also be used for a vehicle in planetary orbit, by accelerating it to escape conditions. This strategy is the NEP equivalent of "jettisoning" a spent propulsion stage after use: leave it where it is, and accept the small possibility of a reencounter.

Because interplanetary transfers cross one or more planet orbits, they set up the possibility of either a direct collision or, more likely, a close encounter (within a few planet radii) that creates a gravity-turn and so perturbs the vehicle's original path. The more close encounters, the greater the perturbations, and the greater the possibility of terminating the vehicle's orbit. Termination may be in the form of a collision with a planet, impacting the Sun, or ejection from the solar system. While not all of these are bad, the process is uncontrolled without further human intervention.

Lifetimes of bodies in planet-crossing paths may be estimated with a Monte Carlo simulation technique, such as SAIC's Planetary Encounter Probability Analysis (PEPA) code. This analysis suggests that, with few exceptions, leaving an NEP vehicle in a typical interplanetary orbit produces a risk no greater than the natural risk of collision with one of the Earth-approaching asteroids.

Disposal on an Interplanetary Flight Path

- Typical Earth-Mars low thrust trajectories (outbound or inbound):
  - lie slightly out of the ecliptic plane
  - graze the orbits of Earth and Mars
- If the MTV is left in a typical flight path, Monte Carlo simulation using SAIC's PEPA Code predicts:
  - Mean orbit lifetimes of 10^7 - 10^8 years
  - Chance of collision with Earth in 10^6 years is low in all cases - nearly zero in most
- So, the risk of a nuclear-powered Mars Transfer Vehicle colliding with Earth is of approximately the same order as the risk of colliding with a near-Earth asteroid

Predicted Orbit Lifetimes for Typical Low Thrust Trajectories

The table opposite summarizes the results of several simulation runs, using various points along typical low-thrust trajectories between Earth and Mars, and to a particular heliocentric disposal orbit to be described later. The low-thrust path must be sampled at several points, since the orbital parameters are subject to continuous change during periods of thrusting. Three samples were selected for the Earth-Mars and Mars-Earth transfers, corresponding to post-escape, transfer time midpoint, and target approach just prior to initiating spiral capture.

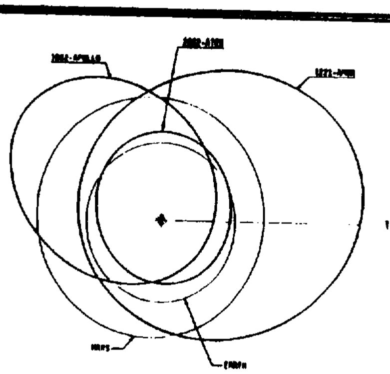
Each row shows a different simulation case: the calculated orbit parameters of interest, namely perihelion, aphelion, and inclination; the mean simulated orbit lifetime in years before termination; the number of trials out of 500 that the simulation resulted in an Earth collision; the mean time to Earth collision for that subset of cases; the probability of an Earth collision in the first one million years after start of simulation. All the times are reassuringly long, and most of the collision probabilities for the first million years are low. The exceptions are those cases just after Earth escape, when the NEP orbit is very close to Earth's orbit.

The following page shows the same statistics for simulation trials with several near-Earth asteroids. The slightly longer expected lifetimes are the result of more highly inclined orbits for the asteroids than for the transfer vehicles. However, the overall risk appears to be of the same magnitude for both groups. We conclude that leaving the NEP vehicle in some unspecified transfer orbit may incur a reasonable risk.

Predicted Orbit Lifetimes for Typical Low Thrust Trajectories

Trajectory Leg		Orbit Size $R_P \times R_A$ (A.U.)	Incl. (deg)	Mean Orbit Lifetime (Years)	Expected Earth Hits in 500 Trials	Mean Time to Hit (Years)	Earth Hit Chance in 10^6 Years
Earth-Mars	Start	0.98 x 1.25	0.0	5.6×10^7	266/500	1.6×10^7	16 %
	Middle	0.85 x 1.64	1.2	4.7×10^7	200	4.4×10^7	3 %
	End	0.61 x 1.51	1.8	4.0×10^7	160	3.1×10^7	2 %
Mars-Earth	Start	0.48 x 1.40	3.0	4.2×10^7	146	3.6×10^7	2.6 %
	Middle	0.50 x 1.89	1.3	4.2×10^7	123	3.3×10^7	1 %
	End	0.51 x 1.02	1.3	9.2×10^7	194	2.2×10^7	5.2 %
Earth-Disposal	Start	0.98 x 1.02	0.1	3.9×10^7	270	1.7×10^7	18 %
	Middle	0.99 x 1.02	0.0	3.9×10^7	266	2.1×10^7	17 %
Mars-Disposal	Start	1.28 x 1.66	2.1	7.5×10^8	148	4.4×10^8	0 %
	Middle	1.22 x 1.61	2.0	6.0×10^8	166	3.5×10^8	0.2 %

Predicted Orbit Lifetimes for Selected Near-Earth Asteroids



Body	Mean Orbit Lifetime (Years)	Expected Number of Earth Collisions (In 500 Trials)	Mean Time to Earth Collision (Years)	Chance of Earth Collision in 10^8 Years
2062 - Aten	5.27×10^7	177/500	4.46×10^7	1.6 %
1862 - Apollo	7.73×10^7	111	2.75×10^7	0.6 %
1221 - Amor	9.88×10^8	128	7.16×10^8	0
1943 - Anteros	7.48×10^8	203	1.98×10^8	0
1982DB	7.68×10^7	264	2.95×10^7	4.4 %
1989ML	3.87×10^8	194	1.95×10^8	0
1980AA	3.89×10^8	200	1.99×10^8	0
1982XB	6.25×10^7	267	3.44×10^7	5.2 %

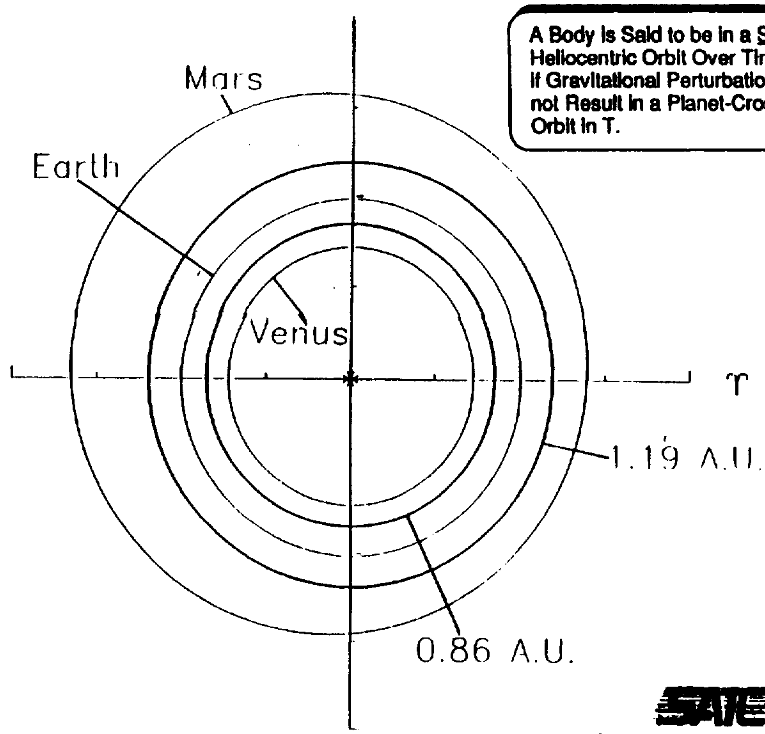
Stable Heliocentric Circular Orbits

The second category of interplanetary orbits was identified by SAIC as a possible permanent storage location for hazardous waste in space.[1] This analysis was one part of a large effort to explore space-based alternatives for nuclear waste disposal conducted during 1977-79. These orbits are of interest because they are predicted to endure for a very long time without becoming planet-crossing orbits. Two bands of these stable orbits have been identified, as shown opposite. The one of most interest for Earth-Mars cases is a circular orbit at 1.19 A.U., between Earth and Mars. The orbit starts out circular, but becomes elliptic "quickly" in the long view of the situation, as shown on the next page.

[1] Friedlander, A. L. and D. R. Davis, "Long-Term Risk Analysis Associated With Nuclear Waste Disposal in Space," SAIC Report No. 1-120-062-T12, prepared under contract NAS8-33022 for NASA/MSFC, December 1978.

STABLE HELIOCENTRIC CIRCULAR ORBITS

A Body is Said to be in a Stable Heliocentric Orbit Over Time T if Gravitational Perturbations do not Result in a Planet-Crossing Orbit in T.



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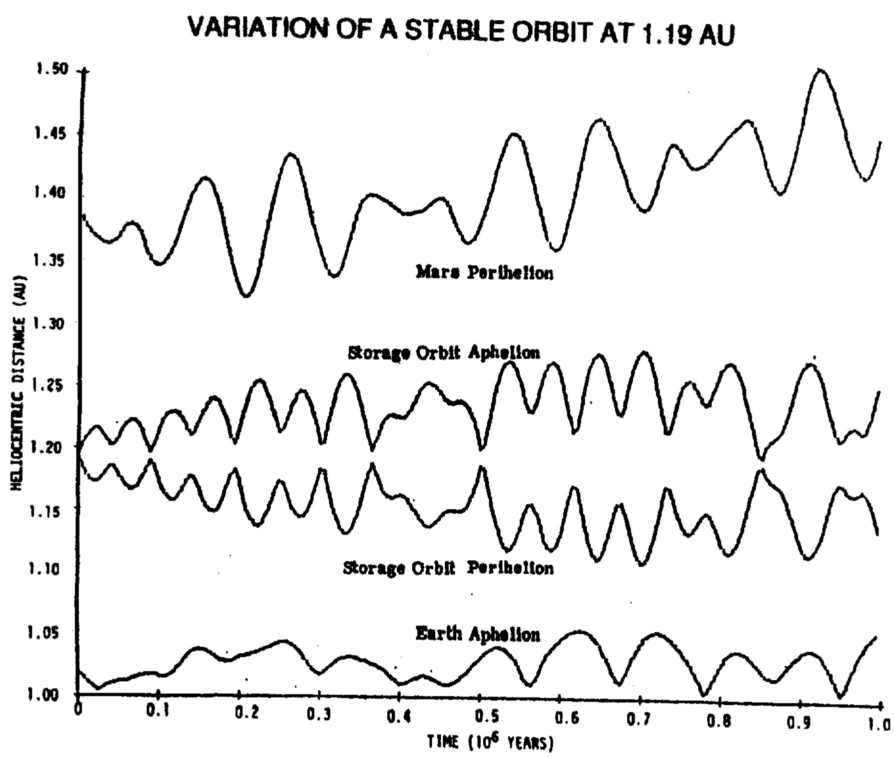
Variation of a Stable Orbit at 1.19 A.U.

This chart plots heliocentric distance as a function of time (note the x-axis scale!) for the periapse and apoapse of the stable orbit. The Mars peripase and Earth's apoapse are also plotted. All four show significant variations over the one million year time frame, but the stable orbit never crosses its closest planetary neighbors' paths. This means that, with no further active management, placing an object in the stable orbit is sufficient to remove the real risk of the on-board radiation hazard.



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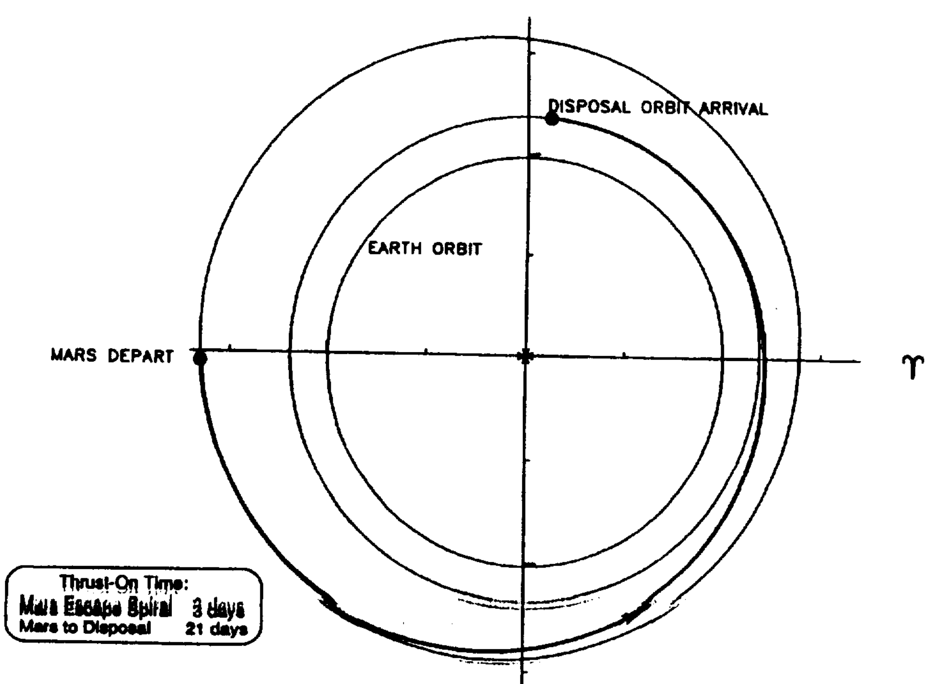
Typical NEP Transfer From Mars to Disposal Orbit

Here is a typical transfer to the stable orbit just described. We have selected a very long flight time to minimize propellant needs and additional thrust-on time. If a transfer vehicle were to leave Mars orbit for the stable disposal orbit, propellant and tankage needs would be a few tonnes, and thrust time would be about 24 days. Faster disposal legs can be traded for increased propellant.

Transfer to NEP Reactor Disposal Orbit (420 days)



Summary of Proposed Disposal Modes

This table summarizes preliminary evaluation of each of the four disposal locations for the cases examined. The comments indicate proposed use as temporary or long-term storage sites, with the preferred long-term selection for each case highlighted by a shaded box.

Earth orbit is recommended as a temporary storage location only, even though boosting the NEP vehicle or some part of it to higher altitude significantly mitigates the real risk. Since perceived risk is not so easily removed, a more distant storage location would be preferable for the baseline. For all cases of normal end of life, we propose that the stable heliocentric orbit be the baseline disposal location. This site could also be used for any partially disabled vehicle that can be moved to the stable orbit. However, recognizing the inherently low risk involved in leaving the vehicle in a transfer flight path, the proposed baseline for total system failures is the interplanetary flight path. Even a modest alternate propulsion system on board could maneuver to a higher inclination, or otherwise reshape the orbit of the derelict vehicle to make reencounter less likely.

Summary of Proposed Disposal Modes

Temp = temporary storage (1-5 years) until permanent disposal is arranged
Long = long-term disposal; "permanent" solution to the potential nuclear risk

NEP Status at Disposal			NEP Reactor Disposal Location: Earth Orbit	Mars Orbit	Interplanetary Flight Path	Heliocentric Stable Orbit
	Normal End of Life	Earth Approach	No	–	Temp - ok Long - ?	
		Earth Orbit	Temp Only	–	Temp - ok Long - ?	
		Mars Orbit	–	Temp - ok Long - ?	Temp - ok Long - ?	
	Propulsion System Failure	Earth Orbit	Temp Only	–		Long Option
		Earth-Mars Cruise	–	–		Long Option?
		Mars Orbit	–		Long - ?	–
		Mars-Earth Cruise	–	–		Long Option?

■ Proposed Baseline Disposal Mode

Disposal Mode Impact on Vehicle Performance

This chart is the companion to the previous one, showing the cost in propellant and thrust time to achieve some of the disposal locations of interest. In every case, the impact is very modest. The largest requirement shown opposite is for an Earth escape spiral to remove a fully operational NEP vehicle from Earth orbit. If the system has failed in Earth orbit and is to be moved, the cost will depend on the nature of the failure - full or partial - and selection of any additional propulsion that may be needed. Note that transfer to the stable orbit from Earth orbit calls for a thrust interval of about 10% of the expected thruster lifetime, so there may be some additional cost in thruster changeout.

Disposal Mode Impact on Mission and Vehicle Performance

NEP Status at Disposal			NEP Reactor Disposal Location: Earth Orbit	Mars Orbit	Interplanetary Flight Path	Heliocentric Stable Orbit
NEP Status at Disposal	Normal End of Life	On Earth Approach	--	--	None	M_{PROP} = 7.1 t ΔTh = 13 days
		In Earth Orbit	Small ΔV to raise orbit*	--	M_{PROP} = 18 t ΔTh = 36 days	M_{PROP} = 27 t ΔTh = 49 days
		In Mars Orbit	--	None	M_{PROP} = 2 t (1% of IMLEO) ΔTh ~ 1.4 days	M_{PROP} = [illegible] t (6% of IMLEO) ΔTh ~ 24 days
	Propulsion System Failure	In Earth Orbit	Small ΔV to raise orbit*	--	Depends on Implementation	
		Earth-Mars Cruise	--	--	[illegible]	
		In Mars Orbit	--	None		
		Mars-Earth Cruise	--	--	None	

M_{PROP} = propellant & tank mass penalty for disposal

ΔTh = Incremental NEP thrust-on time for disposal

* ~ 150 m/s to transfer from 700 x 700 km to 1,000 x 1,000 km

Recommended Approach for Disposal

The next two charts summarize the recommended approach to managed disposal of NEP reactors or transfer vehicles. These are to be viewed as a preliminary recommendation for further evaluation, concurrent with more detailed understanding of operational and performance impacts.

The stable heliocentric orbit is generally easy to reach, and is the most conservative risk management approach evaluated. Selecting this disposal mode for nominal end-of-life seems to greatly reduce both real and perceived risk for very little additional cost.

If a transfer vehicle should become completely disabled, its interplanetary path is almost certainly acceptable as a temporary storage location. It may also be adequate for long-term storage, especially if on-board auxiliary propulsion can be used to control the path.

Earth orbit need not be used for long-term disposal, thus avoiding additional controversy over use of nuclear energy in space. The operational orbit selected appears to support temporary [illegible]ge readily. However, the NEP module design should incorporate sufficient auxiliary propulsion to handle orbit raising burns over a limited number of years. This could be further supplemented by a design that could separate a disabled reactor from the rest of the vehicle to increase the lifetime of the most critical subsystem, and to reduce propellant required to boost just the reactor to a higher orbit.

As a final precaution, some independent orbital transfer vehicle, possibly the Lunar Transfer Vehicle, could be available to push a derelict NEP to escape conditions, or to a stable orbit.

Recommended Approach for Disposal - 1

Location:

- Pick the <u>**stable heliocentric orbit for nominal missions**</u>
  - Modest propellant requirements for all cases examined
  - Conservative approach to risk management avoids programmatic problems
- Use interplanetary path disposal for a completely disabled vehicle
  - Every case we considered shows a predicted orbit lifetime of 10^7 years or better
  - Reencounter probability for most cases is of the same order as near-Earth asteroids
  - No ΔV required
- <u>**Earth orbit for temporary storage only**</u>; not for long-term disposal
  - 700 km altitude seems a reasonable compromise among: launch capability, predicted lifetime for typical configurations, and on-going operations
  - Include independent propulsive capability to raise orbit of MTV
  - Avoid most controversial location for long-term storage

Recommended Approach for Disposal - 2

Transfer Vehicle Design:

- **Include auxiliary propulsion system** in baseline 5 MWe module design
  - Sufficient to raise Earth orbit from 700 km to 1000 km (ΔV - 150 m/s)
  - System design and propellant required depends on how much of the module is boosted to the higher orbit

- Consider adding capability to **separate a disabled reactor** from the rest of the module; auxiliary propulsion remains with the reactor

Transportation Infrastructure

- Assured removal from Earth orbit may require a separately deployed orbital transfer vehicle - possibly an LTV or similar element

N93-26976

Nuclear Electric Propulsion Operational Reliability and Crew Safety Study

NEP Systems / Modeling Report

22 October, 1992

Presented By:

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Presented At:

1992 Nuclear Propulsion - Technical Interchange Meeting
NASA Lewis Research Center
Sandusky, OH

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This work was accomplished under contract NAS3-25809, Mod 22 for the NASA Lewis Research Center Nuclear Propulsion Office, and under the technical direction of Michael Doherty.

The project manager for this contract was Michael Stancati. The technical work effort was led by Joseph R. Fragola, Vice President and Manager, Advanced Technology Division. James J. Karns led the reliability analysis task and overall systems engineering effort. Dennis Pelaccio was responsible for nuclear and propulsion systems engineering, Lloyd Kahan for reliability modeling, Peter Appignani and Richard McFadden for identifying and developing surrogate reliability data bases, and Darrel Walton for administrative support.

We would like to thank Mike Doherty and Jim Gilland of the Nuclear Propulsion Office for their invaluable expertise and assistance in performing this task.

STUDY OBJECTIVES:

- Determine the range of reliability figures of merit required for a successful NEP manned Mars mission.
- Provide design insights:
  - design achievevability, given existing technology;
  - alternative design approaches or concepts to enhance reliability, crew safety;
  - allocation of research and development resources.

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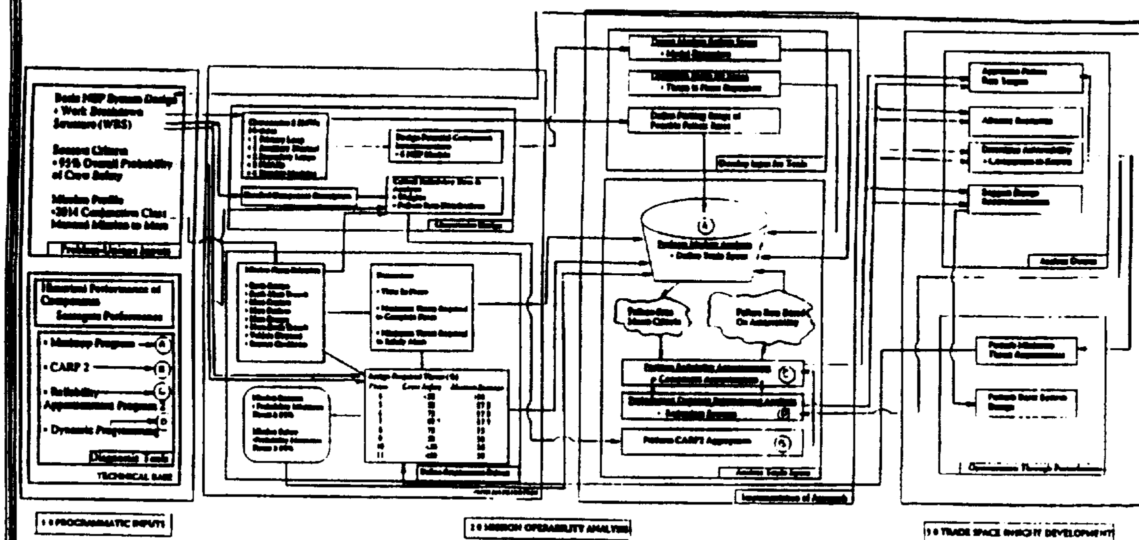
The objective of this study was to establish the initial quantitative reliability bounds for nuclear electric propulsion systems in a manned Mars mission required to ensure crew safety and mission success. Finding the reliability bounds involves balancing top-down (mission driven) requirements and bottom-up (technology driven) capabilities. In seeking this balance we hope to: (1) provide design insights into the achievability of the baseline design in terms of reliability requirements, given the existing technology base; (2) suggest alternative design approaches which might enhance reliability and crew safety; and (3) indicate what technology areas require significant research and development to achieve the reliability objectives.

STUDY OVERVIEW



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This study was broken down into three broad areas: the processing of programmatic inputs; performing the mission operability analysis; and analyzing the trade space for design insights. The processing of **programmatic inputs** began with identifying, soliciting, obtaining, and processing the required *program unique inputs*. These included the basic NEP system design, the top-level mission and crew safety success criteria, and the mission profile. Next, the existing *technology base* was examined to identify and obtain data on the historical performance of NEP and NEP-related (surrogate) components, and to determine the set of diagnostic tools appropriate to this analysis.

The **mission operability analysis** consisted of problem definition and implementation of the selected analysis approach. *Problem definition* included characterizing the design in terms appropriate to the selected diagnostic tools, and defining the reliability requirement drivers in the NEP system for the selected mission. *Implementation of the approach* consisted of developing the input for the various diagnostic tools, and analyzing the reliability trade space developed by the tools. The process of **trade space insight development** included *analyzing the trade space output* and seeking design insights by looking for improvements in system reliability when the basic design is altered, or *optimization through perturbations.*

CONCEPT OF ACHIEVABILITY

- Achievability: The ratio of required performance to achieve performance.
  - Measures how far a design has to go.
    - Achievability Index = 1: Design is achieved.
    - Achievability Index = 0: Design cannot be achieved with existing technology.
- Incorporates uncertainties in:
  - Particulars of design,
  - Relevance of historical performance.
- Should therefore be presented as a range of values.

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A core concept in this analysis is the idea of achievability -- how well the existing technology base will support the NEP mission and design as given. Achievability is formally the ratio of the required performance to the readily achieved performance, given the state of the technology base. Since there are uncertainties in both the particulars of the design, and in the relevance of historical performance to NEP - Manned Mars Mission performance; and since there is significant variability in the measured performance of historical (surrogate) elements, the achievability should be presented as a range of values.

Due to time and funding limitations on this study, a rigorous development of the distribution of achievability values is not presented. Instead, point values of the limits on achievability are found.

ACHIEVABILITY DEFINITION

$$\phi\,(AchIcomponent) = \frac{\phi\,(\lambda pportioned\ Component)}{\phi\,(\lambda Surrogate\ Component)}$$

$$\Phi\,(AchISystem) = Aggregate\ (\phi\,(AchIComponent))\ \Big|_{All\ Components}$$

$\phi\,(AchIComponent)$ Distribution of achievability index (AchI) for a component.

$\Phi\,(AchISystem)$ Distribution of AchI for a system.

$\phi\,(Apportioned\ Component)$ Distribution of apportioned failure rates required for component.

$\phi\,(Surrogate\ Component)$ Distribution of likely failure rates for component based on surrogate performance.

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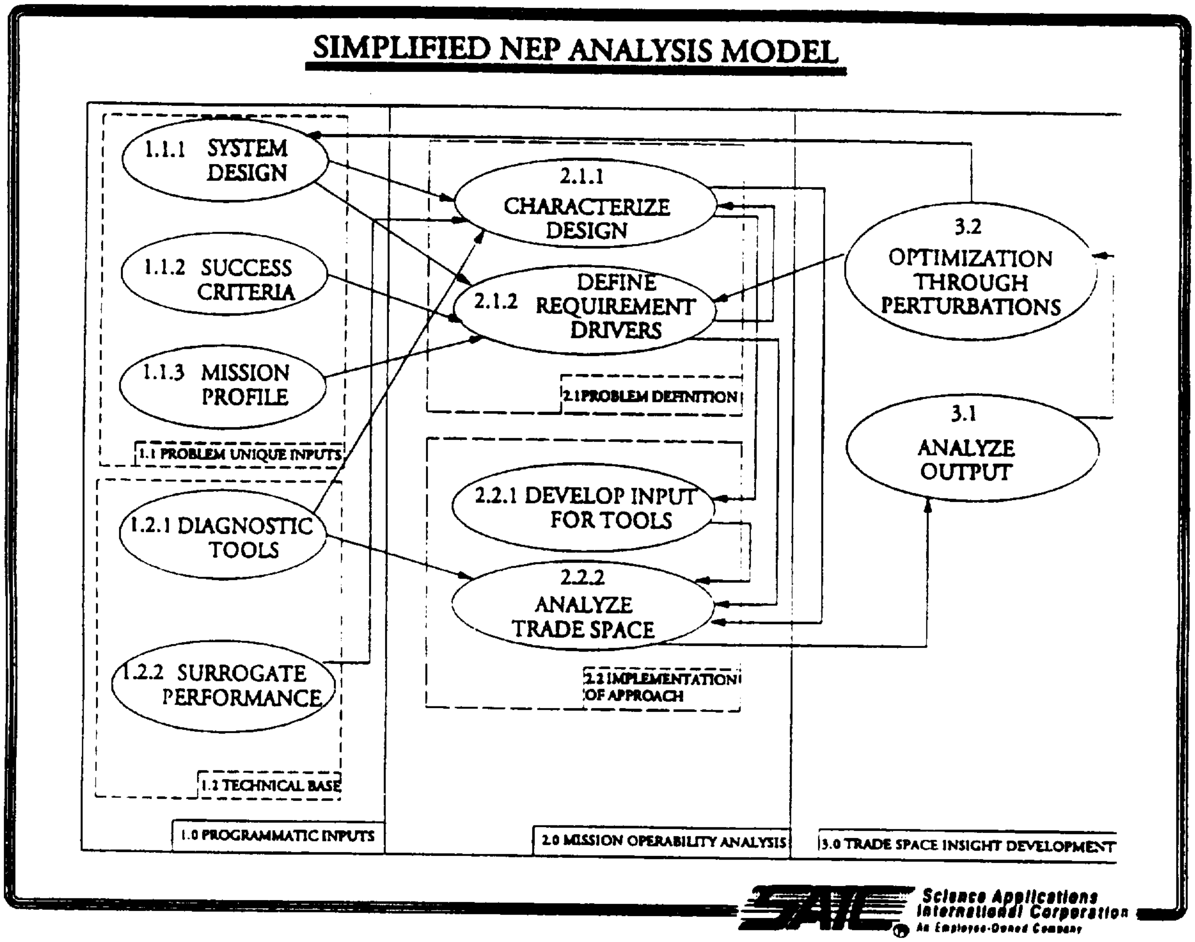
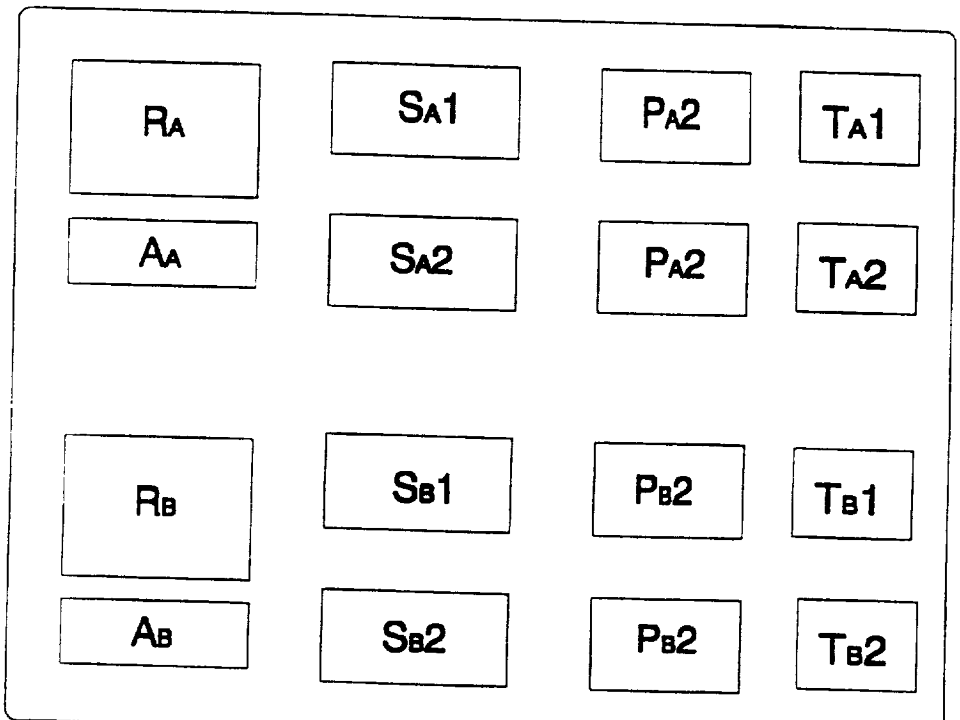
Achievability is measured in terms of an achievability index (*AchI*), which is measured in terms of the measurable figure of merit for this study, random failure rate (λ). The distribution of *AchI* for a component is the ratio of the distribution of failure rates apportioned to the component based on design and mission requirement parameters, and the distribution of failure rates associated with surrogates of the component from the technology base. The distribution of *AchI* for the entire NEP system is the aggregate of component *AchI* distributions.



The analysis process began with characterizing the system design at a high level in terms appropriate to the analysis tools.

BASIC NEP SYSTEM MODEL -- AS GIVEN



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We were provided a simple model of the NEP system, consisting of two essentially independent modules. Each module consisted of a Primary Heat Source Loop (R), an Auxiliary Thermal Subsystem (A) two Secondary Loops (S), two Power Management and Distribution Assemblies (P), and two Thruster Assemblies (T).

This basic top level design representation was extended and altered somewhat to provide various design concept bases for analysis.

NEP SYSTEM MODEL

- Two 5MWe NEP Modules:
  - Each 5MWe NEP module:
    - 1 Primary heat source subsystem (R)
    - 1 Auxiliary thermal management system (A)
    - 2 Secondary subsystems (S)
    - 2 Power Management And Distribution (PMAD) subsystems (P)
    - 4 half-Thruster module subsystems (T)
      - The "given" thruster modules were split, as analysis indicated two halves essentially independent.

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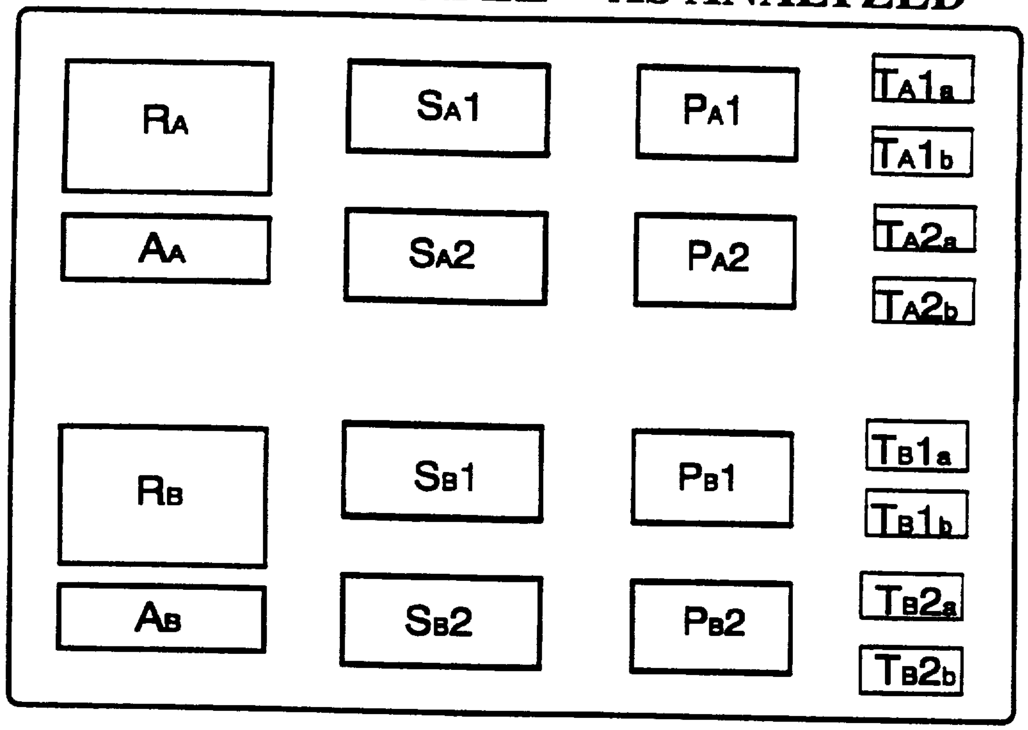
No comment required.

NEP SYSTEM MODEL -- AS ANALYZED



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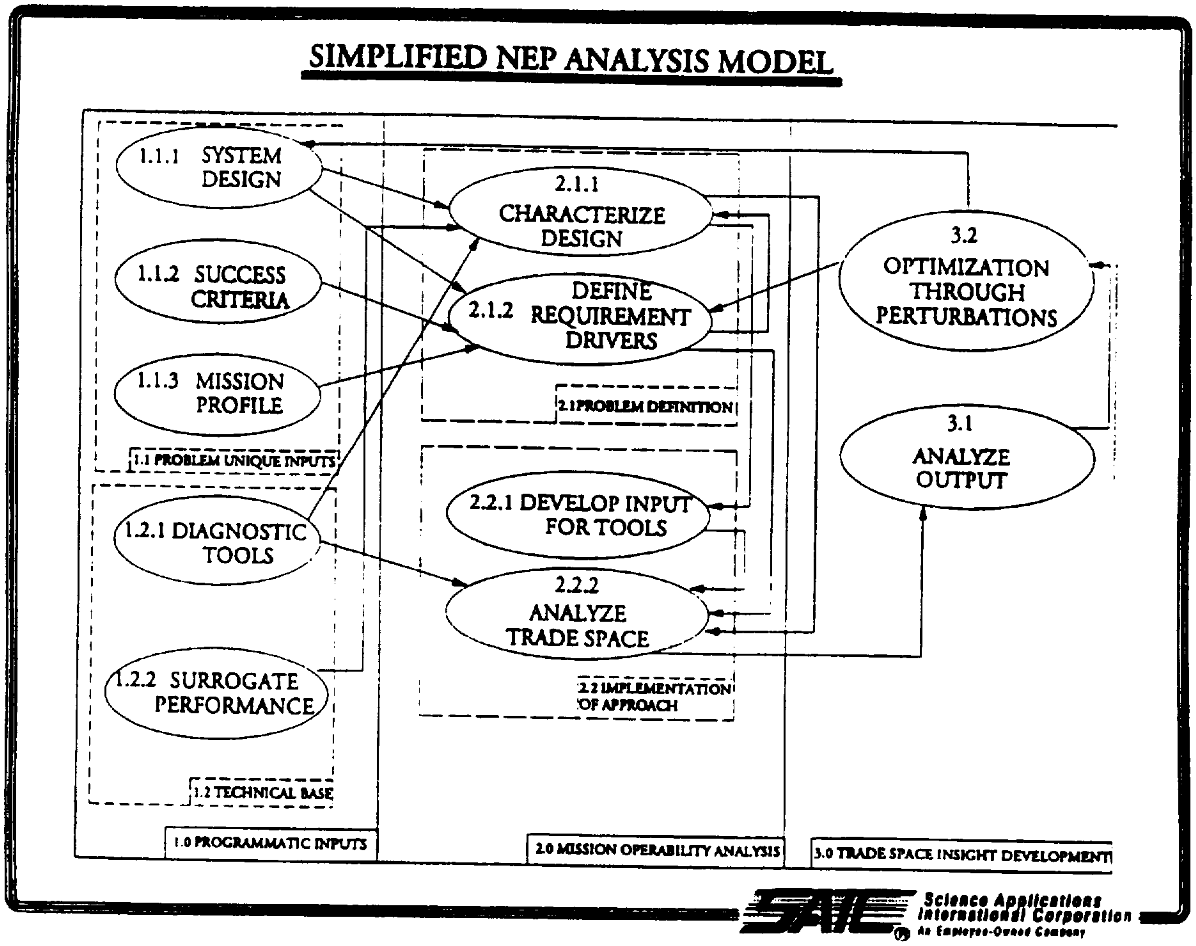
It was noted that each Thruster assembly had two essentially independent halves, so the model was modified slightly to make this apparent.



The next step in the analysis process was to identify and characterize the measurable success criteria for the mission.

NEP MANNED MARS MISSION SUCCESS CRITERIA

- 99% Probability of Crew Safety.
  - Aborts possible,
  - System need not reach Mars, but
  - Must return to Earth in or before nominal mission time frame.
- 95% Probability of Mission Success.
- Criteria applied to NEP System Only!
  - Overall mission probabilities must account for all other systems:
    - Life Support,
    - GNC, EPS (distribution), Thermal, TT&C, C&DH, etc.,
    - Ascent / Descent modules,
    - Earth Crew Capture Vehicle.

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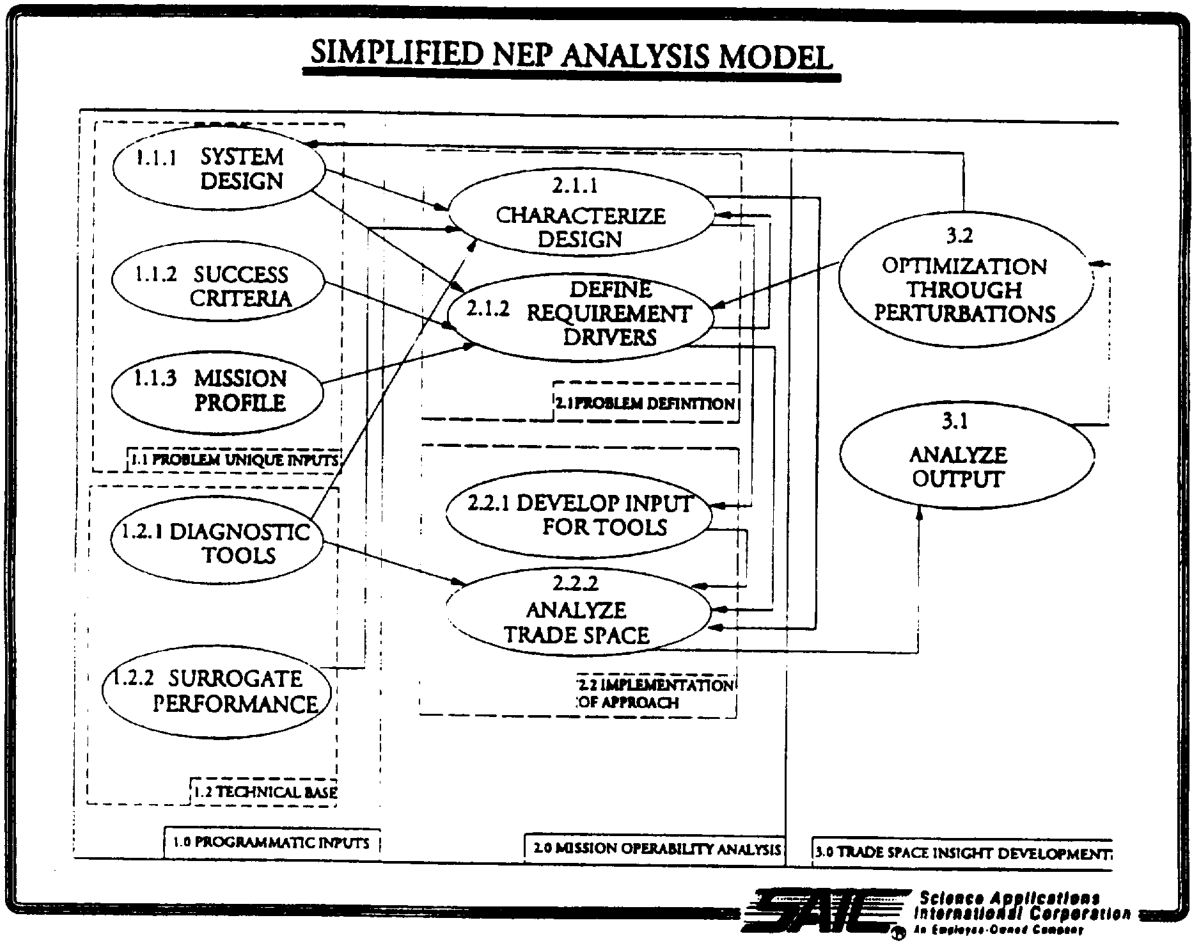
At a top level, the success criteria was given as 99% probability of crew safety, and 95% probability of mission success. It should be noted that this criteria was interpreted to apply only to the NEP system, not to other, equally vital, systems.

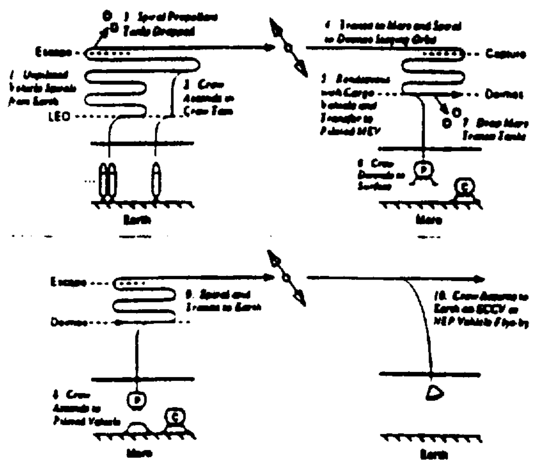


The last aspect of the Problem Unique Inputs portion of the analysis problem was to identify and define the Mission Profile.

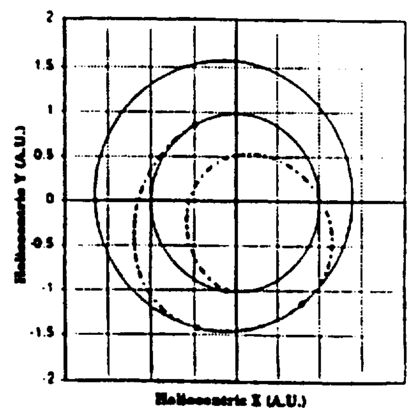
C-5

BASELINE MISSION CHARACTERISTICS

Mission Profile

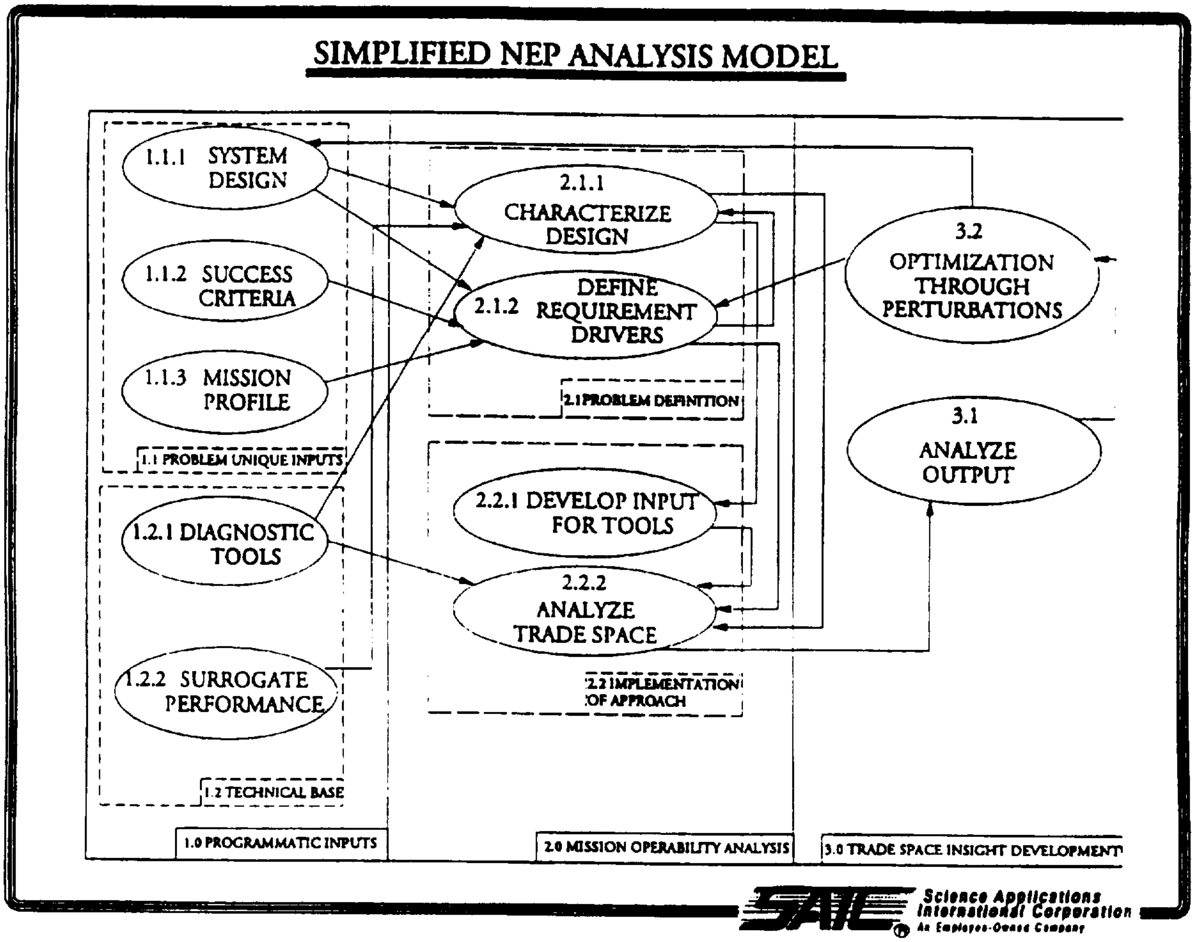


Orbit Plot



- Imelco - 350 MT
- Minimum Heliocentric Distance- 0.50 - Air

The mission analyzed was a 2014 conjunction class Manned Mars Mission.



After obtaining and characterizing the Program Unique Inputs, the technology base was then examined to determine the diagnostic tools appropriate to the analysis problem.

DIAGNOSTIC TOOLS

- Markapp(TM) -- Dynamic Markov Chain analysis program.
  - Determine top-level reliability figure(s) of merit (FOM).
- RAP2(TM) -- Reliablity Approtionment Program.
  - Apportion top-level FOM to component level.
- Dynapro(TM) -- Dynamic Integer Programming
  - Non-linear "optimization" of redundancy complement.
- CARP(TM) -- Computerized Aggregation of Reliability Parameters.
  - Combine historical reliability performance data from multiple sources.

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The analytical tools selected were Markapp™, RAP2™, Dynapro™, and CARP™.

Markapp™ is a dynamic Markov-Chain analysis program. This tool allows the system to be modeled as a set of discrete states, based on the number and types of components that will fail. The probability of the system being in each of the states at any time in the mission can be calculated based on the failure rates associated with the components. This tool is used to determine what set(s) of top-level failure rates will result in achieving the mission success criteria.

RAP2™ apportions top-level reliability goals to lower-level components based on a variety of apportionment strategies. Dynapro™ is a Dynamic Integer Programming tool used in conjunction with RAP2™ to determine optimum allocations of, and limits on, spare allocation.

CARP™ -- Computerized Aggregation of Reliability Parameters is used to combine or aggregate distributions of failure rates from components similar to NEP components to define an appropriate surrogate distribution for each of the NEP components.

MARKAPP(TM) MARKOV CHAIN ANALYSIS

- The Markov chain is a discrete state - continuous time analytical model.
  - Used to determine sets of functional element failure rates that meet success criteria.
- A state is a unique configuration of NEP functional elements
  - 2 Pri, 2 AuxTherm, 4 Sec, 4 PMAD, 8 Thruster
- Transition between states i and j occurs at transition rate λ_{ij}.
- Markapp(TM) calculates probability that the system is in each state -- a function of:
  - Previous state of the system,
  - Failure rates of functional elements,
  - Time in mission.

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The Markov model is comprised of a description of the NEP system in terms of its functional elements, a list of operational states of the system in terms of whether each of the components is operational or failed, and the rate at which the system transitions from one state to another. The transition rates are expressed in terms of the failure rates of the functional elements of the system.

Markapp™ solves the Markov model for the probabilities that the system is in each defined operational state as a function of time in the mission. These probabilities can be combined with the knowledge of which states meet the mission success criteria at each phase of the mission to determine the probability of the system meeting the success criteria. That information, in turn, indicates whether the input (trial) failure rates for the functional components will meet the mission objectives.

THE MARKOV PROCESS

$$\mathbf{x}\,(t + \Delta t) = \Delta t \begin{bmatrix} \lambda_{11} & \lambda_{12} & \cdots & \lambda_{1N} \\ \lambda_{21} & \lambda_{22} & \cdots & \lambda_{2N} \\ \vdots & \vdots & \vdots & \vdots \\ \lambda_{N1} & \lambda_{N2} & \cdots & \lambda_{NN} \end{bmatrix} \mathbf{x}(t)$$

$\mathbf{x}(t) = \left[x_i(t)\right] =$ Vector of probabilities that system is in state i.

$$\lambda_{ij} = a_{ij}\lambda_{Primary} + b_{ij}\lambda_{AuxTherm} = c_{ij}\lambda_{Secondary} + d_{ij}\lambda_{PMAD} + f_{ij}\lambda_{Thruster}$$

$\lambda_{Primary}$, $\lambda_{AuxTherm}$, $\lambda_{Secondary}$, λ_{PMAD}, $\lambda_{Thruster}$: Failure rates of functional elements.

N, a_{ij}, b_{ij}, c_{ij}, d_{ij}, f_{ij},: Parameters determined by the system design.

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These equations describe the mathematics of the Markov Process.

RAP2(TM) RELIABILITY APPORTIONMENT

- RAP2(TM) apportions reliability from top-level to component level.
- Simplified apportionment equation:
  - $R_i\text{Apportioned} = R_{Goal}^{\frac{W_i}{\Sigma W}}$
- 3 apportionment methods:
  - Simple -- based on history of like components:
    - $W_i\text{Simple} = R_i\text{Surrogate} = e^{-\lambda_{i_{Surrogate}}}$
  - AGREE -- based on part count (complexity) and criticality:
    - $W_i\text{Agree} = \#\text{Parts}_i * \text{Criticality}_i$
  - Weighted Nth-Root -- based on physical characteristics of component:
    - $W_i\text{NthRoot} = a_1 W_{iComplexity} + a_2 W_{iStateofArt} + a_3 W_{iType} + a_4 W_{iQuality}$

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The RAP2™ Reliability Apportionment Program is used to apportion the top-level (functional-level) failure rates arrived at using the Markov analysis to the lower level components of the NEP system. The program uses three algorithms, each of which provide unique insight into the apportionment problem. The Simple apportionment algorithm is based strictly on the historical performance of like components, and indicates most directly how much the system reliability requirements will push the technology base. The AGREE algorithm is based on subjective assessment of the component relative importance, and on the component complexity. AGREE therefore provides a simple and much less rigorous way of apportioning based on mission requirements (criticality) than the Markov model. The weighted Nth Root method apportions reliability based on subjective evaluation of the relative difficulty in achieving high reliability for the components. Comparing relative differences between the Simple and Weighted Nth Root algorithms provides a first approximation of what is available versus what the analyst believes ought to be available.

CARP(TM) SURROGATE AGGREGATION

- Identify likely failure rate range of new component based on aggregation of similar components:
  - Similar in function;
  - Similar in application;
  - Similar in stress environment.
- Failure rate distribution incorporates:
  - Inter- and Intra-source Variability;
  - Uncertainty in similarity of function, application, or environment.
- Surrogate data sources:
  - NPRD-91, DSR-4, IEEE 500, CREDO, various NUREGs.
- No similar historical surrogate => establish range by "reality boundary".

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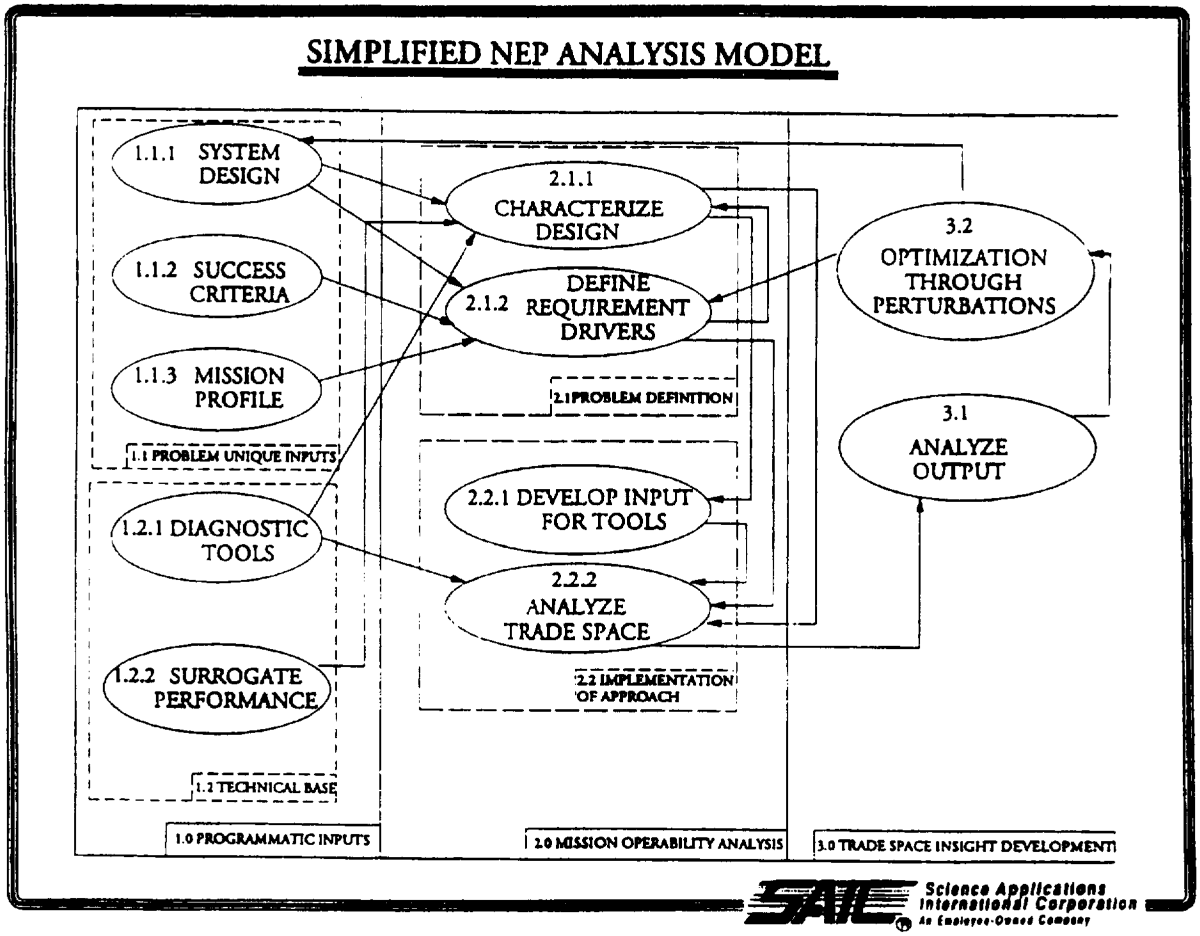
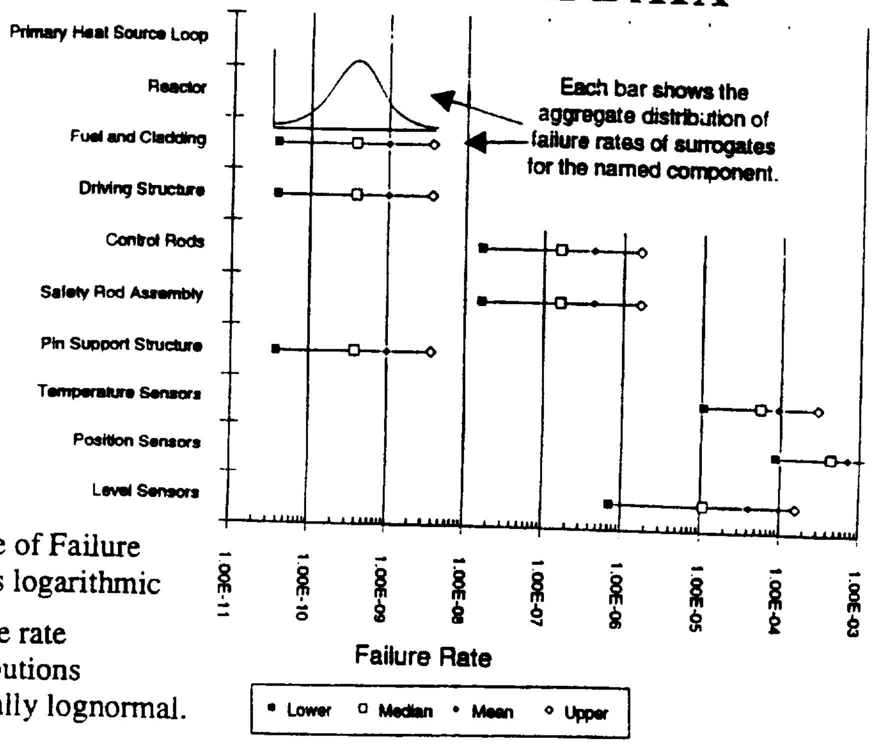
Finding the failure rates of components similar in function, application, and environment to the NEP components involves searching multiple sources. From each source a distribution of failure rates reflecting the variability in the historical components is obtained. CARP combines a number of these sources into a single, surrogate distribution representative of the anticipated performance of similar components in the NEP system.

If sufficiently similar components cannot be found in historical data references, a surrogate distribution for the NEP component is obtained by estimating the bounds within which the failure rate must fall, based on the physics of the component and the comparison of the unknown component with well-known components.



The selection and analysis of surrogates for NEP component performance was the next step in the analysis of the technology base.

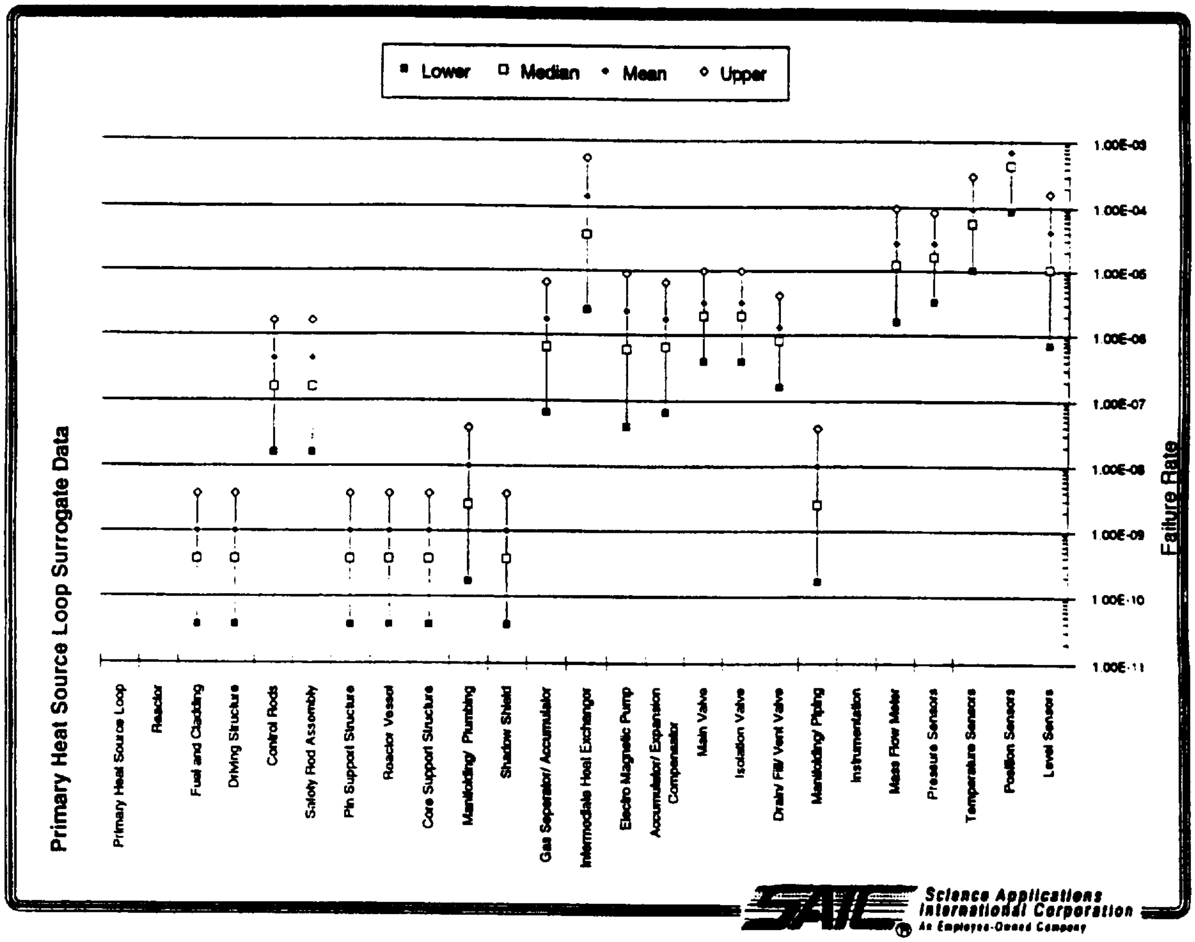
READING SURROGATE DATA



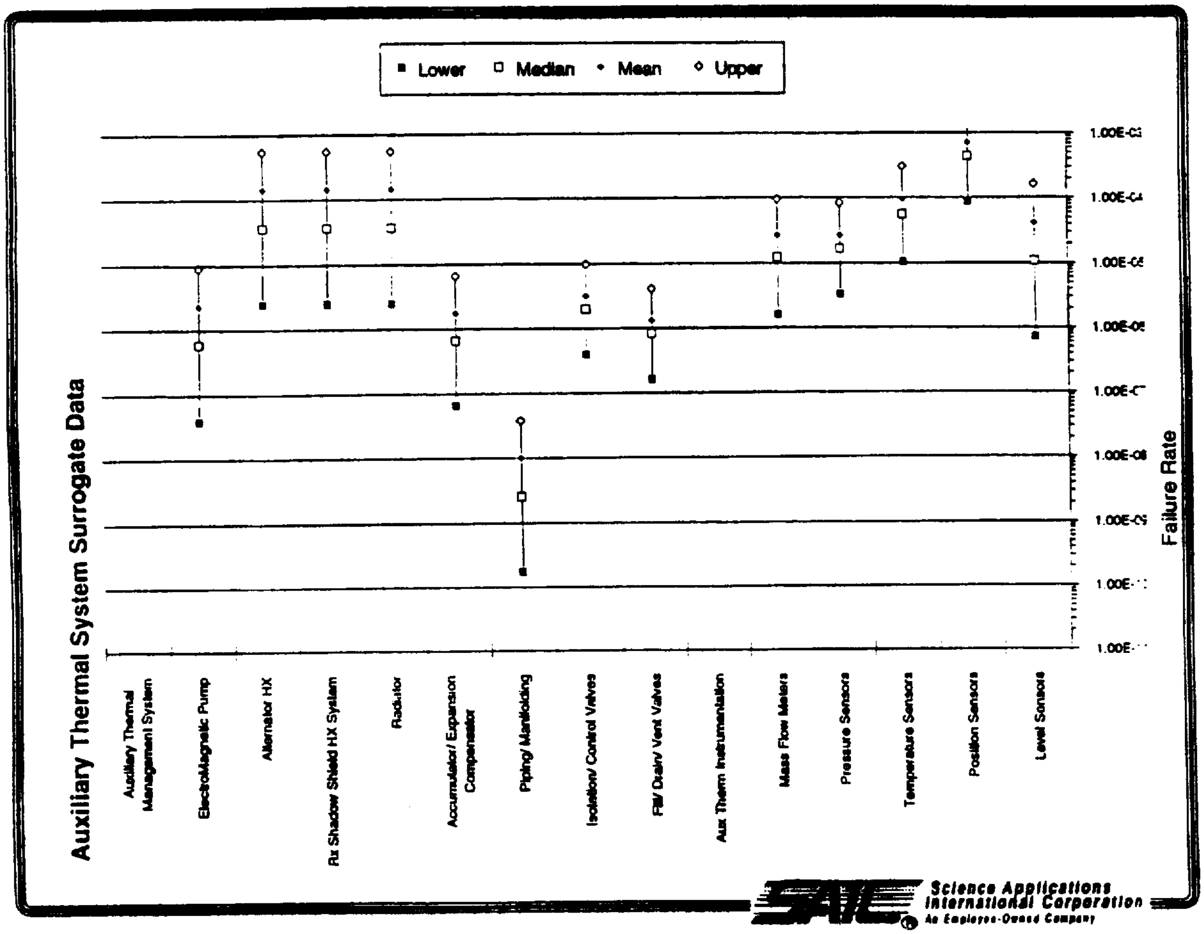
- Note: Scale of Failure Rate axis is logarithmic
  - Failure rate distributions generally lognormal.

For each component, the distribution of representative (surrogate) failure rates is depicted as indicated. The upper and lower bounds of the indicated distributions are in fact the 5th and 95th percentiles. The mean and median are both shown because these distributions are generally left-skewed rather than normal, so the mean and median are different.

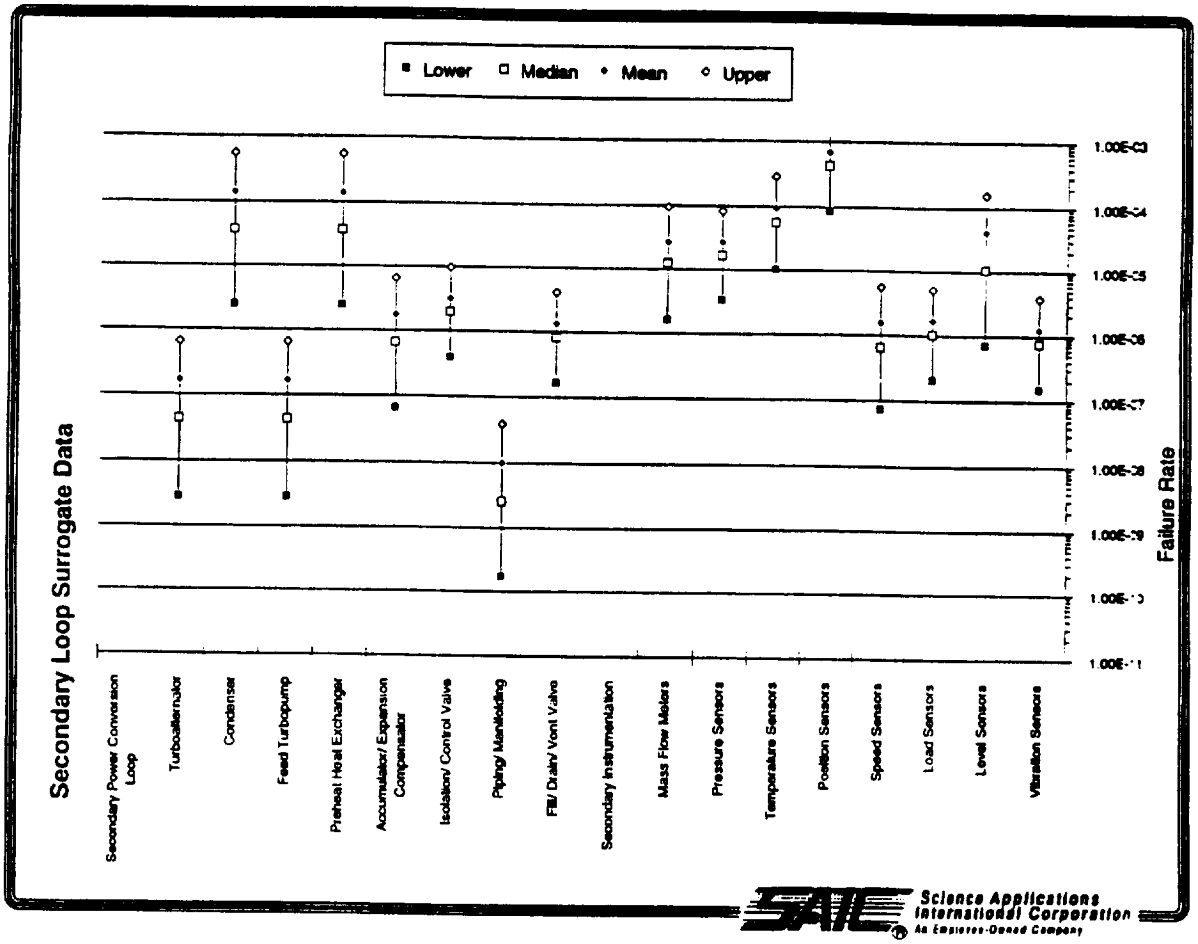
The x axis of this plot is logarithmic, so the distributions (which appear symmetric on this graph) are in fact lognormal.



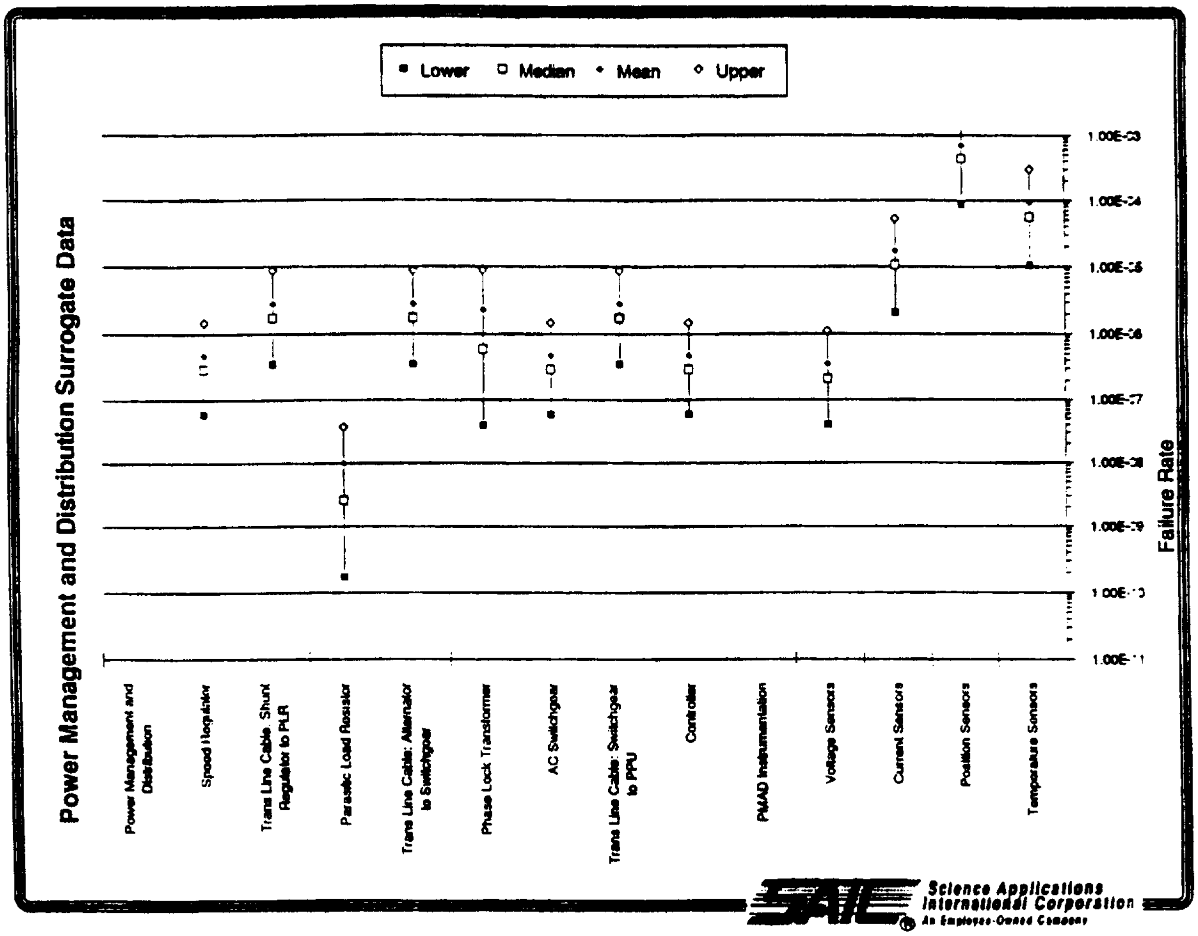
Surrogate failure rate distributions for components in the primary heat source loop.



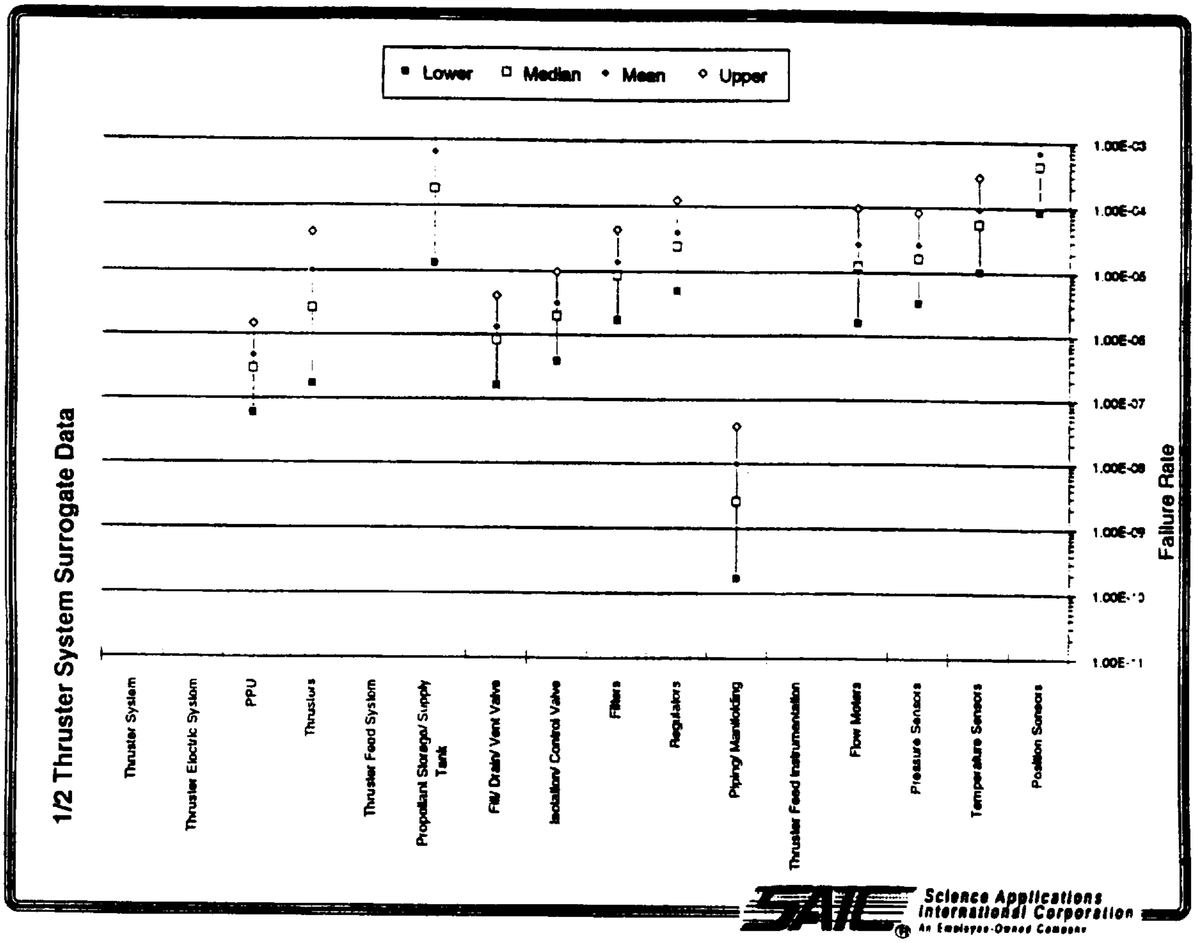
Surrogate failure rate distributions for components in the Auxiliary Thermal Management system..



Surrogate failure rate distributions for components in the Secondary Loop system.



Surrogate failure rate distributions for components in the Power Management and Distribution system.



Surrogate failure rate distributions for components in the Thruster module.

INTERPRETATION OF SURROGATE DATA NARROW SURROGATE DISTRIBUTIONS:

- Cause:
  - Little variability among components in class;
  - Little uncertainty in similarity between surrogate class and NEP application.
  - Generally mature, well understood component.
- Implication:
  - These components unlikely to change their nature through evolutionary design or wishful thinking.
- Candidate NEP components:
  - Valves, Cables, Switchgear, Sensors, Regulators, ...
- Required performance > attained performance?
  - Fundamental redesign of function.

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Narrow distributions in the surrogate data indicate that the component exhibits little variability in historical applications, and that there is little uncertainty in the application of this surrogate to the NEP application.

A narrow distribution is generally indicative of a mature component whose essential nature is well understood and generally not a good candidate for improvement in reliability, except through very fundamental redesign.

INTERPRETATION OF SURROGATE DATA BROAD SURROGATE DISTRIBUTIONS:

- Causes:
  - High variability in surrogate component population.
  - Significant uncertainty in applicability of surrogate data to NEP.
- Implication:
  - Requires close attention in design, specification, and selection.
  - High developmental risk.

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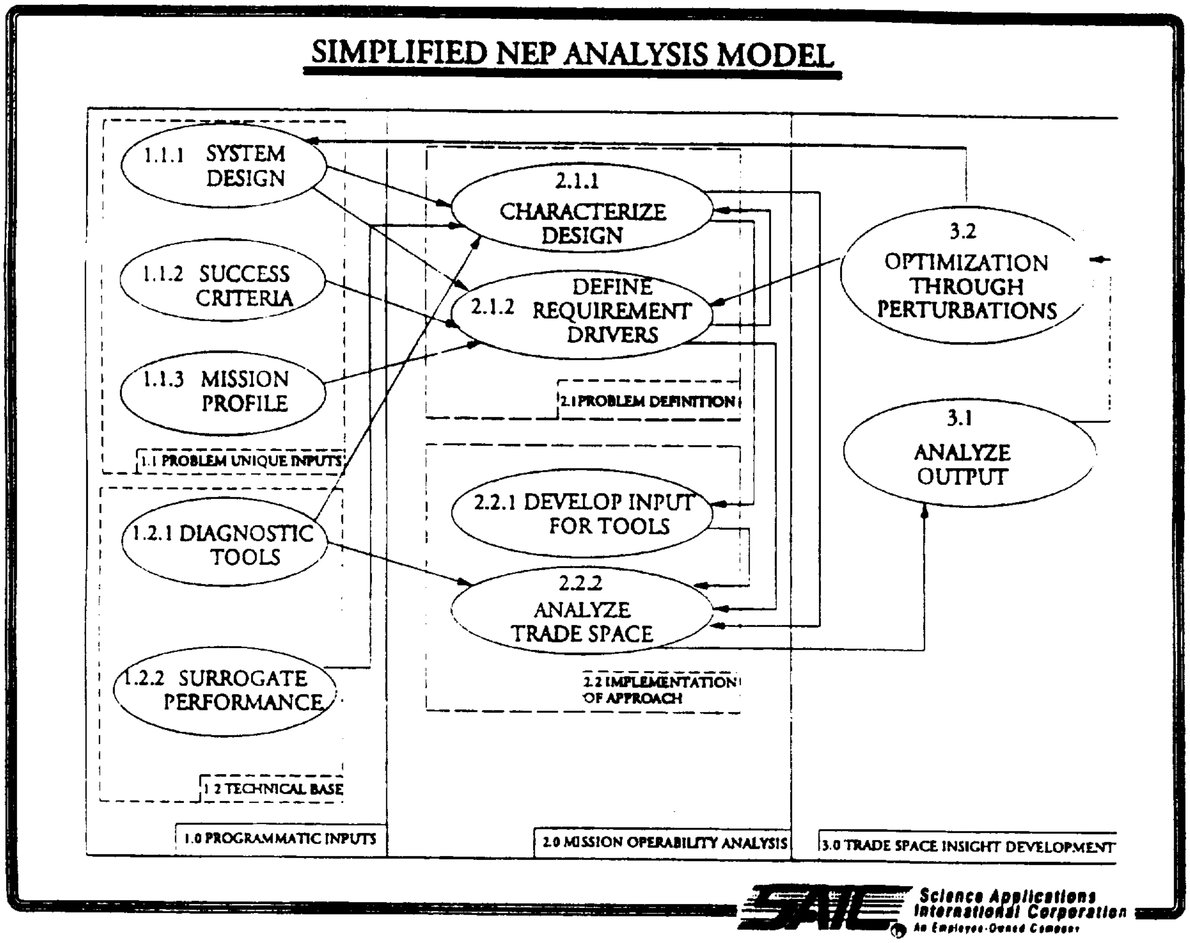
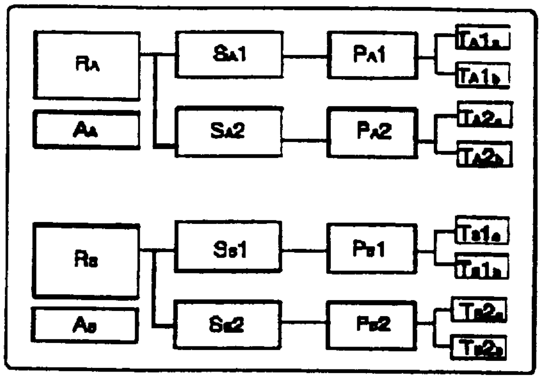
Conversely, wide distributions of surrogate failure rates indicate significant variability, uncertainty, or both. Wide distributions indicate that this component may be a high risk item.

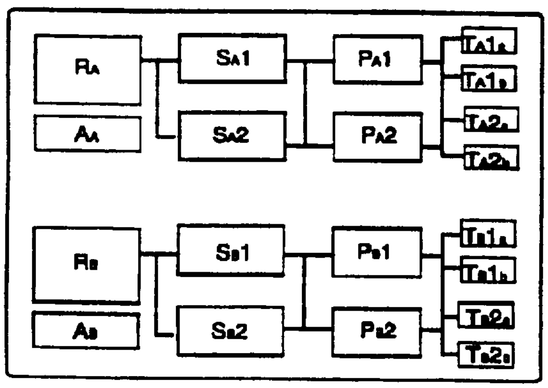


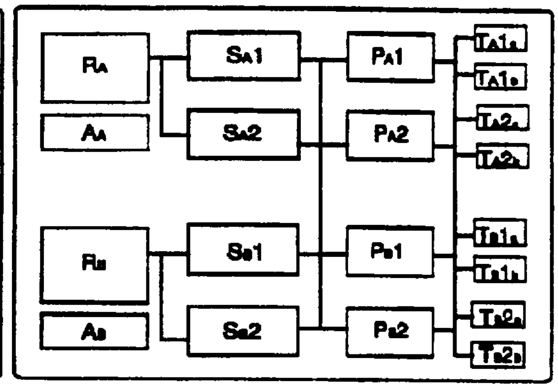
In the problem definition phase of the analysis, the first step was to characterize the design.

NEP MARKOV MODELS - PHYSICAL CONFIGURATION

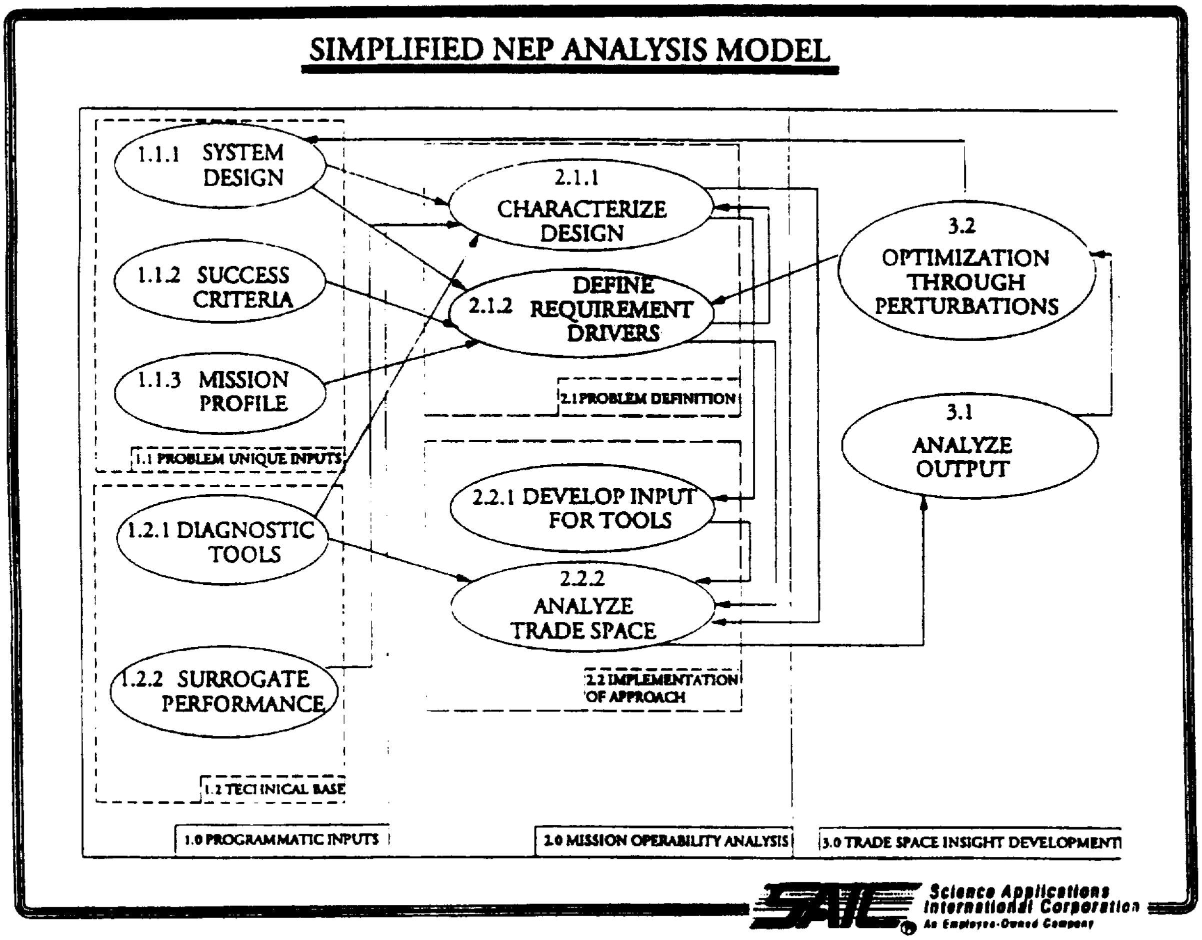


- Three physical configurations of basic model examined:
  - No Cross-Connection
  - Electrical Cross Connection w/in 5MWe module
  - Electrical Cross Connection accross 5MWe modules





There were essentially three different ways to functionally connect, or "wire" the basic design we were provided in the program input phase. Each of the connection strategies embodied a different level of inherent resiliency.



The next step in problem definition was to define the requirement drivers within the context of the model.

QUANTIFY SUCCESS CRITERIA

- Possible quantitative interpretations of success criteria:
  - Simple Reliability -
    - Probability that NEP system performs to specified capacity throughout mission > 0.99.
    - Specified capacity = Full capacity
    - Mission success and crew safety equivalent.
  - Probability of available thrust > minimum thrust required.
    - Minimum thrust required varies with mission phase.
    - Minimum thrust to complete mission generally not equal to Minimum thrust for crew safety (abort).
  - Expected value of thrust.

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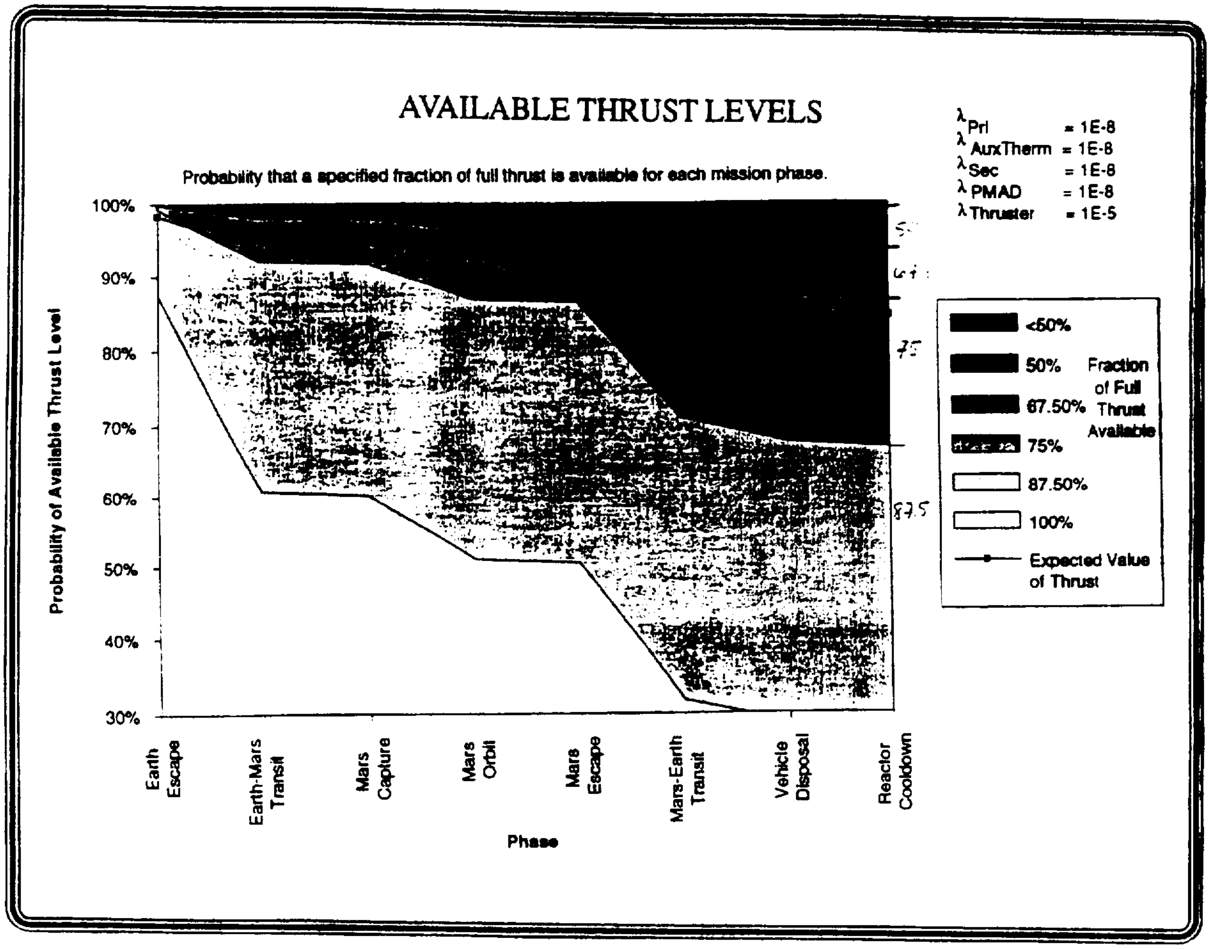
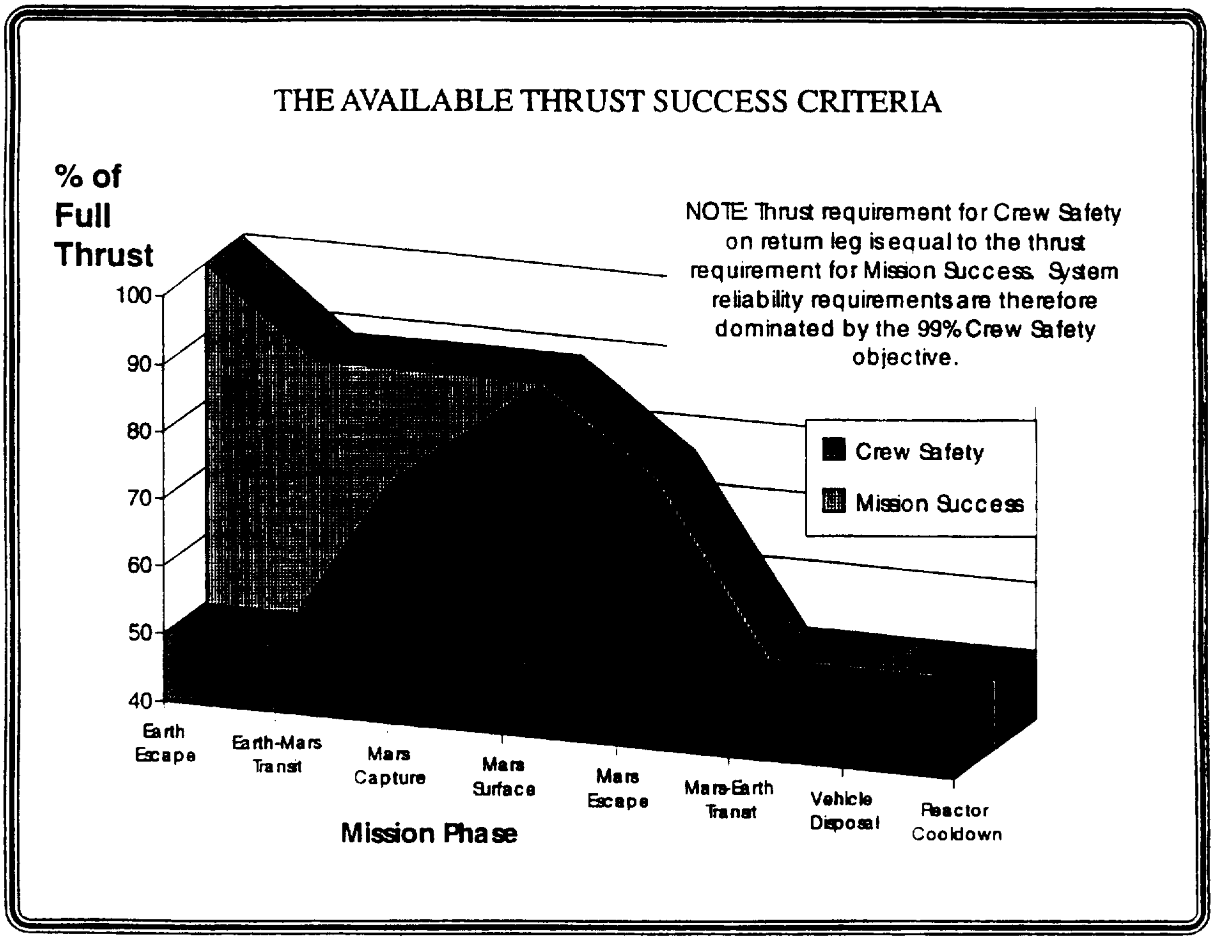
At least three different interpretations could be applied to the basic mission success criteria. The interpretation applied in this study was to determine the minimum thrust required in each phase of the mission for crew safety and for mission success, and to select reliability parameters so that the probability of achieving those levels of thrust was greater than 0.99 (crew safety) and 0.95 (mission success).

An important element of this interpretation is the idea that the thrust required to complete the mission successfully is not necessarily equal to the thrust required to return the crew safely.



This graph depicts the probability that the NEP system will be able to deliver at least the indicated fraction of full thrust (100%, 87.5%, 75%, ...) as a function of mission phase, given the subsystem failure rates indicated in the upper right corner. These failure rates were chosen to produce an exemplary graph, not because the are realistic.

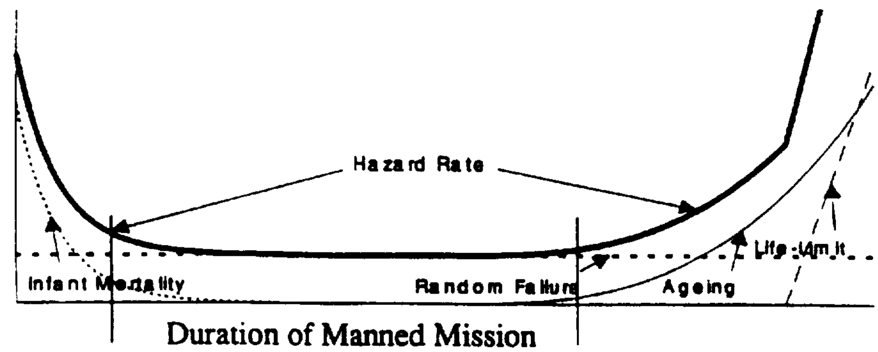


The preceding graph provided the probability that discrete levels of thrust would be available during each mission phase, half of the information required to determine the probability of meeting crew safety and mission success objectives. This curve show the other half of the information required -- specifically, what level of thrust is required in each phase to complete the mission and to ensure crew safety.

While these values were selected with some care, they are not the result of rigorous mission and orbit analysis. They are intended to represent a starting point for further investigation. Note that the values selected imply that the thrust required to ensure crew safety is the same as the thrust required for mission success throughout the return leg of the mission. The implication of this, if it correctly reflects the actual system, is that for most combinations of subsystem reliability parameters the 99% crew safety requirement dominates the 95% mission success requirement.

SELECTING RELIABILITY FIGURE OF MERIT

Hazard Rate "Bathtub" Curve



- Manned mission phases occur after Earth escape spiral "shakedown".
  - Infant mortality not an issue during manned phases.
- Sound design practice is assumed:
  - Crew return before ageing becomes issue.
- Reliability Figure of Merit = Random Failure Rate.

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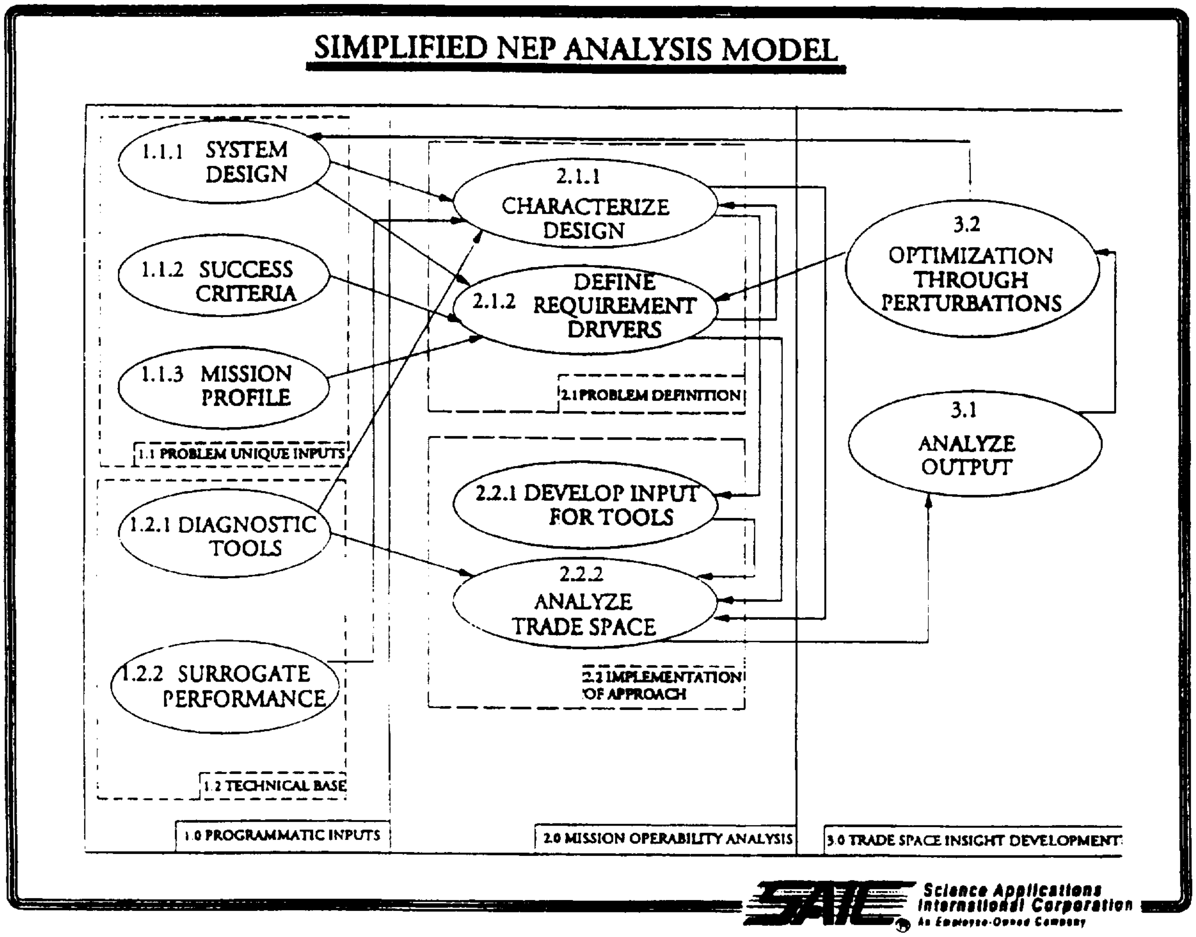
The rate at which failures occur is referred to as the hazard rate. In general, hazard rate is a time-varying quantity and is frequently separated into components which reflect the behavior of the hazard rate over time. These components are: (1) infant mortality, the hazard rate starts high and decreases over time as latent defects are "shaken out" of the new system; (2) random failure, the hazard rate is approximately constant; (3) aging, hazard rate increases as components weaken; and (4) life-limit, hazard rate increases rapidly (to 1) for components with a deterministic, observable depletion mechanism.

The constant random failure rate was the only component of hazard rate analyzed in this study based on the assumption that the manned portion of the NEP mission would occur in that domain.



The next phase in the analysis was to develop the inputs for the selected tools.

DESIGN ALTERNATIVES TESTED

NEP Model Number Achi Static Reliab	Minimum Thrust Required in Limiting Phase				Min Equip. List	Repair / Salvage
	87.5%	75.0%	67.5%	50.0%		
No Cross Connection	1	2	-	-	1MEL	4
Electrical Cross Connection Within 5 MWe Module	5	5T	-	-	-	-
Electrical Cross Connection Between 5 MWe Modules	6	6T	-	-	-	-
Fluid / Mechanical Cross Connection Between 5 MWe Modules	-	-	-	-	-	-
Minimum Equipment List Approach to Safety	1MEL	-	-	-		
Repairable / Salvageable System	4	4T1	4T2	4T3		

- Matrix of achievability analysis experiments.
- Cells contain:
  - Experiment Number

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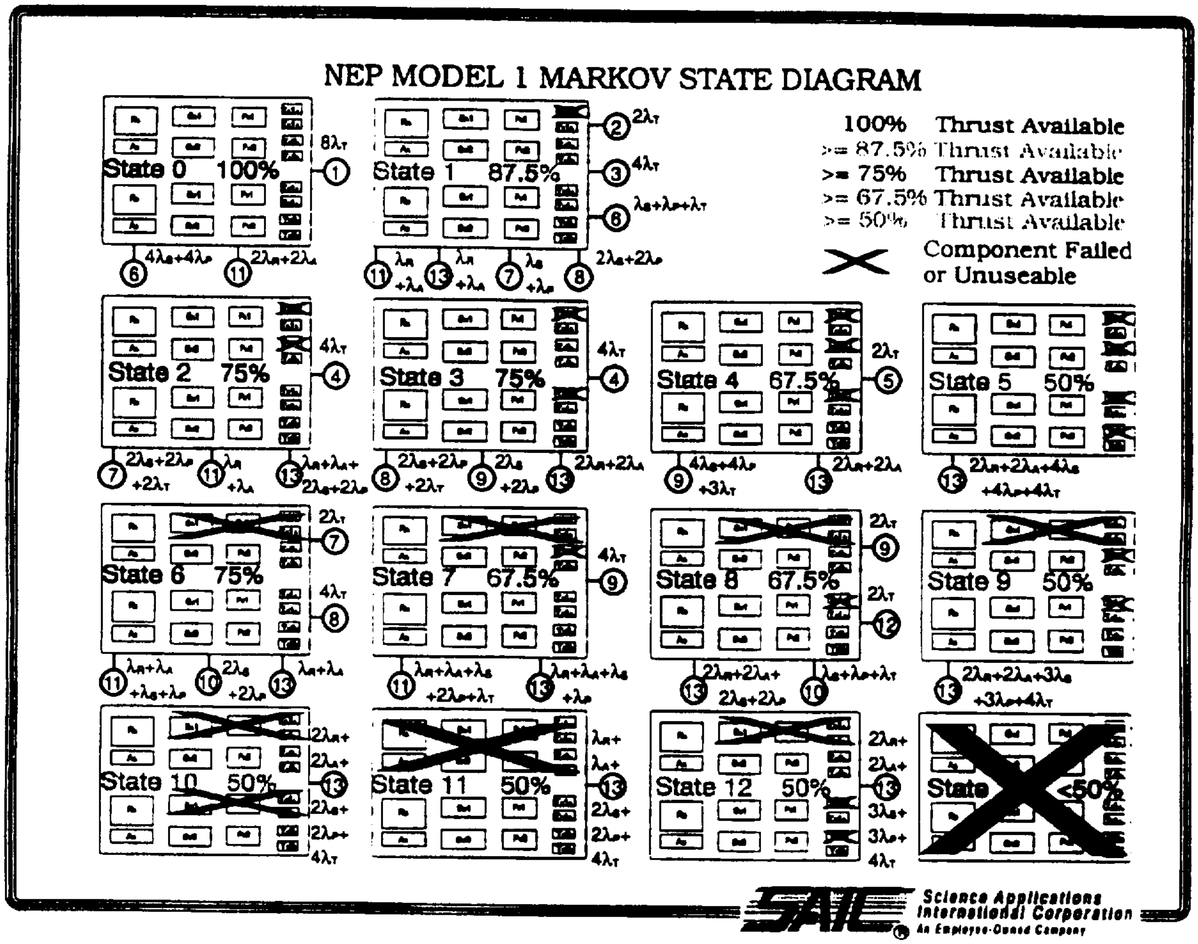
Although the analysis was limited to a single core design concept, a wide variety of perturbations or interpretations of the design could be applied. This matrix depicts the alternatives that were analyzed.

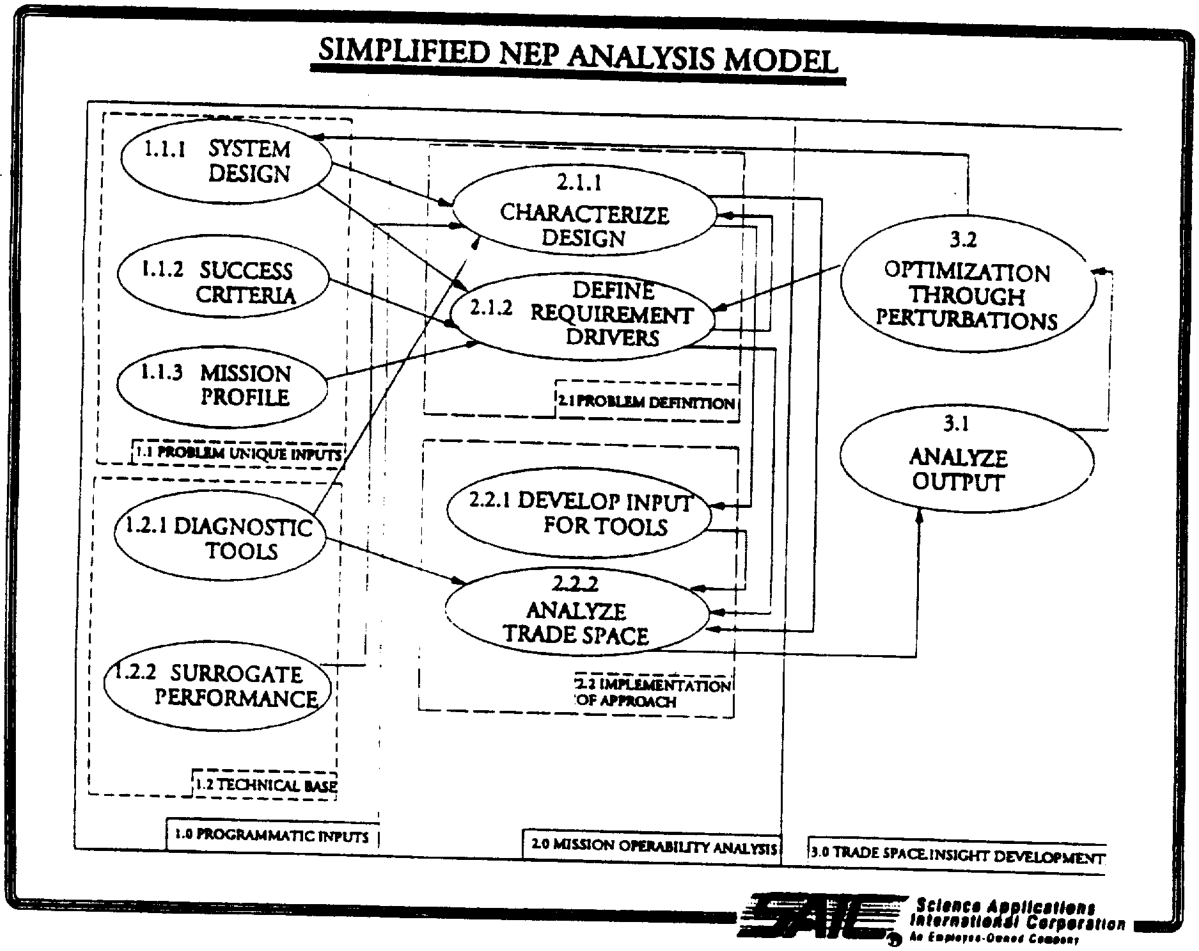


The simplest analytical model of the system allowed no cross connection between subsystems on different legs within a 5MWe module, or across modules. This diagram depicts the system states used in the Markov analysis for this model.

State 0 depicts the system with all modules operational. State 1 is the system with a single failed thruster module, state 2 has two failed thrusters - one in each leg of the same 5MWe module. For this analysis all conditions resulting in less than 50% of total thrust available were lumped into the same state, since we assumed that all such states led to mission failure and loss of the crew.

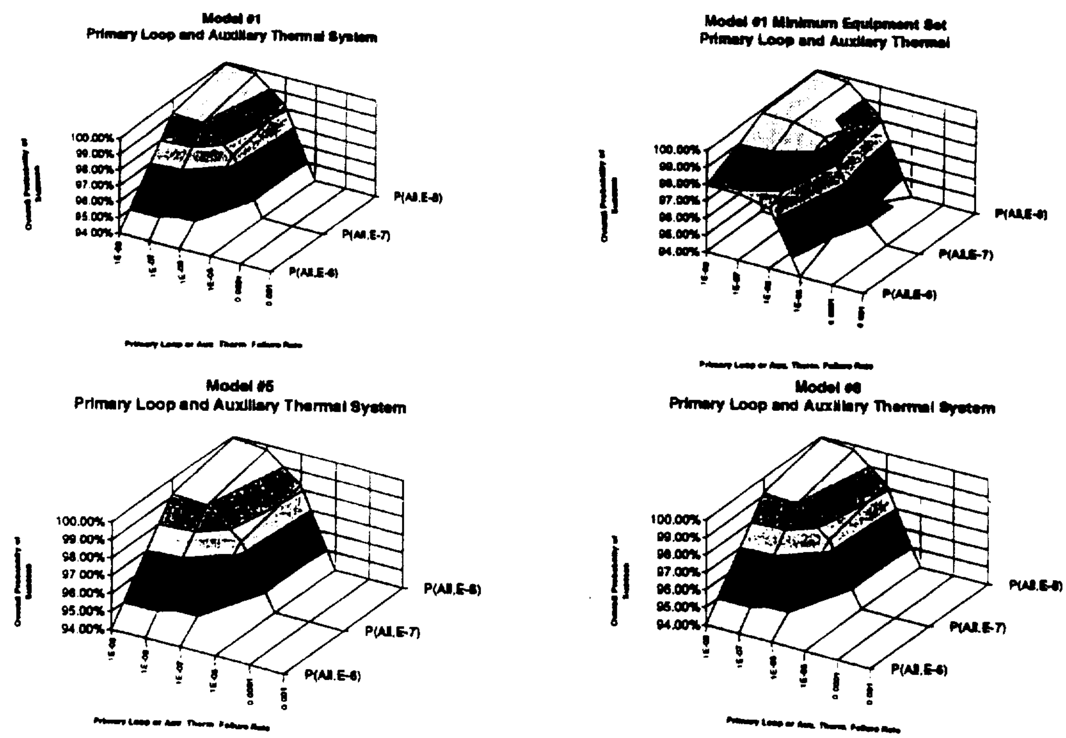
The rate at which this system (model) transitions from one state to another is indicated in terms of the failure rates of the subsystems. Ultimately, the Markov analysis is used to find the set subsystem failure rates that result in the success criteria being met. The thrust levels associated with each system state are also indicated on this diagram.

The other models are not depicted in this fashion because the number of states was too high.



The final step in implementing this study approach was to analyze the subsystem failure rate trade space resulting from the Markov analysis.

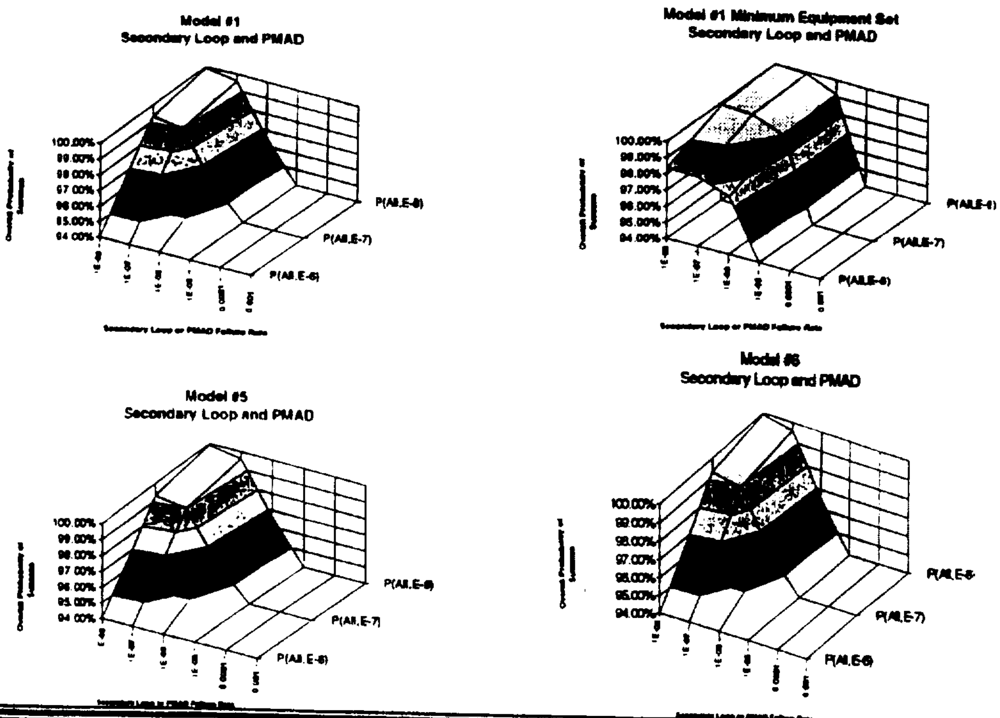
Primary and Auxiliary Model Comparison



The Markov model associates sets of failure rates with the probabilities that the system will be in each state at any time in the mission. Combining this with the knowledge of the thrust available in each state, and the thrust required for mission success and crew safety, we can determine the probability that the system will meet the success criteria as a function of the subsystem failure rates.

These graphs depict the "success probability" of the system as a function of the failure rate of the Primary Loop and the Auxiliary Thermal subsystems versus the failure rates of all other subsystems. Primary Loop and Auxiliary Thermal are lumped together because if either fails, the system is reduced to 50% thrust capacity -- a failure in any mission phase. This means that the Primary Loop and Auxiliary Thermal subsystems are equally important to the system -- from the success requirements point of view their failures are indistinguishable -- therefore the successful failure rates associated with them are the same. The different graphs depict different models which vary primarily in the arrangement of interconnections. Note that the failure rates required for the Primary and Aux. Thermal subsystems is essentially independent of the degree of interconnection, since any failure of these systems results in mission failure.

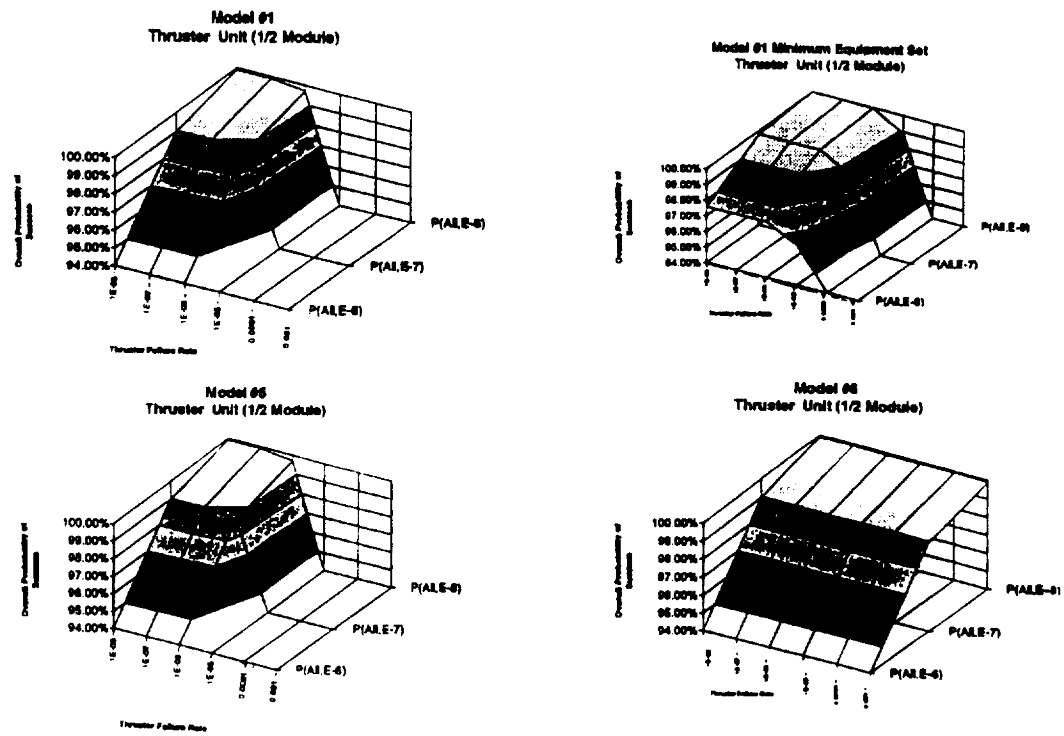
Secondary & PMAD Model Comparison



Like the Primary and Aux. Thermal subsystems, the PMAD and Secondary subsystems are of equal importance. Since a failure of either of these subsystems would reduce available thrust to 75%, and since (for these models) the thrust required for crew safety and mission success is 87.5% during the Mars escape spiral, any PMAD or Thruster failure prior to Mars escape would result in mission failure and generally (given the model assumptions) loss of the crew. The required failure rates for PMAD and Secondary given these model assumptions are therefore essentially the same as those required for the Primary and Aux. Thermal subsystems, very high, and independent of degree of interconnection. We will show in other models which assumptions need to be relaxed to permit more reasonable failure rates for these subsystems.

The Minimum Equipment Set model will be described later, but it should be noted here that in that model the 95% mission success criteria generally dominates the 99% crew safety requirement, so the set of "successful" failure rates in that model are those that result in "Overall Success Probability of >95%, rather than 99% which is the case in the other models.

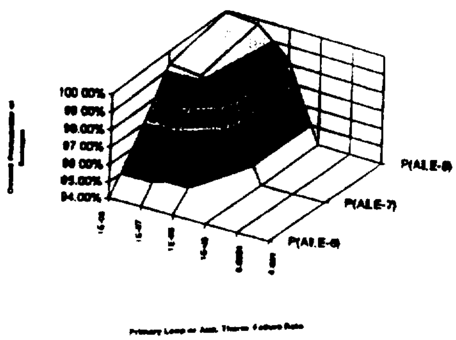
Thruster Model Comparison



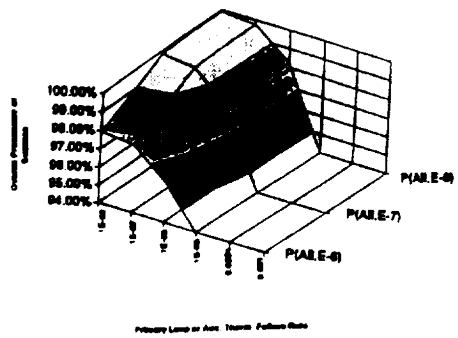
Thruster failures only remove 12.5% of the full thrust capacity, so a single failed thruster results in a successful system state at any phase of the mission, and in most phases, several Thruster failures can occur and still result in mission success. Thrusters are also very sensitive to the degree of interconnection between components.

Model 4 Primary and Auxiliary Thermal Comparison

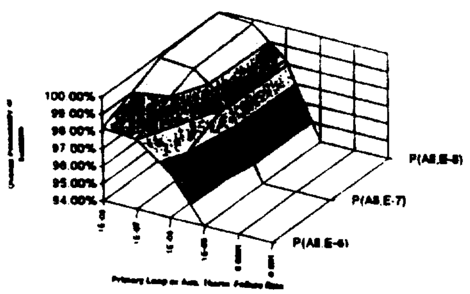
Model #4
Primary Loop and Auxliary Thermal System



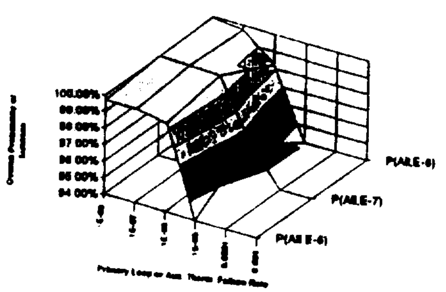
Model #4T1
Primary Loop and Auxliary Thermal System



Model #4T2
Primary Loop and Auxliary System

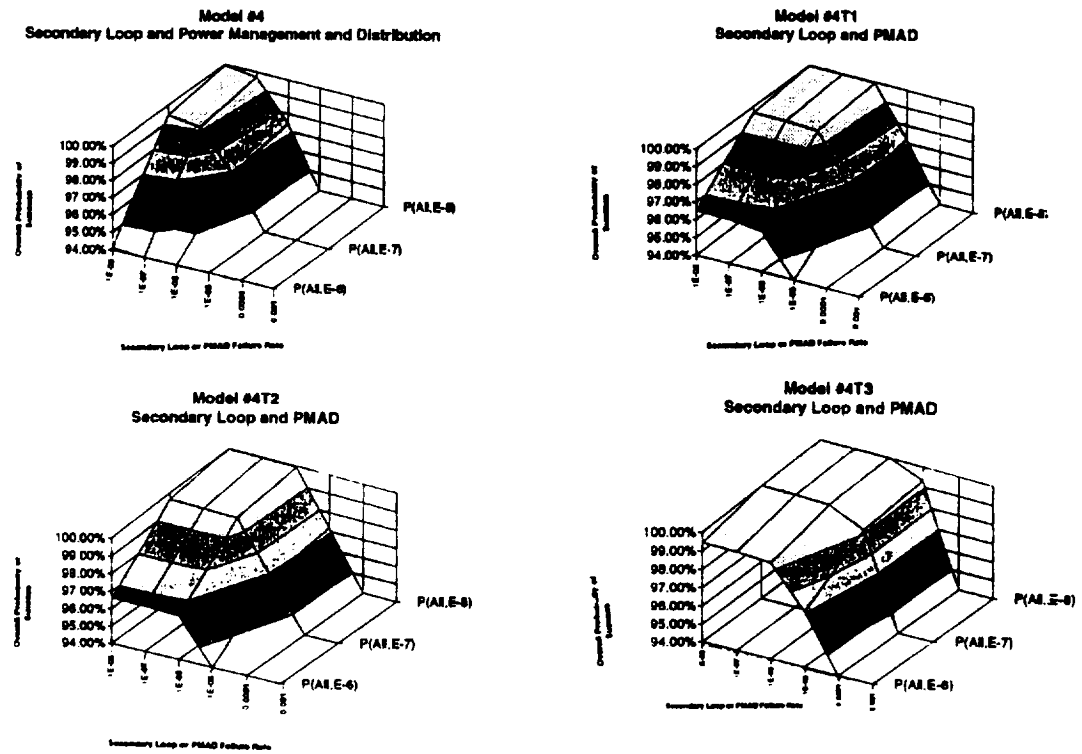


Model #4T3
Primary Loop and Auxiliary Thermal Loop



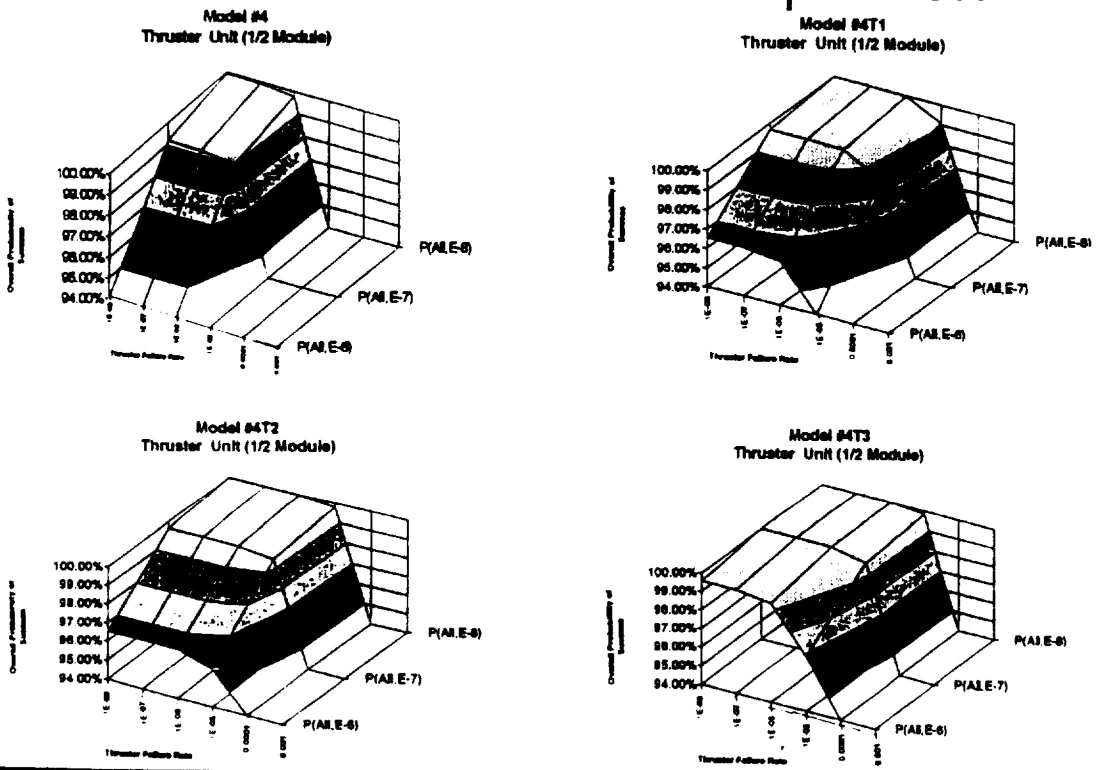
In Model 4 some degree of repair or salvage is allowed in systems other than the Primary, specifically, 25% of the first failures that occur in those subsystems are assumed to be repairable, and all the second failures are repairable, since one of the two failed systems could be used to salvage the other. The different models depicted here show the impact of lowering the highest minimum thrust requirement from 87.5% (Model 4) to 50% (Model 4T3) in 12.5% increments.

Model 4 Secondary & PMAD Comparison



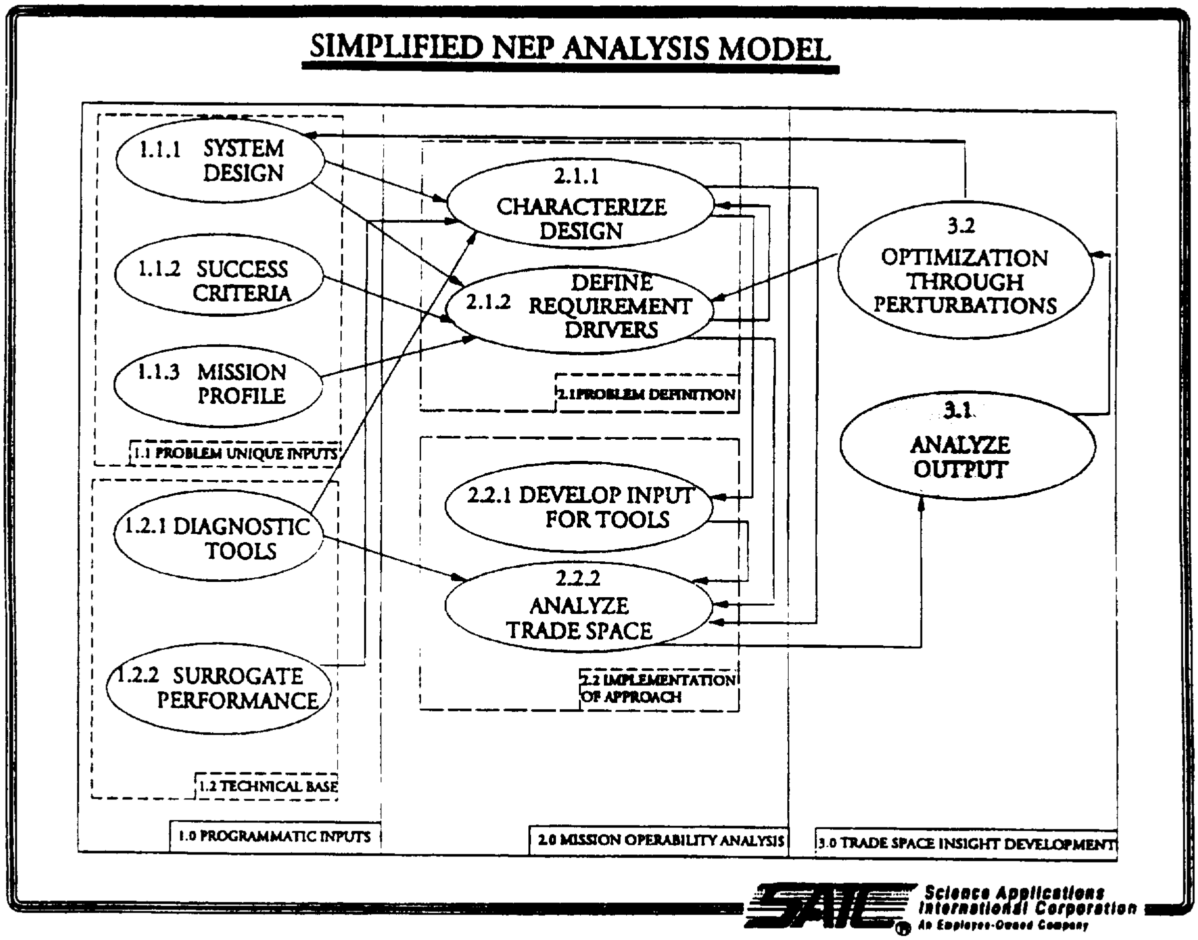
The benefit of reducing the minimum thrust requirement to thresholds which allow the failure of a subsystem without causing system failure are evident in these graphs. When the required thrust is reduced from 87.5% to 75% the required failure rates for Secondary and PMAD subsystems are reduced by an order of magnitude. Further reduction to 67.5% results in no change since Secondary and PMAD failures reduce available thrust in 25% increments. Reducing the required thrust to 50% gains another order of magnitude in required failure rate.

Model 4 Thruster Comparison



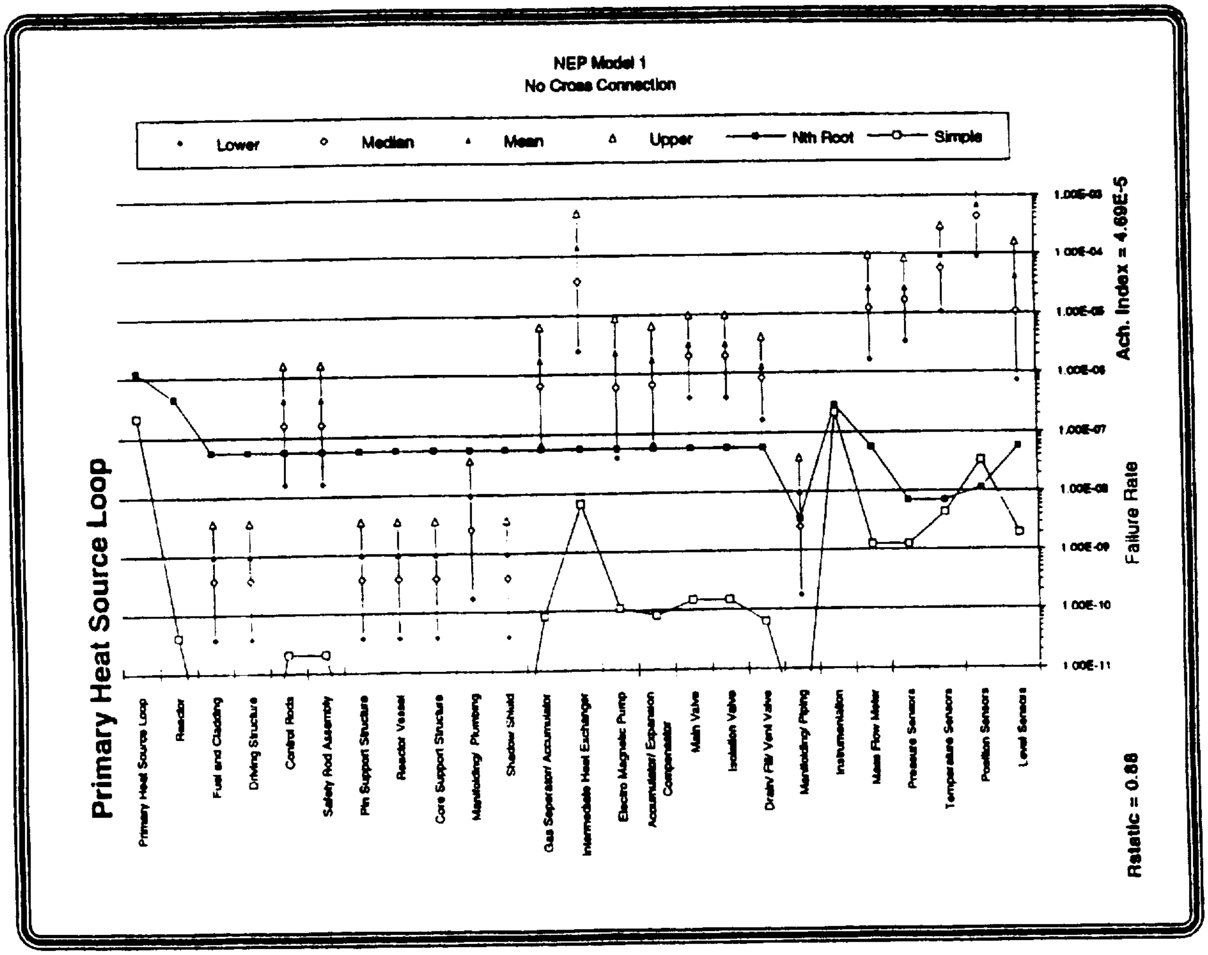
Like the Secondary and PMAD, required Thruster failure rates are significantly reduced as the maximum required thrust is reduced. Since Thruster failures only remove 12.5% of the total thrust capacity, each 12.5% reduction in required thrust has an associated relaxation of Thruster failure rate requirements.

Physically the effect of reducing the maximum required thrust in the model can be achieved without increasing the total power of the system. The reduction of thrust requirements corresponds to designing the Secondary, PMAD, and Thrusters so that they can operate at higher nominal loads. For example, if the Secondary and PMAD were designed to operate at 150% of nominal capacity, half of the failure impact of a unit could be absorbed by the other unit in the 5MWe module. Instead of reducing the thrust capacity of the system by 25%, the failure of a Secondary or PMAD would only reduce the capacity by 12.5%. Similar gain is achieved by designing the Thruster module to operate at 125% of nominal capacity. This effect is enhanced by maximizing the cross-connectivity between subsystems.



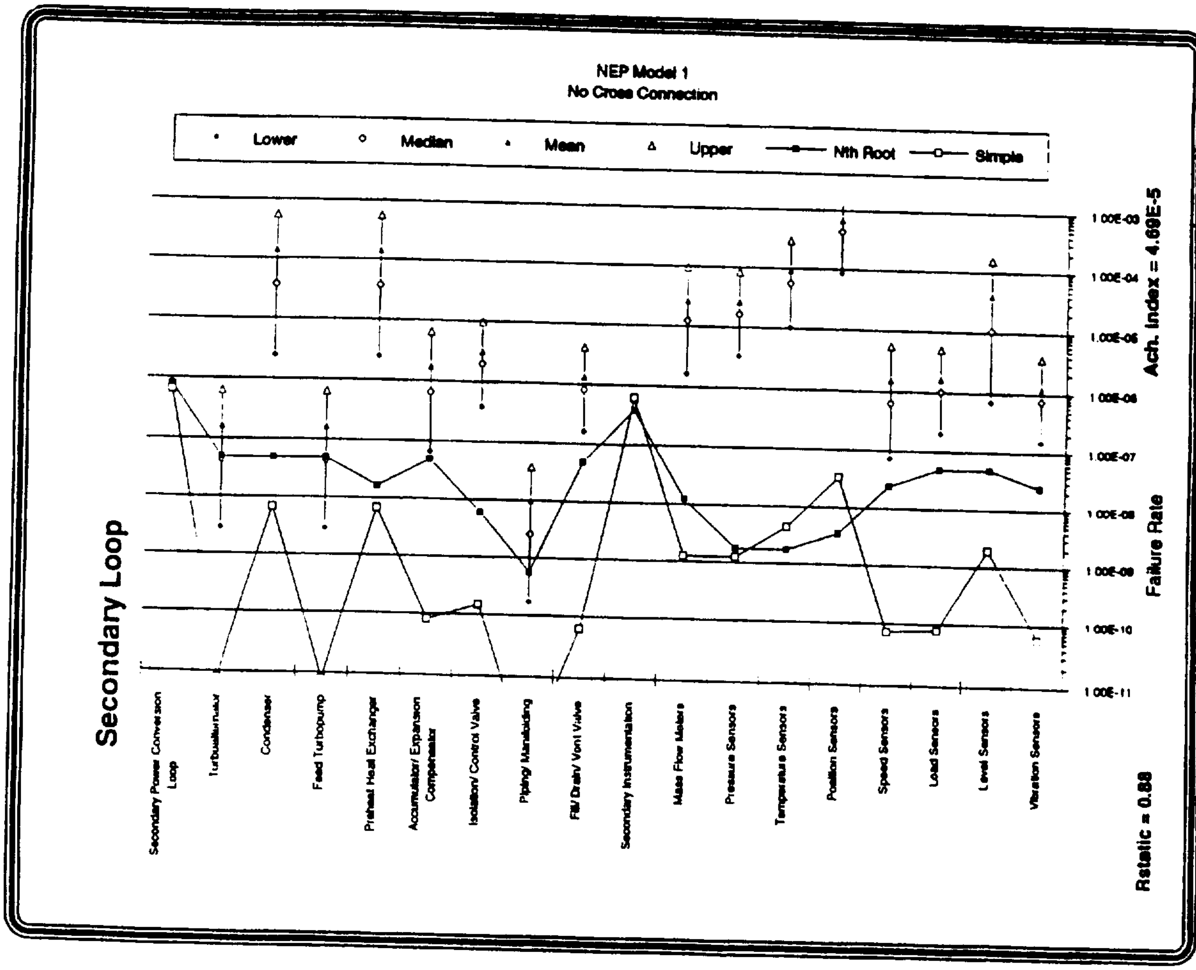
To complete the analysis the sets of subsystem-level failure rates which meet the success criteria are apportioned down to the component level for comparison with surrogate data. The RAP2™ computer code is used to accomplish this apportionment. Only two of the RAP2™ apportionment algorithms (the Simple algorithm and the Weighted Nth Root algorithm) were applied in this analysis to establish the bounds within which component failure rates would need to lie in order for the system to achieve the success criteria. The Simple algorithm establishes the worst case bound, and the Weighted Nth Root method, the best case.

A complete analysis would extend the material presented here in two respects. First, an "optimum" set of component failure rates would be sought by seeking the set of requirement driven subsystem level failure rates which minimize the aggregate achievability index (AchI). This would require extensive iteration which was not possible in this analysis. Second a distribution of apportioned failure rate and AchI would be developed, rather than the mean values presented here. The apportioned failure rates presented here are a solution, but by no means the best solution, to the problem.

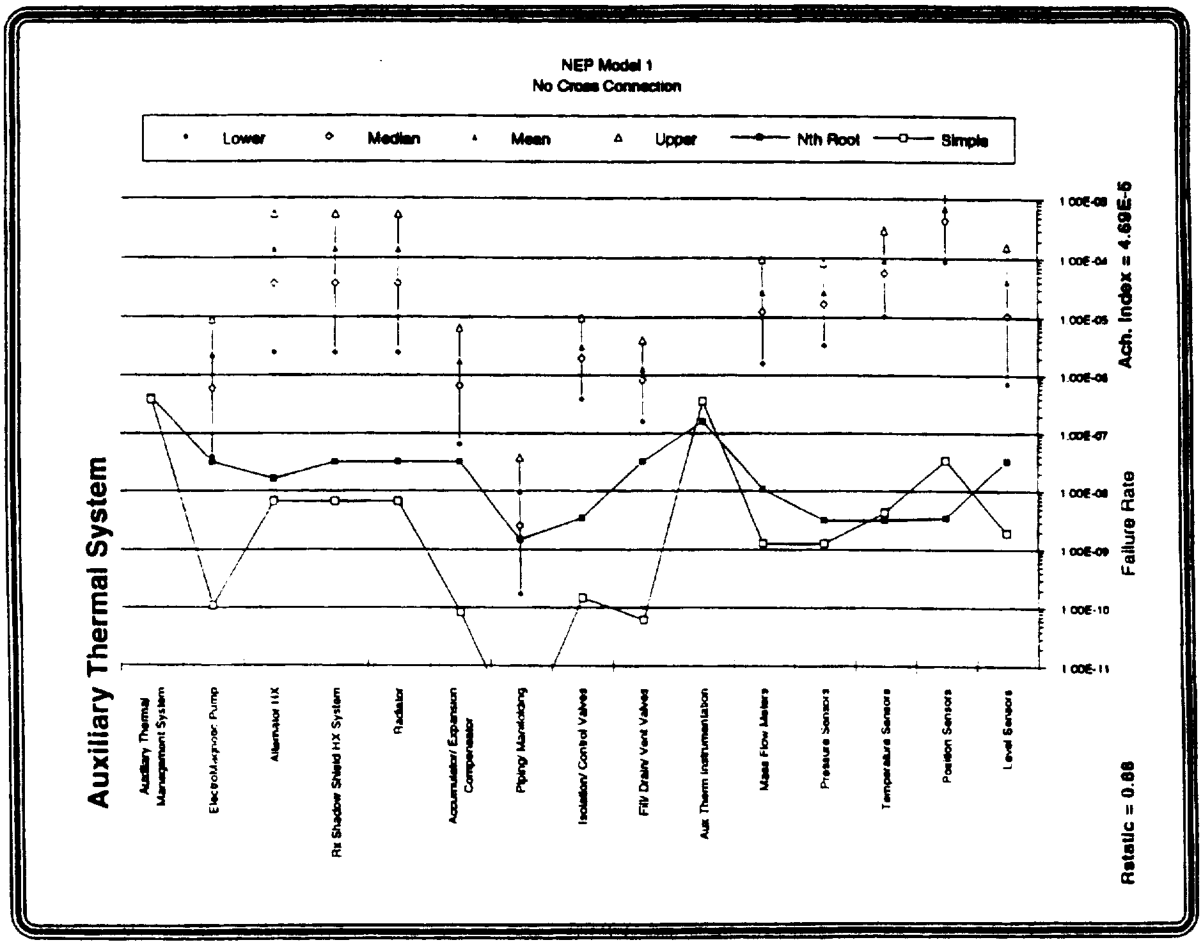


This graphic depicts the apportioned failure rate values for the Primary Loop subsystem along side the surrogate distributions obtained from the historical performance of similar components. The achievability index (AchI) is represented by the distance between the surrogate distributions and the apportioned values. The point estimate of AchI for this model in the upper right corner is the ratio of the Simple method apportioned values to the mean of the surrogate distributions. This value is essentially an outer bound on the achievability of the system for Model 1.

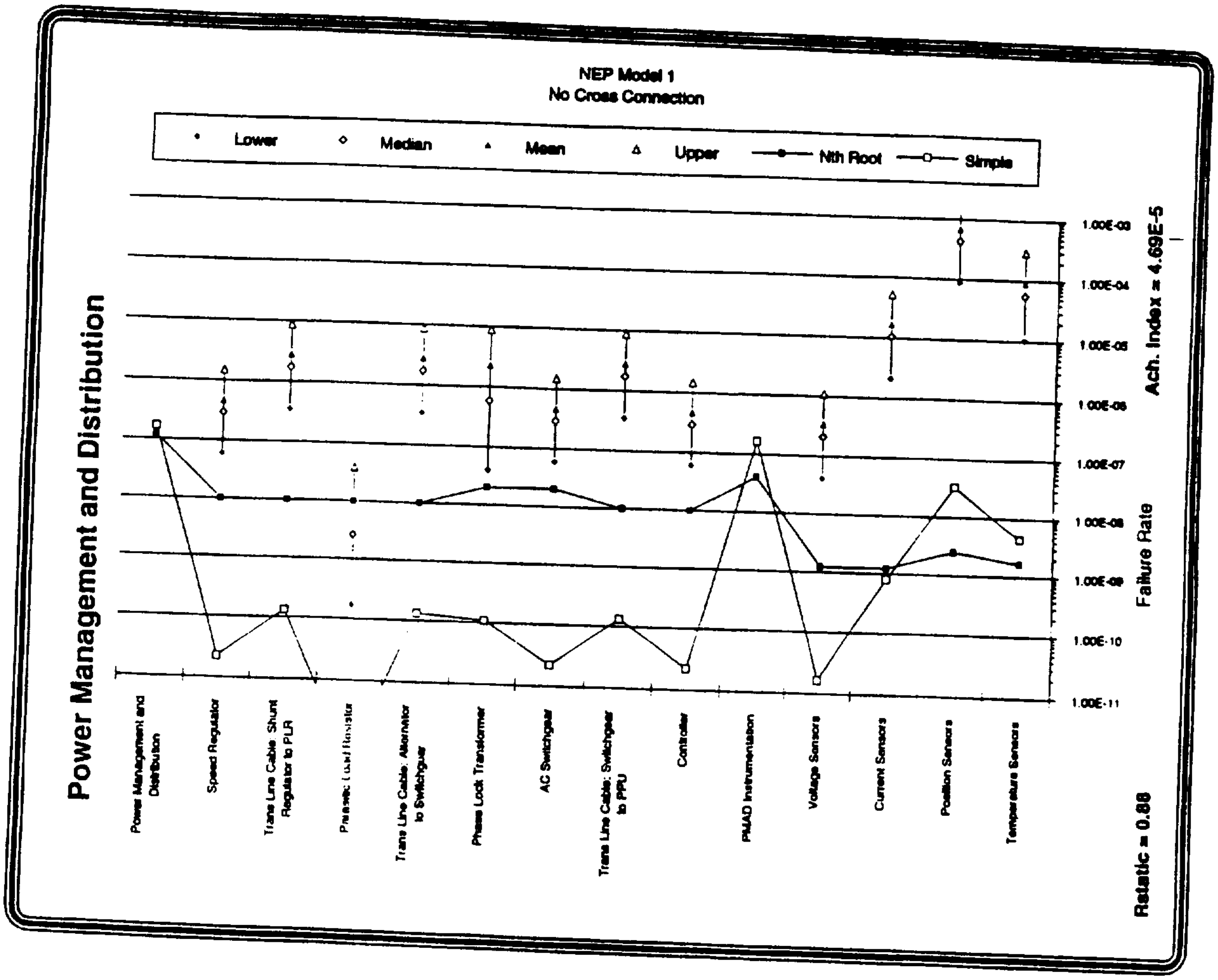
Model 1 was the simplest configuration analyzed, with no resiliency through subsystem cross-connection, and using the worst case (87.5%) required thrust criteria.



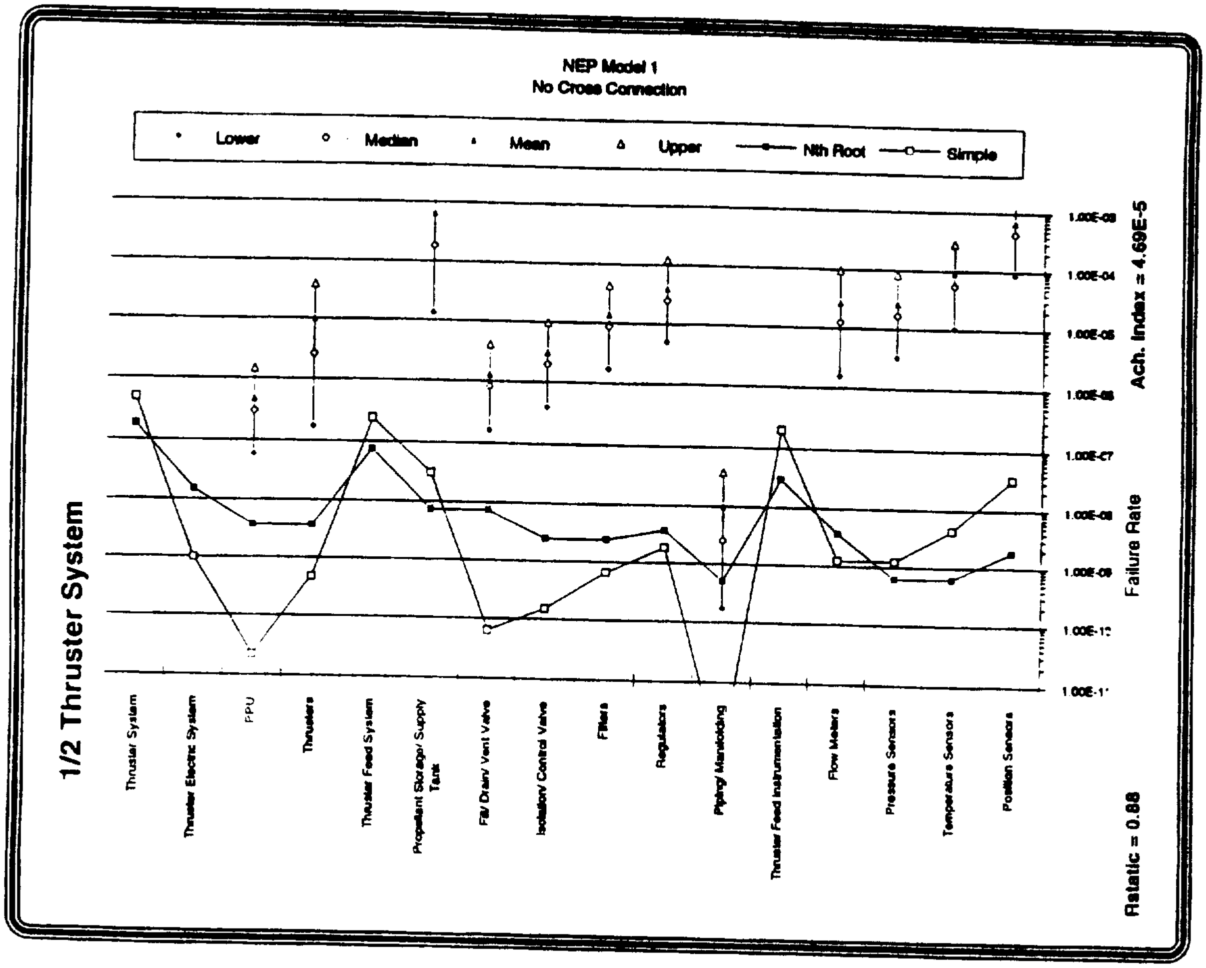
This graphic depicts the achievability of the Secondary system for Model 1. The distance between the Simple apportionment values and the surrogate distributions (the mean values of the surrogate distributions) is the same as it was for the Primary Loop subsystem. This will be true of all components because of the nature of the Simple algorithm. The Weighted Nth Root apportioned values are farther from the surrogates. This is a result of selecting a priori weighting values which indicated that, in general, high reliability would be more difficult to achieve in the Primary subsystem than in the Secondary.



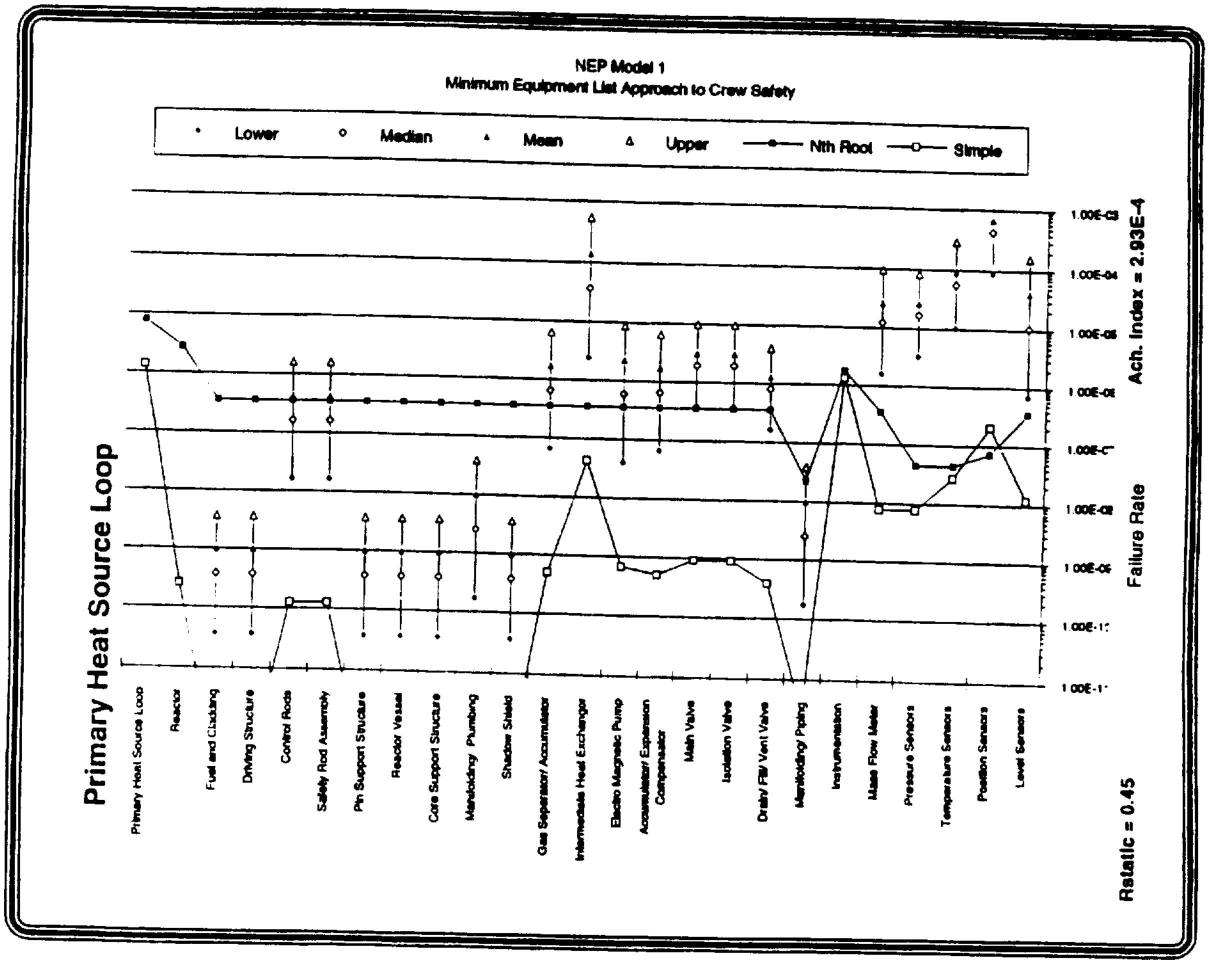
Note that the heat exchangers and the sensors in the Auxiliary Thermal system have significantly higher surrogate failure rates than is required. Also, the sensors have fairly tight distributions, indicating that these are probably fairly mature components with little variance or uncertainty in applicability. These factors indicate that these components should receive special attention. This is particularly true of the sensors, which are found in every subsystem. Sensors are discussed in more detail later.



Sensors, particularly the position sensors, appear to be the limiting PMAD component.



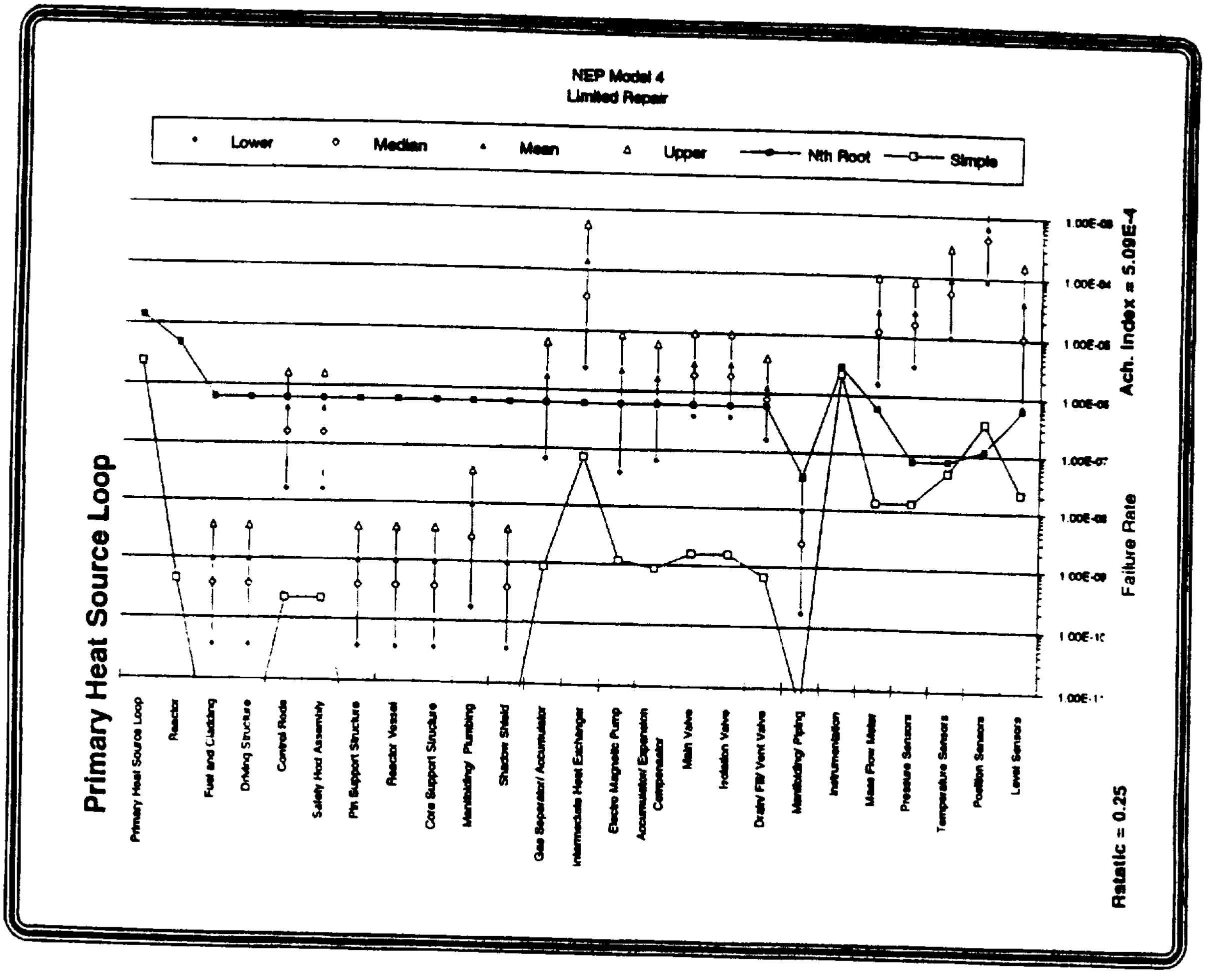
The Thruster Feed System, sensors, filters and regulators are the limiting Thruster components.



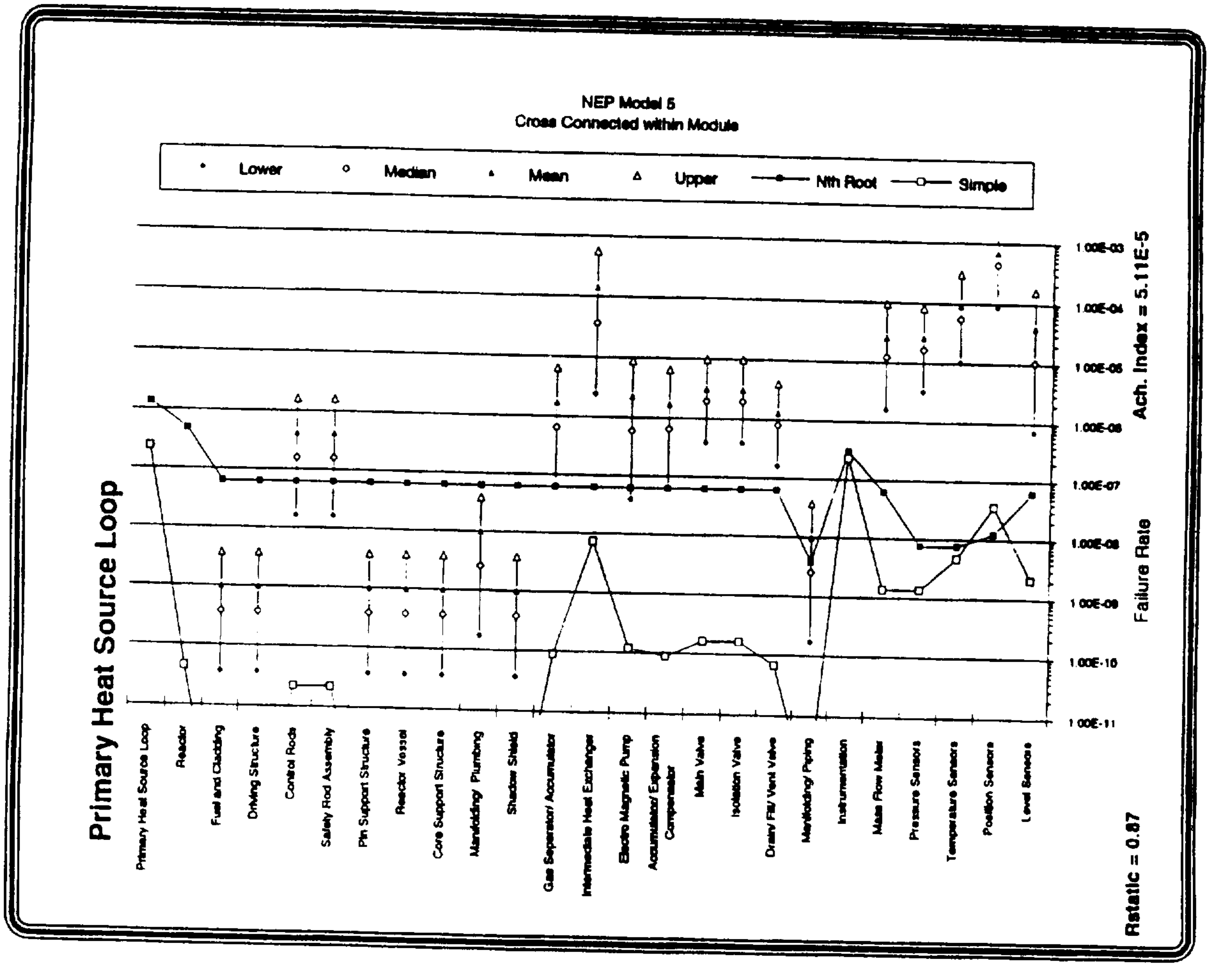
This diagram depicts the apportionment results using a model which reflects a "Minimum Equipment List" approach to crew safety. In this model, it was assumed that the decision to abort would be continuously analyzed based on the operability of a Minimum Equipment List for the NEP system. In this approach, if the system does not have sufficient operating equipment at the start of a phase to complete the mission with a 99% probability of crew safety, then an abort would occur. The set of equipment required to ensure crew safety varies from phase to phase, and is referred to as the Minimum Equipment List.

Applying this standard allows "restarting" the reliability clock with respect to crew safety at the start of each phase. The mission success reliability clock continues to run, so the 95% mission success criteria generally dominates the 99% crew safety criteria in this model.

Note that this approach improves the achievability index by a factor of almost 20 -- from $4.7 * 10^{-5}$ to $2.9 * 10^{-4}$.

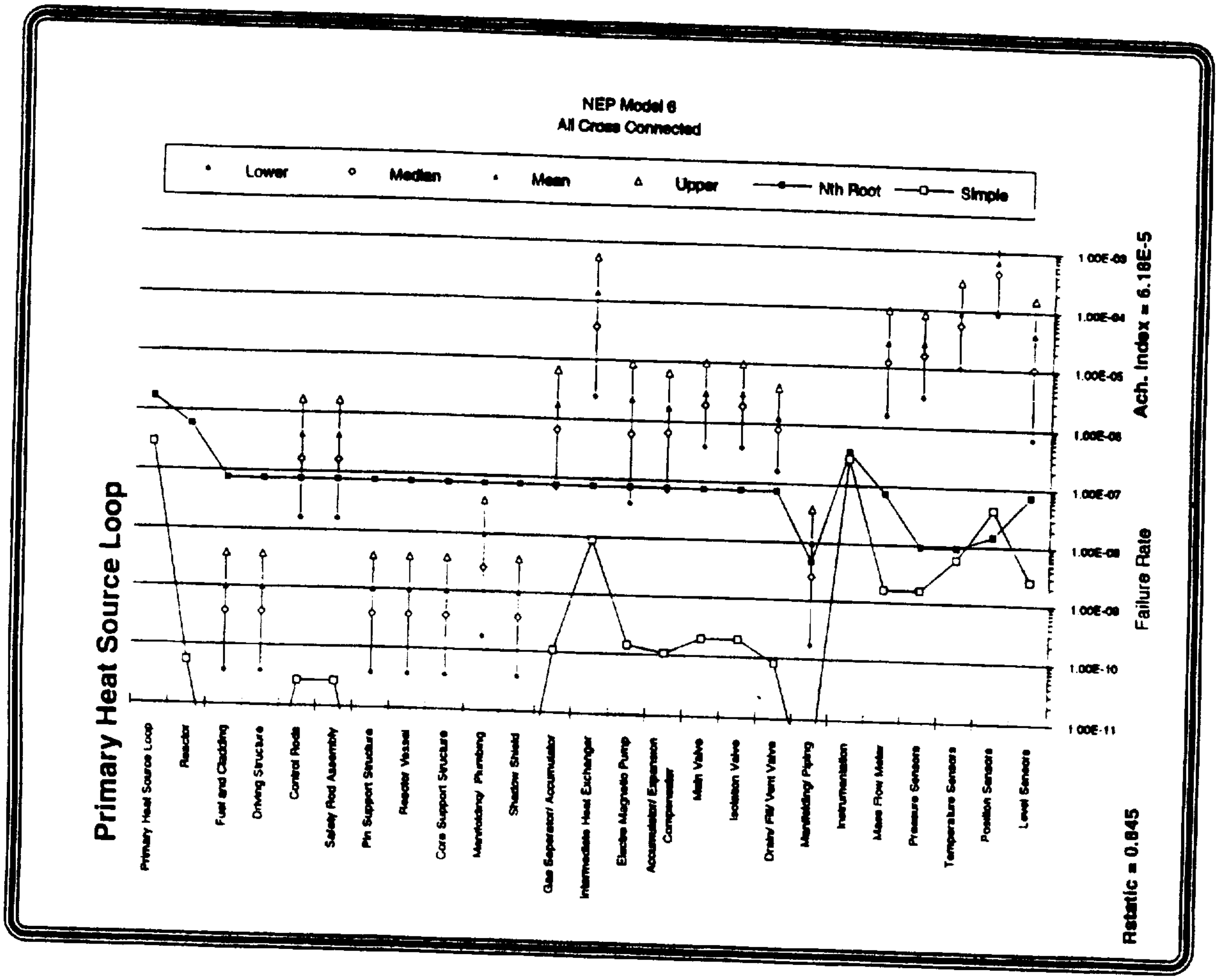


Model 4 (discussed previously) allowed limited repair / salvage. Note that the achievability index is approximately a factor of 10 better than the base case (model 1).



This model allowed cross-connection of the subsystem elements within a 5MWe module. This approach affords little improvement in achievability for these models because of the high importance of the subsystem modules. Any failure other than a Thruster resulted in the system producing less thrust than was required for the Mars escape spiral (87.5%). Therefore, no amount of interconnectivity compensates for a subsystem failure.

Limited cross-connection examined in this model is expected to provide significant benefit if the importance of the subsystems is lowered, either by requiring a smaller minimum thrust, or by providing excess capacity in the components as discussed previously.



This model, which allows for cross-connection of all electrical components -- even across 5MWe modules -- suffers from the same problem that the more limited cross-connection model does. The minimum thrust requirement is set too high to allow the resiliency of the design to have any real impact. What improvement there is in achievability ($6.2 * 10^{-5}$ versus $5.1 * 10^{-5}$) is due to the fact that the thrusters are operating in a six out of eight redundancy configuration for the portion of the mission requiring 75% thrust or less for crew safety.

ACHIEVABILITY OF NEP DESIGN

- Achievability is related to distance between apportionment curves and surrogate distributions.
- Simple and NthRoot Methods provide very different results:
  - NthRoot apportions to function
  - Simple apportions to individual component
  - Where a function has many identical components, Simple lies farther from surrogate.
- Actual solution lies between curves.

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To recap, the achievability index is the measure of the distance between what is required of the system, and what is demonstrably attainable. The surrogate date indicates what is attainable, and failure rates apportioned from top-level reliability requirements establish what is required. The two apportionment methods used here were selected to bound (at least to first order) the failure rates that would actually be required for the NEP system components.

DESIGN ALTERNATIVES ACHIEVABILITY MATRIX

NEP Model Number AchI Static Reliab	Minimum Thrust Required in Limiting Phase 87.5%	75.0%	67.5%	50.0%	Min Equip. List	Repair / Salvage
No Cross Connection	1 4.7E-5 0.88	2 6.8E-5 0.83	-	-	1MEL 2.9E-4 0.45	4 5.1E-4 0.25
Electrical Cross Connection Within 5 MWe Module	5 5.1E-5 0.87	5T	-	-	-	-
Electrical Cross Connection Between 5 MWe Modules	6 6.2E-5 0.84	6T	-	-	-	-
Fluid / Mechanical Cross Connection Between 5 MWe Modules	-	-	-	-	-	-
Minimum Equipment List Approach to Safety	1MEL 2.9E-4 0.45	-	-	-		
Repairable / Salvageable System	4 5.1E-4 0.25	4T1	4T2	4T3		

- Matrix of achievability analysis experiments.
- Cells contain:
  - Experiment Number
  - Central Value of Simple method achievability index.
  - Equivalent reliability for a static system.

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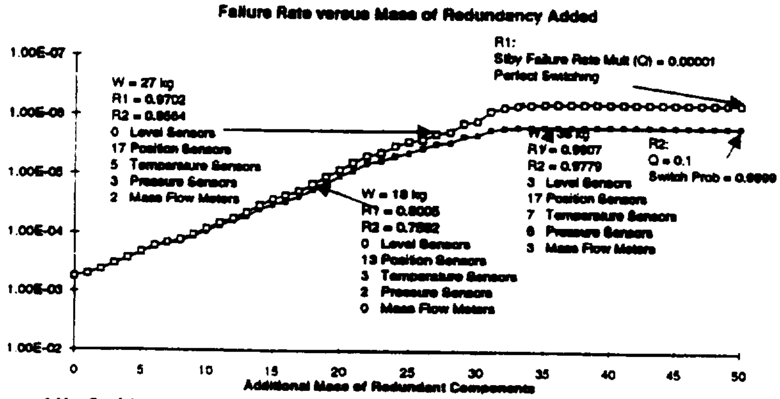
This matrix shows again the different models that were compared, along with the associated achievability index (AchI), and the equivalent static reliability value which would result if the apportioned failure rates for that model were used in a static reliability model of the NEP system.

ADDING RELIABILITY THROUGH REDUNDANCY



- "Optimal" failure rate versus mass of redundancy for Primary Loop Instruments found using Dynapro(TM).
- Note that there is a limit to the reliability that can be added through redundancy.
- Typical levels of redundancy improve functional failure rate by factor of 2.

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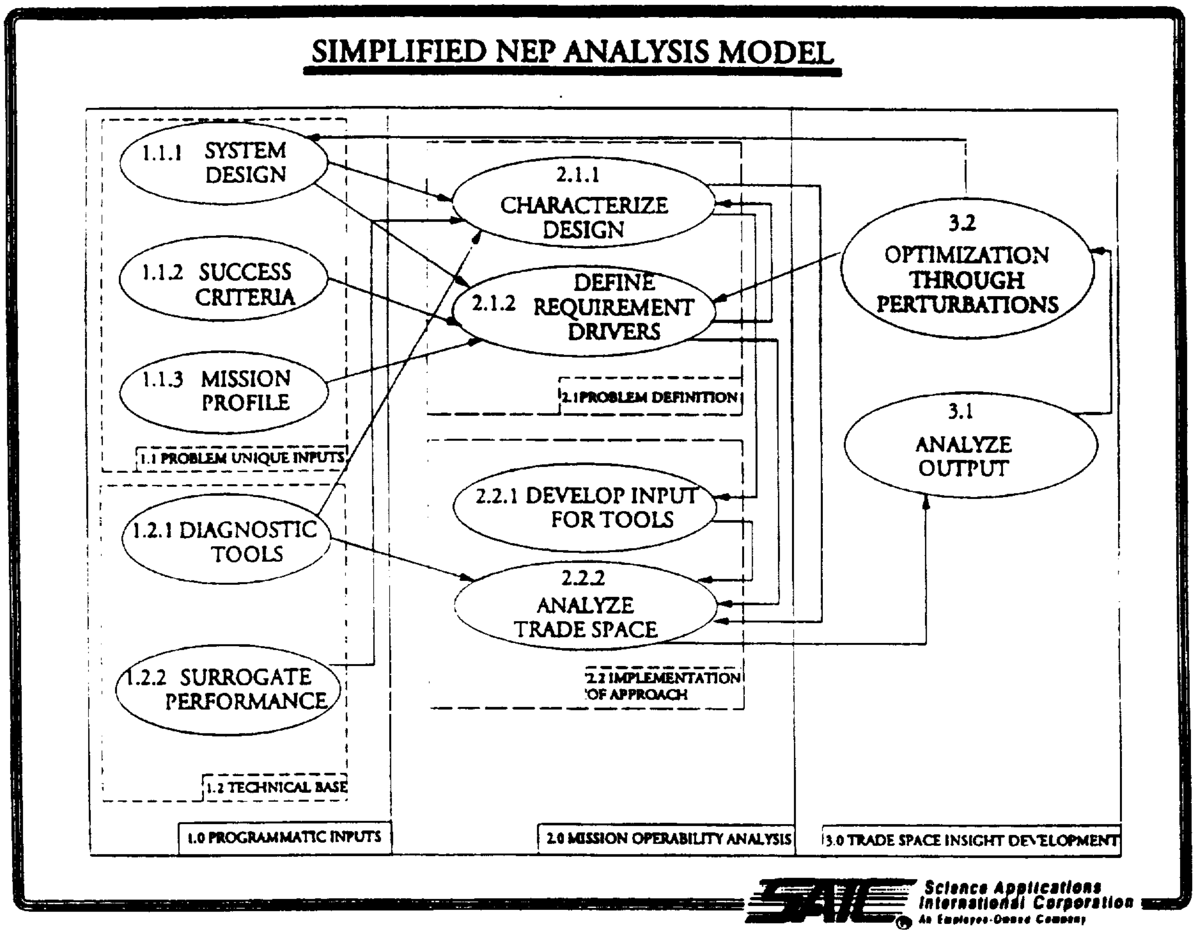
A common fallacy is that any level of reliability can be achieved by adding enough redundancy. To determine the extent to which this true we used Bellman's dynamic integer programming algorithm as implemented in Dynapro™ to find the mathematical "optimum" redundant combinations of sensors in the Primary Loop. Here "optimum" is the highest reliability that can be obtained in a "M out of N" configuration for a specified increase in mass. We added up to 50 kg of mass for redundancy, almost an order of magnitude more than the mass of the single-string sensor suite, and checked the reliability for the "optimum" combination of sensors at that mass increment.

The curve illustrates that, while a very significant improvement in reliability -- three orders of magnitude -- can be obtained, there is a limit. Moreover, the mass penalty for improving reliability solely through redundancy is excessive.

Typically, double or triple redundant systems improve functional failure rate by a factor of two.



Finally we examine the various models to determine what lessons were learned from this analysis.

DESIGN INSIGHTS

- Design for Salvage / Repair is the single best strategy to maximize Probability of Crew Safety, Mission Success.
- Design & plan for refurbishment prior to Mars transfer orbit.
- Design to maximize robustness:
  - Maximize element interconnection.
  - Size system so return is possible with major element failure -- keep element importance < mission threatening.
  - Design to remain operating after major failures -- "Post-*Thresher*" approach to system safety.
- Use Minimum Equipment List approach to mission and abort planning.

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The first order conclusions of this study are fairly simple. (1) In a manned environment where there is a need for the system to operate near its capacity at very high reliability even late in the mission, no single reliability strategy is more effective than designing the system to allow for salvage and repair. (2) Since radiological concerns will probably preclude full scale operation of the system and "burn in" prior to launch, infant mortality will be a factor. (3) Within the basic design parameters specified there are a number of ways to combine the system components to maximize the robustness of the system. (4) The Minimum Equipment List approach to mission and abort design can be used to prevent the very stringent requirement for probability of crew safety from setting unrealistic reliability goals.

DESIGN FOR SALVAGE / REPAIR

- Ability to salvage / repair improves achievability by an order of magnitude or more.
- Keys to salvage are:
  - Modular, repairable design;
  - Element importance < mission threatening.
- Parts on hand governed by:
  - Element importance;
  - Failure probability -- Pareto rule;
  - Commonality.

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Designing the system for salvage and repair does not mean that the crew should be able or required to replace any failed part in the system. It does mean that, as a last resort, the crew should be able to replace critical, highly stressed parts, and should be able to change connections or move modules to jury rig a single working element from two or more that have failed.

PLAN FOR REFURBISHMENT

- Infant mortality failures <u>will</u> occur during Earth escape spiral "shakedown".
  - Take advantage of the shakedown opportunity, rather than be victimized by it.
    - Infant mortality is excellent predictor of random failure performance.
      - 1st month failure rate = 4 to 20 times random (mean = 7 * Random failure rate)
      - Distribution of failures among subsystems / component type approximately constant.
  - Factor in time for minor redesign and on-orbit refurbishment prior to heliocentric transfer.

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Early failures attributed to infant mortality have played a role in nearly every space system. Since the manned portion of the NEP Mars mission does not begin until after the NEP system has accumulated significant operational time, it is highly probable that some failures will have occurred before the crew boards. By designing and planning for minor refurbishment prior to the start of the manned portion of the mission, NEP planners can minimize the possibility that the crew will start the mission with less than a full redundancy complement. Moreover, since infant failures are predictors of the types of failures which will occur during the operational phase, the unmanned "shakedown cruise" can actually be used to significantly enhance the probability of mission success -- through procedure development, work-around strategies, and possibly even minor component redesign -- prior to the actual start of the mission.

MAXIMIZE ROBUSTNESS

- Element interconnection
  - Reduce / remove probability that element failure will prevent use of other elements in string.
- Element importance -- impact of element failure on system.
  - Size system elements so major element failure does not jeopardize crew return.
- "Post-*Thresher*" approach to safety -- System response to component failure determined solely by maximizing probability of returning the crew alive.
  - "Safeing system" generally = leave it alone / operating.
  - e.g.: Reactor may continue operation w/ open control loop (no instrumentation) -- but restart w/out instrumentation difficult or impossible => no shutdown (SCRAM) on instrument / control failure.

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Maximizing the robustness of the NEP system involves three elements. First, minimize the extent to which the failure of one element in a string impacts the other elements in the string. Second, maximize the extent to which an operating element can compensate for the loss of a like element. Third, ensure that no element in the system is made more important to the system than is absolutely required. For example, an irrecoverable failure in the Primary instrumentation which results in the shutdown (SCRAM) of the reactor would result in the loss of the crew in most mission phases. Almost any level of risk associated with continuing to operate the reactor, despite the failure of a critical sensor, is preferable to that alternative.

MINIMUM EQUIPMENT LIST

- Minimum Equipment List (MEL) -- the minimum set of equipment required to complete mission.
    - Varies with time in mission.
    - Points where MEL changes are abort decision points.
        - Determined by Markov or other dynamic analysis:
            - MEL state = minimum state vector that accomplishes success criteria?
        - Actual system state < MEL state => abort.
- In general, changes limiting reliability criteria from 99% $P_{(CrewSafety)}$ to 95% $P_{(Mission\ Success)}$.
    - Improves achievability by factor of 5 or more.
- May have other mission planning benefits -- staging, etc.

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Applying the Minimum Equipment List approach to the mission and system design will enhance crew safety while limiting the burden of very high system reliability goals associated with crew safety.

IMPACT OF DESIGN INSIGHTS ON ACHIEVABILITY

	AchISimple	Cummulative
· Baseline (No Cross Connection)	· $4.7 \cdot 10^{-5}$	$4.7 \cdot 10^{-5}$
· Redundancy (*2)	· $9.5 \cdot 10^{-5}$	$9.5 \cdot 10^{-5}$
· Salvage / Repair (*10)	· $5.1 \cdot 10^{-4}$	$9.5 \cdot 10^{-4}$
· Element Importance < Mission Threatening (Primary and Auxiliary Thermal not included)	· $6.8 \cdot 10^{-5}$	$1.5 \cdot 10^{-3}$
· Remain Operating After Failure (No instruments, sensors in critical failure path)	· $2.4 \cdot 10^{-4}$	$7.5 \cdot 10^{-3}$
· Minimum Equipment Set (*5.1)	· $2.9 \cdot 10^{-4}$	$3.8 \cdot 10^{-2}$

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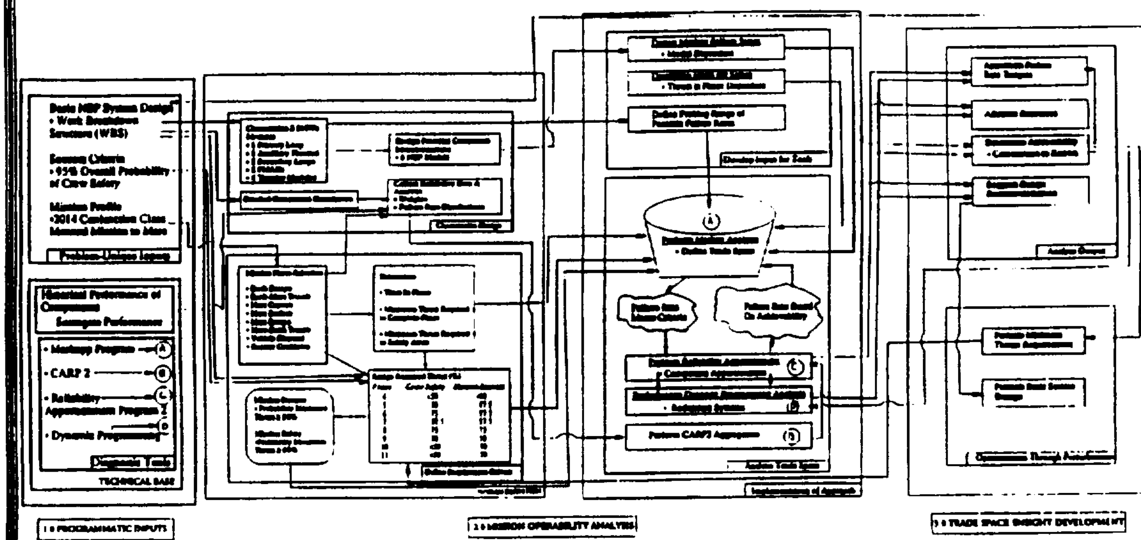
The design insights gained from analyzing the different models (design concepts) are generally not correlated, so to a significant degree their effect (if applied) is cumulative. This table shows that, taken together, the reliability enhancing design alternatives analyzed here improve the outer boundary of overall achievability for the NEP system by three orders of magnitude. Since the range of achievability index spans at least two orders of magnitude, the final AchI value of 4 * 10-3 is within the range of achievable using current technology.

This conclusion does not imply that meeting the quantitative operational reliability goals for this system will be easy, or that new technologies should not be examined for potential reliability improvements. On the contrary, several critical functions, notably heat exchangers / radiators, and sensors should be examined carefully to determine if there is an intrinsically more reliable way to accomplish the function than using existing technology.

SUMMARY



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The process and conclusions of this study have been discussed at some length. This study deliberately only examined the boundaries of the problem and the conclusions should be considered more qualitative (with extensive quantitative backup) than quantitative. We did not attempt, for example, to find optimal or near optimal component failure rate requirements. To do so would require refinement and extensive recursion of the models and tools we have demonstrated.

CONCLUSIONS
CONCEPT OF ACHIEVABILITY:

- Quantifies how far a design has to go with respect to success criteria.
  - A powerful method for
    - assessing design alternatives;
    - assessing developmental risk;
    - directing R&D effort.

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The concept of achievability was used in this study to measure the distance between the required and the attainable. This concept proved to be very powerful and is recommended for use in quantitative analyses of any performance dimension which pushes the state of the art.

CONCLUSIONS
DESIGN ALTERNATIVES:

- Several promissing design strategy alternatives were analyzed.
  - Repair / Salvage.
  - Maximizing Robustness:
    - Cross-Connection
    - Reducing element importance < mission threatening.

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This study examined only a few design alternatives within a fairly rigid basic design envelope. While several promising reliability-enhancing strategies were identified and examined, there is clearly more that could be done.

CONCLUSIONS
DESIGN ACHIEVABILITY:

- Overall achievability for simple, no cross-connection design is very low ~ 10-4 even with redundancy factored in.
- However, simple design alternatives presented here give a cummulative 3 order of magnitude increase in achievabiliity.
- While challenging, NEP achievability is within striking distance of realization.

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It is the conclusion of this study that the existing technology base could support the quantitative reliability requirements of a manned Mars mission.

NUCLEAR ELECTRIC PROPULSION

TECHNOLOGY

N93-26977

SPACE PROPULSION TECHNOLOGY DIVISION

NEP TECHNOLOGY - FY 92 MILESTONES
(NASA LERC)

THRUSTERS

- o ESTABLISH 100 H TEST CAPABILITY FOR 100 KW MPD THRUSTERS
- o DEMO LIGHTWEIGHT 20-KW KRYPTON ION THRUSTER
- o OPTIMIZE THE DESIGN OF LOW-MASS POWER PROCESSOR TRANSFORMERS

NEP FACILITIES

- o COMPLETE EPL'S TANK 5 CRYOPUMP UPGRADE

Presented by: Jim Sovey
NASA Lewis Research Center

SPACE PROPULSION TECHNOLOGY DIVISION

NEP TECHNOLOGY - FY92 RESOURCES
(NASA LERC)

THRUSTERS

- o $129K, MPD THRUSTER TECHNOLOGY
- o $18K, TANK 5 CONSUMABLES
- o $23K, ION OPTICS
- o $30K, WITH $35K (BASE R&T) FOR PPU MAGNETICS, UNIVERSITY OF WISCONSIN

NEP FACILITIES

- o $40K, TANK 5 CRYOPUMP UPGRADE

NEP - ION THRUSTER TECHNOLOGY
(NASA LERC)

ACCOMPLISHMENTS.............................THRUSTER

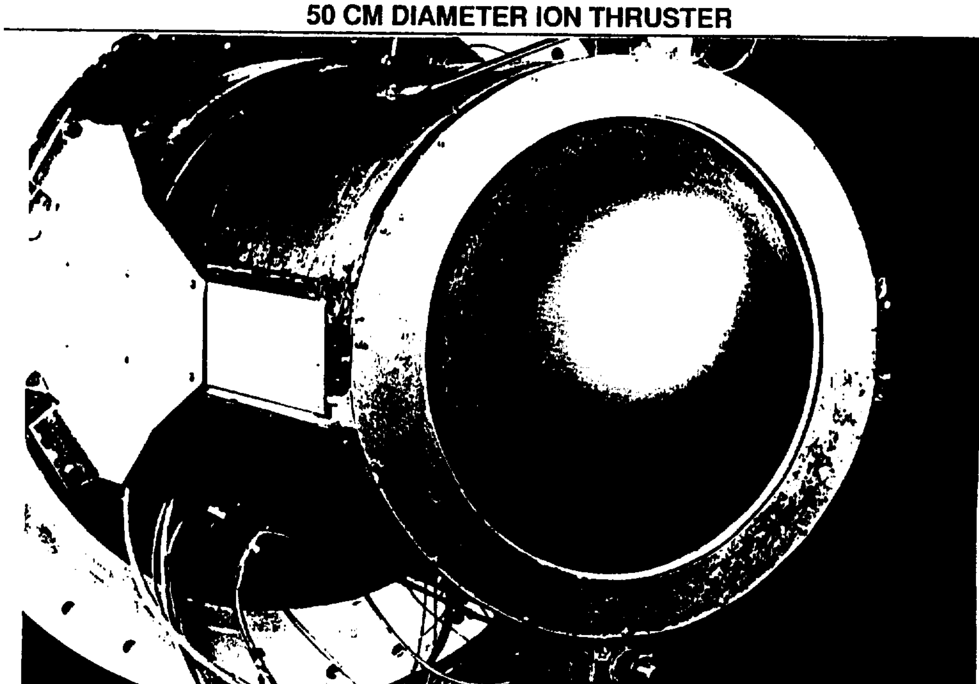
- o PERFORMANCE OF VIBRATION WORTHY 50-CM DIAMETER THRUSTER DESIGN COMPARABLE TO SOA DESIGNS
- o LIGHTWEIGHT 30-CM THRUSTER ASSEMBLED UNDER BASE R&T PROGRAM
- o 16 PAIRS OF DISHED ACCELERATOR GRIDS ARE NOW BEING FABRICATED.......... TESTING SCHEDULED FOR FEBRUARY 1993.

POWER PROCESSOR

- o ANALYSIS OF FULL-BRIDGE, LOW VOLTAGE DC/DC CONVERTER COMPLETE
- o DETAILED ANALYSIS, TRADE-OFFS, AND DESIGN OF TRANSFORMERS COMPLETE
- o FOLLOW-ON WILL PROVIDE CONVERTER HARDWARE

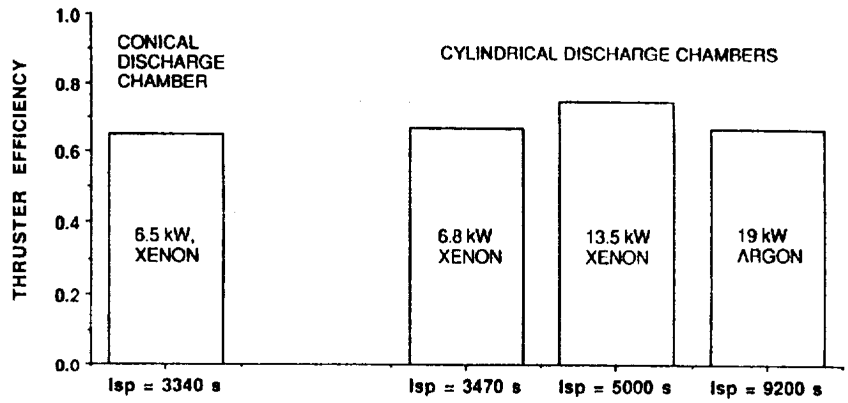
ATD AEROSPACE TECHNOLOGY DIRECTORATE — SPACE PROPULSION TECHNOLOGY DIVISION — NASA Lewis Research Center

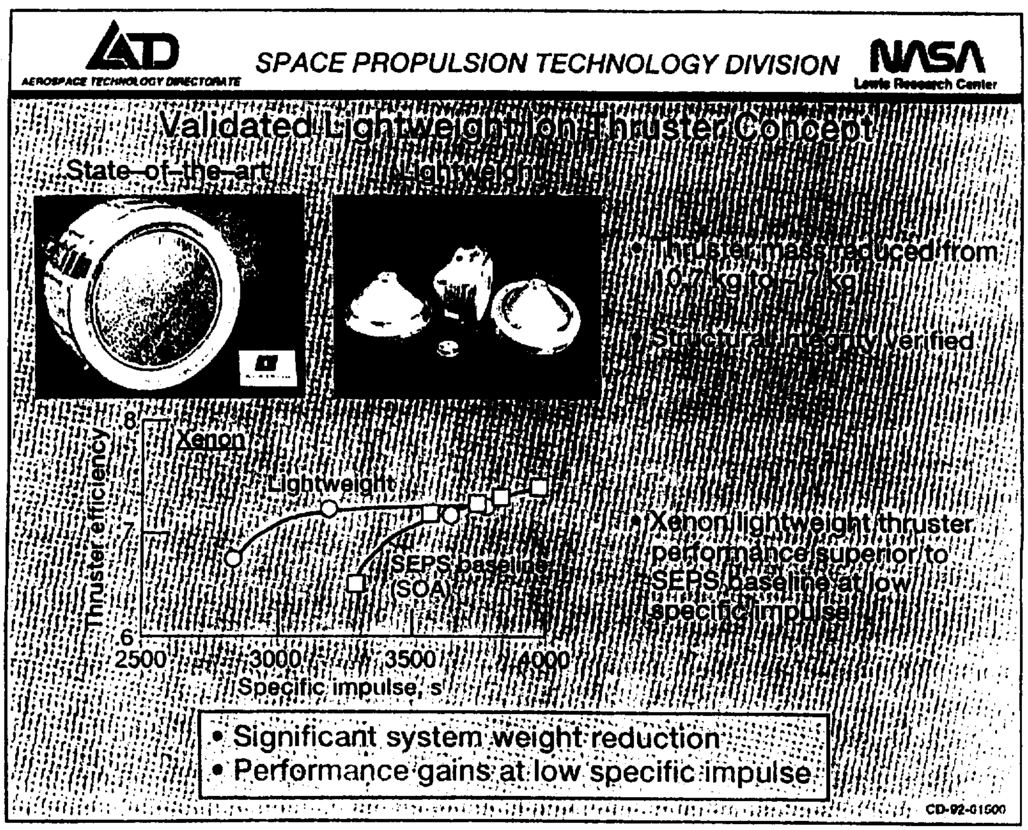
50 CM DIAMETER ION THRUSTER



50 CM DIAMETER ION THRUSTER PERFORMANCE

VIBRATION WORTHY CONICAL DIACHARGE CHAMBER DESIGN HAS PERFORMANCE COMPARABLE TO SOA CYLINDRICAL DESIGN





LERC/JPL COORDINATED ION PROPULSION PROGRAM SUPPORTED UNDER BASE R&T STARTING FY93

LERC/JPL COORDINATED ION PROPULSION TECHNOLOGY PROGRAM

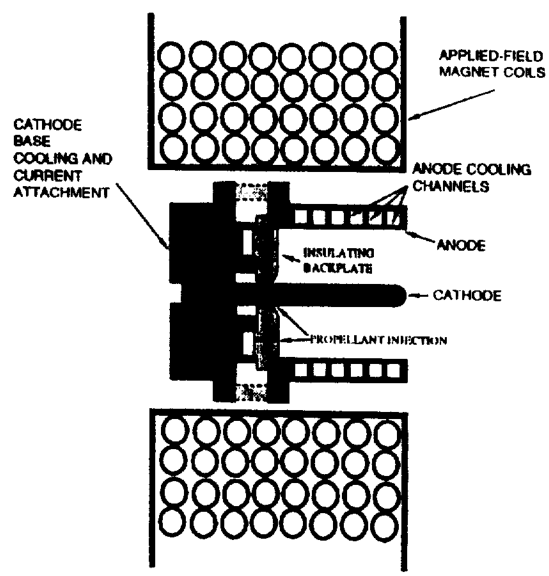
	FY93	FY94	FY95	AGENT (L: LERC, J: JPL)
1. THRUSTER DEVELOPMENT				
o LIGHTWEIGHT 30 CM	BOEING ^	^ AEROSP. ^ VIB, WEAR		L
o POWER CONSOLE DEL.	^ BOEING	^ AEROSP. ^ CSTAR		L
o SEG. THR. SYS. EVAL.	^ COMPLETE			J
o 5 KW SEG. THRUSTER	EXP EVAL ^ ^ 1-SEG WEAR	^ 5-KW WEAR		J
o LIGHTWEIGHT 50 CM	^ Hi-Isp, 25 kW	^ FAB COMPL.		L
o DOWNSELECT THR. FOR SEP OR NEP			^	L, J
2. CATHODE DEVELOPMENT				
o PROTOCOLS		^ DEFINE	^ LIFE	L (SSF)
o DIAGNOSTICS/MODELS	^ THERMAL	^ PLASMA		L (W, MPD)
3. GRID DEVELOPMENT				
o CARBON-CARBON		^ 15 CM	^ 30 CM EVAL.	J
o 30 & 50 CM MOLY		^ HI PERV.	LOW WEAR ^	L
o DOWNSELECT			^	L, J
o LASER DIAGNOSTICS	CONTOURS ^	^ EVAL. HOLOGR.		L
o CHARGE EXCH. STUDY	^ PRELIM MODEL	^ IMPROVED MODEL	LIFE PRED ^	L, J
4. POWER PROCESSOR				
o COMPONENT TECH.	LITE MAG. ^	^ HV INVERTER	HI POWER ^	L
o SIMPLIFIED PPU	LAB DEMO ^	LV BB'S ^ BB DEMO ^	^ COMPL. INTEG.	L
o PACKAGED PPU		SOW ^	^ATP ^	L
5. BB FEED SYSTEM LIFE				J
6. DIAGNOSTICS				
o THRUST STAND	^ COMPLETE			L, J
o BEAM DIAGNOSTICS		^ CHG. STATE	^ T-VECTOR S/C EFF. ^	L, J

NEP - MPD THRUSTER TECHNOLOGY

FY 92 Milestone: Establish 100 hr test capability at 100 kW

Background:

- Base Technology Program supported extensive testing of
  - argon MPD thrusters to 240 kW
  - hydrogen thrusters to 100 kW
- Extensive performance data base established



Applied-Field MPD thruster schematic
Anode and cathode lengths of 7.6 cm. Cathode radius =0.64 cm, anode radii of 2.54, 3.81, and 5.1 cm. Thrust exit plane was even with solenoid exit plane.

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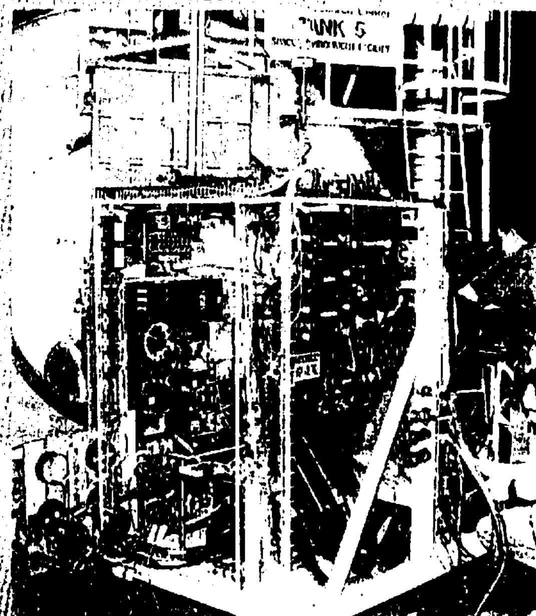
High Power Electric Propulsion
MPD Thruster Technology

- New facility established
  - Helium cryopumping
  - 350 kW power
  - Plume diagnostics
  - Electrode power diagnostics
- MPD thruster tested to 240 kW



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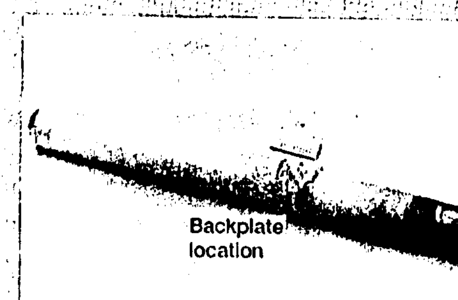
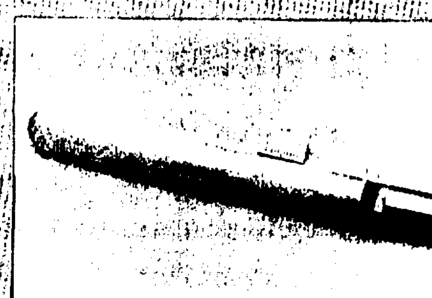
MPD Thruster Lifetime
Cathode Erosion



- Low purity Argon (99.995%)
- No vacuum purge



- High purity Argon (99.999%)
- With vacuum purge

Major cause of cathode erosion eliminated

CD-92-61459

Applied-Field MPD Thruster Geometry/Operation Point Selection

Cathode

- Testing showed hollow cathode temperature was ~ 1000 K below rod cathode

Boron Nitride Backplate

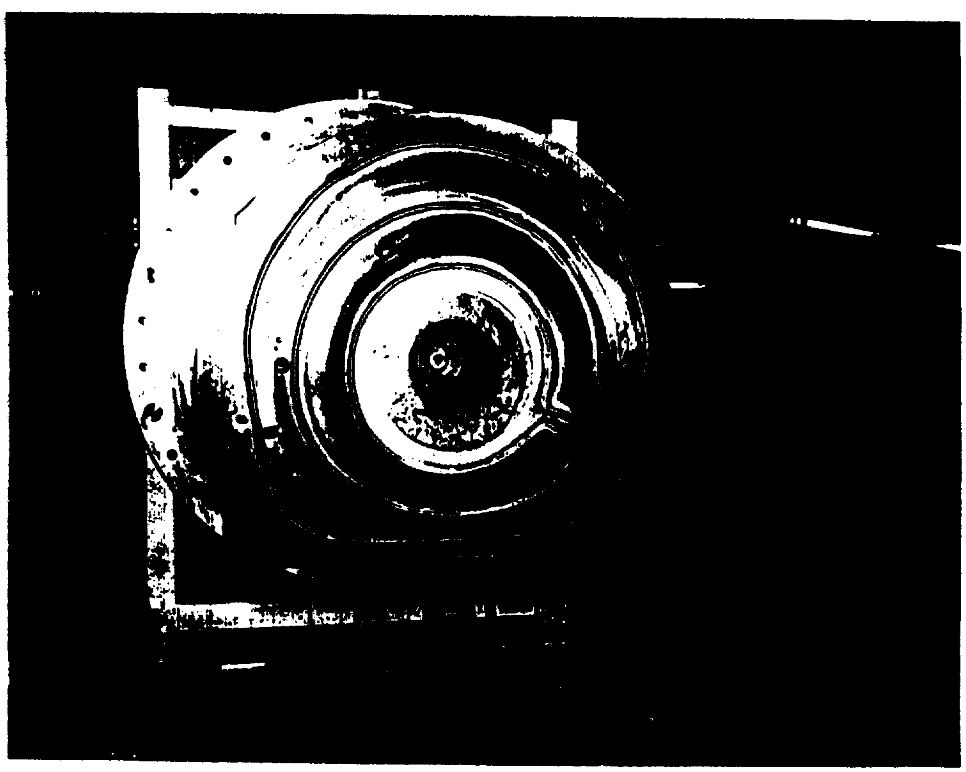
- Increasing cathode-to-backplate separation improved insulator life

Anode

- 5.1 cm radius, 15 cm long anode to reduce power density

Operating point

60 kW: 1400 amps, 47 volts
0.14 g/s argon



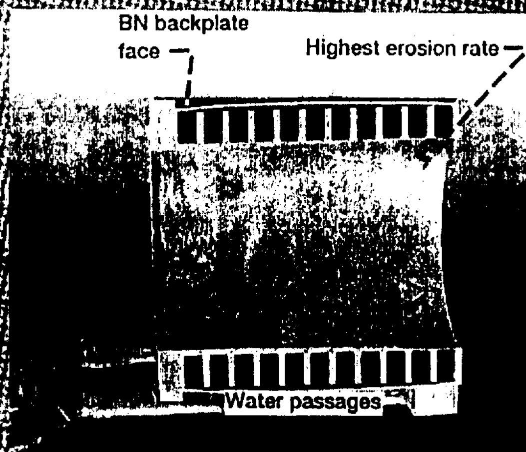
AEROSPACE TECHNOLOGY DIRECTORATE

SPACE PROPULSION TECHNOLOGY DIVISION

NASA Lewis Research Center

MPD Thruster Lifetime
Anode Erosion

- Extended test conducted to identify first-order lifetime limits



- Sputtering by argon propellant identified as major cause of erosion
  - fundamental limit for Isp's of interest

Program emphasis shifted toward light propellants and refractory metal anodes

CD-92-61460

N93-26978

AEROSPACE TECHNOLOGY DIRECTORATE **POWER TECHNOLOGY DIVISION** NASA Lewis Research Center

NUCLEAR PROPULSION

TECHNICAL INTERCHANGE MEETING

OCTOBER 20-23, 1992

Power Management and Distribution Technology

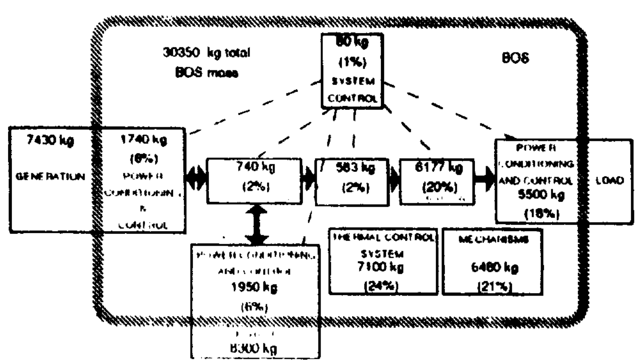
John Ellis Dickman

OCTOBER 21, 1992

APPLICATIONS AND SYSTEMS DEFINITIONS

OBJECTIVES:

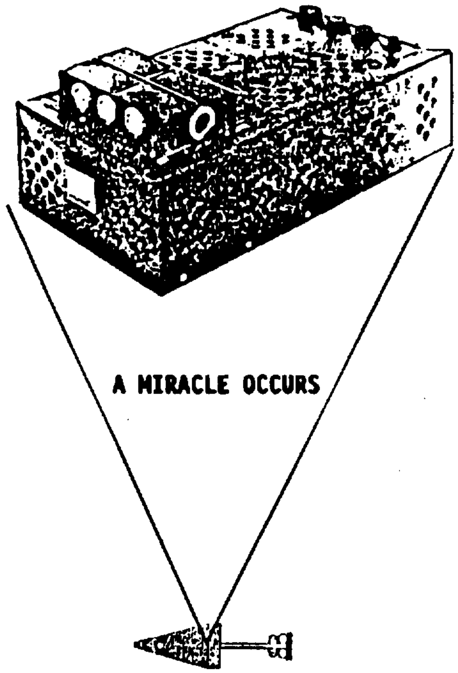
- DEFINE PMAD TECHNOLOGY REQUIREMENTS FOR ADVANCED SPACE MISSIONS, e. g. SSF EVOLUTION, LUNAR/MARS BASES, ADVANCED SPACECRAFT, PLATFORMS AND VEHICLES.



ACCOMPLISHMENTS:

- DEVELOPED MASS DATABASE OF EXISTING AND SOA SPACE SYSTEMS
  - PMAD MASS RANGES FROM 40 TO > 220 kg/kW
  - NEW CLASS OF "SPACE UTILITY" POWER SYSTEMS EVOLVING
  - "BALANCE OF SYSTEM" (PMAD, THERMAL, MECHANICAL) ARE MAJOR MASS CONTRIBUTORS (e. g. BOS IS 2/3 OF SSF POWER SYSTEM MASS)

POWER TECHNOLOGY DIVISION



POWER PROCESSING, CONTROLS, AND DISTRIBUTION

STATE-OF-THE-ART

25-100 KG/kWe

PILOTED MARS NEP VEHICLE

TOTAL

5-10 KG/kWe

POWER TECHNOLOGY DIVISION

HIGH PERFORMANCE COMPONENTS

- **TECHNOLOGY DEVELOPMENT CHALLENGES**
  - To establish the technology base in power electronics that will enable or significantly enhance future NASA missions
    - Survive adverse environments
    - Improved performance, mass, and reliability
    - Enable advanced system architectures

- **TECHNOLOGY DEVELOPMENT APPROACH**
  - Assemble complete program out of individual programs focused on customer needs
    - Base R&T: High temperature components
    - Nuclear Propulsion High temperature components
    - CSTI HCP: Radiation tolerant power switches; Fiber optic sensors
    - OSMQ, T. Standards: NASA Space Wiring
  - Form strategic alliances with other component development efforts
  - Build commercial capability in advanced parts

POWER TECHNOLOGY DIVISION

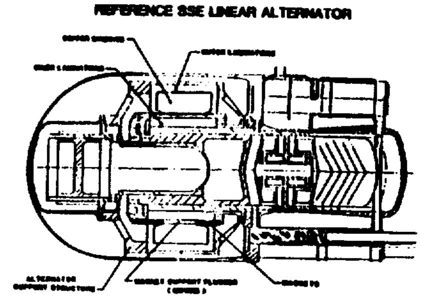
HIGH CAPACITY POWER/CSTI (586-01)

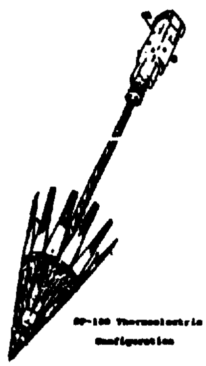
OBJECTIVE:

DEVELOP ENABLING ELECTRIC COMONENT AND CIRCUIT TECHNOLOGY FOR SP-100

- > 600 K
- > 1 mRAD GAMMA, 10^{13} NEUTRON FLUENCE
- FAULT TOLERANT
- STIRLING LINEAR ALTERNATOR

REFERENCE SSE LINEAR ALTERNATOR





APPROACH:

o INVESTIGATE 10-100 kW INVERTER/CONVERTER CIRCUITS
- MAPHAM SWITCH COMPARISON (IN HOUSE)
- CASCADE SCHWARTZ INVERTER (U. TOLEDO)

o COMPONENTS
- DETERMINE DEGRADATION OF H.P. S.S. SWITCHES IN HIGH TEMPERATURE AND NUCLEAR ENVIRONMENTS
- CHARACTERIZE AND DEVELOP TRANSMISSION LINES, CAPACITORS AND TRANSFORMERS/INDUCTORS

CSTI HIGH CAPACITY POWER

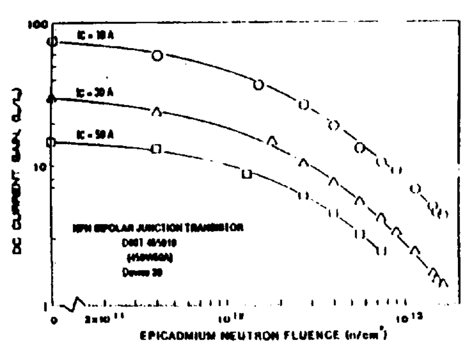
NEUTRON & GAMMA RAY EFFECTS ON SOLID STATE POWER SWITCHES

OBJECTIVE: DETERMINE AND ASSESS THE EFFECTS OF GAMMA RAYS AND NEUTRONS ON COMMERCIAL AND DEVELOPMENTAL-TYPE SOLID STATE SWITCHES

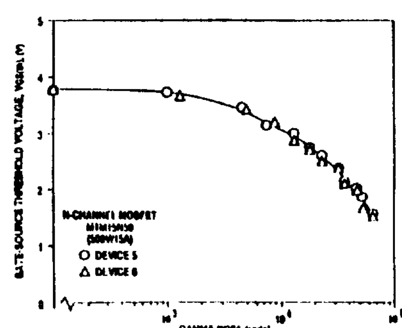
APPROACH: MEASURE SENSITIVITY OF SWITCH PARAMETERS TO GAMMA AND NEUTRON IRRADIATION UNDER IN-SITU CONDITIONS AT ROOM AND ELEVATED TEMPERATURES

STATUS: POWER BJTs, MOSFETs AND SITs TESTED AND EVALUATED TO NEUTRON FLUENCES $\geq 10^{13}$ n/cm^2 AND GAMMA DOSES $\geq 10^6$ rads

CURRENT GAIN @ VCE = 2.5V vs EPICADMIUM NEUTRON FLUENCE
FLUX = 7.6×10^8 n/cm^{2}s FLUENCE = 1.7×10^{13} n/cm^2



GATE-THRESHOLD VOLTAGE vs GAMMA DOSE
DOSE RATE = 6.8 krad/hr DOSE = 73 krads



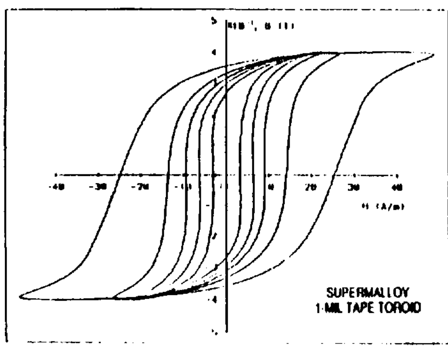
HIGH TEMPERATURE, HIGH FREQUENCY SOFT MAGNETIC MATERIAL'S CHARACTERIZATION

OBJECTIVE: DETERMINE AND ASSESS THE COMBINED EFFECTS OF TEMPERATURE, FREQUENCY AND EXCITATION WAVEFORM ON COMMERCIAL SOFT MAGNETIC MATERIALS

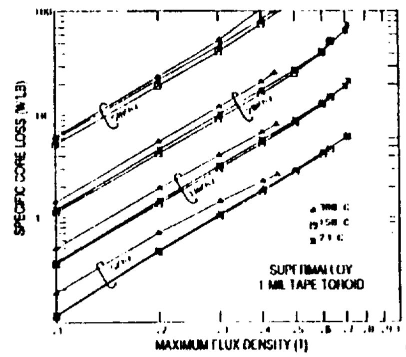
APPROACH: DEVELOP TEST SYSTEM TO ACCURATELY MEASURE, RECORD AND PLOT SPECIFIC CORE LOSS AND DYNAMIC B-H HYSTERESIS LOOPS TO TO 300C AND 50 kHz UNDER SINE- AND SQUARE-WAVE VOLTAGE EXCITATION

STATUS: 80-20 Ni-Fe, 50-50 Ni-Fe, 3% Si-Fe AND AMORPHOUS MAGNETIC ALLOYS TESTED UNDER SINEWAVE VOLTAGE EXCITATION TO 300C AND $f \geq 20$ kHz

FREQUENCY-CLUSTER B-H LOOPS AT B_M = 0.4 T AND T = 300C
f = 1 kHz (INNER LOOP), 5, 10, 20 AND 50 KHZ (OUTER LOOP)
SINEWAVE VOLTAGE EXCITATION



SPECIFIC CORE LOSS vs FLUX DENSITY, FREQUENCY & TEMPERATURE
SINEWAVE VOLTAGE EXCITATION



GES90 010 3

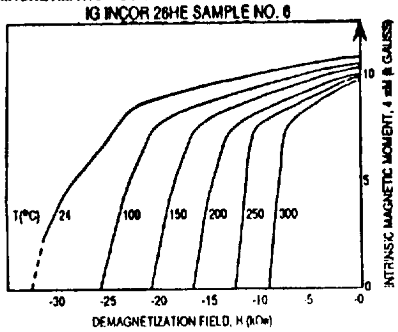
HIGH TEMPERATURE RARE EARTH PERMANENT MAGNET CHARATERISTICS

OBJECTIVE: CHARACTERIZE RARE-EARTH PERMANENT MAGNETS TO 300°C AND INVESTIGATE LONG-TERM AGING EFFECTS

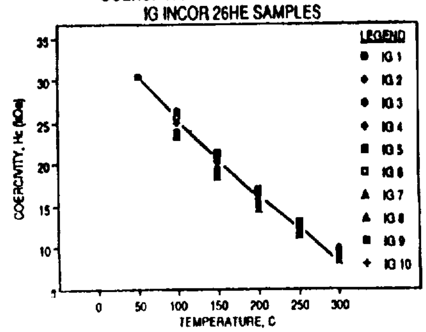
APPROACH: MEASURE REVERSIBLE, IRREVERSIBLE, AND PERMANENT LOSS OF MAGNETIC PROPERTIES DUE TO SHORT AND LONG TERM EXPOSURE TO ELEVATED TEMPERATURES

STATUS: 50 SAMPLES OF Sm_2Co_{17} FROM 5 VENDORS (10 PER VENDOR) TESTED TO 300°C TO INVESTIGATE SHORT-TERM TEMPERATURE EFFECTS

DEMAGNETIZATION CURVES AT SELECTED TEMPERATURES
IG INCOR 26HE SAMPLE NO. 6



COERCIVITY VERSUS TEMPERATURE
IG INCOR 26HE SAMPLES



GES90 006 1

FIBER-OPTIC SENSORS FOR POWER DIAGNOSTICS

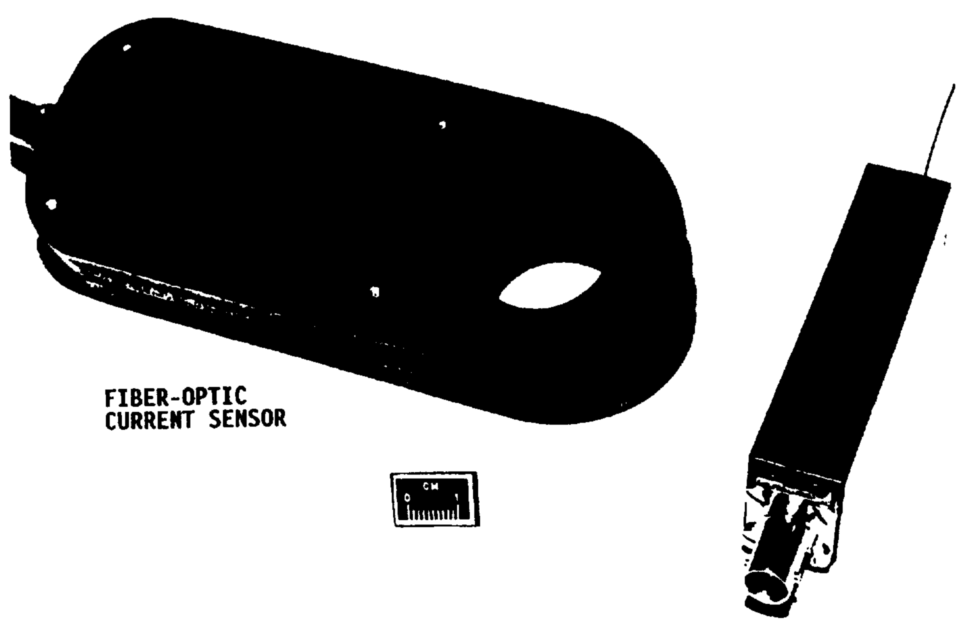
SHOWN
- Fiber Optic Current Sensor and Voltage Sensor.

OBJECTIVE
- To provide accurate electrical sensors with very high electrical isolation and immunity to electromagnetic interference (EMI).

ACCOMPLISHMENTS
- Developed fiber-optic current sensor with very high EMI immunity and electrical isolation. Operation between - 65 to + 125° C. Survived 17g vibration tests.
- Developed fiber-optic voltage sensor. Working to reduce sensitivity to vibration for voltage sensor.

BENEFITS
- Accurate electrical measurements at locations somewhat remote from central electronics, such as in aircraft wings or in conjunction with electromechanical actuators. High EMI immunity. Very high isolation with low mass. Very applicable to industrial operations.

APPLICABLE MISSIONS
- Lunar and Mars surface power, aircraft (especially with electro-mechanical actuators), Vehicle Health Management systems, electric utility industry.



FIBER-OPTIC
CURRENT SENSOR

FIBER-OPTIC
VOLTAGE SENSOR

POWER TECHNOLOGY DIVISION

AEROSPACE TECHNOLOGY DIRECTORATE — NASA Lewis Research Center

NASA WIRING TECHNOLOGY

GOAL: DEVELOP SAFE AND RELIABLE POWER WIRING SYSTEMS FOR FUTURE NASA SPACE MISSIONS

APPROACH:

- o EVALUATE POSSIBLE METHODS OF ACCOMPLISHING GOAL
  - QUANTIFY/UNDERSTAND BREAKDOWN MECHANISMS IN PRESENT WIRING SYSTEMS
  - ASSESS LIMITATIONS OF PRESENT WIRING SYSTEMS FOR PROPOSED MISSIONS
  - IDENTIFY AND EVALUATE CANDIDATE ADVANCED MATERIALS AND WIRE DESIGNS
  - RESOLVE WIRING SYSTEM ISSUES
- o PRIORITIZE APPROACHES: COST, LIMITATIONS, ETC.
- o IMPLEMENT DEVELOPMENT PROGRAM

POWER TECHNOLOGY DIVISION

AEROSPACE TECHNOLOGY DIRECTORATE — NASA Lewis Research Center

HIGH TEMPERATURE POWER ELECTRONICS

- REQUIREMENTS, TRADE STUDIES AND GOALS DEFINITION:
  - Define system requirements and applications environments for NASA space missions
  - Assess system mass and volume drivers
  - Identify opportunities and benefits of specific technology developments
- HIGH-TEMPERATURE CHARACTERIZATION:
  - Experimentally determine the efficiency, reliability, and upper limit on operating temperature for advanced power electronic components as a function of power level.
- HIGH EFFICIENCY, ELEVATED TEMPERATURE POWER ELECTRONICS:
  - Establish a high efficiency, elevated operating temperature advanced power electronics technology base
  - Build a 95% efficient inverter power circuit operating at 125°C

POWER TECHNOLOGY DIVISION

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HIGH TEMPERATURE POWER ELECTRONICS PROGRAM

COMPONENTS R&D:

INDUCTORS

- DESIGNED AND TESTED MOLY-POWDERED-PERMALLOY CORE (MPP) INDUCTORS VERSUS FREQUENCY AND TEMPERATURE.
- INDUCTORS PERFORMED SATISFACTORILY UP TO 200° C, UNDER LOW BIAS @ 50 Hz-100 kHz.
- PROCUREMENT OF LARGE MPP CORES IS COMPLETE.
- TESTING TECHNIQUES UNDER FULL BIAS ARE BEING INVESTIGATED.

TRANSFORMER

- DEVELOPMENT OF 200°C COAXIALLY-WOUND TRANSFORMER IS UNDERWAY AT THE UNIVERSITY OF WISCONSIN.

CAPACITORS

- THERMAL AGING TESTS (200°C, 2000 HOURS) WITHOUT ELECTRICAL BIAS OF CERAMIC, TEFLON CAPACITORS ARE COMPLETED. LIFE TESTING UNDER FULL BIAS IS UNDERWAY.
- MOUNTING OF THERMOCOUPLES ON CAPACITORS IS COMPLETE FOR FUTURE TEMPERATURE RISE MEASUREMENTS.
- PROCUREMENT OF POWER CAPACITORS IS UNDERWAY.

SWITCHES

- DEVELOPMENTAL EFFORTS OF HIGH TEMPERATURE SWITCH TECHNOLOGY ARE BEING MONITORED.

POWER TECHNOLOGY DIVISION

AEROSPACE TECHNOLOGY DIRECTORATE — NASA Lewis Research Center

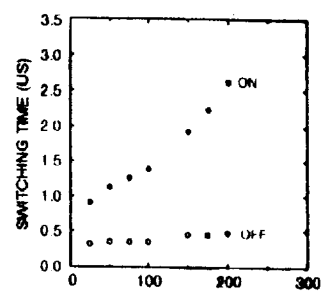
200°C-BASEPLATE ELECTRONICS

SURVIVES SEVERE ENVIRONMENTS AND LIGHTENS RADIATORS

GOAL: BUILD & TEST ASSEMBLY

- ACHIEVABLE (100°C > SOA)
- UNCOVERS MISSING TECHNOLOGY
- EXCEEDS LUNAR TEMPERATURE (130°C)
- REDUCES RADIATOR AREA > 2
- BROAD SPINOFFS

H. T. TEST LAB



MCT SWITCHING TIMES

- SUNY/AUBURN GRANTS INITIATED
- COMPONENTS TESTED
  - MCT
  - CAPACITORS
  - INSULATION
- LABS SET UP
- CUSTOM COMPONENTS ORDERED

H. T. COMPONENT CHARACTERIZATION

SHOWN:
- 200°C inductor, transformer and capacitors

OBJECTIVE:
- Experimentally determine the efficiency, reliability and upper limits on operating temperature for advanced power electronic components as a function of power level

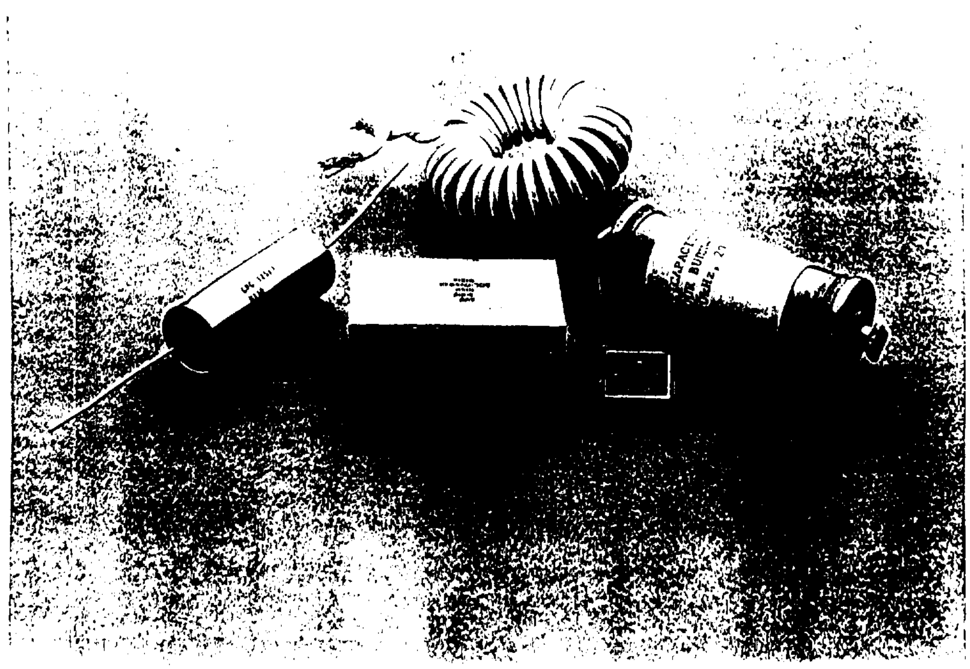
APPROACH:
- Acquire SOTA commercially available and/or developmental power electronic components
- Test performance as a function of temperature
- Conduct aging studies at maximum acceptable temperature. Repeat performance tests

ACCOMPLISHMENTS:
- Acquired and completed performance testing of three types of capacitors to 200°C. Aging tests are on-going
- Built and completed performance test on four types of inductors to 200°C
- Completed high temperature characterization of power switching devices

BENEFITS:
- Simplifies and lightens thermal management system
- Enhanced tolerance of hostile environments
- Improved reliability and efficiency

MISSION:
- Lunar base, advanced platforms; nuclear & solar-dynamic power
- Engine integrated electronics

C-91-10310



POWER TECHNOLOGY DIVISION — NASA Lewis Research Center

H.T. COAXIAL TRANSFORMER

SHOWN:
- Coaxially wound transformer for 50 kW converter
- 50 kW soft switched, dc-dc converter

OBJECTIVE:
- Develop very light, very low loss topologies and components for high power space systems (Megawatt Inverter Program)
- Develop high temperature coaxial transformer

APPROACH:
- Grants to U. Wisconsin

ACCOMPLISHMENTS:
- Developed and demonstrated the coaxially wound transformer, a new concept that improves the converter's power density
- Demonstrated 0.24 kg/kW converter
- Grant underway for development of high temperature transformer
- Applied to induction heating on robotic production lines (Miller Electric Co.)
- Applied to zero-force power transfer into μgravity experiment pallet

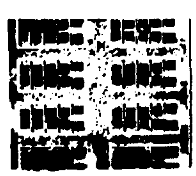
BENEFITS:
- Lighter weight, higher efficiency power electronics, and simplified thermal management
- Unique features allow design innovations

National Aeronautics and Space Administration Lewis Research Center — INSTRUMENTATION & CONTROL TECHNOLOGY DIVISION — NASA

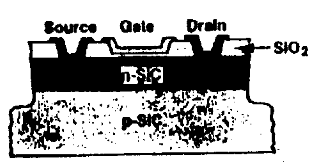
SILICON CARBIDE MOSFET

Milestone: Develop and demonstrate a high temperature, (400 °C), 6H-SiC metal-oxide-semiconductor field effect transistor (MOSFET)

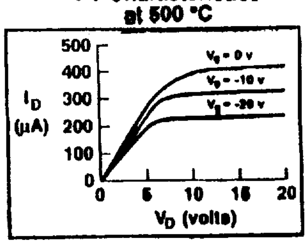
MOSFET Array



SiC MOSFET Structure



I-V Characteristics at 500 °C



Accomplishments: A depletion-mode silicon carbide MOSFET has been developed and successfully demonstrated at an operational temperature of 500 °C.

Benefits: Silicon carbide MOSFETs (switches) provide the most basic active electronic device from which integrated circuits can be developed.

NUCLEAR PROPULSION

TECHNICAL INTERCHANGE MEETING

OCTOBER 20-23, 1992

RADIATOR TECHNOLOGY

ALBERT J. JUHASZ

OCTOBER 21, 1992

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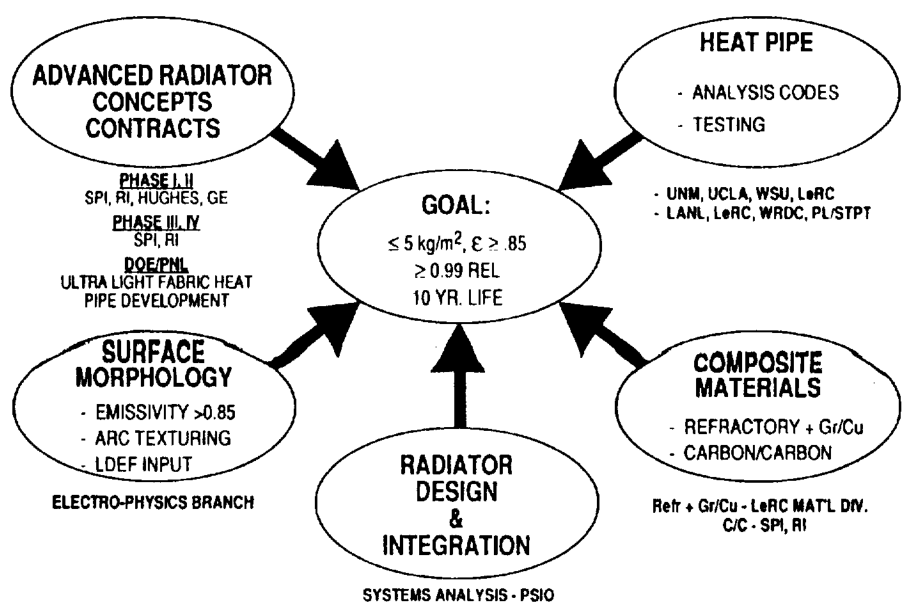
HIGH CAPACITY POWER

NASA

CSTI HIGH CAPACITY POWER - THERMAL MANAGEMENT PROJECT ELEMENTS



AJ/92-002 2

EXTERNAL PROGRAM SUPPORT FOR FY92

FUNDING SOURCE/AMOUNT	FOCUSED TASK
1. NASA PHASE I SBIR (50 K)	R&D ON HEAT PIPE WORKING FLUID ALTERNATIVES TO Hg (500K - 700K) CANDIDATES: SULFUR-IODINE; ORGANICS
2. AIR FORCE PL/STPT (50 K)	HEAT PIPE CODE DEVELOPMENT - WSU & VALIDATION
3. SDIO (30 K)	HEAT PIPE CODE DEVELOPMENT - UNM & VALIDATION
4. NEP PROGRAM (40 K)	HIGH CONDUCTIVITY FIN DEVELOPMENT VIA INTEGRAL WOVEN FIBER APPROACH
(36 K)	ALTERNATE HEAT PIPE WORKING FLUIDS RESEARCH FOR 500K - 700K RANGE

AJJ-91q 01 4

HIGH CAPACITY POWER NASA

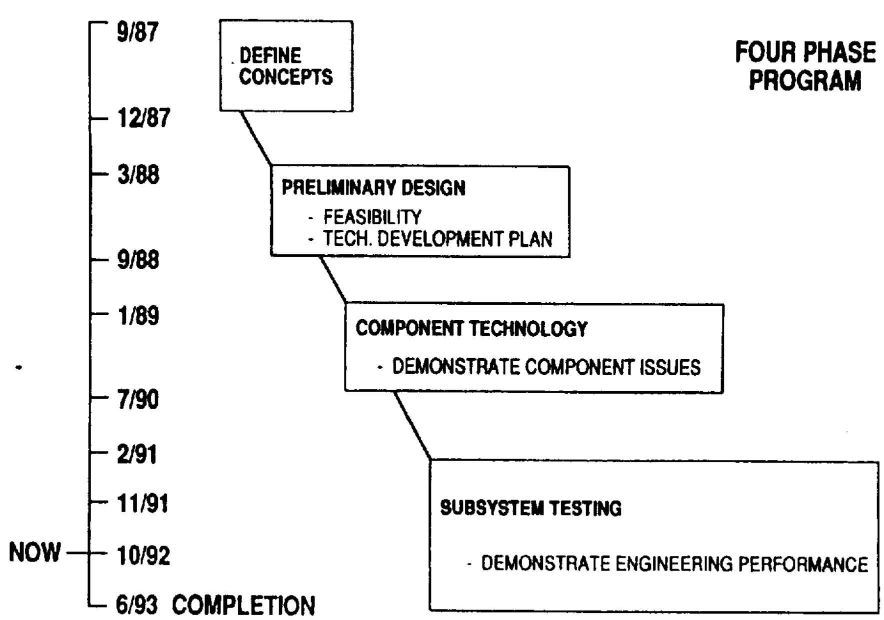
SP-100 ADVANCED RADIATOR CONCEPTS PROJECT

ADVANCED RADIATOR CONCEPTS PROJECT OBJECTIVES

- **IDENTIFY ADVANCED SPACE RADIATOR CONCEPTS TO MEET THE FOLLOWING REQUIREMENTS**
  - TECHNICAL GOALS
    - **SPECIFIC MASS OF 5 kg/m^2; EMISSIVITY $\geq$0.85**
    - **0.99 RELIABILITY**
    - **10 YEAR LIFE**
  - APPLICATIONS
    - **RADIATORS SIZED FOR POWER SYSTEMS WITH A 2.5 MWt HEAT SOURCE**
      - THERMOELECTRIC POWER SYSTEM AT 875 K (Area = 106 m^2, Qr = 2.4 MW$_t$; P = 100 kWe)
      - STIRLING ENGINE POWER SYSTEM AT 600 K (Area = 335 m^2; Qr = 1.7 MW$_t$; P = 800 kWe)
- **DEVELOP THE TECHNOLOGY NEEDED FOR THE IDENTIFIED CONCEPTS BY:**
  JANUARY 1992 (ORIGINAL PLAN)
  JUNE 1993 (NEW PLAN)

AJJ92-002 3

ADVANCED RADIATOR CONCEPTS PROJECT FLOW CHART



FOUR PHASE PROGRAM

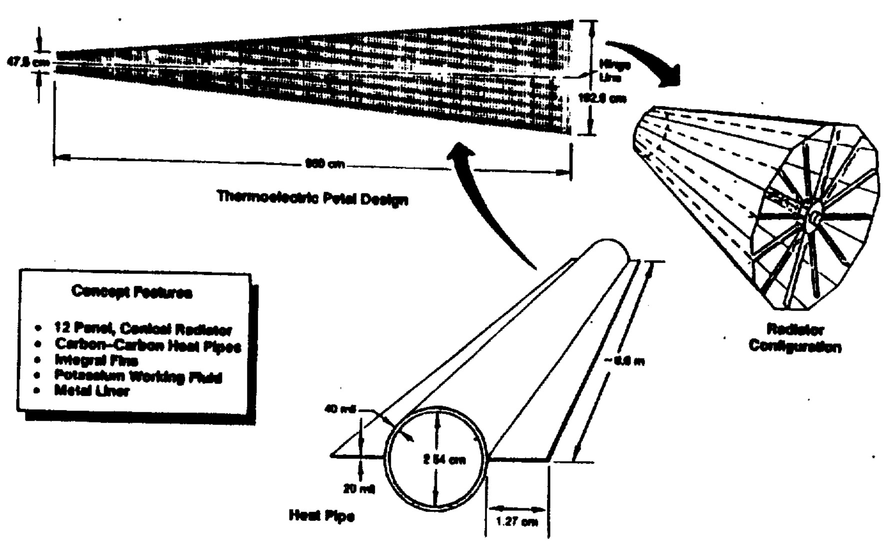
AJJ92-002 4

ADVANCED RADIATOR CONCEPTS
ROCKWELL APPROACH

- TWO-SIDED FLAT PLATE RADIATOR PANELS
- MONOLITHIC C-C PIPE CONSTRUCTION
- EFFORT EMPHASIZING MATERIALS; GEOMETRY SECONDARY
- TECHNOLOGY IMPACT
  - INTEGRAL C-C PIPE/FIN CONSTRUCTION
  - CVD METAL LINED C-C TUBES
- BRAZE DEVELOPMENT FOR METAL LINED C-C TUBES
  - C-C COMPOSITE HEAT PIPE FABRICATION & TESTING

AJJ92-002 7

SP–100 Advanced Radiator Concept



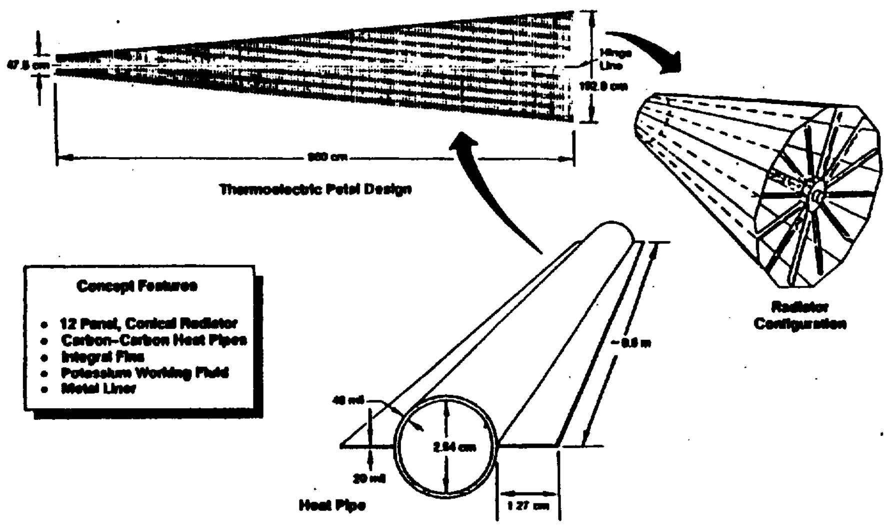
Rockwell International
Rocketdyne Division

V6020-5

HIGH CAPACITY POWER

NASA

SP-100 Advanced Radiator Concept



Rockwell International
Rocketdyne Division

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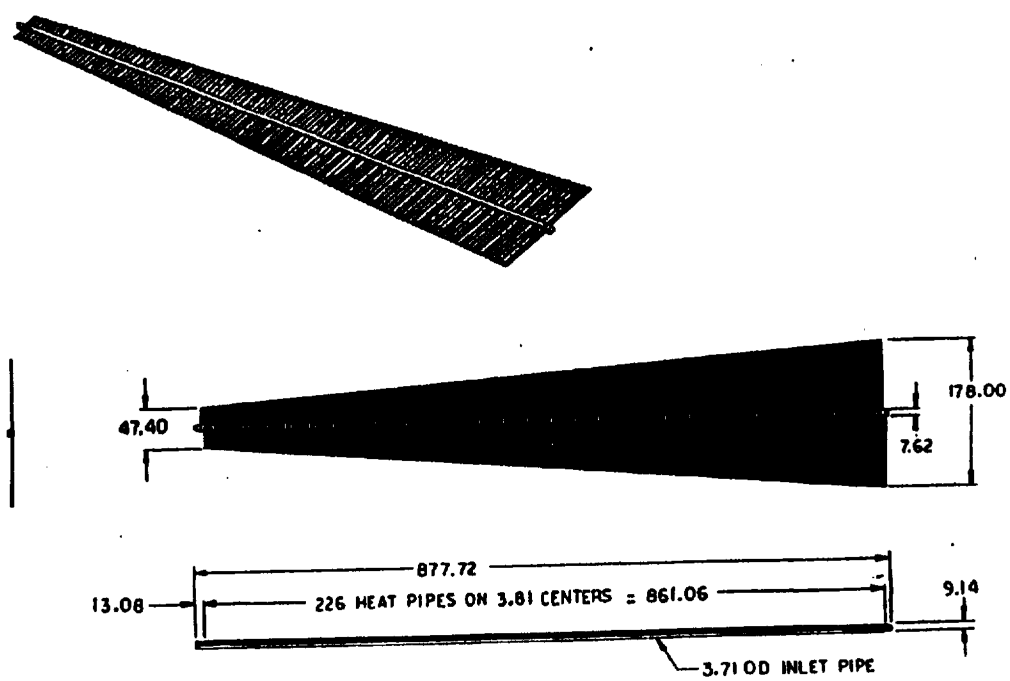
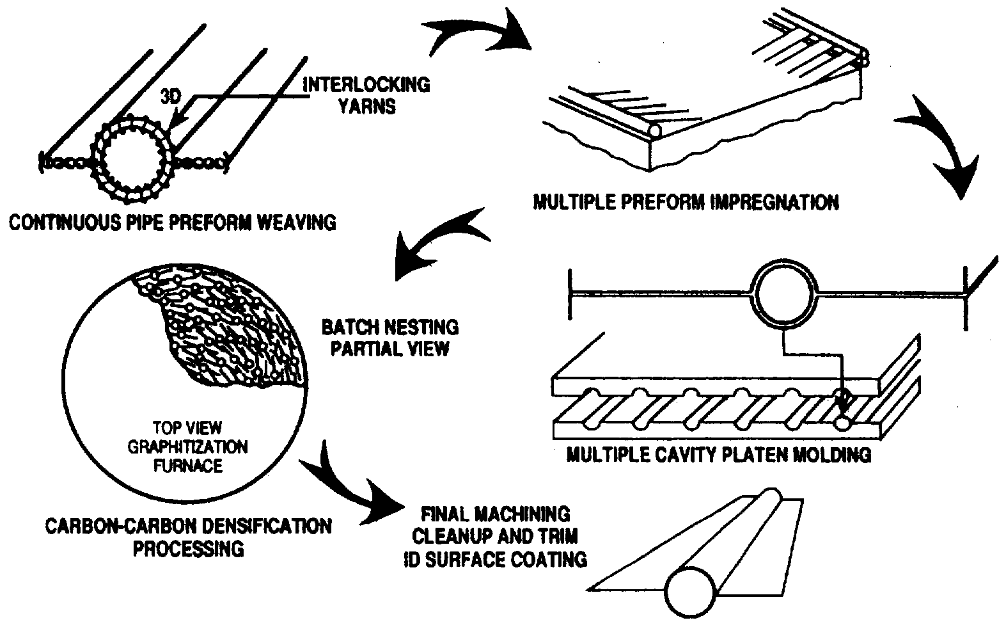
HIGH CAPACITY POWER

NASA



RADIATOR WITHOUT BUMPER ARMOR

ARC - ROCKWELL CONCEPT



Criteria for Selection of Braze Alloys

- Brazing temperature (generally 22-28K above T_L) must be above maximum operating temperature (875K) of heat pipe to ensure in-service life
- Braze alloy compatibility with carbon-carbon substrate & thin-metallic liner
- Good wettability of carbon-carbon & metallic liner
- Longevity & stability

7 Commercial Braze Alloys Evaluated

Alloy	Composition (wt %)	Foil Thickness (in.)	$T_{liquids}$ (°K)	T_{braze} (°K)
Copper ABA	92.7 Cu/3 Si/2 Al/2.25 Ti	0.002	1297	1311
Silver ABA	Bal Ag/5 Cu/1.25 Ti/1Al	0.002	1185	1200
Palcusil 15	65 Ag/20 Cu/15 Pd	0.002	1173	1186
Gapasil 9	82 Ag/9 Pd/2 Ga	0.002	1153	1178
Ticusil 70	68.8 Ag/26.7 Cu/4.5 Ti	0.002	1123	1144
Cusil ABA	65 Ag/30 Cu/2 Ti	0.002	1078	1100
Cusil	70 Ag/28 Cu	0.002	1053	1075

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Rocketdyne Division

91c-9-451
M387

HIGH CAPACITY POWER

NASA

7 Commercial Braze Alloys Evaluated With CP-Ti

Alloy	Success	Failure	General Observations
Copper ABA		X	Braze alloy dissolved CP-Ti sheet
Palcusil 15		X	Limited wettability of C-C
Silver ABA	X		Good wetting of both C-C & CP-Ti
Gapasil 9		X	Limited bonding to C-C
Cusil ABA	X		Good adhesion to both C-C & CP-Ti
Cusil	X		Good intimate contact between surfaces
Ticusil 70		X	Good bonding but Ti interface eroded

Braze Alloy Used With Nb-1% Zr

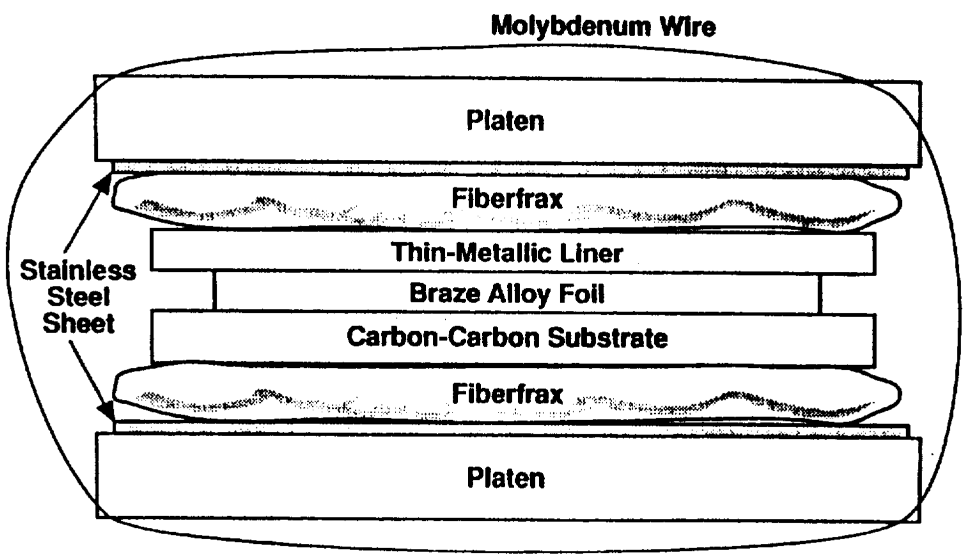
(Nb-1% Zr sheet thickness = 0.001 in.)

Braze Alloy	Success	Failure	Observations
Silver ABA	X		Good wetting & adhesion
Cusil ABA	X		Good wetting & adhesion

Rockwell International
Rocketdyne Division

91c-9-455
M397

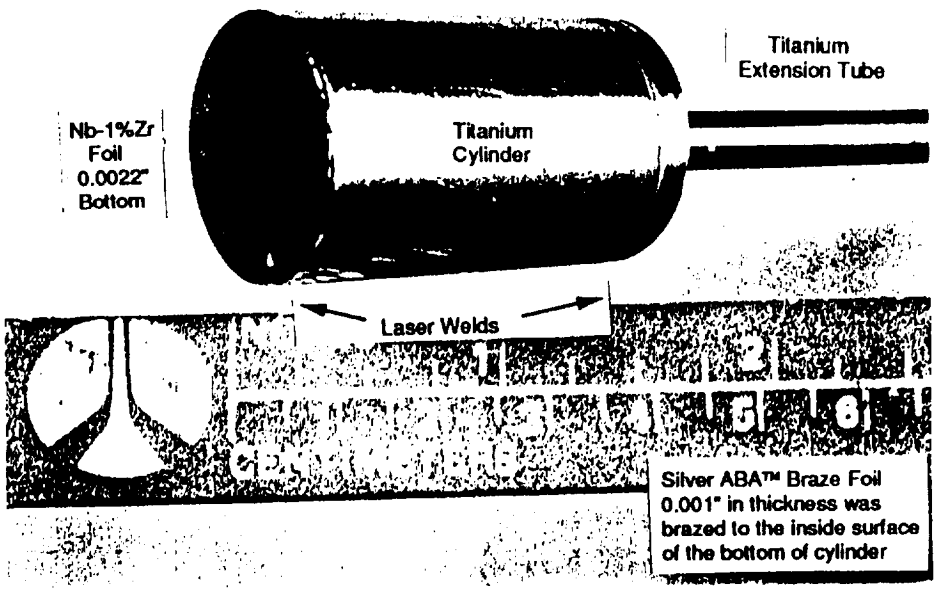
Illustration of Braze Test Fixture



Rockwell International
Rocketdyne Division

91c-9-452
M397

LIQUID POTASSIUM MATERIAL COMPATIBILITY TEST SPECIMEN



HIGH CAPACITY POWER

NASA

ROCKWELL ADVANCED RADIATOR CONCEPTS
FY 1992 ACCOMPLISHMENTS

SUCCESSFULLY DEMONSTRATED THE ABILITY TO FABRICATE A METAL LINED C-C HEAT PIPE WITH INTEGRAL FINS

- CARBON-CARBON TUBE FABRICATION
  - COMPLETED FABRICATION OF EIGHT FEET OF T-300 C-C TUBE WITH INTEGRAL WOVEN FINS
  - INITIATED WEAVING OF C-C PREFORM USING ONLY HIGH THERMAL CONDUCTIVITY P95-WG FIBERS AND ALL PITCH DENSIFICATION

HIGH CAPACITY POWER

NASA

ROCKWELL ADVANCED RADIATOR CONCEPTS
FY 1992 ACCOMPLISHMENTS

- LINER FABRICATION
  - COMPLETED FABRICATION OF Nb-1%Zr LINER TUBES WITH INTEGRAL EVAPORATOR SECTION VIA UNISKAN (PNL) METHOD
  - COMPLETED FABRICATION OF ALTERNATE LINERS (Nb-1%Zr AND Ti) BY DEEP-DRAW/CHEMICAL ETCHING TECHNIQUE
- HEAT PIPE FABRICATION
  - SUCCESSFULLY WELDED Nb-1%Zr END CAPS WITH FILL TUBES TO EVAPORATOR (~20 mil) AND CONDENSER (~3 mil)
  - SUCCESSFULLY FABRICATED PERFORATED FOIL WICK MATERIAL AND ESTABLISHED WELD PARAMETERS
  - SUCCESSFULLY DEMONSTRATED BRAZING OF A THIN METAL LINER INTO A FINNED C-C TUBE

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HIGH CAPACITY POWER

NASA

ROCKWELL ADVANCED RADIATOR CONCEPTS
FY 1992 ACCOMPLISHMENTS

- HEAT PIPE FABRICATION (Continued)
  - SUCCESSFULLY DEMONSTRATED THE ABILITY TO UNIFORMLY CVD COAT THE INSIDE OF A 12 INCH TUBE
  - SUCCESSFULLY DEMONSTRATED THE ABILITY TO COT AND MACHINE THE TUBE CUSP AREA CREATING A SMOOTH TUBE INTERIOR
  - SUCCESSFULLY DEMONSTRATED THE BRAZING OF A THIN METAL LINER INTO A C-C TUBE
- GENERAL
  - COMPLETED COUPON AND TUBE THERMAL CONDUCTIVITY TESTS
  - COMPLETED 30, 60, AND 180 DAY THERMAL DIFFUSION TESTS - Nb-1%Zr SAMPLES SHOW NO CARBON OR BRAZE DIFFUSION, Ti SAMPLES SHOW BRAZE DIFFUSION INTO LINER
  - UPDATED SP-100 HEAT REJECTION DESIGN INCORPORATING C-C HEAT PIPE CONCEPT

AJJ92-002 11

ROCKWELL FY 93 TASKS

- FABRICATE METAL LINED C-C HEAT PIPE WITH INTEGRAL FINS FOR SP-100 (820 K) RADIATOR
  - INSTALL ANNULAR FOIL WICK
  - PERFORM POTASSIUM FILL-PURGE OPERATION
- PERFORM HEAT PIPE TESTING AT SIMULATED SP-100 HEAT REJECTION CONDITIONS

AJJ92-002.17

LeRC C-C AND COMPOSITE MATERIALS PROGRAM FOR SPACE RADIATORS

IN-HOUSE	- Gr/CU COMPOSITES FOR HEAT PIPE FINS (Gr/Al COMPOSITES BEING DEVELOPED UNDER WRDC CONTRACTS)
	- ARC TEXTURING FOR EMISSIVITY ENHANCEMENT
CONTRACTS	- ARC (ADVANCED RADIATOR CONCEPTS)
SPI-SAN JOSE, CA	- VGCF (VAPOR GROWN CARBON FIBER) MATERIAL FOR VERY HIGH SPECIFIC CONDUCTIVITY HEAT PIPE FINS
RI - CANOGA PARK, CA	- C-C TUBE WITH INTEGRAL WOVEN FINS AND INTERNAL METALLIC LINERS FOR POTASSIUM HEAT PIPES
PNL - (PACIFIC NORTHWEST LABS) - RICHLAND, WA	- LIGHTWEIGHT FLEXIBLE CERAMIC FIBER HEAT PIPES WITH METAL FOIL LINERS

AJJ92-002

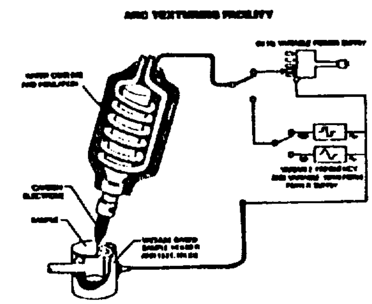
ADVANCED RADIATOR SURFACES

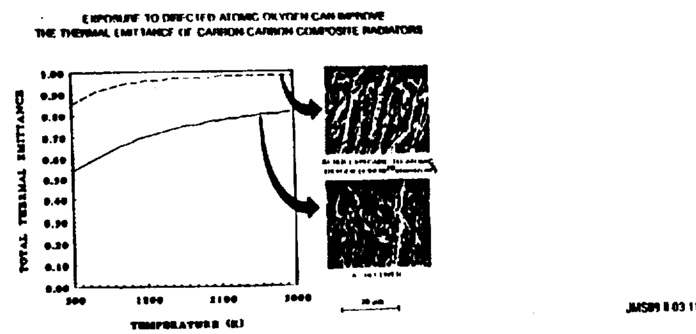
OBJECTIVE: DEVELOP DURABLE, HIGH TEMPERATURE, HIGH EMITTANCE RADIATOR SURFACES

ACCOMPLISHMENTS:

DEMO EMITTANCE >.85 @ 500K
FOR TYPICAL RADIATOR MATERIALS
PRELIMINARY DATA ON ATOMIC OXYGEN

STATUS: ON GOING

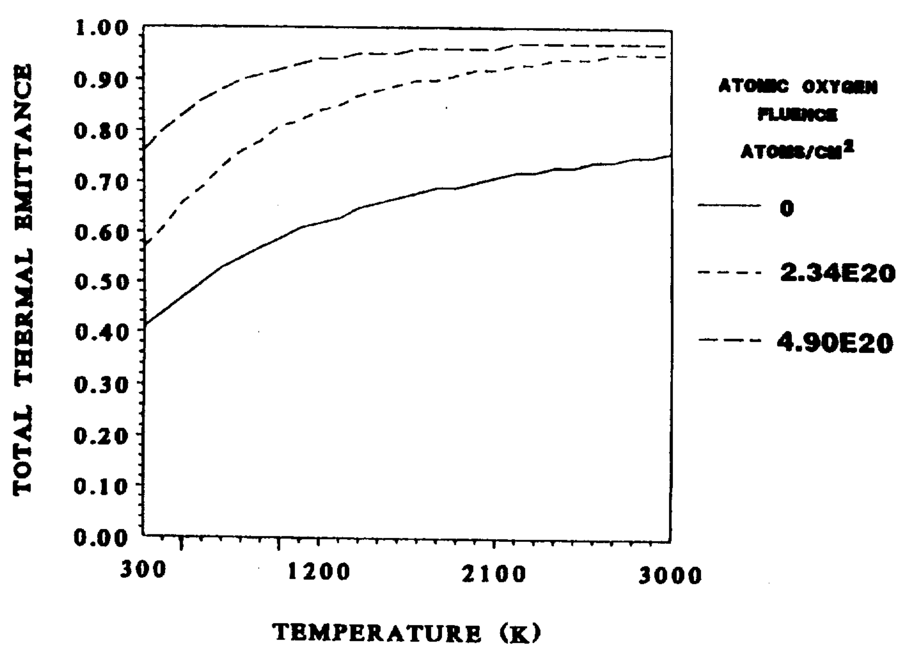




HIGH CAPACITY POWER

NASA

EMITTANCE VS TEMP. FOR ROCKETDYNE C741C C-C COMPOSITE WITH A/O FLUENCE



• RADIATOR DESIGN & INTEGRATION

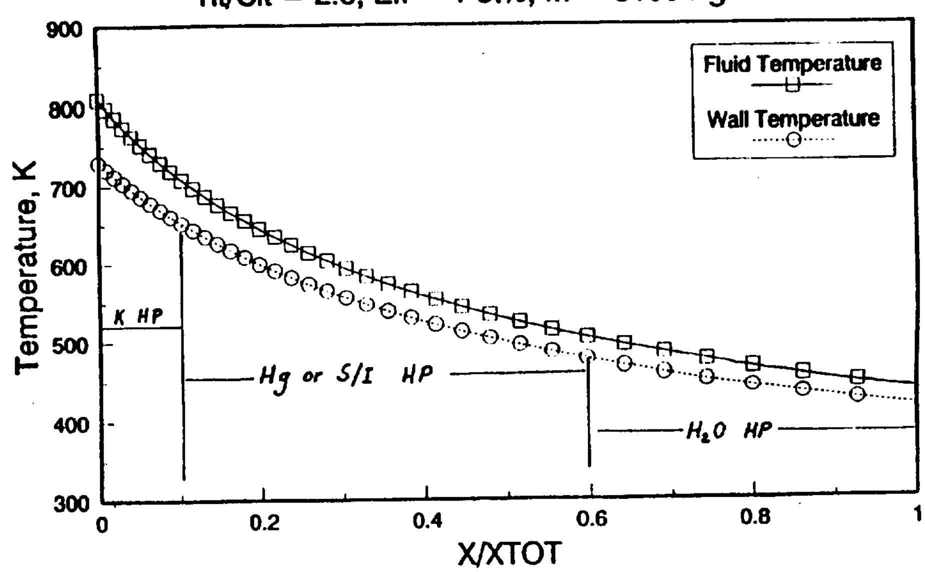
HIGH CAPACITY POWER

NASA

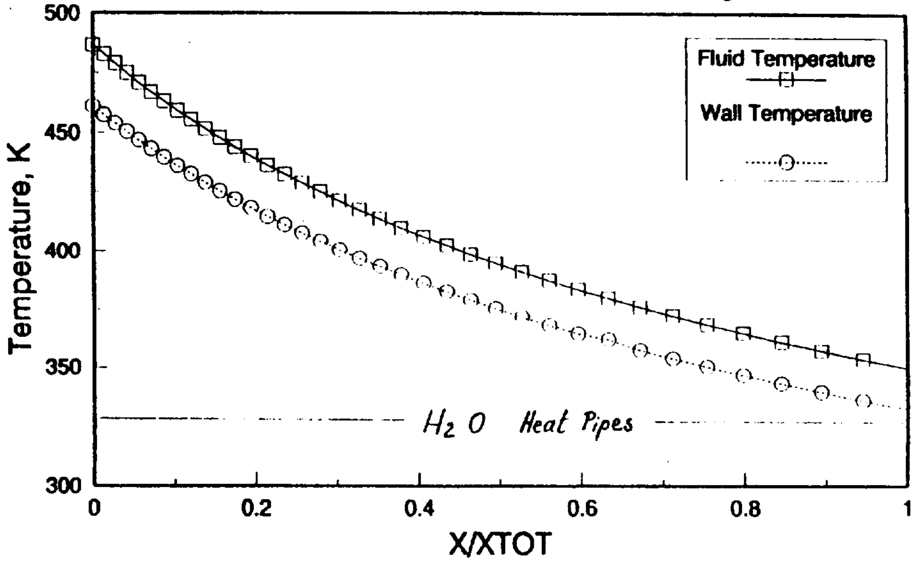
100 kWe CBC Radiator

Tit = 1140 K; Pr = 2.7; ERG = 0; A = 130 m^2

Tit/Cit = 2.6; Eff = 1 8.%; M = 3100 Kg

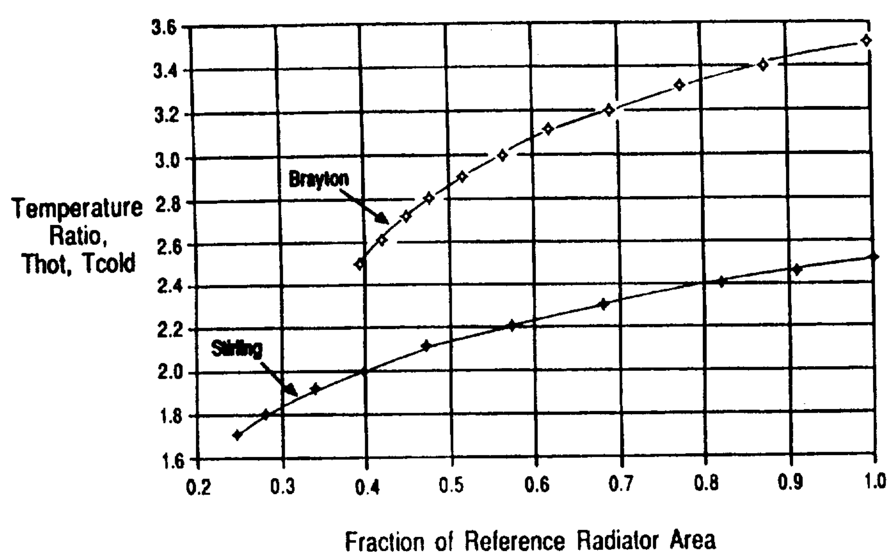


100 kWe CBC Radiator
Tit = 1140 K; Pr = 1.85; ERG = 95%; A = 184 m^2
Tit/Cit = 3.26; Eff = 37.5 %; M = 3600 Kg

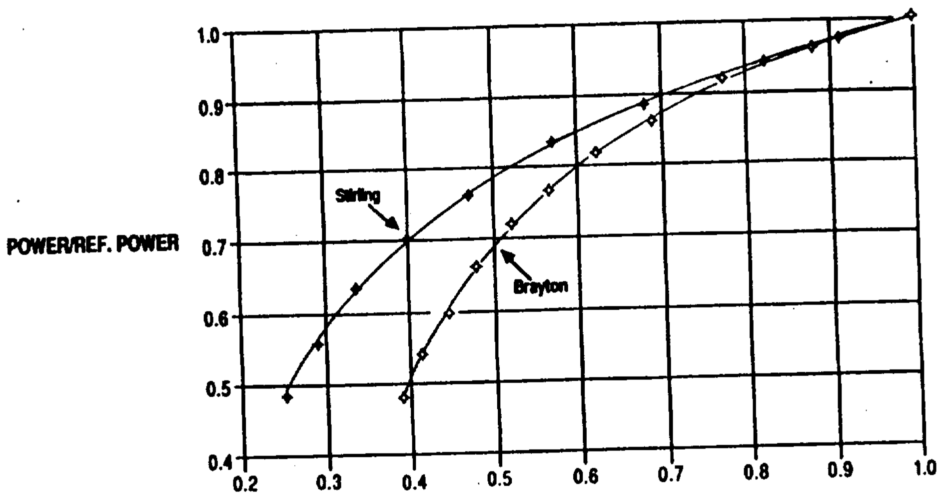


HIGH CAPACITY POWER NASA

EFFECT OF REDUCTION IN RAD. AREA ON STIRLING & BRAYTON TEMP. RATIOS
(Constant Heat Rejection, Thot = 1050 K, Sink Temp. = 250 K)



EFFECT OF REDUCTION IN RAD. AREA ON STIRLING AND BRAYTON POWER
(Constant Heat Rejection, Thot = 1050 K, Sink Temp. = 250 K)

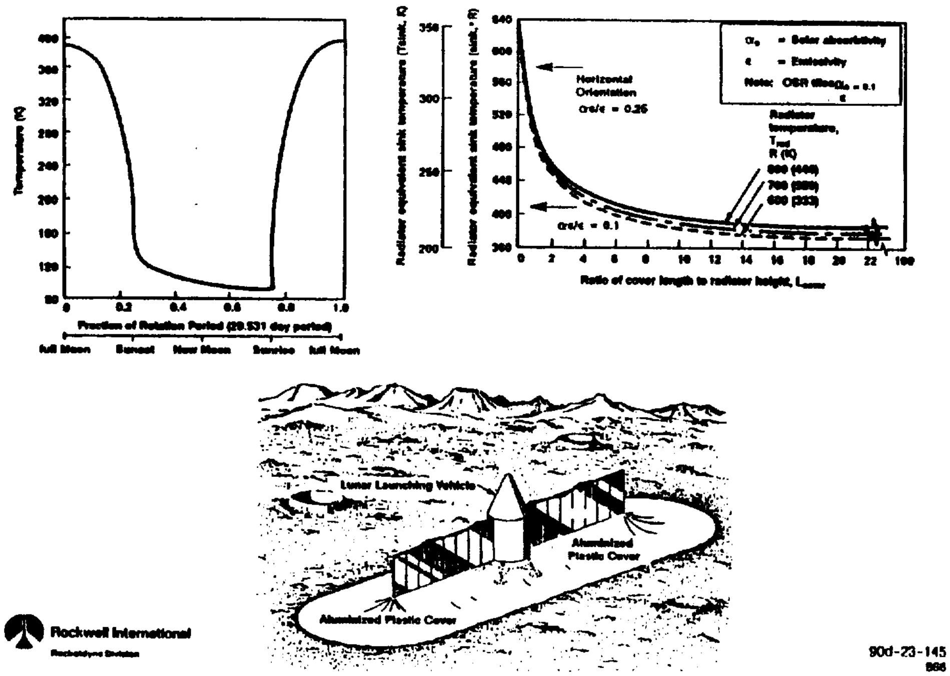


HIGH CAPACITY POWER

ARC TECHNOLOGY POTENTIAL APPLICATIONS

- NUCLEAR POWERED LUNAR BASE
  - SP-100 OR DERIVATIVE
  - MW TO MULTI MW POWER OUTPUT
- SOLAR DYNAMIC POWER SYSTEM FOR LUNAR BASE
  - IN-SITU (REGOLITH) THERMAL STORAGE
  - 25 TO 100 kWe POWER PLANT
- GEO BASED COMMUNICATIONS SATELLITE
  - SD PCS - 3 TO 5 kWe
- NUCLEAR ELECTRIC PROPULSION
  - 10 MWe CLASS PCS: TI, LMR, TE OR CBC

Lunar Surface Sink Temperature



POWER TECHNOLOGY DIVISION

NEP POWER SYSTEM HEAT REJECTION

TEMPERATURE RANGES OF INTEREST

POWER SYSTEM CANDIDATES		PEAK CYCLE TEMP (K)	HEAT REJECTION TEMP (K)
IN CORE THERMIONIC	- TI	2200	1000
LM RANKINE	- LMR	1450	950
THERMOELECTRIC - SP100	- TE	1300	850
CLOSED CYCLE BRAYTON	- CBC	1500	320 - 800
STIRLING FPSE	- ST	1300 1050 900	550 - 600 450 400

ATD AEROSPACE TECHNOLOGY DIRECTORATE — POWER TECHNOLOGY DIVISION — NASA Lewis Research Center

NEP POWER SYSTEM RADIATOR TECHNOLOGIES

POWER SYSTEMS (10 MWe)		$Q_R = P\,(^{1-\eta}/_{\eta})$ HEAT REJECTED MWt	RADIATOR PARAMETERS TEMP K	AREA m^2	TECHNOLOGY	kg/m^2	kg/kWt
THERMIONIC	$\eta_t = .15$	57.0	1000	1600	SS/Na HP	10	0.2
LIQUID METAL RANKINE	$\eta_t = .18$	45.5	950	1230	SS/Na HP	10	0.3
THERMOELECTRIC	$\eta_t = .05$	190.0	850	7600	Ti/K HP	5	0.2
CLOSED BRAYTON	$\eta_t = .30$	23.3	400 - 800	4800	Ti/K HP C-C/K HP C-C/H2O HP	4	0.8
STIRLING - FPSE	$\eta_t = .30$ $= .33$	23.0 20.0	600 450	3500 11200	SS/Hg HP C-C H2O HP Li/NaK LOOP	10 2 5	0.9

JEC90 007 4

ATD AEROSPACE TECHNOLOGY DIRECTORATE — POWER TECHNOLOGY DIVISION — NASA Lewis Research Center

NEP POWER SYSTEM RADIATOR TECHNOLOGIES THRUSTS

POWER SYSTEMS (10 MWe)		HEAT REJECTED MWt	RADIATOR TECHNOLOGIES TEMP K	NEAR TERM	MID TERM	FAR TERM
THERMIONIC	$\eta_t = .15$ $\eta_t = .20$	57.0 40.0	1000 1050	SS/Na HP 10 kg/m^2	CC/Na HP * 5 kg/m^2	LSR, Fiber HP 2 kg/m^2
LIQUID METAL RANKINE	$\eta_t = .18$	45.5	950	10 kg/m^2 SS/Na HP	5.0 kg/m^2 C-C/Na HP	2 kg/m^2 LSR, Electrostatic 3 kg/m^2
THERMOELECTRIC	$\eta_t = .05$	190.0	850	9 kg/m^2 Nb Zr/K HP	5.0 kg/m^2 Ti-SiC/K HP	C-C HP 3 kg/m^2
CLOSED BRAYTON	$\eta_t = .30$	23.3	800 - 400	10 - 15 kg/m^2 MP Loop Mixed HP	Mixed HP Ti, C-C 5 kg/m^2	Fiber Fabric/H2O 1-2 kg/m^2
STIRLING - FPSE	$\eta_t = .33$	20.0	500 - 450	10 kg/m^2 MP Loop Hg HP	Li-NaK Loop 5 kg/m^2	Fiber Fabric/H2O 1-2 kg/m^2

* ALL C-C HEAT PIPES HAVE INTERNAL COATING COMPATIBLE WITH WORKING FLUID

JEC90-007 5

THERMAL MANAGEMENT BASELINE BUDGET

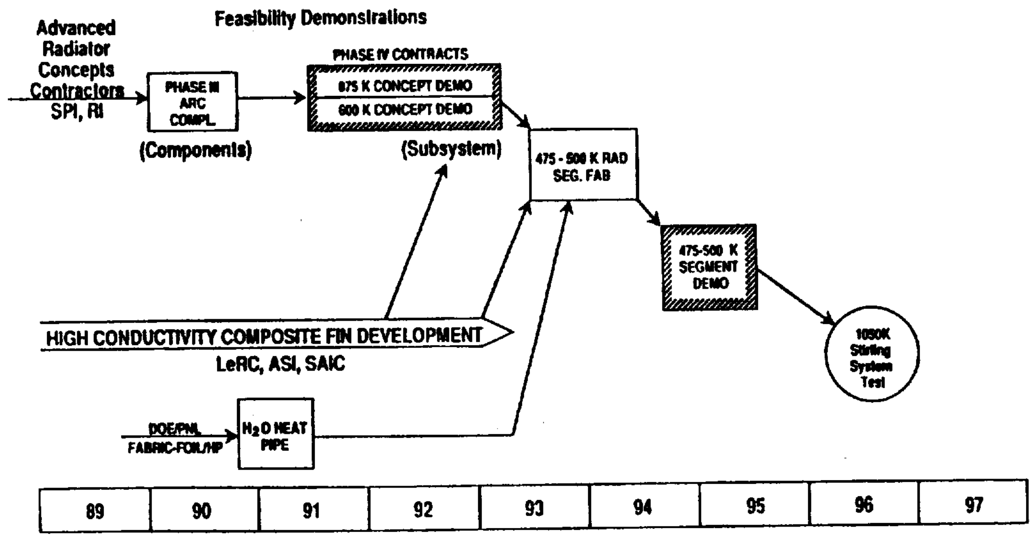


Figure 9

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HIGH CAPACITY POWER

CONCLUDING REMARKS

- PROGRAM ON TIME AND WITHIN BUDGET
- PROGRAM BROADLY COORDINATED WITH OTHER PROGRAMS THROUGHOUT THE THERMAL MANAGEMENT COMMUNITY
- CSTI/HCP TM PROGRAM $\equiv$ SP-100 TM PROGRAM
- TECHNOLOGY BEING DEVELOPED HAS BROAD APPLICATION
  - SP-100
  - SOLAR DYNAMIC
  - LUNAR/MARS INITIATIVE

JMS89 8 03 16

N93-26980

JPL NUCLEAR ELECTRIC PROPULSION TASK

Tom Pivirotto
Keith Goodfellow
Jay Polk

JPL

Nuclear Propulsion Technical Interchange Meeting
NASA Lewis Research Center/Plum Brook Station
October 20-23, 1992

JPL

LITHIUM MPD THRUSTER TECHNOLOGY DEVELOPMENT AT JPL

- Funded by NPO in FY92 to develop a lithium feed system
  - Reservoir and vaporizer designed and under construction
  - Flow rate calibration system design complete, components under construction
- Test facility design nearly complete, construction to be completed in FY93
  - 6' x 15' double-walled stainless chamber with 27' long extension to be used as a beam dump pumped by a 20" diameter oil diffusion pump
- Initial testing of 100 kWe-class radiation-cooled engine to begin in FY93

C-6

9

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COMPARISON OF MEASUREMENTS WITH THEORY FOR MERCURY PHASE SEPARATOR

- DATA OBTAINED WITH A SMALL DEVICE AND AT LOW TEMPERATURES
- FOR LITHIUM MPD REQUIRED TEMPERATURE AND FLOW AREA MUST BE GREATER

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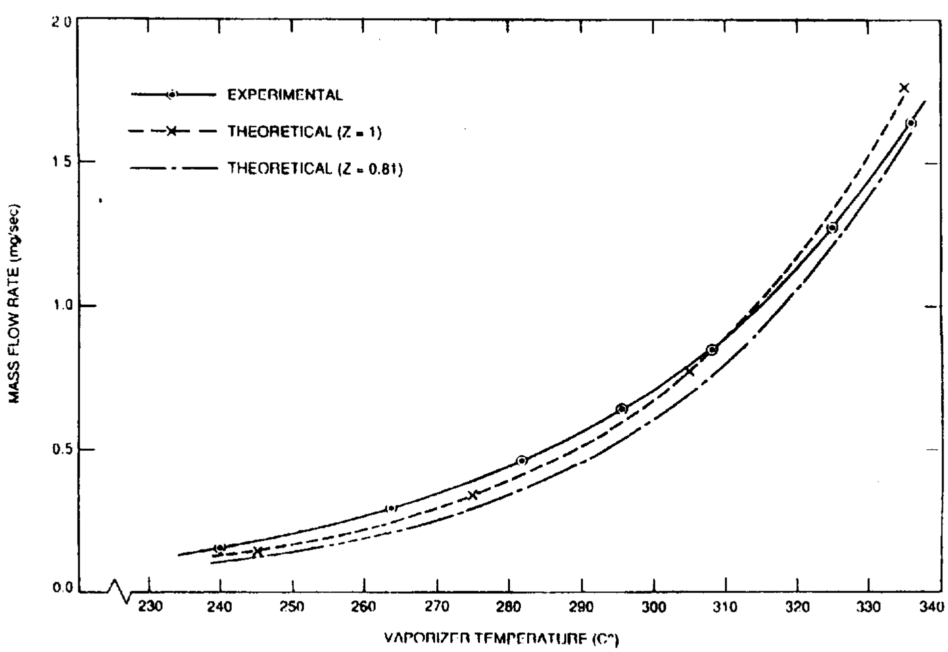
MERCURY VAPOR MASS FLOW CONTROL



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INITIAL EXPERIMENTAL HARDWARE DESIGN

- HIGH TEMPERATURE WILL BE CONFINED TO THIN LITHIUM LIQUID SHEET BETWEEN HOUSING AND SEPARATOR
- CAN EASILY REPLACE SEPARATOR

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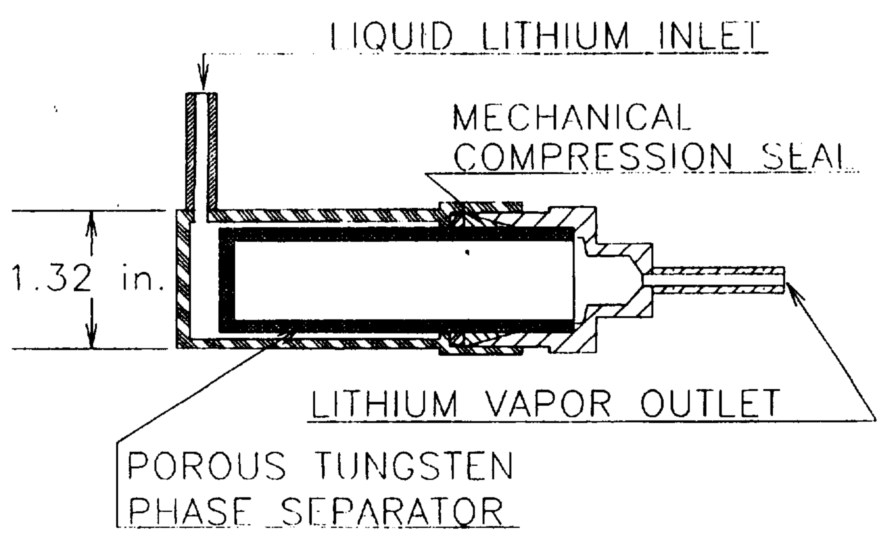
POROUS TUNSTEN VAPORIZER AND HOUSING



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EXPERIMENTAL SETUP

- VAPOR COLLECTOR WILL BE LIGHT
- HEAT OF CONDENSATION WILL BE REMOVED THROUGH OIL BATH
- LIQUID PRESSURE AT SEPARATOR WILL BE KEPT WITHIN ACCEPTABLE RANGE WITH REGULATED ARGON PRESSURE

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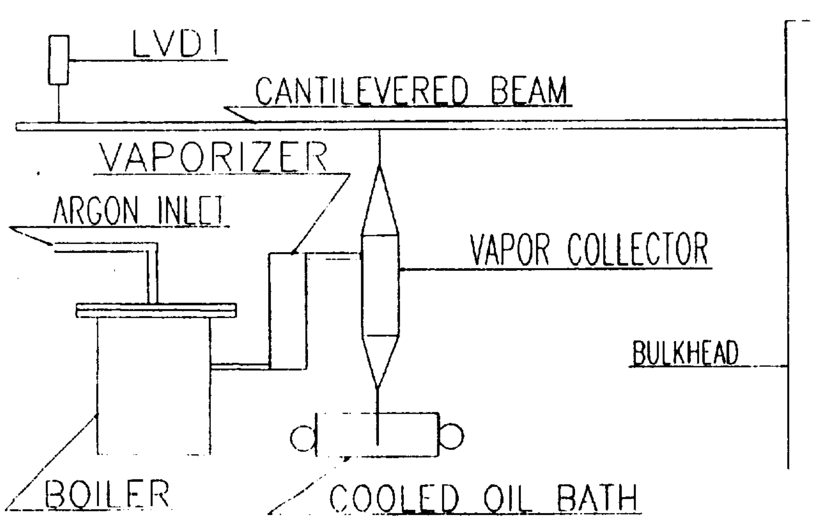
LITHIUM VAPORIZER EXPERIMENT



HEATERS NOT SHOWN

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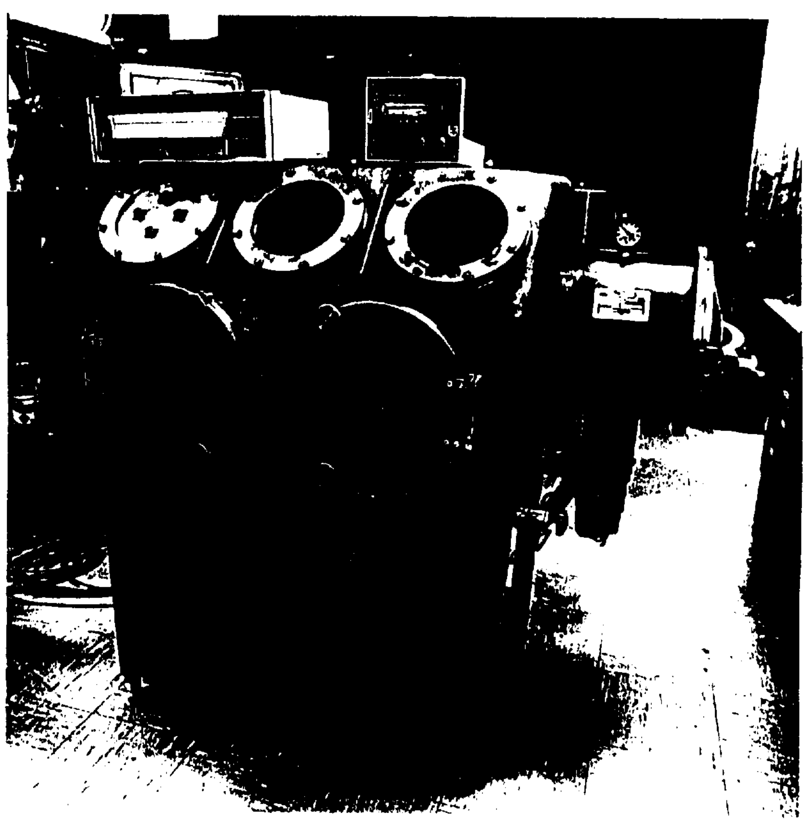
DRY BOX FOR HANDLING SOLID LITHIUM

• ZERO CONTACT BETWEEN SOLID LITHIUM AND AIR



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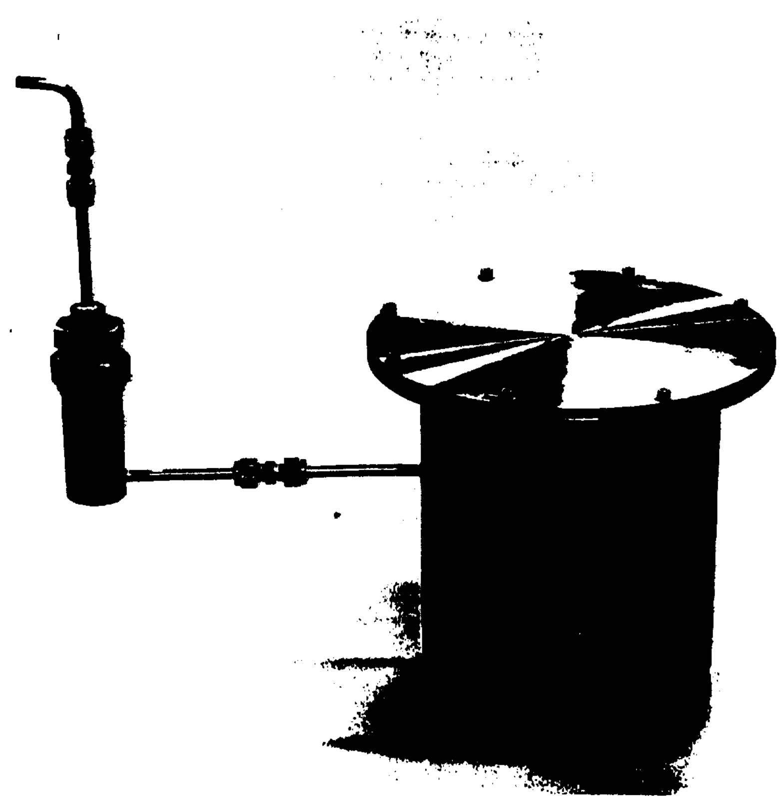
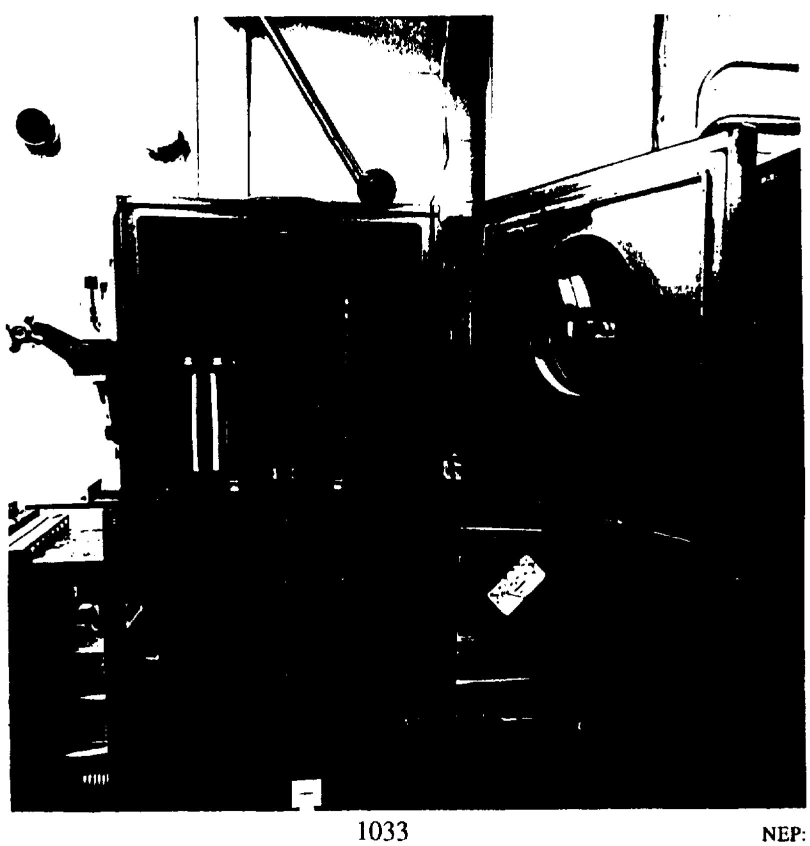
EXPERIMENTAL HARDWARE

- BOILER CAN HOLD 900 G OF LITHIUM
- HARDWARE EASILY DISASSEMBLED FOR CLEANING



TEST FACILITY

- VACUUM TANK IS 45 x 45 x 80 CM
- PUMP OUT PRESSURE TO LESS THAN 1 MTORR



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MPD THRUSTER ELECTRODE MODELLING

- Cathode - Emphasis is on lifetime assessment:

  Methodology
  Modelling
  Experimental Verification

- Anode - Primary focus is thermal management:

  Impact of anode work function
  Assessment of heat rejection methods

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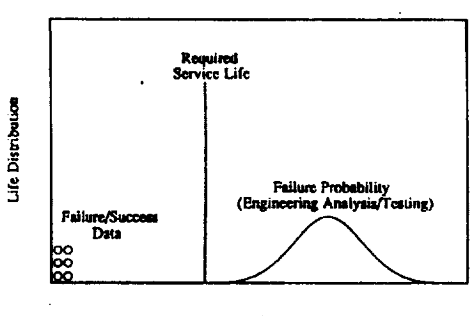
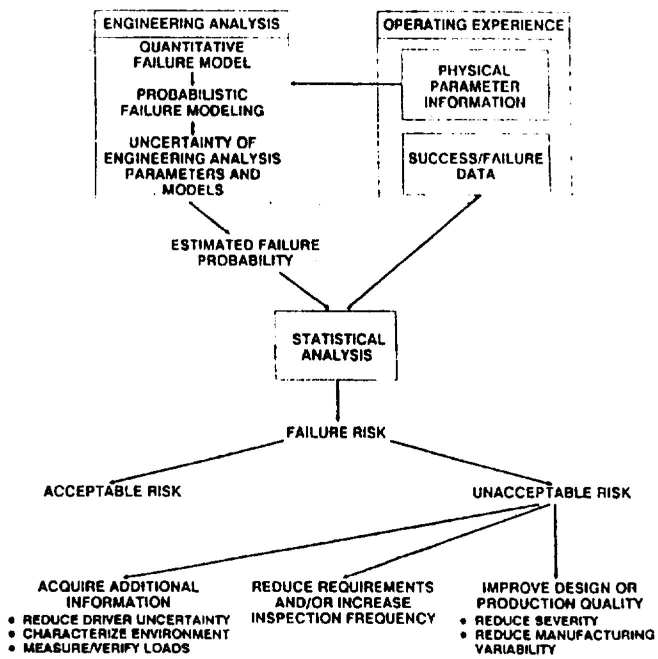
DEFINING ENGINE LIFETIME



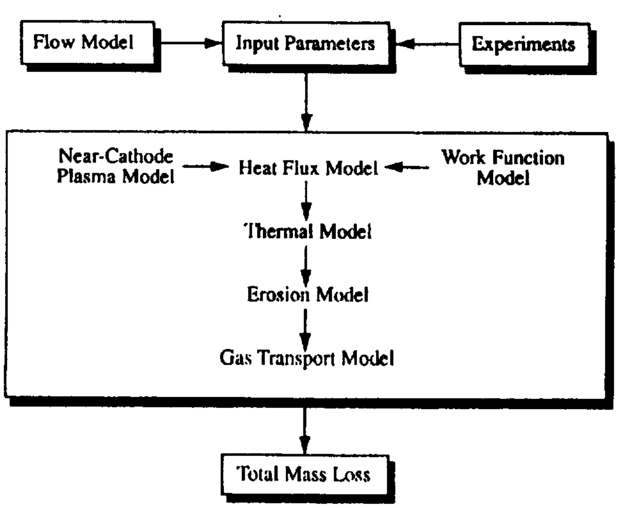
Engine lifetime, requirements and operating experience

- CURRENT STATUS
  - Required service life is not well defined
  - Critical failure modes have not been identified
  - No theoretical or experimental characterization of life distribution
- IMPORTANT OBSERVATIONS
  - Life distribution characterization by system-level operating experience is not feasible
  - Engine lifetime is inherently probabilistic

PROBABILISTIC FAILURE ASSESSMENT



QUANTITATIVE CATHODE FAILURE MODELLING



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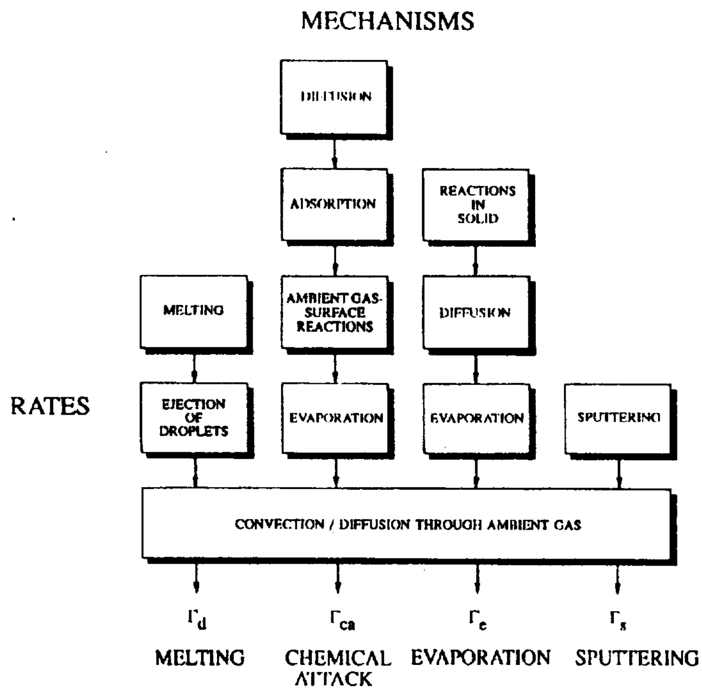
CATHODE EROSION MODELLING



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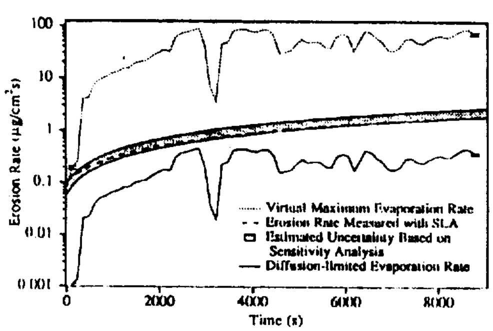
COMPARISON OF CALCULATED AND MEASURED CATHODE EROSION RATES



Cathode erosion measurements performed with Stuttgart thruster NCT-1 at 2500 A, 1.0 g/s of argon, 71 kWe and 20 Torr ambient pressure

- Diffusion-limited evaporation of tungsten is the dominant mechanism
- Model underpredicts erosion rate by a factor of 6, reflecting uncertainties in transport rate through concentration boundary layer
- Calculated erosion rates are based on measured temperatures--thermal model required for fully predictive capability

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CATHODE THERMAL MODELLING

- HT9: 1-1/2 D thermal model with variable grid spacing and non-linear thermal and electrical conductivity. Allows specification of radiation, conduction, convection and arc attachment boundary conditions on ends and inner and outer radii.
- AFEMS: Commercial 2D finite-element model with nonlinear material properties. Very flexible solid modeller for geometry specification, but definition of boundary conditions is more cumbersome than in HT9.
- Fully 2D version of HT9 to be developed in FY93.

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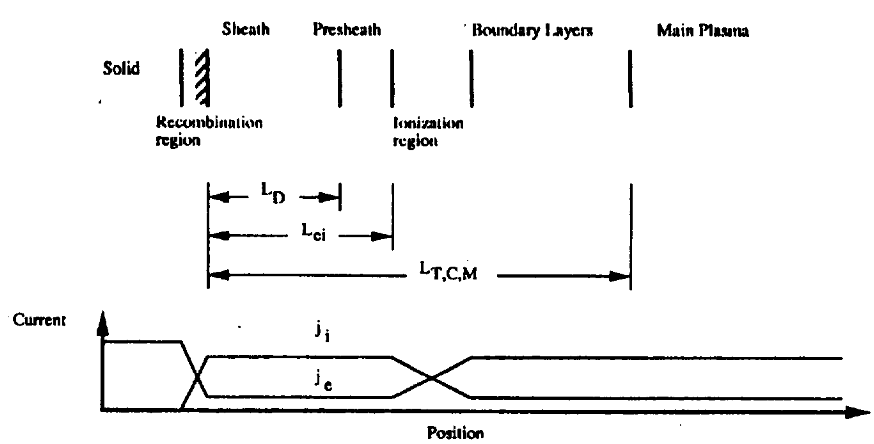
NEAR-CATHODE PLASMA MODEL REGIONS



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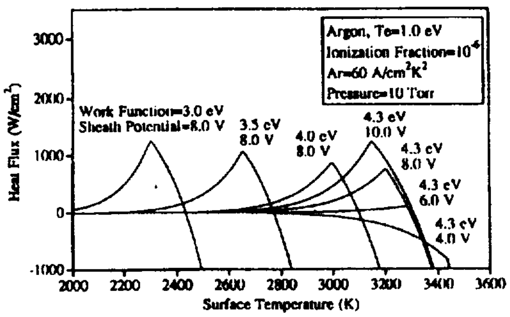
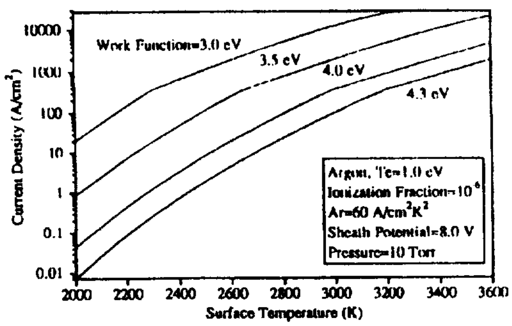
NEAR-CATHODE PLASMA MODELLING





- The model describes the electrostatic sheath, presheath and ionization zones
- Current and heat fluxes are calculated as functions of gas properties, thermionic properties, surface temperature and sheath potential
- Terms normally neglected in high-pressure noble gas arc models are included to allow accurate modelling of low-pressure alkali metal arcs

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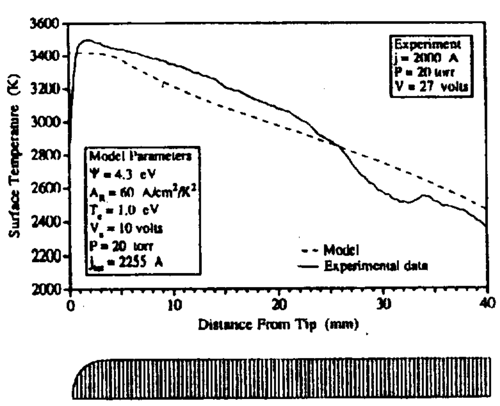
COMPARISON OF CALCULATED AND MEASURED TEMPERATURE DISTRIBUTIONS



Cathode model geometry and results

- The model includes radiation, conduction out the base and heat input over the first 5 mm from the near-plasma model
- The model reproduces the tip temperature and shaft behavior for reasonable values of the input parameters
- Errors may be due to experimental data not in equilibrium and thorium effects on spectral emissivity

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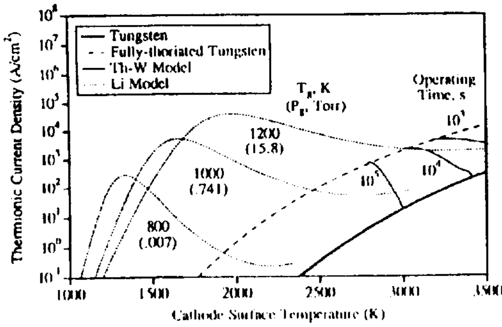
CATHODE WORK FUNCTION MODELLING



Emission capability of tungsten metal with Th and Li adsorbed on the surface.

- "Activator" may be electropositive material in the cathode bulk or in the propellant
- Two models were developed for cathode additive transport and propellant-surface interaction
- Th-W effect on work function is limited by depletion of thorium additive
- Li supply from propellant is unlimited, but surface coverage depends on gas pressure and temperature
- There is considerable uncertainty in model input parameters

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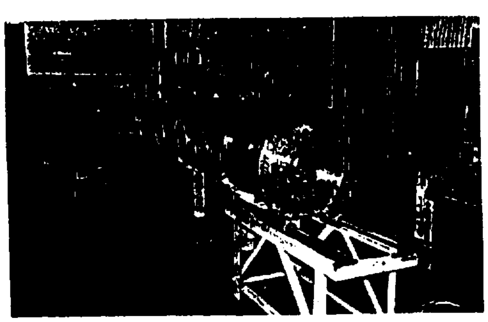
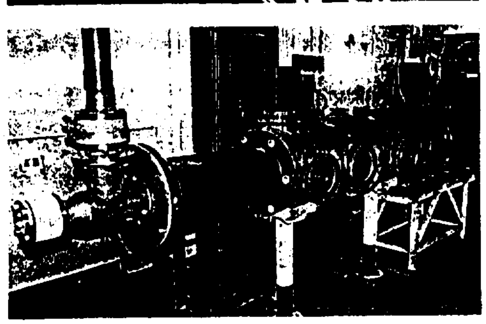
CATHODE TEST FACILITY





- Demonstrate feasibility of new cathode concepts
- Measure cathode temperature distributions and erosion rates to validate models
- Measure model input parameters
- Collect success/failure data in long endurance tests

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IMPACT OF ANODE WORK FUNCTION

Two limiting cases examined:

- Strong positive anode sheath, $V_s >> kT_e/e$

  Thermionic current can be neglected, heat transfer rate is lower for a low work function anode.

- Negative anode sheath

  Preliminary sheath model results indicate lower anode heat transfer rate for low work function anodes at moderate temperatures (Example: For 100 A/cm^2, $n_e = 10^{14}$ cm^{-3} (Argon), T_e = 1 eV, an anode with a work function of 3.5 eV has lower heat transfer rates than one at 4.5 eV for temperatures below about 2600 K.)

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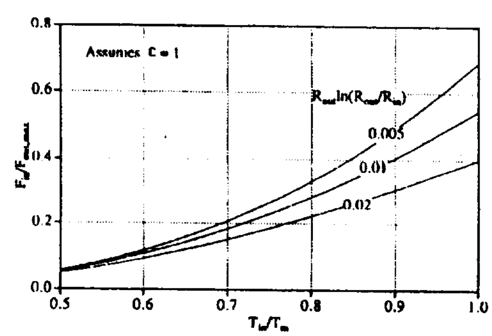
ASSESSMENT OF RADIATION-COOLED ANODES



Analytical model of thin-walled, cylindrical anodes.

T_{in} = Temperature on inner surface
T_m = Melting temperature of material
F_{in} = Power/unit axial length
$F_{out,max}$ = Maximum possible radiated power/unit length from exterior, σT_m^4

- Analytical model of thin-walled anodes completed--neglects axial conduction, internal radiation and Joule heating.
- Example: 10 cm dia. tungsten anode with 10 mm wall thickness and maximum allowable T_{in}=0.8 T_m can reject 18 kW of power per cm of length.
- Effect of axial heat conduction and Joule heating is being studied with finite element analysis.
- Comparison between thin-walled anodes and anodes with large radiators is being performed using finite-element analysis.

LOS ALAMOS RESEARCH IN NOZZLE BASED COAXIAL PLASMA THRUSTERS

Kurt F. Schoenberg

Presented to the Nuclear Propulsion Technical Interchange Meeting

October 21, 1992

LOS ALAMOS THRUSTER RESEARCH

Colleagues and Collaborators

- Richard Gerwin
- Robin Gribble
- Ivars Henins
- John Marshall
- Ron Moses
- Jay Scheuer
- Glen Wurden
- Dorwin Black, N.C. State
- Rob Hoyt, U. Washington
- Tom Jarboe, U. Washington
- Robert Mayo, N.C. State

LOS ALAMOS THRUSTER RESEARCH

Outline

- Colleagues and Contributors
- History: Where we're coming from
- Our Perspectives on High-Performance EP
- Approach
- On Going Research Activities
- Plans

LOS ALAMOS THRUSTER RESEARCH

Historical Perspective

Los Alamos has conducted continuous research in coaxial plasma accelerators since their inception.

- Pioneered by John Marshall in the late 50's
- A rich history of applications:
  - Propulsion (1960's)
  - Plasma Fueling (1960's)
  - Radiation Source (1960's)
  - Space Plasma Injection (Birdseed) (1970's)
  - Magnetic Fusion Research (1980's)
  - SDI Research (1980's)
  - Propulsion (in collaboration with NASA LeRC) (1990's)
  - Materials Processing (1990's)
- Recent focus on steady-state operation (pioneered by Morozov)

LOS ALAMOS THRUSTER RESEARCH

Approach

Can electrodynamic-based thrusters achieve the performance required for space missions of interest?

- **Optimize** large-scale, multi-megawatt electrodynamic thruster performance.
- **Ascertain** performance scaling in terms of size and power.
- **Engineer** performance at power levels applicable to NASA or DOD "near term" missions like orbital transfer or robotic exploration.
  - In steady-state
  - For adjustable duty-cycle (pulsed) operation

LOS ALAMOS THRUSTER RESEARCH

Approach

Why Study Large, High Power Devices?

- There is a minimum "buy-in" for high performance operation!
- How high and how large is under investigation.
- Pulsed operation may be our "evolutionary approach".

Efficient MPD Operation

Perspectives

In addition to frozen flow losses, efficiency is limited by two processes:

- Macro plasma acceleration and detachment
  - Efficient operation ⇒ High grade plasma
  - High grade plasma ⇒ Ideal MHD
  - Ideal MHD ⇒ Economy of scale
- Electrode phenomena

These processes are coupled by the Electrical Effort (Morozov Hall parameter) *

$$\Xi \equiv \left(\frac{m_i}{e}\right)\frac{I}{\dot{M}} \approx \left(\frac{c}{\omega_{pi}}\right)\frac{1}{\Delta}$$

* Schoenberg, et al., AIAA 91-3770 (1990)

MMWe ELECTRIC PROPULSION

Efficacy of Magnetic Nozzles

Dominance of ideal MHD leads to the efficacious use of magnetic nozzles for optimization of:

- Acceleration
- Detachment
- Electrode Phenomena

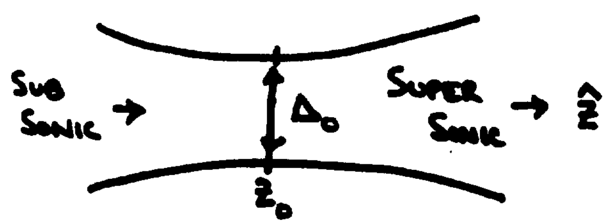
Magnetic nozzle expansion ratios are an important efficiency optimizer

MMWe THRUSTER DEVELOPMENT

Magnetic Nozzles

- Plasma Acceleration in Ideal MHD Requires ($\nabla \times V \times B = 0$):
  - Non-ideal effects
  - Converging-Diverging Flow (Nozzle)

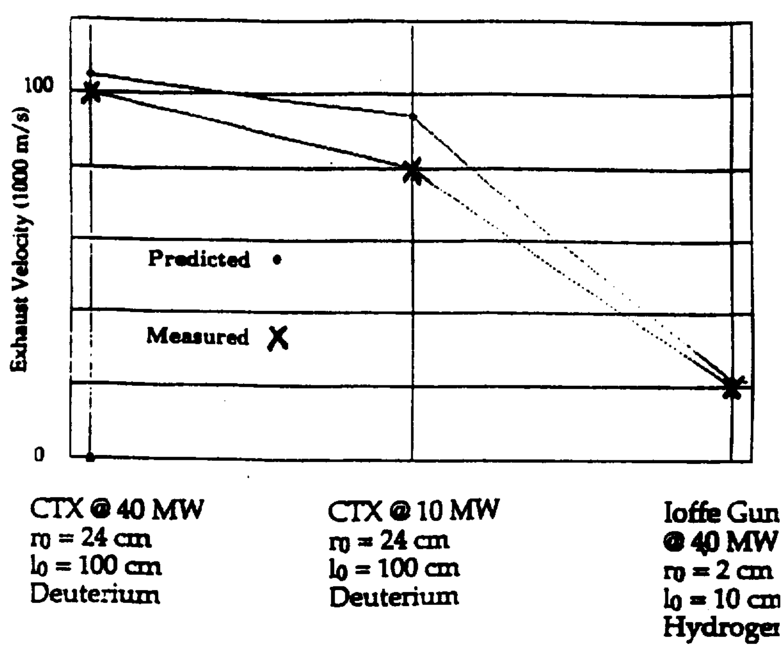
- Hydrodynamic Nozzle Theory has Direct Analogs in MHD (Morozov):



Mach 1 $\equiv$ Magnetosonic Velocity $= \sqrt{C_{so}^2 + C_{Ao}^2}$

COAXIAL THRUSTER PERFORMANCE

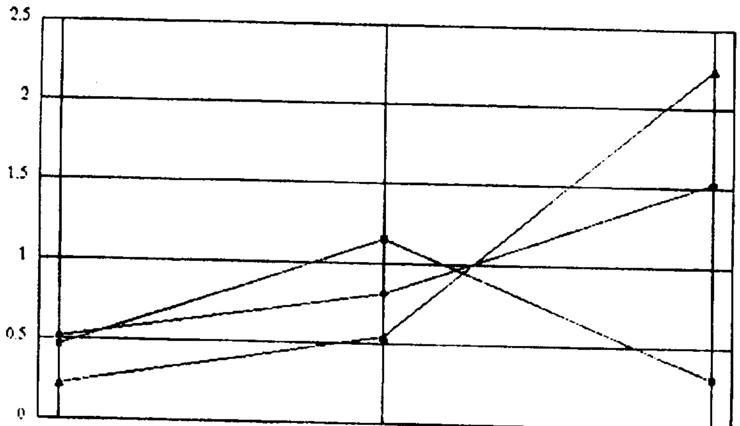
Exhaust Velocity



* Afanas'ev et al., Sov. Phys. Tech. Phys., 36, 505 (1991)

COAXIAL THRUSTER PERFORMANCE

Electrical Effort



* Afanas'ev et al., Sov. Phys. Tech. Phys., 36, 505 (1991)

LOS ALAMOS THRUSTER RESEARCH

FY91 & FY92 As-Was Experiments

- Power range 10-40 MW
- Unoptimized Gun
- Unoptimized 2.5 MJ capacitor bank
  - 1ms, round-top discharges
- Unoptimized $B_{r,z}$ nozzle field
- Wide range of diagnostics
  - Mullti-chord interferometry
  - Temporally and spatially resolved bolometry
  - Temporally and spatially resolved IR calorimetry
  - Langmuir and magnetic probes
  - Neutral particle spectroscopy

LOS ALAMOS THRUSTER RESEARCH

FY91 & FY92 As-Was Experimental Conclusions

- High exhaust velocity achieved (10^5 m/s) in agreement with MHD based theory.
- Thruster operational impedance in agreement with MHD based theory for constant $I^2/\dot{M}$.
- Radiative (frozen flow) losses small ($\leq$ 10%)
- Applied magnetic configuration can affect and control the anode fall.
- Power flux to the electrodes well quantified.
- Power flux to the anode probably dominated by ion flux
- Global electrode power loss probably less than 50 % at high power operation (40 MW).

LOS ALAMOS THRUSTER RESEARCH

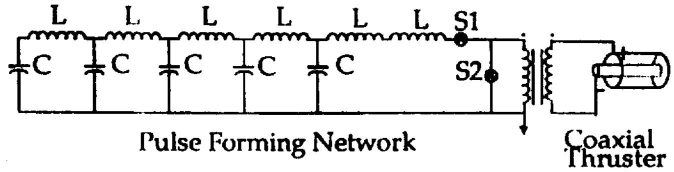
FY93 Optimized Experiments

In FY92, CTX was converted into a "world-class" high power MPD test facility

- PFN controlled 2 MJ, transformer coupled capacitor bank
- 10 ms flat-top discharges at 1 to 50 MW
  (10 - 100 kA and 50 to 1000 v)
- Constant propellant injection at 1 to 10 g/s (deuterium)
- DC control of applied nozzle field
- Electrically isolated test-stand
- PC / Sparc Station control, data acquisition. and analysis
- Full diagnostics capability

Pulse Forming Network

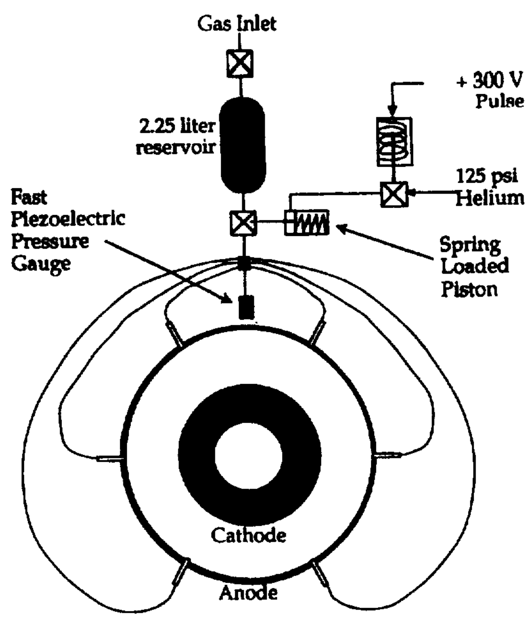
Schematic



- C = 0.8 mF
- L = 0.125 mH
- 5:1 Transformer
- 2.0 MJ Stored Energy
- 10 ms Flat Top Pulse

Long Pulse Gas Valve System

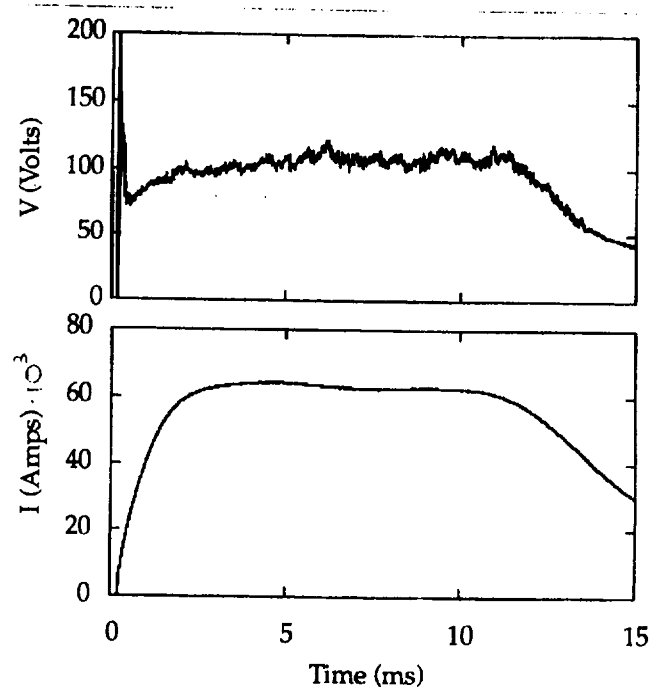
Schematic



- Stainless steel feed lines are of equal length

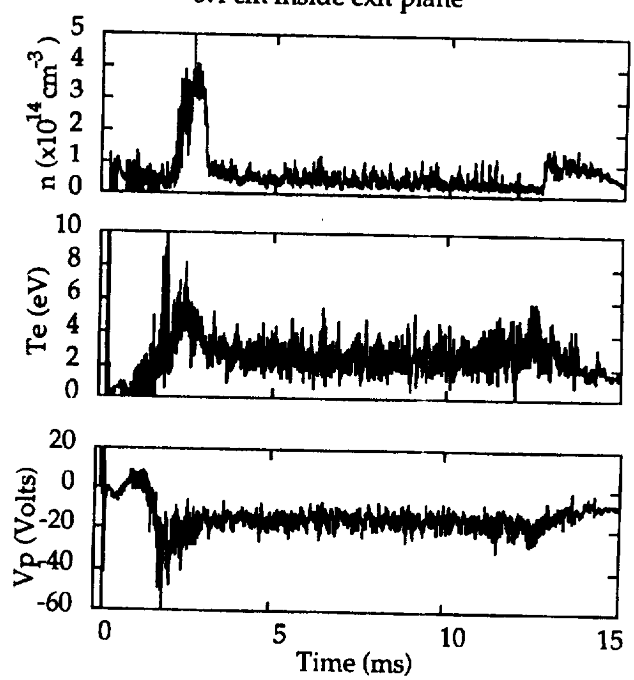
Thruster Current and Voltage

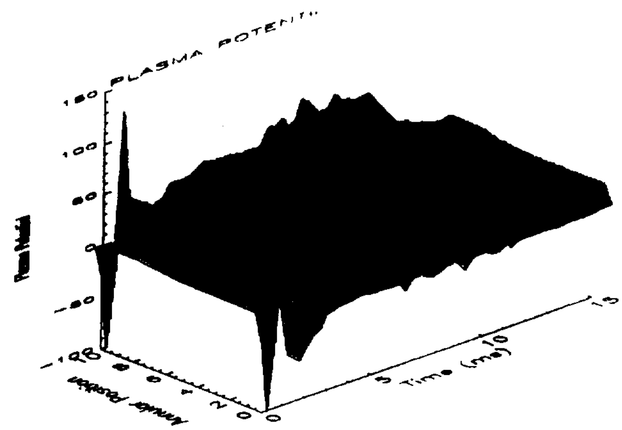
6 g/s helium

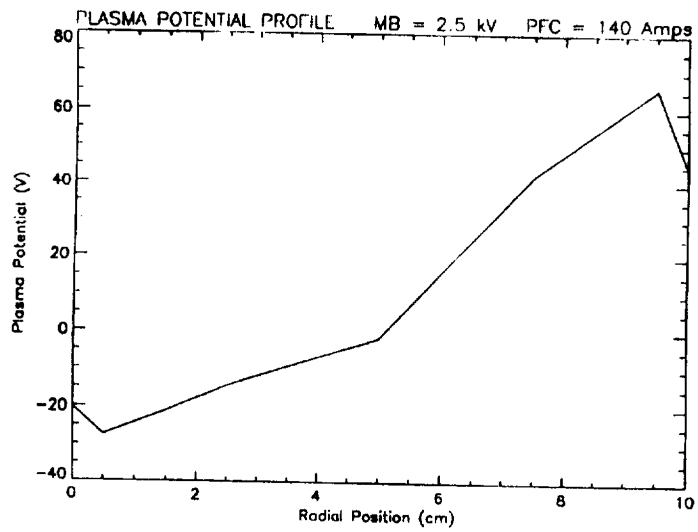


Triple Langmuir Probe Data

5.4 cm inside exit plane

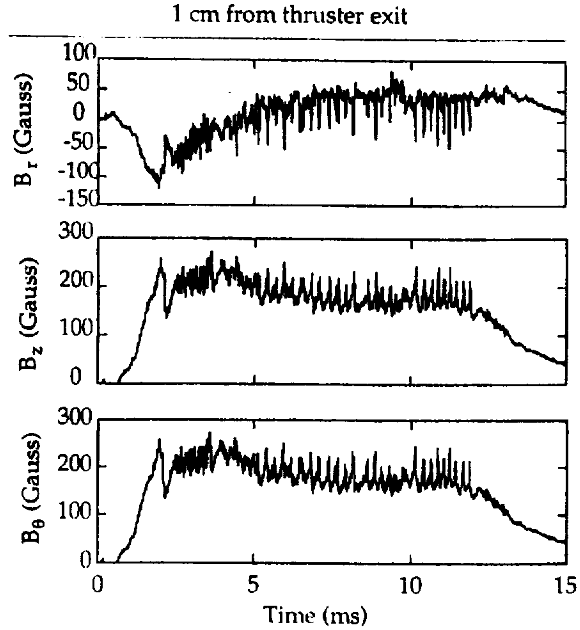


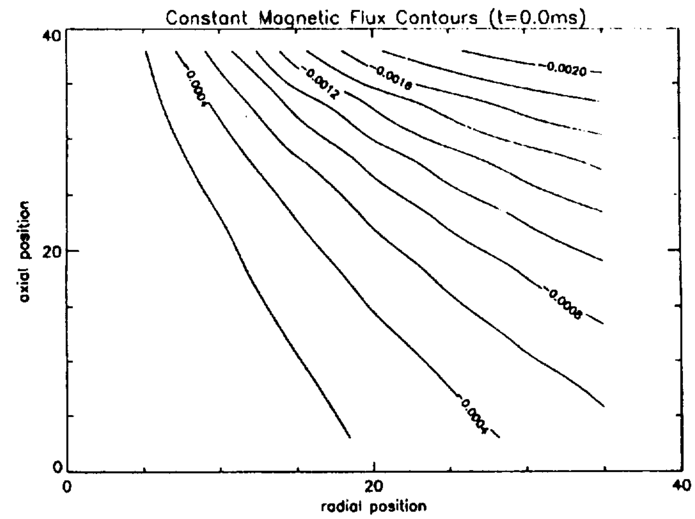




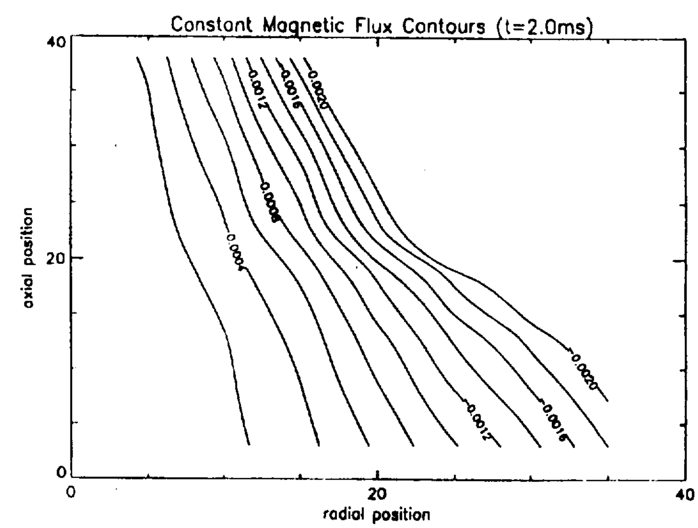
Magnetic Field Fluctuations

1 cm from thruster exit





Constant Magnetic Flux Contours (t=0.0ms)
axial position
radial position
0.0020
0.0016
0.0012
0.0008
0.0004



Constant Magnetic Flux Contours (t=2.0ms)
axial position
radial position
0.0020
0.0016
0.0012
0.0008
0.0004

LOS ALAMOS THRUSTER RESEARCH

Plans

- With quasi-steady-state capabilities:
  - Experiments to repeat electrode loss, plasma flow, power balance, and spatial magnetic field measurements on the unoptimized coaxial gun.
  - Control of anode fall by applied field.
  - Estimate of thruster efficiency through power balance.
- Design and construct an optimized applied field thruster.
- Repeat performance assessment.
- Apply research conclusions to MPD thruster design.

LOS ALAMOS THRUSTER RESEARCH

Concluding Remarks

Will the National Labs be advancing the state-of-the-art in electric propulsion in FY 94?

N93-26982

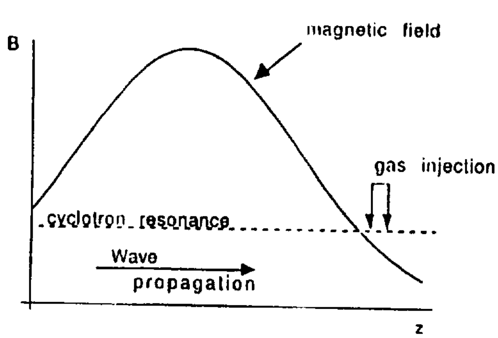
Electron Cyclotron Thruster
New Modeling Results
Preparation for Initial Experiments

E. Bickford Hooper
Lawrence Livermore National Laboratory

Presented at
Nuclear Propulsion Technical Interchange Meeting
NASA-LeRC Plum Brook Station
October 20-23, 1992

Whistler-Based ECRH Thruster — Concept

- A thruster using ECRH has no electrodes and, is thus less sensitive to materials problems than arc-based thrusters such as the Magneto-Plasma Dynamic (MPD) arc.



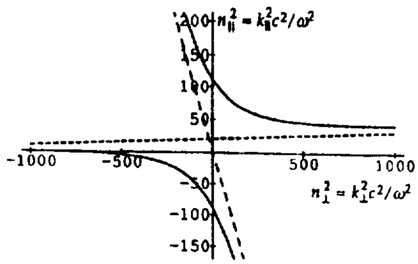
- Rear wall bombardment can be minimized, by a large mirror ratio between the resonance and peak field. (The flow across the mirror is reduced by approximately the mirror ratio from that downfield.) This:
  - o Maximizes efficiency by minimizing energy loss to the wall
  - o Maximizes lifetime by minimizing material damage

Cross-field Coupling in the Helicon Approximation

- Coupling is expected to be strongest if the magnetic field has a small gradient. Thus, we consider coupling at the peak of the magnetic mirror. There, ω_c/ω, $\omega_p/\omega \gg 1$. We illustrate the coupling at $\omega_c/\omega = 10$, $(\omega_p/\omega)^2 = 1000$. This is the helicon regime, with

$$\frac{k^2c^2}{\omega^2} \approx 1 - \frac{\omega_p^2}{\omega(\omega - \omega_c \cos\theta)} \approx \frac{\omega_p^2}{\omega\,\omega_c \cos\theta}$$

- The wave characteristics can be seen from a plot of the squared parallel vs perpendicular indices of refraction

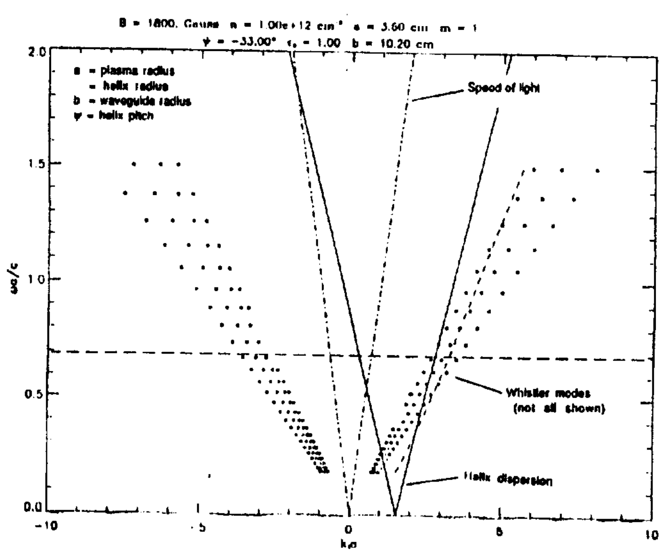


- Waves in the upper-right quadrent are propagating both along z and radially. These are the waves of interest
- There are two such waves at a given parallel index of refraction, but one is at very large perpendicular index of refraction and not of interest in the finite-radius plasma column
- The finite-radial geometry will pickout particular values of $n_\perp$

EBH 1/30-31/92

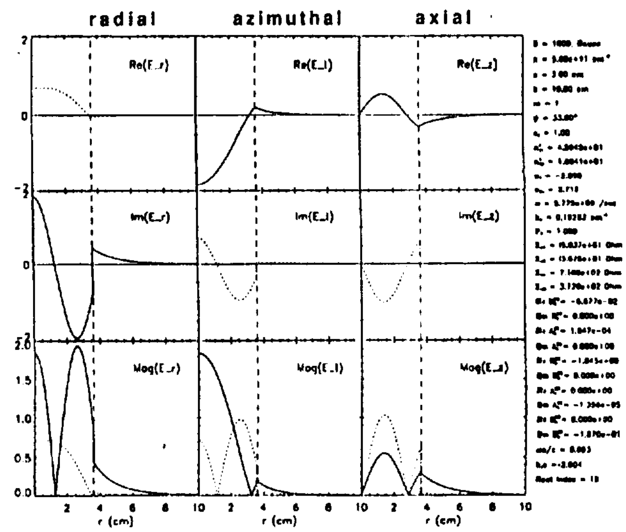
Wave propagation: Waveguide with helix and plasma column

- Several modes with different radial structure propagate in the system



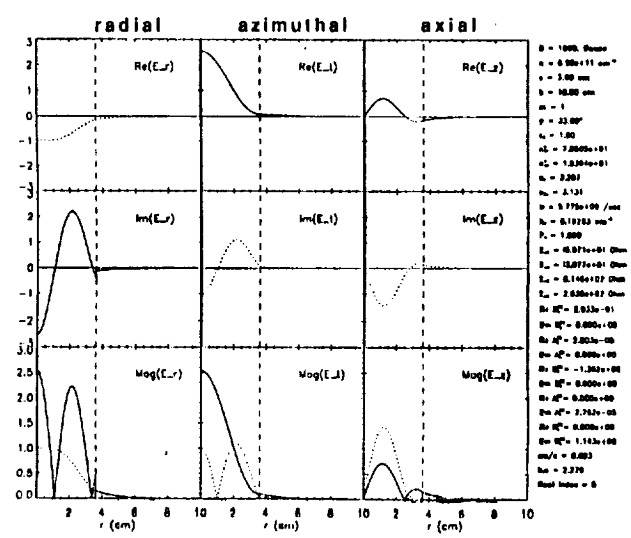
Wave structure: Low impedance mode

- Electric field = solid lines, magnetic field = dashed lines
- Note jump in magnetic field corresponding to current flow in helix



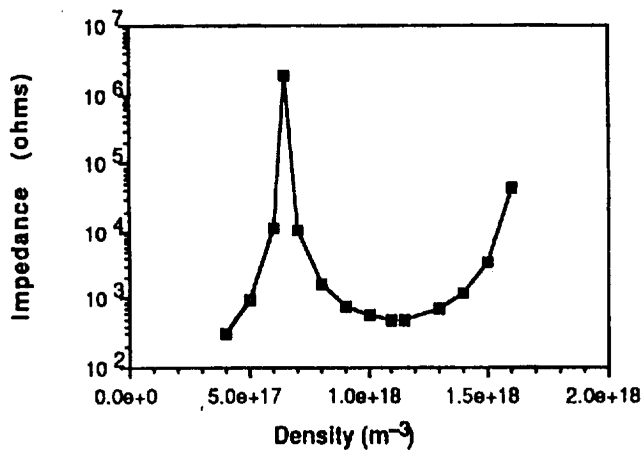
Wave structure: High impedance mode

- Electric field = solid lines, magnetic field = dashed lines
- Note no jump in magnetic field corresponding small current flow



System impedance varies with plasma density

- The experiment is designed to allow tuning of the microwave system



Wave Absorption at the Cyclotron Resonance

- As the whistler wave approaches the cyclotron resonance, the value of $k_{||}$ becomes very large and the phase velocity becomes small

  This has two favorable consequences for absorption:

  o The direction of propagation becomes nearly along the field and at short wavelength so that reflection is very small

  o The phase velocity becomes comparable to the thermal velocity of the particles, so that the Doppler-shifted resonance ($\omega - \omega_c - k_{||}v_{e||} = 0$) couples to the bulk electrons

- Furthermore, there is no electromagnetic plasma mode at high density and $\omega > \omega_c$, so the wave cannot tunnel through the resonance

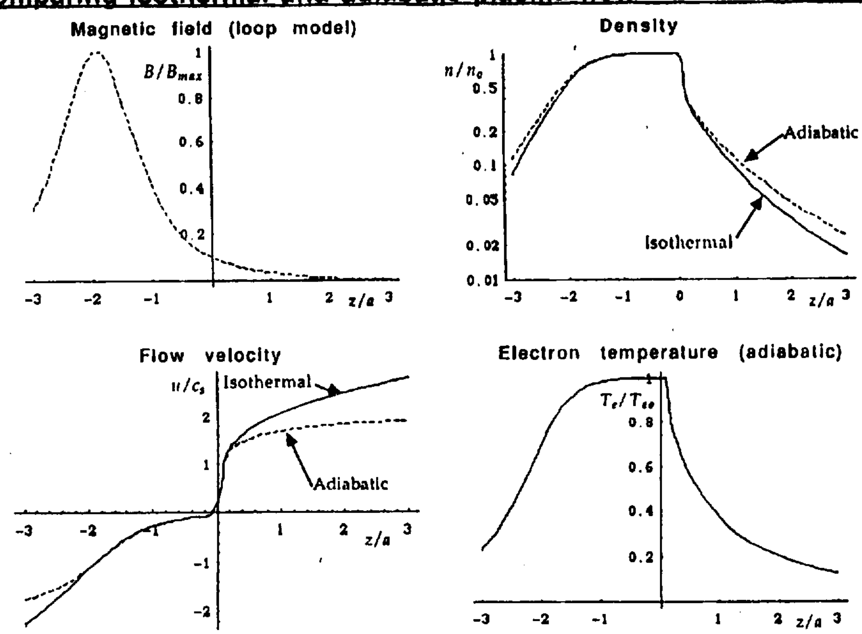
- Absorption is consequently nearly 100% for the whistler wave at the cyclotron resonance

- Absorption at high power will generally generate a nonthermal electron velocity distribution. Calculations are needed to quantify this and its consequences

Flow sensitivity to electron distribution function

- The isothermal and adiabatic limits illustrate the sensitivity of the flow to the thermal conductivity and thus to the electron distribution function
- For ECRH the electron distribution may be anisotropic and nonthermal in nature, with significant consequences for thermal conductivity, particle and energy flow, plasma recycling at the rear wall, etc.
- Understanding the distribution resulting from the heating, as a function of plasma density and microwave power, is thus key to predicting performance.

Comparing isothermal and adiabatic plasma flow



ECR thruster modeling: heating and plasma flow

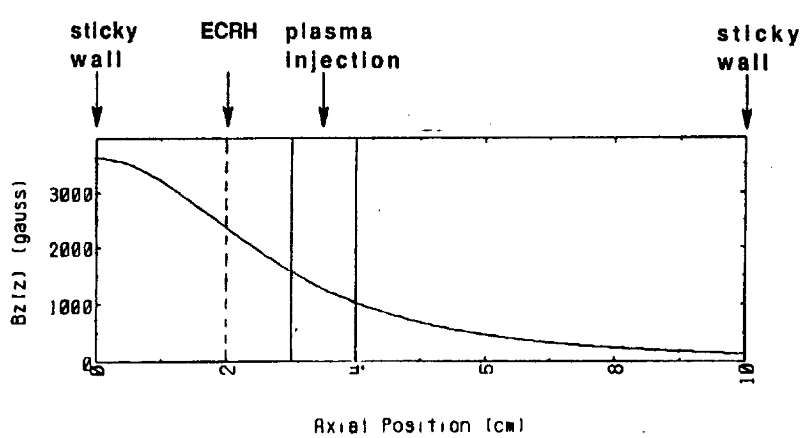
- A particle-in-cell code – ICEPIC – has been used to model the thruster plasma heating and motion along the magnetic field
- Individual particles are followed in the guiding center approximation
  - o Electrons are heated by rf with velocity-space diffusion in the quasilinear approximation
  - o For the present cases, the electrons are weakly collisional
  - o The ion mass is $100m_e$ to speed up calculations
- Plasma is injected on the side of a magnetic hill and heated up the hill from the injection point
- Two cases are compared

	Injected T_e	Injected T_i	ECRH
No ECRH	100 eV	5 eV	None
ECRH	5 eV	5 eV	E_{rf} = 320 V/cm

Geometry for PIC code model

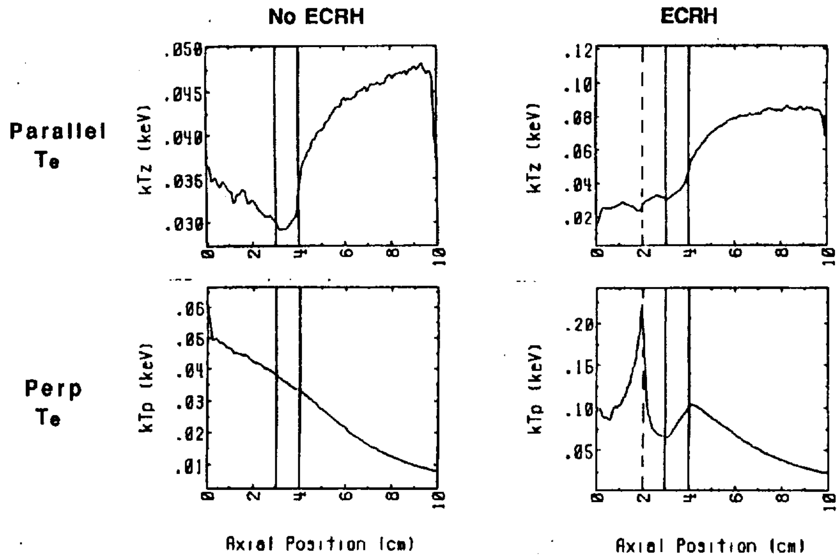
Magnetic field strengths

z(cm)	0	2	3.5	10
B(gauss)	3650	2350	1250	125
B(0)/B	1	1.6	2.9	29



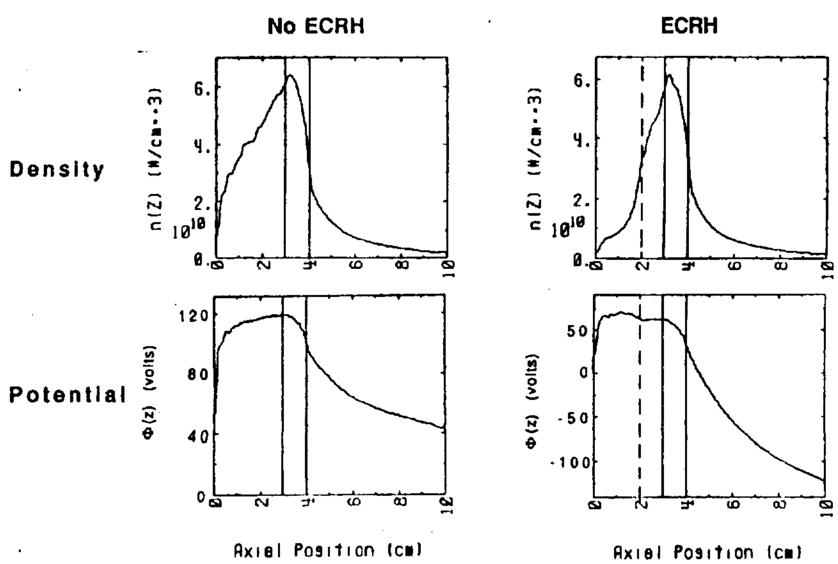
Electron "temperature" moment in the flow

- The electrons are highly anisotropic even without ECRH
- The electron temperature is highly nonuniform along B
- Strong electron heating by ECRH is evident perpendicular to B

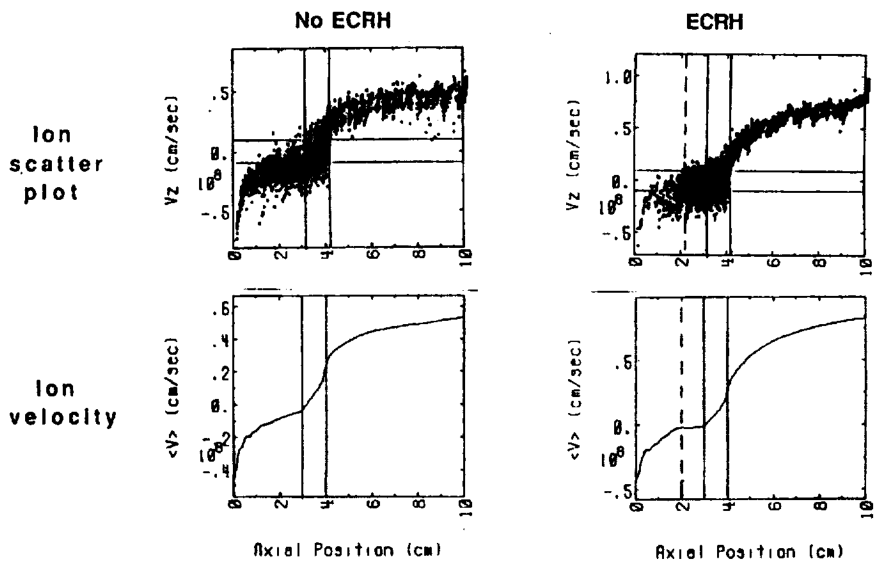


Density and potential are strongly affected by ECRH

- Note the rise in potential upfield of the ECRH. It reduces the flow of ions to balance the $\mu \partial B / \partial s$ force on the electrons and maintain quasineutrality

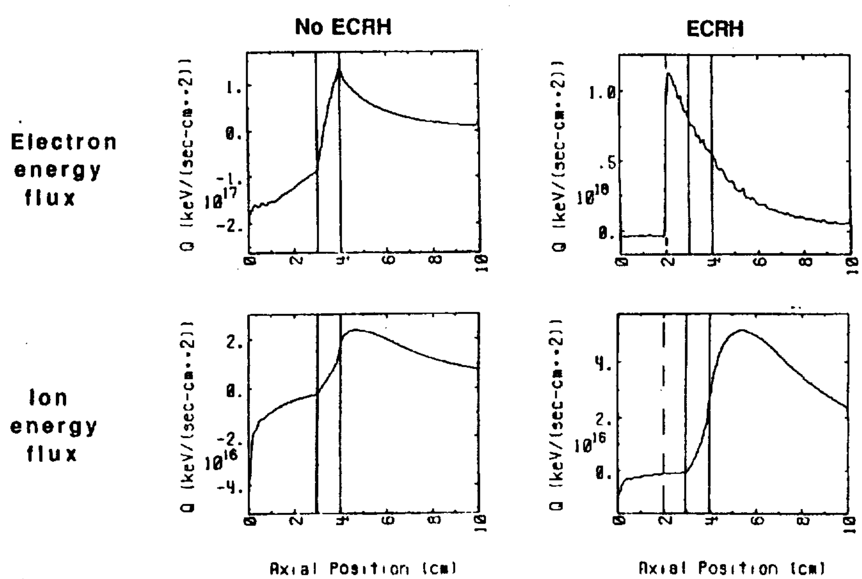


Electron energy is converted into ion flow



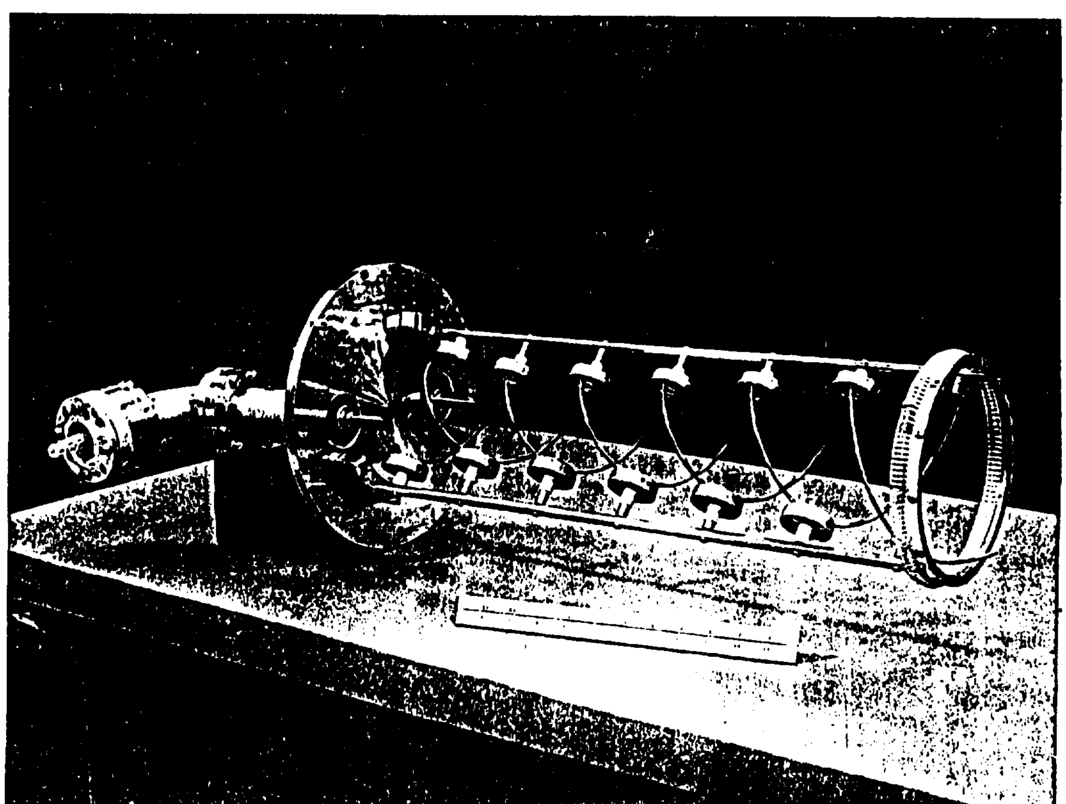
Energy flow up the field is suppressed by ECRH

- The total energy flow is proportional to the flux bundle area, which is a factor of 29 larger at the exit than at the magnetic field peak



Initial experimental tests: preparation

- Initial experiments will be conducted at NASA LeRC (tank 7)
  - o Space has been provided; magnets and SCR controller for pulsing microwave power have been sent to LeRC
  - o Microwave components have been delivered to LeRC
  - o Vacuum vessel, helical coupler, and gas box have been constructed and are undergoing final bench tests at LLNL
- First experiments will be directed to forming the plasma and making preliminary measurements of density, electron temperature
- Subsequent experiments will explore the details of the plasma for comparison with modeling
  - o Electron anisotropy
  - o Suppression of flow to rear wall
  - o Efficiency
- Measurements will also be made of the separation of the plasma plume from the magnetic nozzle



NUCLEAR ELECTRIC PROPULSION

SYSTEMS MODELING

20 kWe NEP FLIGHT SYSTEM

NUCLEAR PROPULSION TECHNICAL INTERCHANGE MEETING

PLUMBROOK STATION

NASA LEWIS RESEARCH CENTER

OCTOBER 22, 1992

Jim Gilland

Sverdrup Technology, Inc./Nuclear Propulsion Office

20 kWe NEP FLIGHT SYSTEM

NASA — LEWIS RESEARCH CENTER

STUDY BACKGROUND

- High Level Interest in Early Flight of SP-100 Based NEP System
  - OAST AA
  - Joint NASA/DOE Space Nuclear Power and Propulsion Team
- Mission Emphasis is Space Science
  - Planet, Asteroid Exploration
  - NASA User: Office of Space Science and Applications

NUCLEAR PROPULSION OFFICE

STUDY BACKGROUND

A low power near term NEP system has been proposed as a useful interim system for near term space exploration. Although the ultimate goal of a 100 kWe class, low specific mass for planetary exploration remains, application of the technologies that are currently mature to earlier missions of interest has grown at the higher levels of NASA. In response to this interest, a study of low power system and mission options has been initiated, with the Nuclear Propulsion Office serving to coordinate system activities. A nominal 20 kWe system using Brayton power conversion has been selected by the joint NASA/DOE Space Nuclear Power and Propulsion team; however, other power levels and system options will be considered. NASA's Office of Space Science and Applications has expressed interest in exploiting NEP's mission capabilities, both in the near term and for more difficult, later missions.

Technologies considered mature for this type of system are the SP-100 reactor, Brayton dynamic power conversion, and 30 cm ion thrusters, all of which have extensive ground demonstration backgrounds.

NASA — LEWIS RESEARCH CENTER

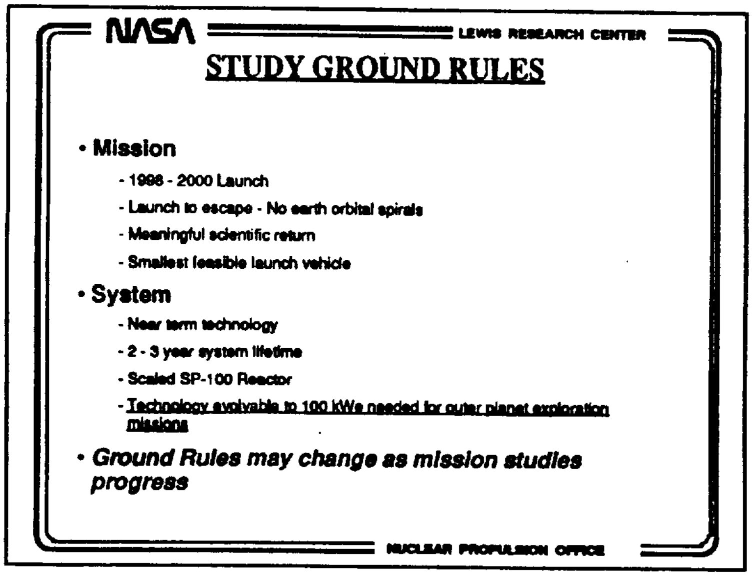
Study Participants

- **NASA HQ**
  - Code RP - Gary Bennett
  - Code SL - Carl Pilcher/Doug Stetson
- **NASA Lewis**
  - Nuclear Propulsion Office
  - Power Technology Division
  - Space Propulsion Technology Division
  - Advanced Space Analysis Office
- **JPL**
  - Mission Design Section
  - Power and Propulsion Section
  - Propulsion and Chemical Systems Section
  - SP-100 Program Office

NUCLEAR PROPULSION OFFICE

Study Participants

The full assessment of a 20 kWe NEP system and its applications has drawn together a team spanning NASA's Codes S and R, including experts from both Lewis Research Center and the Jet Propulsion Laboratory. The team includes mission planners, power system engineers, electric propulsion researchers, and program level managers. Mission design and analysis is primarily the responsibility of Code S, while system design and technology assessment is the responsibility of Code R.



STUDY GROUND RULES

The concept of a near term NEP flight and science mission is based on achieving certain goals in terms of timely delivery of scientific information as well as timely use of mature technologies. In this case, near term means a launch in 1998 to 2000. Some initial ground rules that have been imposed on the study to date are that the mission should leave Earth orbit, and gather data useful to space scientists. On a system level, a power level of 20 kWe and a lifetime of 3 years were mandated for initial studies. The combination of low lifetime and power leads to a mission requirement of launch to escape. In the interest of low cost and easier launch scheduling, expendable launch vehicles are assumed, up to and including a Titan IV/Centaur as the largest option. A further ground rule was that the technology used on this early mission has some bearing on the development of the ultimate 100 kWe outer planet systems.

These are initial ground rules, based on preliminary conceptions of mission performance. As more detailed analysis warrants, these assumptions can change to incorporate improved data.

NASA LEWIS RESEARCH CENTER

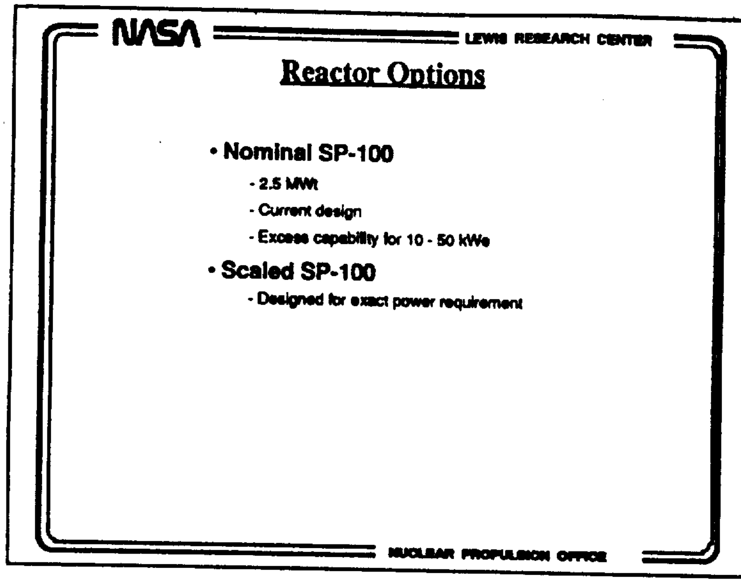
System Analysis Assumptions

- 10 - 50 kWe
- 3 year life
- 2000 V to load
- 15 m reactor-to-payload separation distance
- Payload radiation dose:
  - 1.0×10^{12} neutron/cm^2
  - 5×10^4 rad(Si) γ
- 17° shield half-angle
- 10% excess heat rejection capacity

NUCLEAR PROPULSION OFFICE

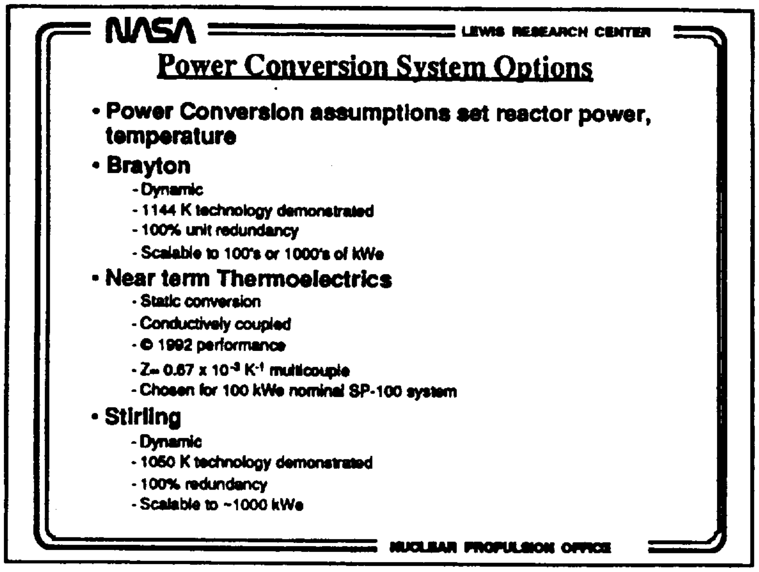
System Analysis Assumptions

System assumptions are shown above. Of primary importance are the separation distance and radiation dose constraints. These are lower than those identified for the 100 kWe SP-100 mission, impacting relative shielding mass. The lower doses are aimed at using near term electronics rather than radiation hard materials. In addition, the lower dosages may ameliorate interference of the power system with scientific instruments. The shorter boom length allows for greater ease of packaging and deployment in expendable launch vehicles. Improved system mass might be achieved through the use of a greater separation distance; however, this must be included in a detailed trade versus technology readiness and packaging concerns. The above assumptions were imposed on all systems designs, regardless of reactor or power conversion selection.



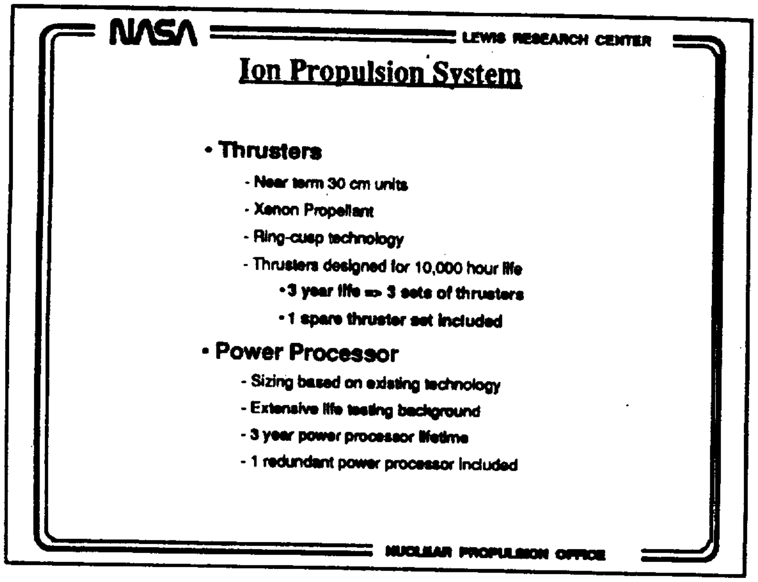
Reactor Options

Two reactor options were considered in these studies: a full power, 2.5 MWt SP-100 reactor, with excess capability for the low power system, and a scaled reactor designed for exactly the thermal power required for a given electric power output. Due to the desire to obtain a minimum mass system, the scaled option has been baselined; however, the full power option would provide experience in fabricating the same reactor that will be used in the later, 100 kWe planetary exploration system. These two options represent an additional trade which will have to be performed to determine the most effective development approach.



Power Conversion System Options

Three power conversion system options were considered: the baseline Brayton, near term thermoelectrics, and a near term Stirling system. The Brayton system is based on the Brayton Rotating Unit (BRU) developed and tested at NASA Lewis Research Center in 1966-1968. Lifetimes of up to 41,000 hours (>4.5 years) were demonstrated at 1144 K with this system. A system redundancy of 100% (1 spare power conversion unit) was assumed in mass estimates. Of the alternatives, the near term thermoelectrics is based upon interim technology thermoelectric elements, based on performance demonstrated in 1992. The thermocouples are the precursors to the elements that are to be used on the 100 kWe nominal system, maintaining an evolutionary link to the ultimate system. The Stirling option is based upon a low temperature technology that has been tested in the laboratory, although not to the level of the BRU.



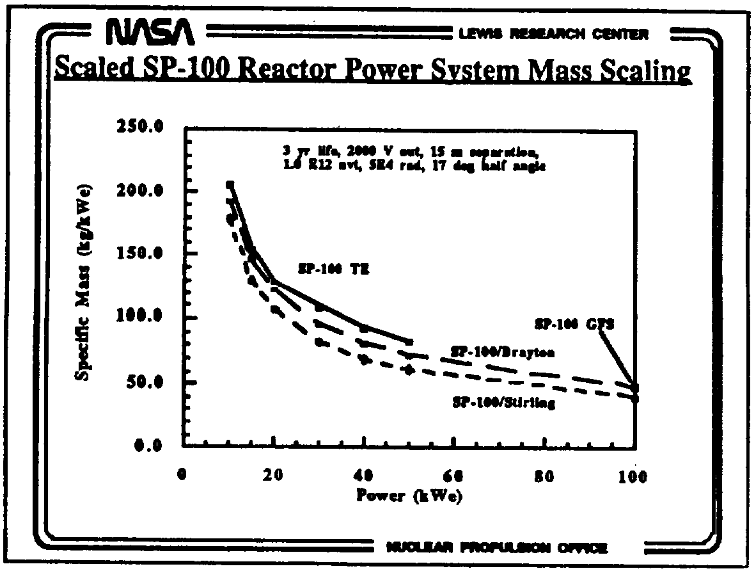
Ion Propulsion System

The electric propulsion system uses 30 cm diameter ion thrusters operating on xenon propellant. Thrusters of this size using xenon have been ground tested extensively, and the thruster designs build on flight testing and development of ion thrusters extending back to the 1960's. Life testing of these thrusters has identified regimes of operation to permit 10,000 hours life, and these regimes have been assumed in thruster system design. Performance parameters have been generated over a range of specific impulses for these thrusters, to allow flexibility in mission analysis and optimization. Thruster masses are based upon flight like thrusters that were constructed in 1992.

The assumed electric propulsion power processing electronics share a heritage with the thrusters. System mass estimates have been based on scaling equations taken from actual flight systems and designs. Power processors have demonstrated lifetimes more than adequate for the full mission life assumed in this study.

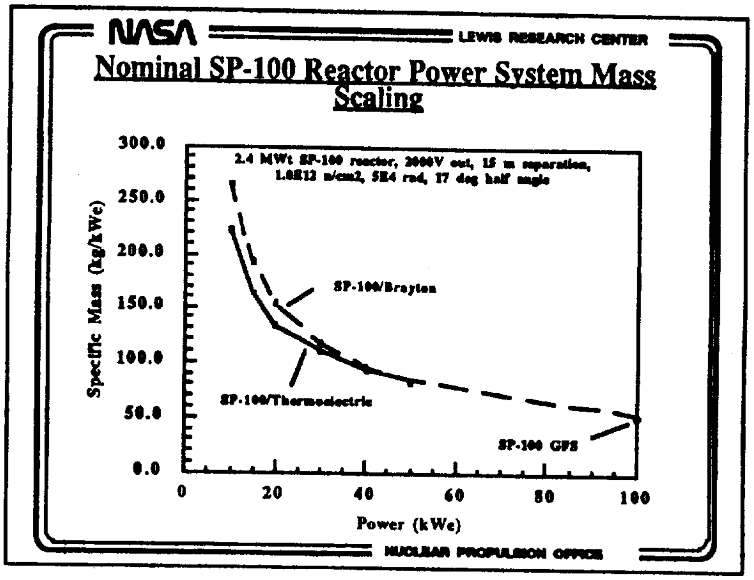
In order to meet system lifetime requirements, several sets of thrusters are required. Three years of life is 26,280 hours, requiring 3 sets of thrusters to ensure suitable lifetime. An entire redundant set of thrusters has been included in the system mass to provide an additional level of reliability. Each thruster in a set is assumed to have its own power processor; however, In the case of the power processor, a single unit should operate for the entire life of the mission. One set of spare units is included for additional reliability.

As mission analyses mature, the exact number of thrusters and power processors required will be determined and more exact system designs can be developed.



Scaled SP-100 Reactor Power System Mass Scaling

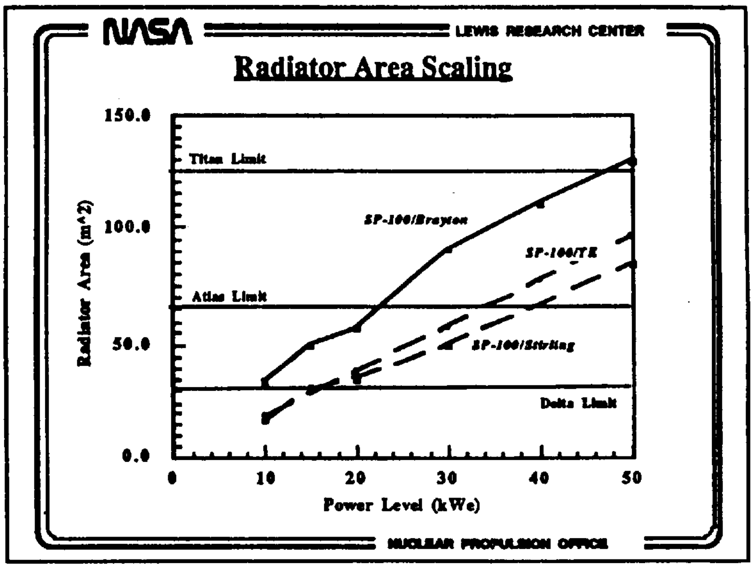
Results of power system analysis are shown above for the case of the scaled SP-100 reactor. Specific mass includes boom and transmission to the spacecraft bus. Electric propulsion specific mass is not included, as this will vary with specific impulse as well as power. A significant penalty in specific mass is seen at power levels below 30 kWe, due to the limits in scaling of the reactor and shield. However, some launch vehicle payload mass and volume considerations may restrict the system to these lower powers.



Nominal SP-100 Reactor Power System Mass Scaling

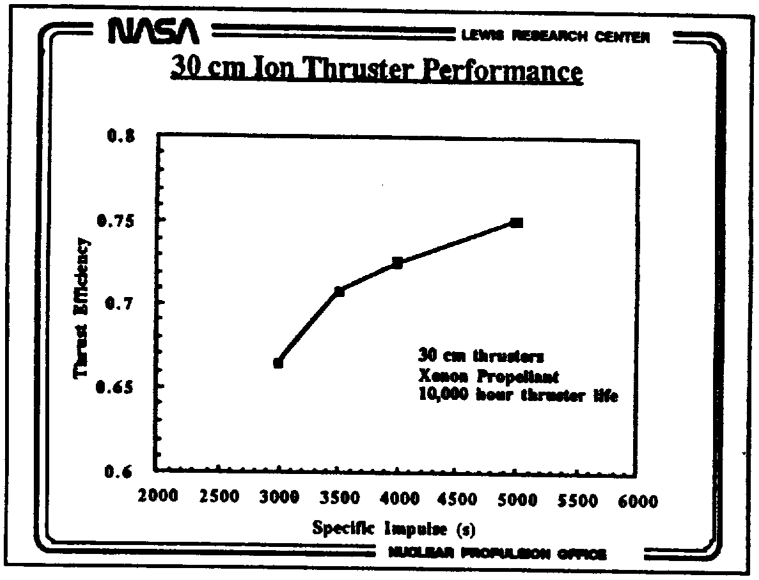
Comparable results are shown for the case using the nominal 2.5 MWt reactor. At 20 kWe, there is approximately a 25 kg/kWe penalty for using the larger reactor. Again, mission and development cost analyses are needed to determine the impact of this difference on the implementation of the early NEP system.

N93-26984



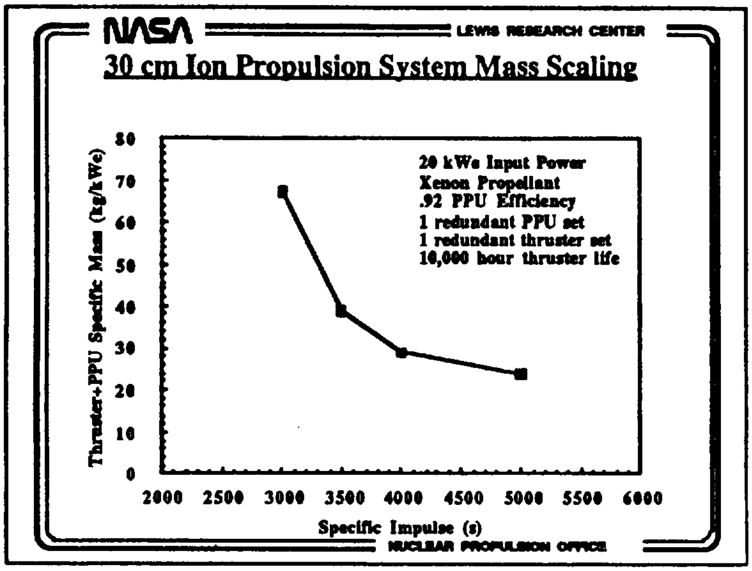
Radiator Area Scaling

Radiator area scaling is shown for the three options, with corresponding launch vehicle volumetric limits provided for reference. Volume limits are for the entire launch vehicle shroud, with no allowance for upper stage. The trade between Brayton and thermoelectrics is shown in the relative area for the two. The higher rejection temperature of the thermoelectrics allows a reduced radiator area. System specific masses are comparable, however, due to the higher efficiency of the Brayton power conversion. System and mission analysis will ultimately be based on three primary points: mission performance (specific mass), development time, and launch vehicle compatibility.



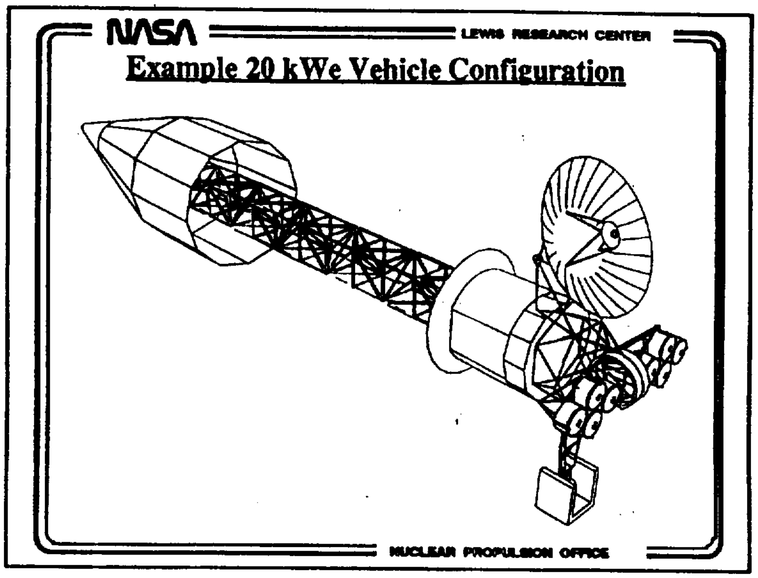
30 cm Ion Thruster Performance

Projected ion thruster performance is shown in terms of thrust efficiency and specific impulse. These data are necessary for trajectory and system optimization, in order to determine the proper design point in terms of thruster specific impulse and system power.



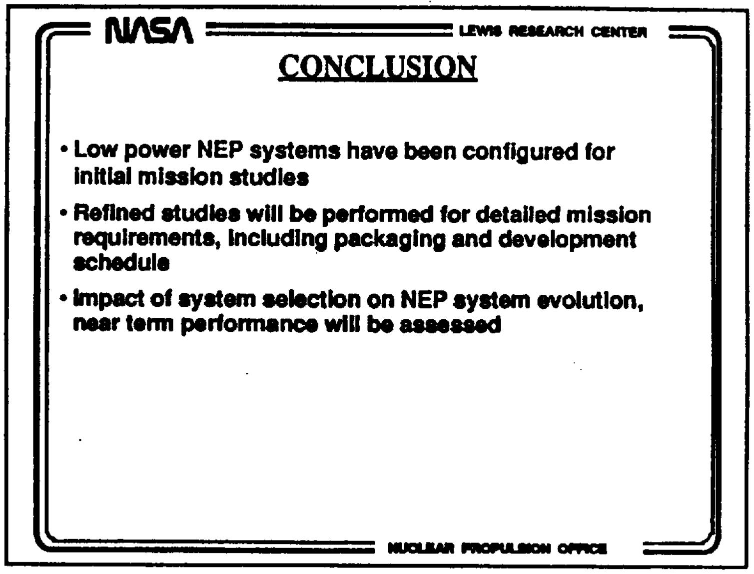
30 cm Ion Propulsion System Mass Scaling

The ion propulsion system includes thrusters, gimbals, power processors and associated thermal control. The above system is for a fixed input power to the power processor of 20 kWe. Specific mass decreases with specific impulse because of the decrease in the number of thrusters required to process the power. Included in the specific mass budget are an extra set of thrusters and power processors (PPU). The system is designed to last 30,000 hours, or almost 3.5 years. These data, in addition to specific masses for other lifetimes, have been provided to the mission analysts for more detailed trajectory analysis.



Example 20 kWe Vehicle Configuration

A conceptual design of a 20 kWe NEP vehicle configuration is shown above. Of key interest at this stage of the analysis is the design of the radiator and the location of the thrusters. These components have the potential for the greatest amount of interaction with the payload and launch vehicle. Overall vehicle integration will require detailed assessments of the configuration of these components. In addition, thruster location determines vehicle trajectory and steering capabilities. Placement of thrusters and their electronics will also impact transmission line designs. Currently, system designs assume that the thrusters are mounted as shown above, with the greatest distance between power processors and power conversion.



CONCLUSION

A range of low power NEP system performance parameters have been defined for initial scoping mission studies. Following the initial mission assessment, more refined studies will be developed. Included in these studies will be a development schedule and cost analysis for the system of interest, including the flight system. Trade studies of system options, such as the nominal versus scaled reactor options, will continue in parallel with mission analysis.

100 - 500 kWe NEP Systems

Nuclear Propulsion Technical Interchange Meeting
LeRC Plum Brook Station
October 22, 1992

Jeff George
Advanced Space Analysis Office

NASA Lewis Research Center
Advanced Space Analysis Office

100 - 500 kWe NEP Systems

- Use 2.4 MWt SP-100 reactor / dynamic power conversion
- Enhancing to 100 kWe thermoelectric SP-100
- Serve as interim step between 100 kWe and multimegawatt NEP
- New NEP mission/performance regime

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System/Technology Assumptions

- SP-100 Reactor
  - fast spectrum, lithium-cooled, pin type
  - 2.4 MWt
  - 1375 K out
  - 7 yr life
- Dynamic Power Conversion
  - 1100 K Brayton
  - 1300 K Brayton
  - 1300 K Rankine
  - 1 to 4 100-125 kWe "modular" power conversion loops
  - 2000 V to load
- Heat Rejection
  - 10 kg/kWe (SP-100 program)
- Krypton Ion Thrusters
  - 50-100 cm
  - 3000-7000 sec Isp
  - 50-150 kWe/thruster
  - 6 kg/kWe

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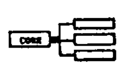
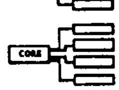
Electrical Output Power of Modular Dynamic Power Conversion Systems

Conversion Loops	Low Temperature Brayton Cycle 100 kWe Loops	High Tempetature Brayton Cycle 125 kWe Loops	Rankine Cycle 125 kWe Loops
	100	125	125
	200	250	250
	300	375	375
	400	500	500

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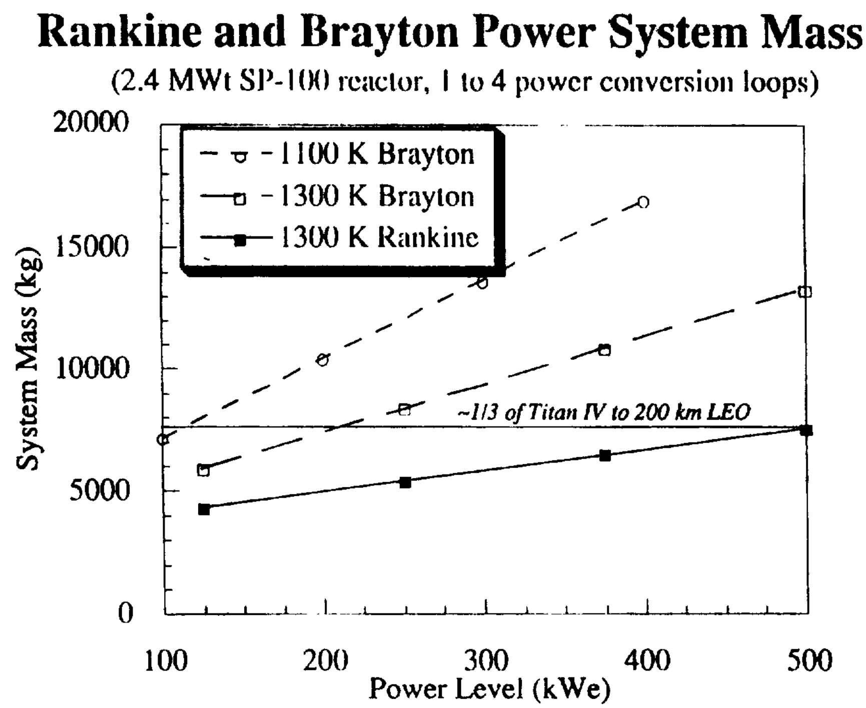

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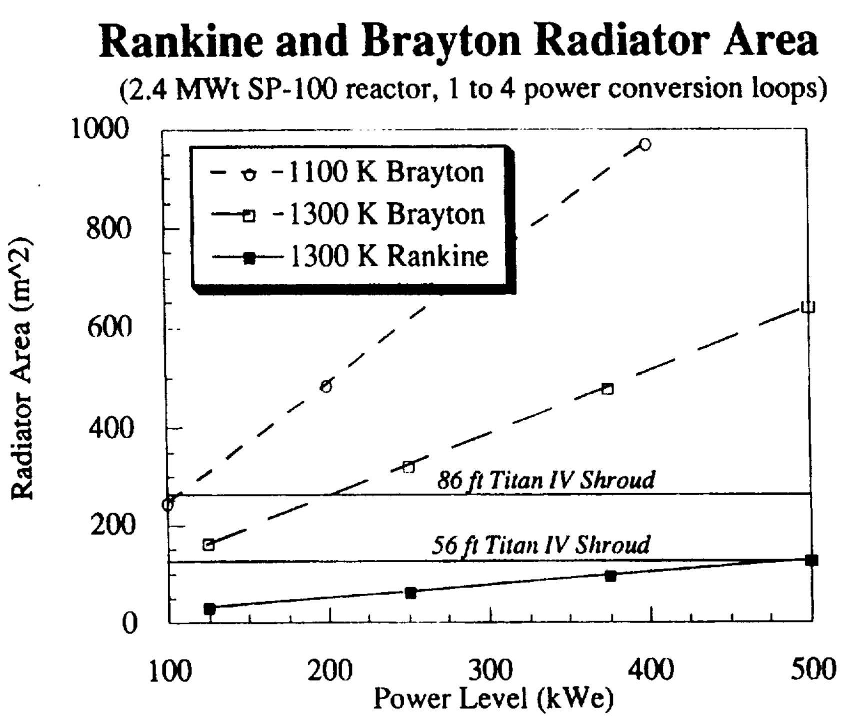

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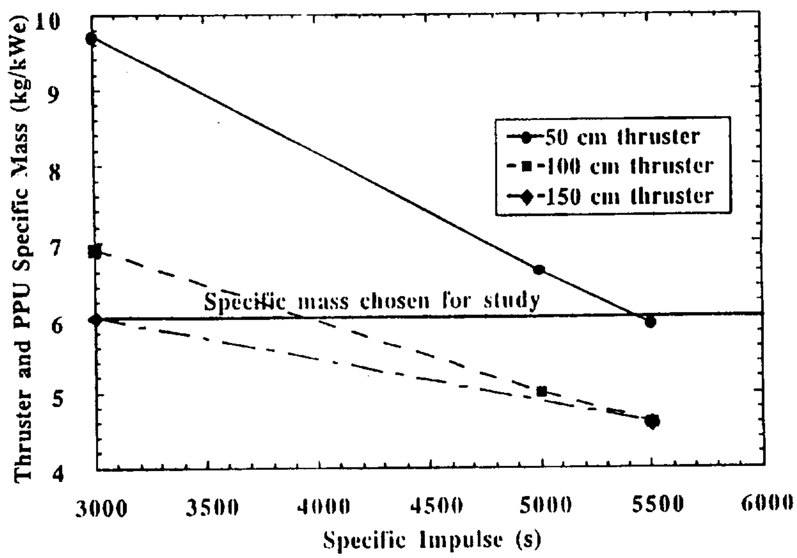
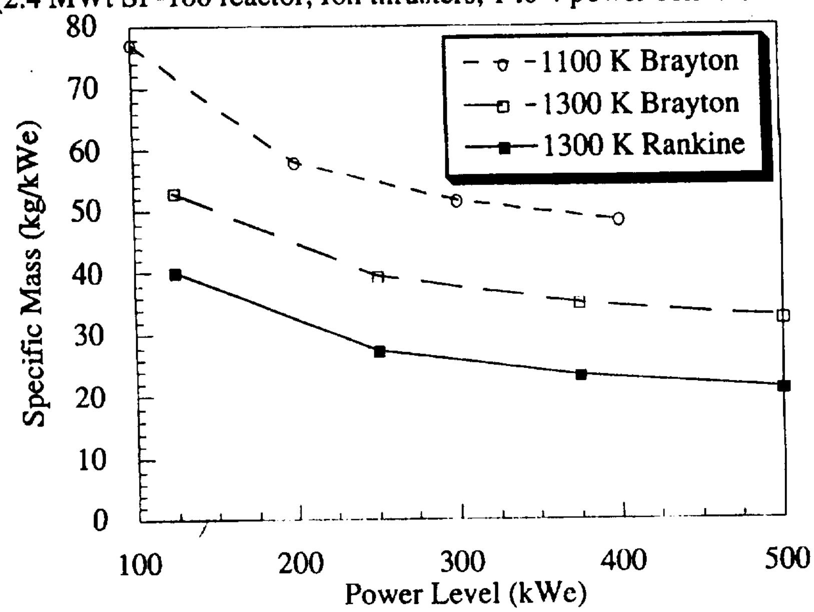
KRYPTON ION THRUSTER MASS SCALING (500 KWe)

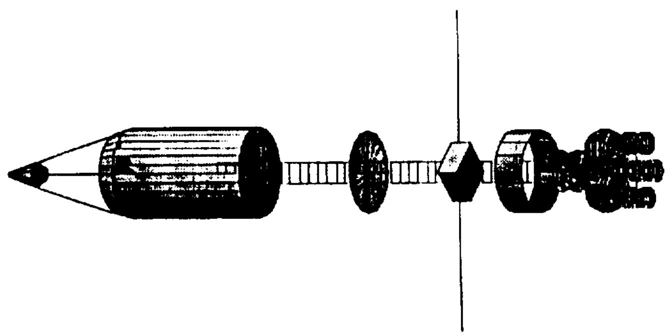


NEP System Specific Mass for Rankine and Brayton Power Conversion

(2.4 MWt SP-100 reactor, Ion thrusters, 1 to 4 power conversion loops)



500 kWe SP-100/K-Rankine/Ion NEP Vehicle



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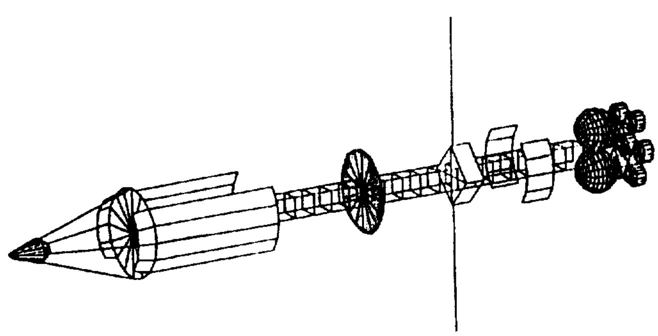
250 kWe SP-100/K-Rankine/Ion NEP Vehicle



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NEP MISSIONS

- **Lunar Cargo**
  - **Scenario:**
    - Depart LEO (400 km)
    - Spiral to Moon, Capture at Moon
    - Spiral down to Low Lunar Orbit (LLO)
    - Return Empty
  - Payload:
    - 40 MT to lunar surface
    - 39.5 MT lunar lander
  - Trip Time:
    - Round trip time < 1 year
    - **Trip Time = Reactor, thruster operating time**
  - Reference Cargo Vehicle:
    - Cryogenic LOX/LH2
    - Isp: 468 seconds
    - IMLEO: 267 MT
    - Trip Time: 3 days

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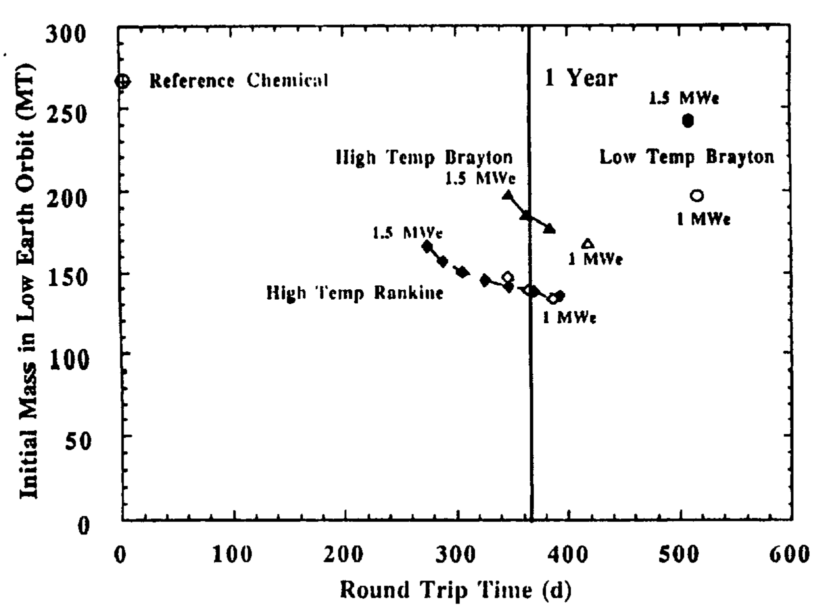
EARLY TRACK NEP LUNAR CARGO MISSION PERFORMANCE



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RESULTS

- **1350 K Rankine, Brayton provide system beneficial to SEI objectives**
- **Lunar Cargo:**
  - 1350 K power systems at 1- 1.5 MWe allow 90 - 130 MT savings over chemical vehicle (up to 50% reduction)
  - Round trip times: 250 days - 1 Year
- **Mars Cargo:**
  - 1350 K power systems at 1- 1.5 MWe allow mass performance comparable to advanced NTP systems
  - Trip Time: 500 days - 2 Years

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CONCLUSIONS

- **Early Track NEP provides the option for "faster, cheaper" implementation of advanced propulsion for SEI**
- **Other areas of application:**
  - Space Science - significant augmentation to exploration of outer planets and beyond
  - Precursors - Early Track NEP to Mars for robust mapping, sample return, subsurface probing
- **Technology Developments Required:**
  - Dynamic Power Conversion
  - Scaled Krypton Ion Thrusters
    - **MPD Thrusters may also be an option**
  - System integration

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Nuclear Electric Propulsion Options for Piloted Mars Missions

Nuclear Propulsion Technical Interchange Meeting
LeRC Plum Brook Station
October 22, 1992

Jeff George
Advanced Space Analysis Office

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NEP for SEI Mars Missions

- Synergy with Surface Power Technology
- "Fast" Piloted Missions
- Efficient Cargo Delivery
- Fewer and/or Smaller (135 MT) Launch Vehicles
- Continuous Abort Mode
- Continuous Earth Return Window
- Technology:
    - Existing Reactor Technology Program
    - Need Potassium Rankine Power Conversion
    - Need Multimegawatt Ion Thrusters

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Why not NEP?

- Long Earth spiral escape times
  - Impractical piloted lunar missions
  - Chemical crew taxi for piloted Mars
- Long operating times
  - High reliabilities necessary
  - Complications for artificial gravity
- Multiple technologies
  - Reactor
  - Power Conversion
  - Thrusters

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NEP Technologies

- Reactor
  - 2 yr life, 25 MWth SP-100
  - Li cooled, fast spectrum, UN fuel, Nb-1Zr clad
  - Technology developed in current SP-100 program
- Power Conversion
  - 1400 K Potassium Rankine
  - SNAP-50 tested components at 1420 K for 10,000 hours
    - 3-5 life projected from turbine erosion
- Thrusters
  - Argon ion engines, 5000 sec. Isp, 69 % efficiency, 10,000 hour life
  - Efficiency and life demonstrated at Isp but lower power
  - EP will be used on upcoming Telstar IV

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2 x 5 MWe Reduced Life Growth SP-100 NEP System

Reactor	**Li-cooled pin-type fast reactor**
Power Conversion	**Potassium Rankine**
Power Output	**5 MWe/Module**
Full Power Life	**2 yrs**
Propulsion	**Ion**
Cycle Characteristics:	
Turbine Inlet Temp.	**1400 K**
Condenser Temp.	**975 K (Min. mass)**
Thermal-Electric Eff.	**20.5 %**
Reactor:	
Spectrum	**Fast**
Coolant	**Lithium**
Fuel	**UN pins**
Cladding	**PWC-11**
Structure	**PWC-11**
Man-rated Shadow Shield:	
Dose Constraint	**5 rem/yr**
Materials	**W / LiH**
Dose Plane Diameter	**20 m**
Separation Distance	**100 m**
Heat Rejection:	
Type	**Heat Pipe Radiators**
Geometry	**Planar**
Specific Mass	**6 kg/m2**
Total Radiator Area	**693 m2/Module**

System Mass Breakdown:

Reactors	6990 kg
Shielding	12200 kg
Power Conversion	19060 kg
(4+2 T-G units, 50% redundancy)	
Radiators	8320 kg
Power Cond. & Dist.	20000 kg
Ion Propulsion	6000 kg
Total (2 Modules)	72570 kg
Specific Mass	7.3 kg/kWe

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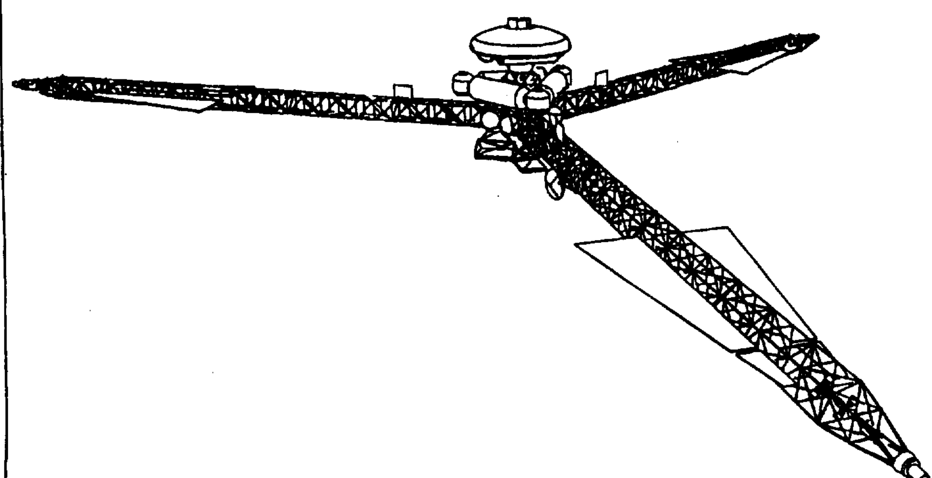
15 MWe Multi-Reactor Nuclear Electric Propulsion Vehicle for a Piloted Mission to Mars



ADVANCED SPACE ANALYSIS OFFICE

Groundrules

- Systems
  - Modular/Multiple Power Systems
    - Growth SP-100 Reactor
    - 1400 K Potassium Rankine Power Conversion
  - Argon Ion Engines
    - 5000 sec Isp
    - 68.9 % efficiency
    - 10,000 hour life
  - 7.3 kg/kWe
  - 10 % Tankage Fraction
  - 10 MT Inerts/Structure Mass
- Orbits
  - SSF Altitude Earth Departure Orbit
  - Crew boards at HEO
  - Areosynchronous Orbit at Mars
  - ECCV return at Mars (9.4 km/sec V_∞ Limit)

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Payload Assumptions

ECCV	7 MT
Transit Habitat	55 MT
Piloted MEV	65 MT
Cargo MEV	65 MT

- Unless otherwise noted - all Piloted NEP missions presented carry return propellant

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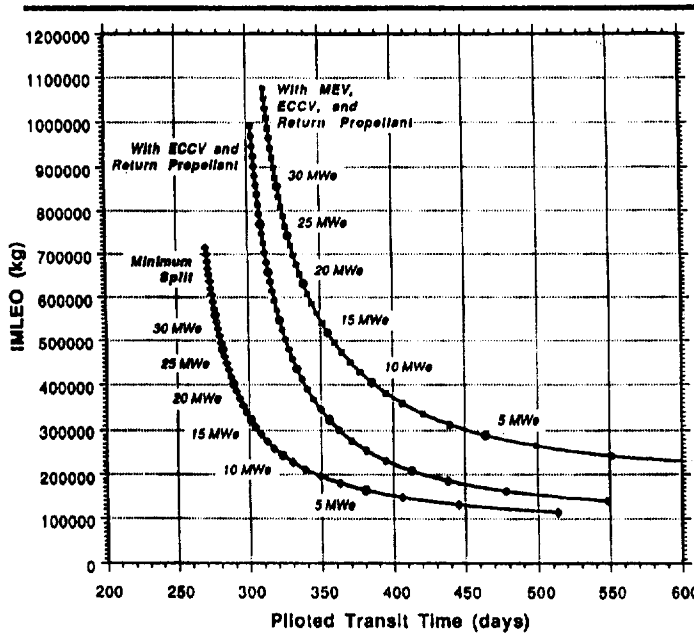
Conjunction Mission Performance for the 2010 Mission Opportunity



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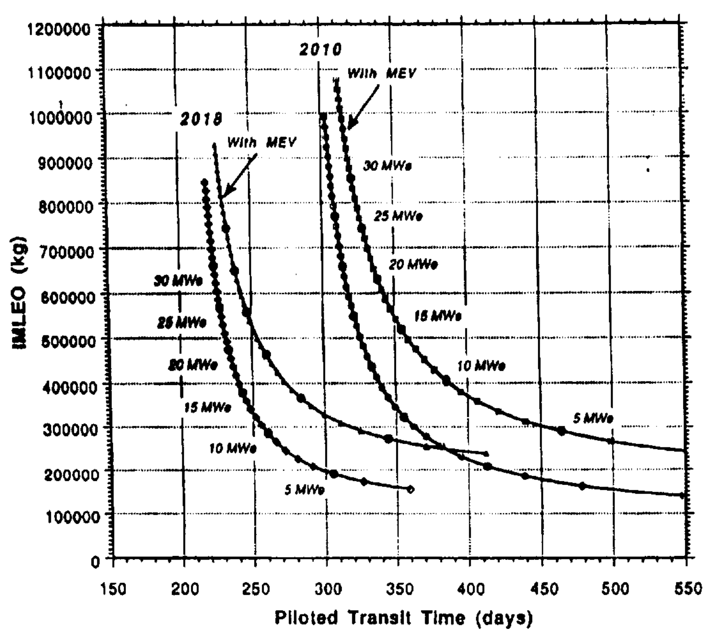
Conjunction Mission Performance over Various Opportunities



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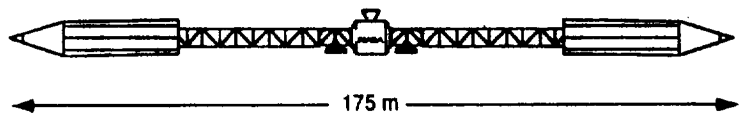
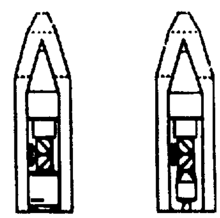
10 MWe Piloted Mars NEP with ECCV

2 x 5.0 MWe Modular "Hydra" NEP Vehicle



2 x 181 MT HLLV Launches



179 MT 41 m x 8 m

181 MT 41 m x 8 m

	2010	2018
Piloted Transit Time:	193 d ±180 d 373 d	154 d ±106 d 260 d
IMLEO:	310 MT	285 MT

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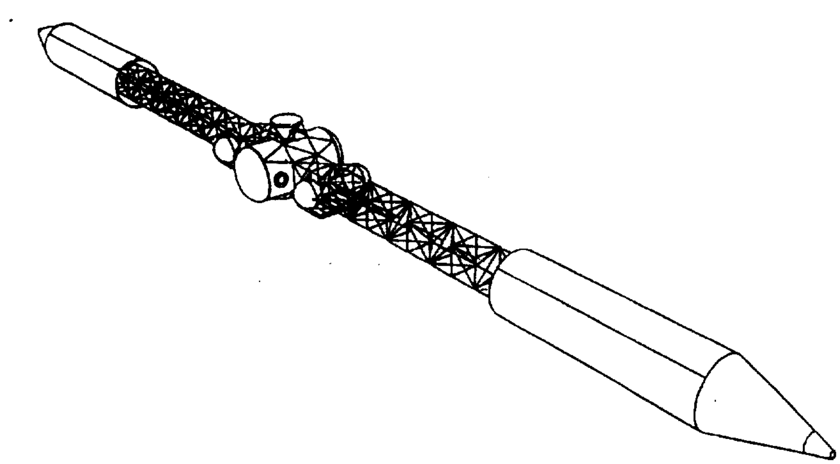
10 MWe Modular NEP Piloted Mars Vehicle



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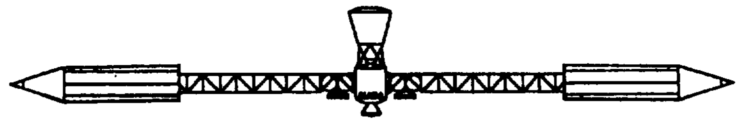
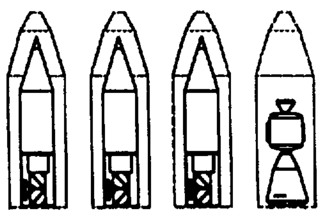
10 & 15 MWe Piloted Mars NEP with ECCV & MEV

2 x 5.0 MWe Modular "Hydra" NEP Vehicle



3-4 x 132 MT HLLV Launches



	2010*	2018
Power:	15 MWe	10 MWe
Piloted Transit Time:	200 d +180 d 380 d	177 d +106 d 283 d
IMLEO:	479 MT	367 MT

* - Optimal leg distribution 221+134=355 d & 518 MT

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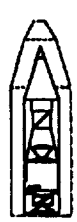
5 MWe Piloted Mars NEP with ECCV

5.0 MWe Piloted NEP Vehicle

1 x 190 MT HLLV Launch



	2010	2018
Piloted Transit Time:	233 d +200 d 433 d	181 d +125 d 306 d
IMLEO:	189 MT	190 MT

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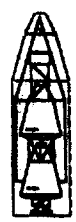
5 MWe Mars Cargo NEP with 2 MEVs

5.0 MWe Cargo NEP Vehicle

← 73 m →

1 x 242 MT HLLV Launch



242 MT

46 m x 12 m

	2007 One-Way
Transit Time:	418 d
IMLEO:	242 MT

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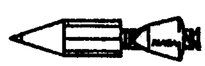
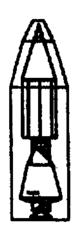
2.5 MWe Mars Cargo NEP with MEV

2.5 MWe Cargo NEP Vehicle



← 46 m →

1 x 135 MT HLLV Launch



135 MT

46 m x 10 m

	2007 One-Way	2007 Round Trip
	405 d	460 d
	±0 d	±209 d
Transit Time:	405 d	669 d
IMLEO:	135 MT	135 MT

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Launch Vehicle Requirements

Launch Vehicle Size	Mission Mode	Piloted	Cargo	Total
"Small" (135 MT)	10 MWe Piloted with ECCV	3	4	7
	10/15 Piloted with MEV	3-4	3	6-7
"Medium" (180 MT)	5 MWe Piloted with ECCV	1	4	5
	10 MWe Piloted with ECCV	2	4	6
	10/15 Piloted with MEV	3-4	3	6-7
"Large" (220 MT)	5 MWe Piloted with ECCV	1	2	3
	10 MWe Piloted with ECCV	2	2	4
	10/15 Piloted with MEV	3-4	2	5-6

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Future Work

- Preliminary trade studies completed
    - EXPO '92 NEP Mars Scenario
- Select reference mission/system scenario
- Perform focused studies
    - System design
        - Krypton propellant
        - Advanced reactor/power conversion technologies
    - Launch manifest
    - Aborts/Window Assessment
    - 10 MWe out/15 MWe back
    - Radiation Protection

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Summary

- NEP meets EXPO trip time requirements (5-10 MWe)
- NEP enables reduction of number and/or size of HLLV's
- NEP has inherent flexibilities and abort capabilities not afforded by high thrust systems
- Synergy exists between NEP, surface, and spacecraft power technologies
- NEP could be ready to support 2010 Mars mission - No technological "show-stoppers" exist

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NEP SYSTEMS MODEL

NUCLEAR PROPULSION TECHNICAL INTERCHANGE MEETING

PLUMBROOK STATION

NASA LEWIS RESEARCH CENTER

OCTOBER 22, 1992

Jim Gilland

Sverdrup Technology, Inc./Nuclear Propulsion Office

Jeff George

NASA LeRC/Advanced Space Analysis Office

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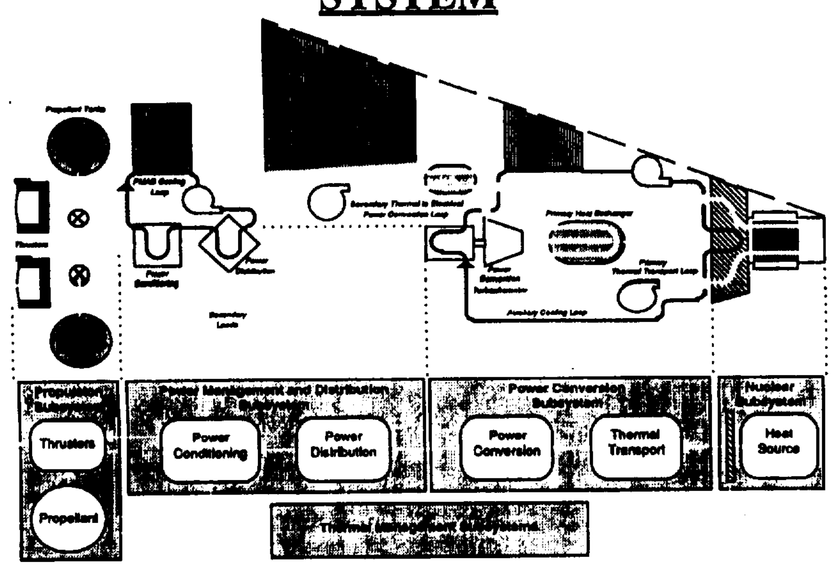
THE NUCLEAR ELECTRIC PROPULSION SYSTEM



Nuclear Electric Propulsion System Schematic
Example High Power Dynamic System for Piloted Missions

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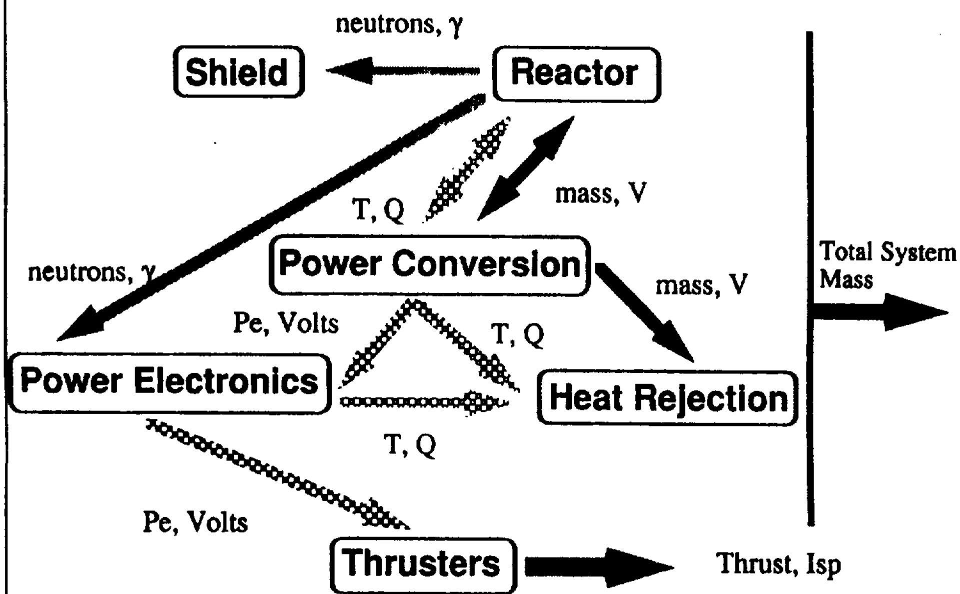
THE NEP SYSTEM ANALYSIS CHALLENGE



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GOALS FOR NUCLEAR ELECTRIC PROPULSION SYSTEM ANALYSIS

- **Design**
  - Develop an effective means of system integration, optimization and design
  - Perform subsystem level trades and sensitivity studies
  - Establish system design for planetary exploration
- **Studies**
  - Develop an effective means of performing integrated system trade studies over a range of technology options
  - Identify most advantageous technologies for next generation NEP systems

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NUCLEAR PROPULSION OFFICE APPROACH TO NEP SYSTEM ANALYSIS

- **NPO's initial purpose was analysis and design of MWe NEP systems for SEI applications**
  - MWe NEP subsystem models not well developed
  - Very little system integration was taking place in NEP studies
  - NPO chose to fund development of broad based component models that
    - **Update MWe subsystem designs**
    - **Allow for integrated system analysis**
- **Current emphasis is on kWe systems**
  - 20 - 100 kWe SP-100 power system definition
  - kWe ion thruster modelling
  - Integrated NEP system, vehicle definition

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NEP SUBSYSTEM MODEL DEVELOPMENT (1992)

- **In House**
  - Improve existing K-Rankine code
  - Develop thruster systems model
    - **Ion**
    - **MPD**
- **Power Conversion - Rocketdyne**
  - K - Rankine
  - Brayton
- **Power Management and Distribution - Rocketdyne**
- **Heat Rejection - Rocketdyne**
- **Reactors - Oak Ridge National Laboratory**
  - Liquid Metal Cooled Fuel Pin
  - NERVA - Derived
  - Liquid Metal Cooled Cermet

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NEP SYSTEMS MODELLING OVERVIEW

- An integrated systems analysis code is the next step for both SP-100 and SEI NEP systems analysis
- Preliminary in-house efforts at systems integration are underway
- Another alternative may be a general systems analysis code that can incorporate NPO system models

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NEP Systems Model

Nuclear Propulsion Technical Interchange Meeting
LeRC Plum Brook Station
October 22, 1992

Jeff George
Advanced Space Analysis Office

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New NEP Systems Analysis Code

- Modular
  - Driver Code
  - Variety of subsystem models
- Five subsystems modelled
  - Reactor/Shield
  - Power Conversion
  - Heat Rejection
  - PMAD
  - Thrusters
- Optimizes for:
  - Minimum mass
  - Minimum radiator area
  - Low mass/low area
- Parameters optimized:
  - Separation distance
  - Temperature ratio
  - (Pressure ratio)
  - (Transmission frequency)

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New NEP Systems Analysis Code, Cont.

- Top level requirements
  - Power level
  - Full power lifetime
  - Payload dose constraint
  - Reactor temperature
  - Turbine inlet temperature
  - Materials
  - Subsystem types/models

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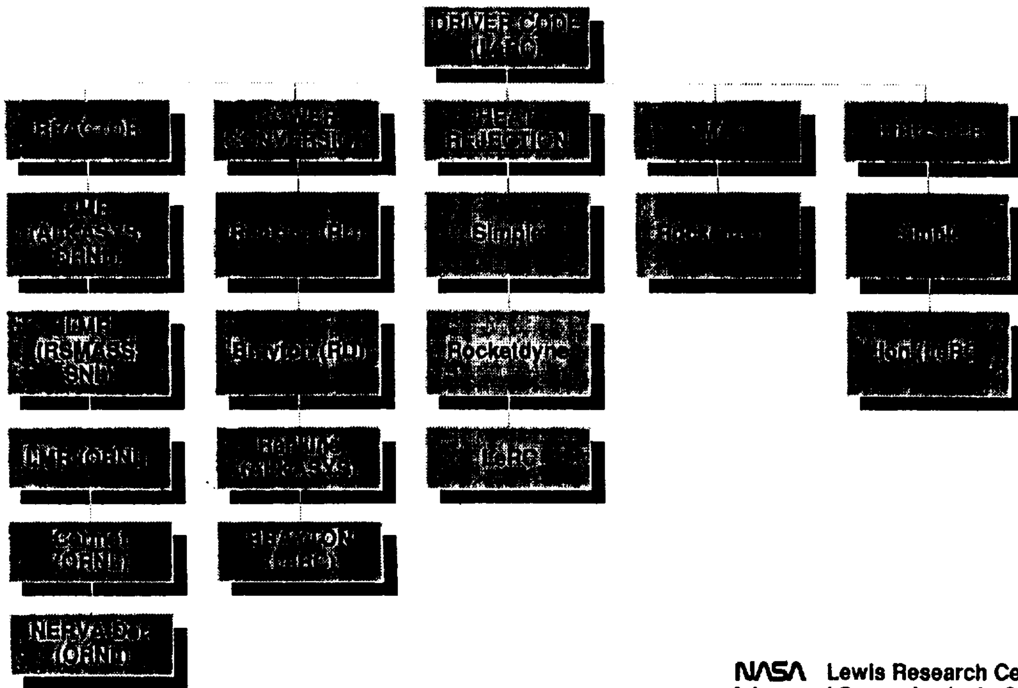
Subsystem Models Library



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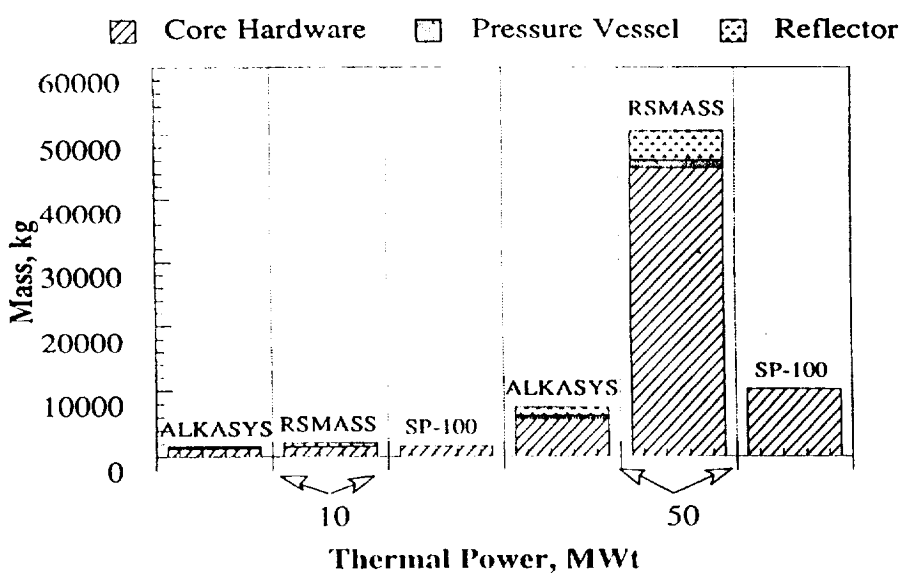
Mass Distribution: ALKASYS v. RSMASS v. GE (SP-100)



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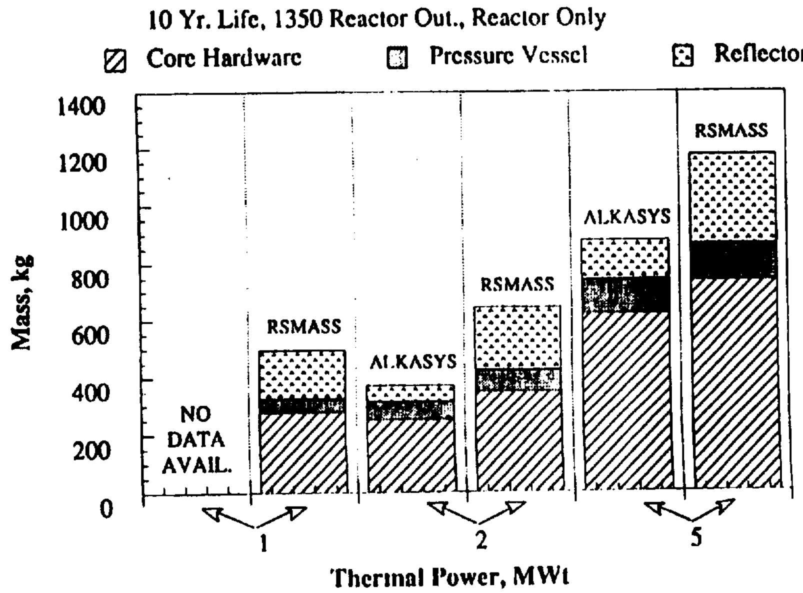
Mass Distribution: ALKASYS v. RSMASS



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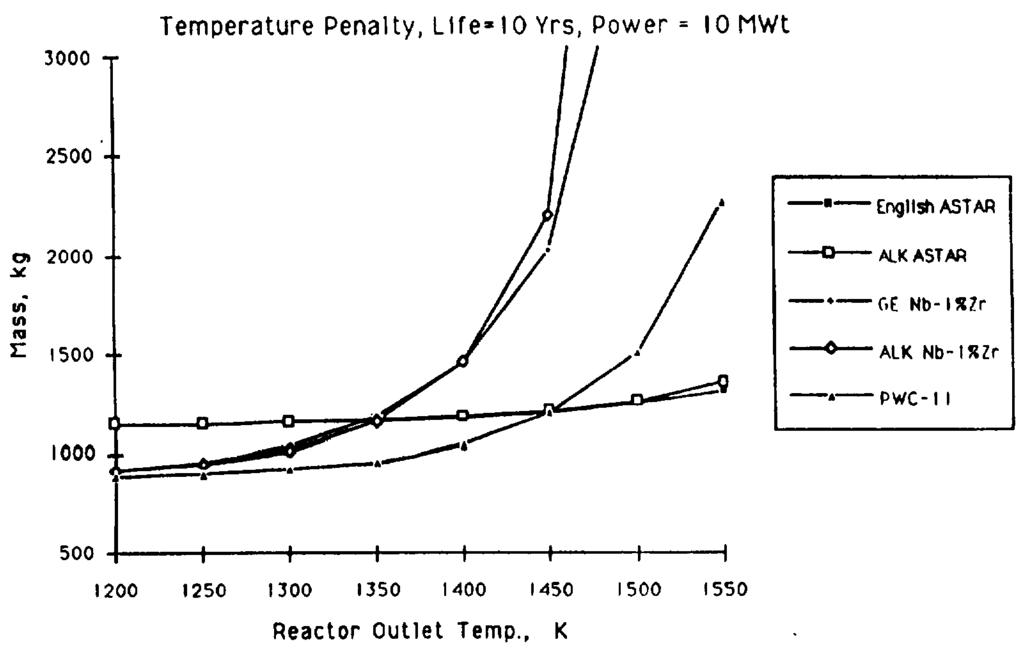
System Mass for Different Materials



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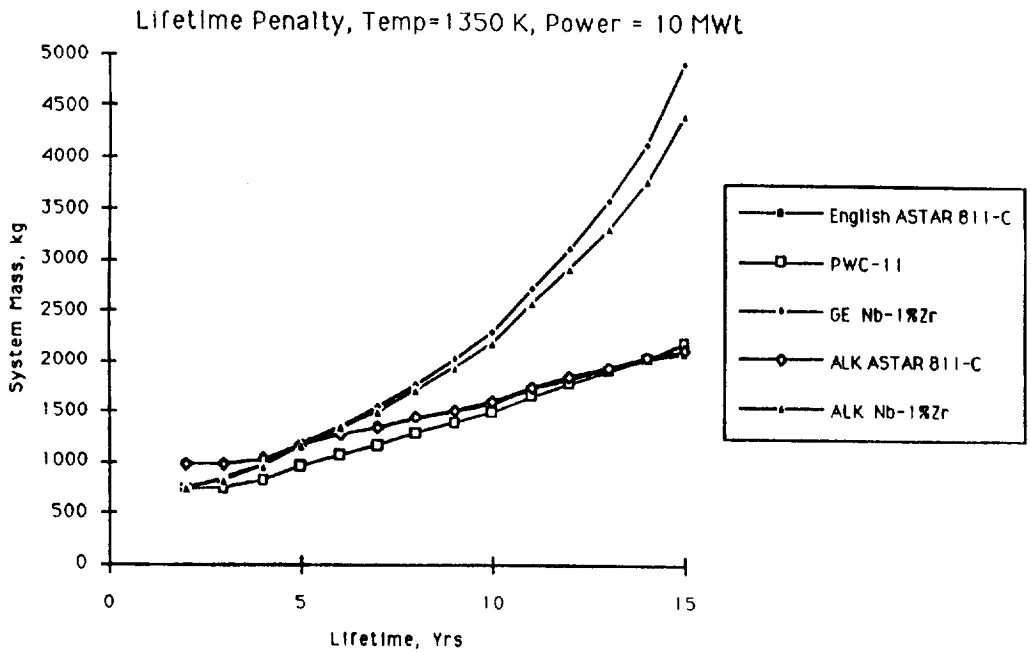
System Mass for Different Materials



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Status

- Two LMR reactor models comopared:
  - ALKASYS better above 2.5 MWt
  - RSMASS better below 2.5 MWt
- Modular systems driver code completed
- LMR/Rankine version undergoing verification & validation
- Various subroutine models collected, under development

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THRUSTER MODELS FOR NEP SYSTEM ANALYSIS

NUCLEAR PROPULSION TECHNICAL INTERCHANGE MEETING

PLUMBROOK STATION

NASA LEWIS RESEARCH CENTER

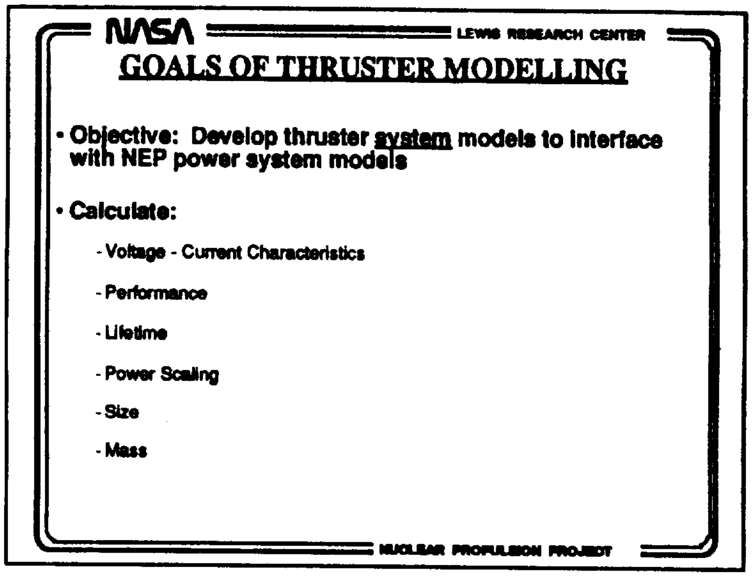
OCTOBER 22, 1992

Jim Gilland

Sverdrup Technology, Inc./Nuclear Propulsion Office

NUCLEAR PROPULSION PROJECT

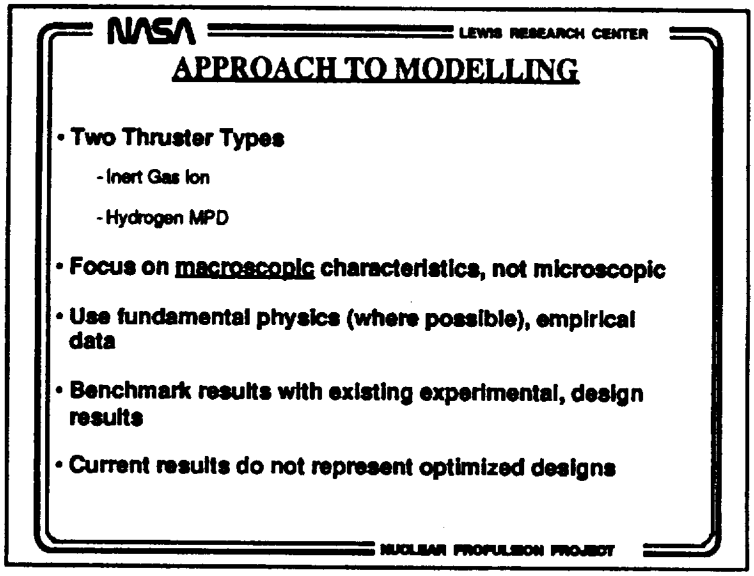
THRUSTER MODELS FOR NEP SYSTEM ANALYSIS



GOALS OF THRUSTER MODELLING

There are currently no thruster modelling codes that can be integrated with power system codes for full propulsion system modelling. Most existing thruster models have been written from a "stand alone" viewpoint, assuming the user is performing analyses on thruster performance alone. The goal of the present modelling effort is to develop thruster codes that model performance and scaling as a function of mission and system inputs, rather than in terms of more elemental physical parameters.

System level parameters of interest are performance, such as specific impulse and efficiency; terminal characteristics, such as voltage or current; and mass. Specific impulse and efficiency couple with mission analyses, while terminal characteristics allow integration with power systems. Additional information on lifetime and operating may be required for detailed designs.



APPROACH TO MODELLING

For this initial effort, the two thruster types with the strongest development background are being modelled: the Magnetoplasmadynamic (MPD) and Ion Thrusters. The emphasis is on modelling these devices as systems; that is, to focus on the macroscopic system level parameters such as power, thrust, specific impulse , rather than on the microscopic parameters such as electron temperature, ionization fraction, and plasma instabilities. Where possible, the fundamental physics of the concept are used, to provide as close an understanding of the underlying processes as possible. Where understanding is incomplete, or too complex for productive system analysis, empirical results have been used. For example, applied field MPD thruster thrust generation is based on experimental measurements, rather than an analytical model.

As these models are developed, they are and will be compared to experimental data and point studies.

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MPD Thruster

- **Description**
  - Cylindrical Anode, Cathode
  - Applied Solenoidal Magnetic Field
  - Heat Pipe/Graphite Radiator
- **Constraints**
  - Cathode Life (Limits Temperature, Current Density)
  - Waste Heat Conduction Through Anode
  - Material Temperature Limits
  - Regenerative Cooling of Hyperconducting Magnet Coil
- **Inputs**
  - Performance: Power, Isp, Efficiency
  - Initial Guess on Dimensions
  - Centerline Applied Field Strength
  - Magnet Coil Size
  - Empirical Thrust Coefficient
  - Electrode Work Functions

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MPD Thruster

The MPD thruster accelerates a plasma propellant through the electromagnetic Lorentz body force. The system considered in this modelling activity is a cylindrical, coaxial thruster, with an external anode and central cathode. Acceleration is provided through the interaction of radial and azimuthal currents with both the self-induced (azimuthal) and applied (axial and radial) magnetic fields. The applied field is generated by a solenoidal coil located externally of the anode. The majority of the thruster's waste heat has been observed to be deposited in the anode, requiring a radiator to reject this energy to space. In this design, the radiator is a set of lithium heat pipes conductively coupled to the anode and transferring the heat from the anode surface to a surrounding circular graphite surface.

Constraints on MPD thruster operation are cathode lifetime due to mass loss, the ability to reject the anode heat, material temperature limits, and the cooling of the hyperconducting magnet coil, which operates at 21 K

- **Inputs range from performance requirements to some system design parameters.**

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MPD Thruster Model Benchmark

Performance: 2.5 MWe, 5000 s, η = 0.5, B0 = 0.5 T

	Design+	Model
Anode Radius (cm)	15	15
Cathode Radius (cm)	2.5	2.5
Anode Length (cm)	30	30
Cathode Length (cm)	10	10
Current (kA)	10	8.5
Voltage (V)	250	295
Magnet Current (A)	2300	2471
Anode Temperature (K)	1400	1861
Anode Fall Voltage (V)	25*	90**
Radiator Area (m2)	1.1	4.4
Mass (kg)	~337#	993
Mass w/o Radiator (kg)	~132#	117

+Myers, et al. "Multimegawatt MPD Thruster Design Considerations," 9th Symposium on Space Nuclear Power Systems, 1992

*Assumed **Estimated #From a Related Study of a Flared Anode Thruster

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MPD Thruster Model Benchmark

An initial benchmarking of the code in terms of system level parameters has been performed. The point design is actually a combination of results from two references: "Multimegawatt Electric Propulsion System Design Considerations,"AIAA 90-2552; and "Multimegawatt MPD Thruster Design Considerations," in the 9th Symposium on Space Nuclear Power Systems, January, 1992. MPD thruster mass was taken from the first reference, which actually used a flared anode, with an inital anode radius of 15 cm flaring to 30 cm at the exit. The second reference is a cylindrical anode of 15 cm radius. The second reference was used for input data to the MPD model.

In terms of terminal characteristics and magnet design, the model results are reasonably close to the point design. Such differences that do exist are due to differences in assumptions of applied field thruster performance, and could be remedied through better empirical parameters in the model.

Model results differ primarily in terms of radiator mass. This is because of the difference in anode heating between the two cases. The reference case assumed a low (25 V) anode drop, whereas the MPD model estimates a 90 V drop. This difference shows up in both the radiator size and the anode temperature. An improved model of MPD thruster loss mechanisms will be required to resolve this difference.

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MPD Thruster Example

Inputs:

Power	2.5 MWe
Magnetic Field	0.1 - 0.5 T
Isp	4000 - 6000 s
Efficiency	0.4 - 0.6

Outputs:

- Anode, Cathode Dimensions
- Current, Voltage
- Electrode Temperatures
- Magnet Size, Current
- Radiator Size, Mass
- Thruster Mass

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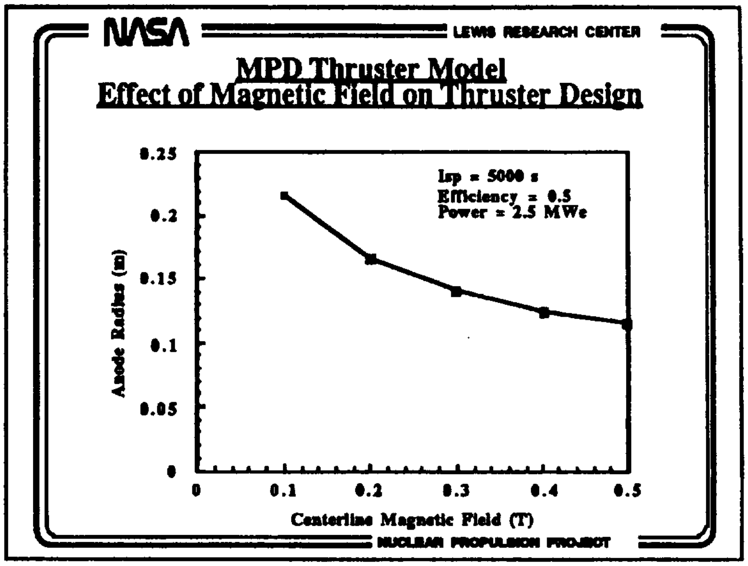
MPD Thruster Example

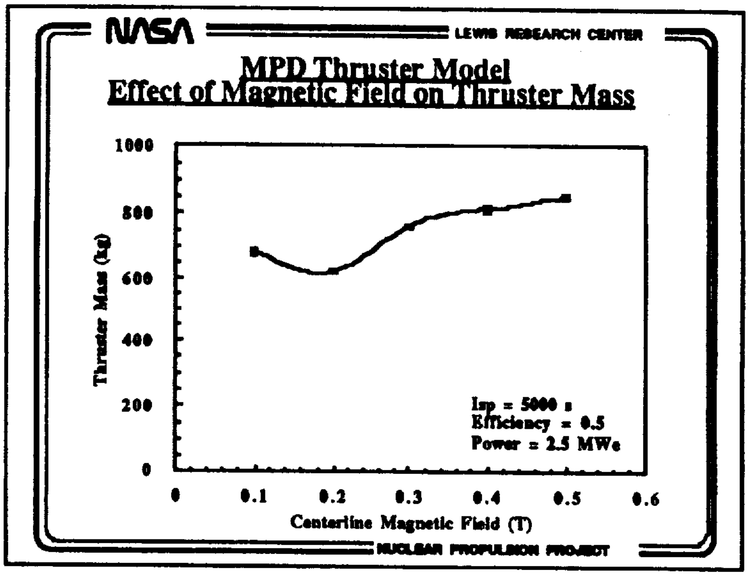
An example of the MPD code results has been generated for a range of pertinent parameters. Although a great many variables are output, only some of the more interesting results are presented herein. The power level, specific impulse, and efficiency are representative of thruster performance useful for lunar or Mars mission applications.



MPD Thruster Model
Effect of Magnetic Field on Thruster Design

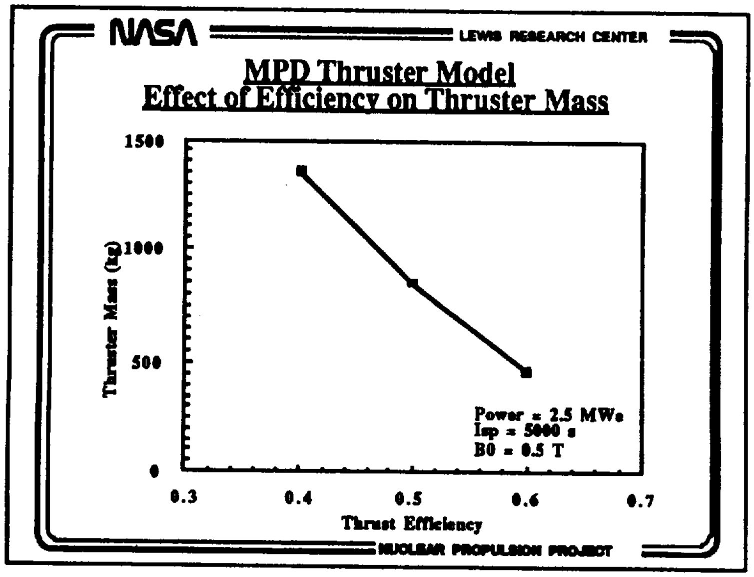
The impact of the applied field upon thruster design is shown in this figure. Increasing the applied field increases its contribution to accelerating the propellant, reducing the need for the self field thrust component. This results in a decrease in anode radius, for conditions of constant power and efficiency. This effect is seen to become less marked at higher fields, indicating that there may be maximal field strength for MPD thruster operation.

This result indicates one benefit of the model: previously, scaling of the thruster with field strength had not been addressed on a parametric basis. Instead, a single design point of field strength and anode radius was selected. It should be noted that this anode radius is also consistent with anode heat rejection and heat conduction constraints.



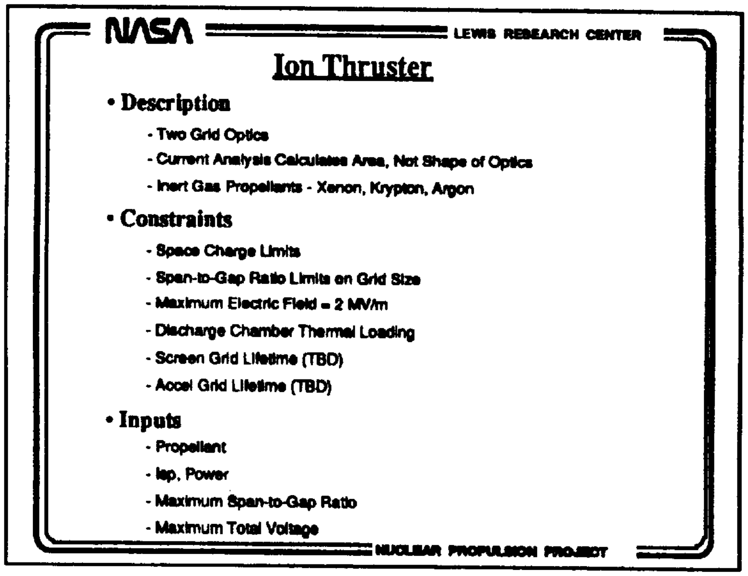
MPD Thruster Model Effect of Magnetic Field on Thruster Mass

The scaling of thruster system (anode, cathode, magnet, radiator) with applied field is shown here. The result indicates a region of field strengths with minimal thruster system mass. In the present model, radiator mass is a dominant segment of the design. The minimum mass point is due to a trade off in decreased anode and magnet size with increased anode losses at higher fields. This behavior is dependent upon the anode loss assumptions, currently an area of experimental and theoretical investigation. An improved anode loss model will ensure the minimum mass point. The MPD model is amenable to incorporating such changes as they become necessary.



MPD Thruster Model
Effect of Efficiency on Thruster Mass

The dominance of radiator mass in the overall system mass is seen in this calculation of thruster mass for varying efficiencies. Increased efficiency is simply decreasing the amount of waste heat delivered to the anode. Additional effects due to thruster or magnet radius are subsumed in the radiator effects.



Ion Thruster

The ion thruster generates thrust through the electrostatic acceleration of a plasma propellant. The electrostatic field is generated via two grids, placed downstream from a discharge chamber in which the plasma is generated. Propellants of choice are the inert gases xenon, krypton, and argon. Propellant choice depends upon the specific impulse and efficiency required.

Ion thrusters operate under several constraints. The primary limit is the space charge limit upon ion beam density. In addition, numerous engineering level constraints upon power density exist, such as grid lifetimes. These considerations are functions of propellant and operating conditions. Of the constraints listed here, all but grid lifetime have been addressed in the thruster model to date.

Some constraints are based on engineering concerns, such as the span-to-gap ratio. This is the ratio of the thruster grid length (the span) to the inter-grid spacing (the gap). Due to thermal and electric deformation, there is a practical upper limit to this ratio for thruster fabrication.

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Ion Thruster Example

Inputs:		**Outputs:**
Propellant	Xe, Kr, A	No. Thrusters
Power	100 kWe	Beam, Discharge Current, Voltage
Isp	3000 - 10000 s	Efficiency
Max Span-to-Gap	850	Area
Max Voltage	3000	Thruster Mass (TBD)
		Lifetime (TBD)

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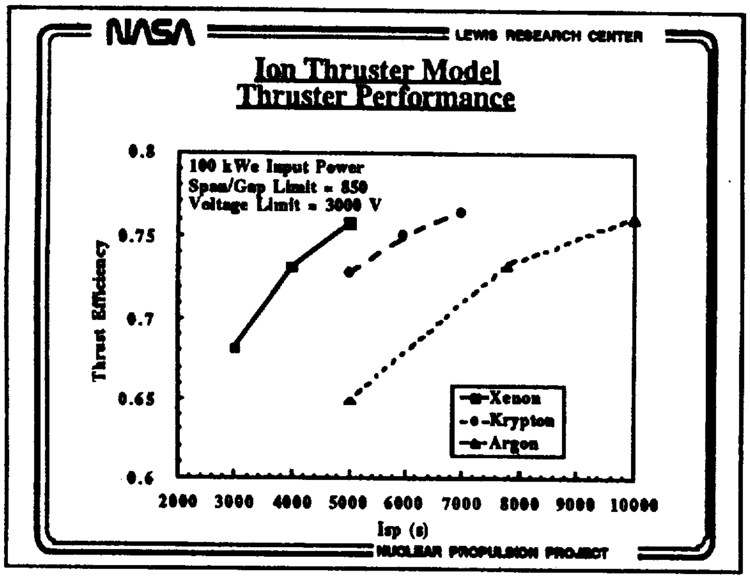
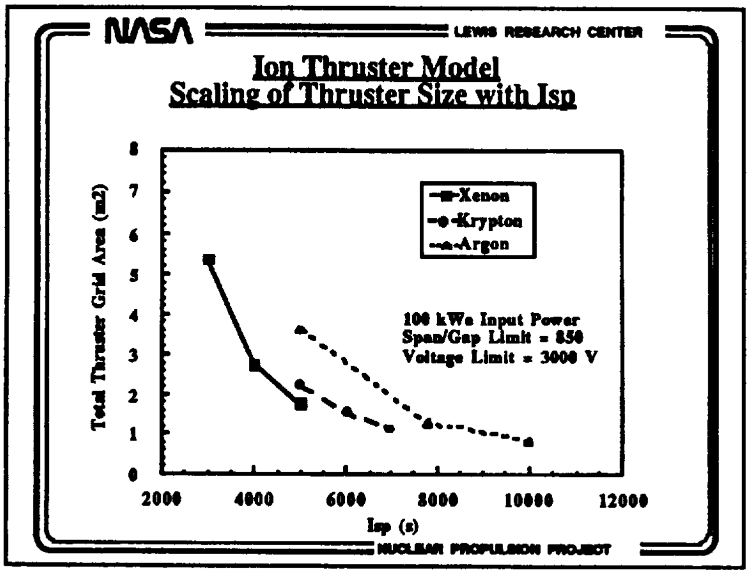
Ion Thruster Example

A sample case of a 100 kWe ion propulsion system has been assessed for this presentation. Inputs are shown above. The ion thruster model was used to calculate system parameters and operating conditions that both met the input requirements and satisfied the constraints. The thruster model will ultimately calculate thruster masses, as does the MPD model.



Ion Thruster Model
Thruster Performance

Ion thruster performance (efficency, specific impulse) is shown for all three propellants. These results are comparable to experimental data for 30 or 50 cm diameter thrusters operated at Lewis Research Center. It should be noted that these data were not generated for fixed thruster dimensions; rather, thruster scaling was an output of the model.



Ion Thruster Model
Scaling of Thruster Size with Isp

Thruster scaling is shown for the three propellants. Total grid area is the area required to process 100 kWe of power, although the number of thrusters changes with specific impulse. The model predicts greater power densities at higher specific impulse, as is seen in experiment. The behavior of these data may change after grid lifetime constraints are imposed.

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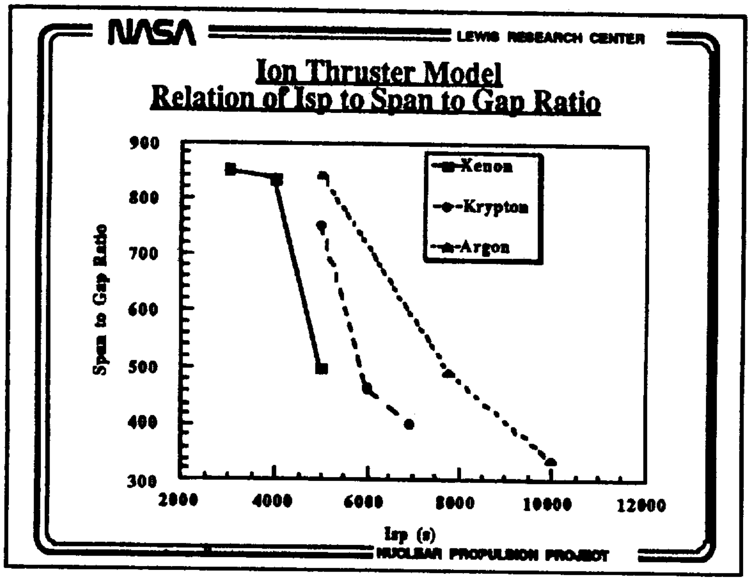
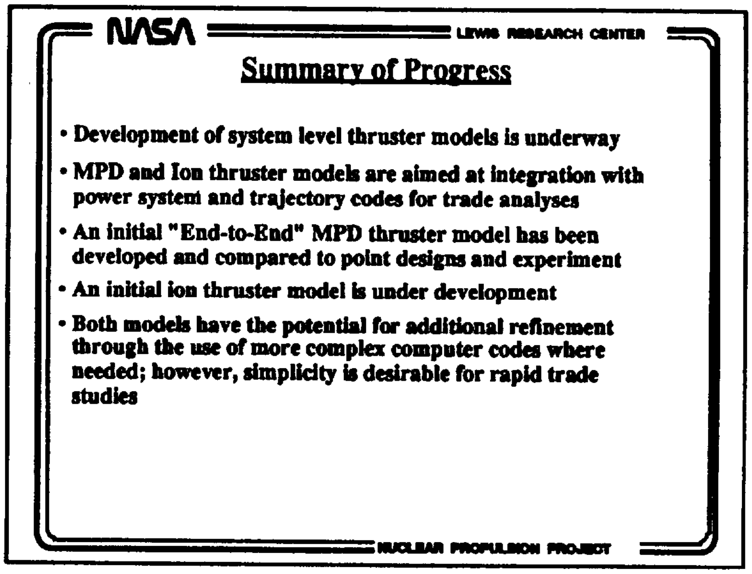


Ion Thruster Model
Relation of Isp to Span to Gap Ratio

The required span-to-gap ratios for each operating point are shown. As power density increases, the total area required decreases, allowing reduced span to gap ratios. This graph is intended as an example of the variations in parameters to be expected in a design study; the variation of other parameters such as number of thrusters, and total voltage would have to be examined in a true system analysis.



Summary of Progress

This presentation is intended as a status report on thruster system modelling efforts currently underway at Lewis Research Center. An evolutionary approach is being taken in developing these models. Refinement of the codes and their component subroutines is expected in the coming months. First order modelling has provided some initial insights into thruster behavior and requirements for effective implementation.

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Further Work

- **MPD Thruster Code**
  - Improved definition of thruster loss mechanisms to allow accurate anode heat deposition calculations
  - Improved definition of radiator subsystem
  - Integration of updated thruster performance data, analyses
- **Ion Thruster Code**
  - Incorporate grid lifetime calculations in design algorithm
  - Thruster component mass definition
- **Both Codes**
  - Investigate optimization techniques

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Further Work

In addition to completing the ion thruster code lifetime and mass models, several areas for improvement of both codes are evident. The impact of the MPD power loss models upon thruster design emphasizes the need for a better understanding, either theoretical or empirical, of dissipation in the MPD thruster. Further refinement of the radiator model is required for effective system design.

In both codes, the potential for internal optimization of certain thruster components is very strong. For example, optimization of the MPD thruster's applied magnetic field strength for minimum thruster system mass might be included in the analysis. Similarly, optimization of the ion thruster voltages, grid spacing, and grid area could be included in the analysis.

Perhaps most important at this stage is that thruster system models are being developed that allow rapid analysis while providing some understanding of the physical processes involved.

N93-26989

INNOVATIVE ELECTRIC PROPULSION THRUSTER MODELING

Presented at the
Nuclear Propulsion Technical Interchange Meeting
NP-TIM-92
NASA Lewis Research Center Plum Brook Station
Cleveland Ohio

October 22, 1992

JPL

Robert H. Frisbee, Ph.D.

Advanced Propulsion Systems Group
Propulsion & Chemical Systems Section
Jet Propulsion Laboratory

JPL

OUTLINE

- Introduction
  - Objective and Approach
  - Related Activities
- Concepts Selected for Modeling
  - C60 Electron-Bombardment Ion Thruster
  - Pulsed Inductive Thruster (PIT)
  - Lithium-Propellant MPD
- Other Concepts Modeled in Previous Studies
- Status and Plans

INTRODUCTION

JPL OBJECTIVES & APPROACH

- Objective
  - Model and evaluate advanced innovative electric propulsion concepts as an aid to performing NEP mission benefits studies
    - Provide scaling relationships for mass, power, efficiency, etc. as a function of Isp, propellant type, etc.
    - Identify technology status / needs
- Approach
  - Select concepts most appropriate for NEP Piloted / Cargo Mars Missions (MMW NEP emphasis)
  - Review relevant literature
  - Identify technology status / needs
  - Formulate scaling relationships
    - Use first-principals modeling approach

INTRODUCTION

JPL INNOVATIVE ELECTRIC PROPULSION RELATED ACTIVITIES AT JPL

- Advanced Propulsion Concepts Studies
  - High-Power Ion, MPD, and ECR Thruster Modeling
  - Microwave Electrothermal (MET) Thruster Modeling
  - MMW SEP / NEP - Ion / MPD Thruster PPU Modeling
- In-House Research in Advanced Electric Propulsion
  - Inert-Gas Ion Thrusters
  - C60 Ion Thrusters
  - Li-MPD Thrusters
  - Arcjets
  - ECR Thrusters (JPL/Caltech)
  - MET Thrusters
- Contract Research in Advanced Electric Propulsion
  - Variable-Isp Thruster Research (MIT)

JPL SUMMARY OF CONCEPTS CONSIDERED

Concept	Typical Isp (s)	Typical Eff. (%)	Typical Pe (MWe)	Likely Application Cis-Lunar	Likely Application Mars	Comments
High-Power Ion thruster	5,000-20,000	85	0.05-2	X	X	• Modeled in FY'91 (APC)
C60 ion thruster	2,000-5,000	75	0.05-5	X	?	• THIS TASK • Good Eff. at Low Isp
Inert-gas MPD	5,000-9,000	60	1-10	X	X	• Modeled in FY'91 (APC)
Li-propellant MPD	5,000-9,000	80	1-10	X	X	• THIS TASK • Good Eff.
ECR	2,000-10,000	70	0.01-2	X	X	• Modeled in FY'91 (APC)
MET	1,000-2,000	60-70	0.001-0.1	X		• Modeled in APC RTOP • Not applicable to Mars
MIT Variable Isp Thruster	1,000-20,000	50	0.1-2	X	X	• Modest Eff.; Only ~ 10-20 % savings w/ variable Isp
TRW PIT	1,000-5,000	60	0.1-2.5	X	X	• THIS TASK • Omnivorous (ETRU ?)
Mass Drivers, Rail Guns	1,000-1,500	90 50	0.1-10	X		• Modeled in FY'89 (ASAO) • Omnivorous; pellet debris

JPL C60 ELECTRON BOMBARDMENT ION THRUSTER MODELING

- Electron-bombardment ion thruster analysis based on a model originally developed by Brophy
  - Propellants:
    - C60
    - Xenon
    - Krypton
    - Argon
  - Span-to-Gap Ratio: 500
  - Minimum Grid Separation: 0.6 mm
  - Maximum Electric Field between Grids: 3000 V/mm
  - Maximum Thruster Diameter: 1m
  - Losses considered:
    - Ion Production Cost
    - Propellant Utilization Efficiency
    - Beam Divergence Loss

C60 ELECTRON BOMBARDMENT ION THRUSTER MODELING

JPL

PROCEDURE

- For a given specific impulse, maximize thrust (power input) of thruster
- Model two regimes:
  - Regime 1: Maximize grid diameter until 1-m limit is reached. Net-to-total voltage ratio R=0.2
  - Regime 2: Keep grid diameter fixed at 1 m, raise net-to-total voltage ratio R from 0.2 to 0.9
- Compute:
  - Total Power Consumption
  - Thrust
  - Thruster Efficiency
  - Thruster Mass
  - Specific Mass
  - Thrust-to-Power Ratio
  - Mass Flow Rate
  - Discharge Current
  - Beam Current
  - Grid Separation
  - Grid Diamenter
  - Beam Voltage
  - Total Voltage

C60 ELECTRON BOMBARDMENT ION THRUSTER MODELING

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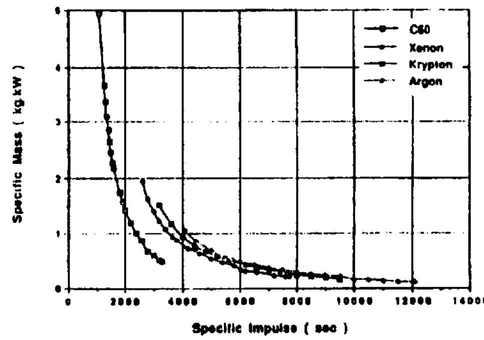
SAMPLE INPUT DATA

Propellant	C60	Xenon
Beam Divergence	0.95	0.95
Ion Production Cost	100 eV/ion	150 eV/ion
Propellant Utilization	0.9	0.9
Discharge Voltage	36 V	36 V
Neutralizer Coupling	20 V	20 V
Grid Open Area Fraction	0.75	0.75
Thruster Chamber Length	20 cm	20 cm

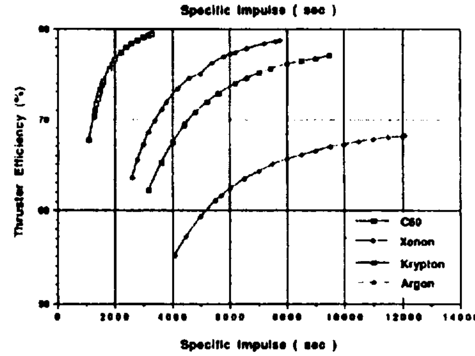
C60 ELECTRON BOMBARDMENT ION THRUSTER MODELING

JPL SPECIFIC MASS & EFFICIENCY vs Isp

- Specific Mass impacts vehicle sizing



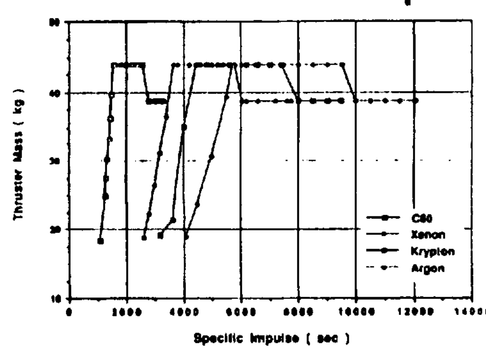
- Efficiency (Pjet/Pe) impacts "jet power" and thrust



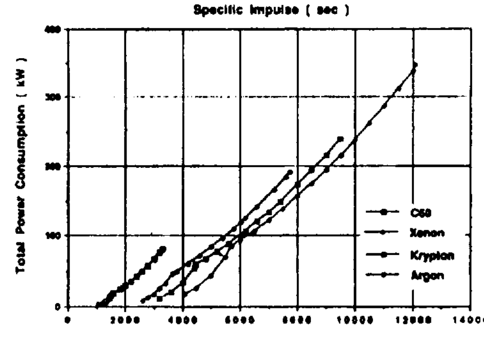
C60 ELECTRON BOMBARDMENT ION THRUSTER MODELING

JPL THRUSTER MASS & POWER vs Isp

- Mass-per-thruster impacts gimbal sizing



- Power-per-thruster impacts PPU sizing



C60 ELECTRON BOMBARDMENT ION THRUSTER MODELING

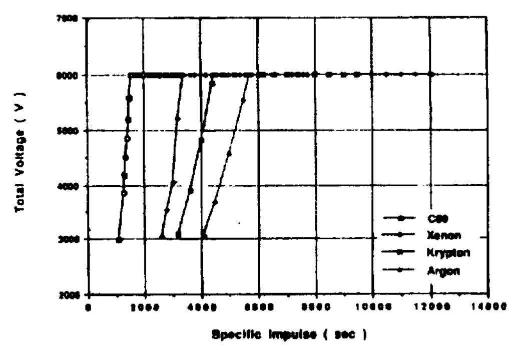
JPL C60/Xe/Kr/Ar-ION THRUSTER SUMMARY

- C60 versus Xe/Kr/Ar
  - For Isp < 4000 lbf-s/lbm, C60 has lower specific mass and higher efficiency than Xe/Kr/Ar
  - Isp of C60 ideal for cis-lunar missions
- Xe vs Kr vs Ar
  - Xe/Kr/Ar have ~ same specific mass
  - Xe/Kr efficiencies higher than Ar
  - High cost of Xe and low eff. of Ar may favor Kr
  - High power-per-thruster (>0.1 MWe) possible

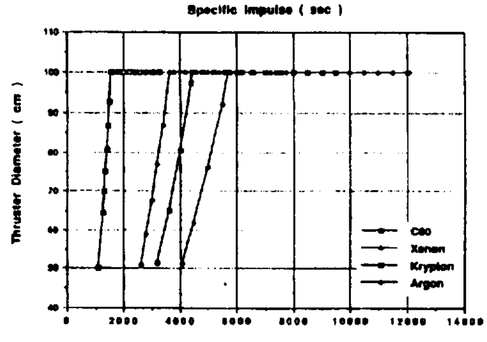
C60 ELECTRON BOMBARDMENT ION THRUSTER MODELING

JPL MAX. VOLTAGE & DIAMETER vs Isp

- Maximum Voltage impacts PPU sizing



- Thruster Diameter impacts vehicle packaging / configuration

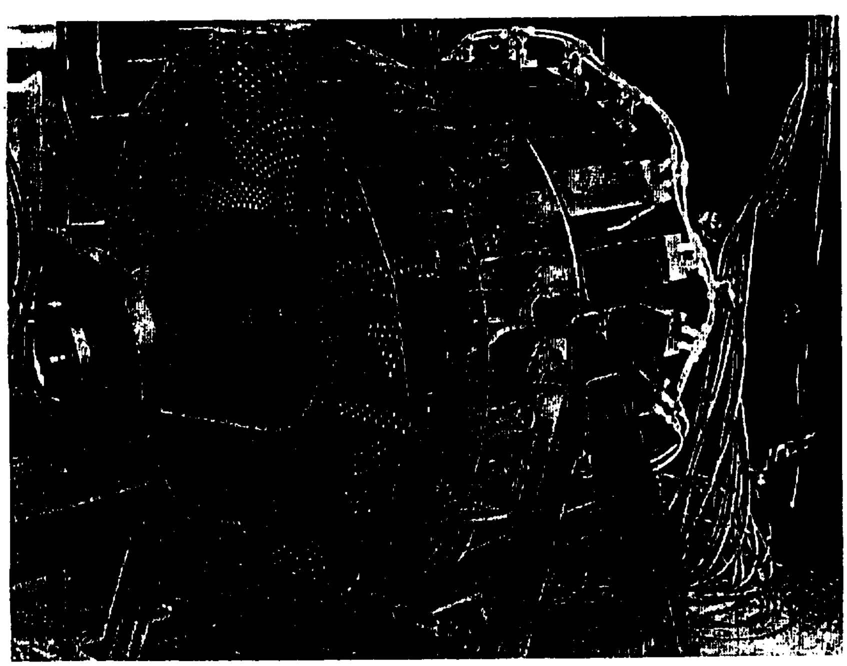


PULSED INDUCTIVE THRUSTER (PIT) MODELING

- Concept
  - Current pulse in flat induction coil (1 m dia) induces ionization and drives plasma current
  - Magnetic (JxB) force accelerates plasma
  - Propellant injected with pulsing valve
- Advantages
  - Electrodeless (minimal errosion)
  - Can operate with a variety of propellants
    - Ammonia, hydrazine, argon, carbon dioxide demonstrated
- Technical Issues
  - Propellant valve lifetime
  - High rep-rate switch and capacitor life-time
  - System performance at high rep-rate

TRW Federal Systems Division
Space & Technology Group

Mark V Front View

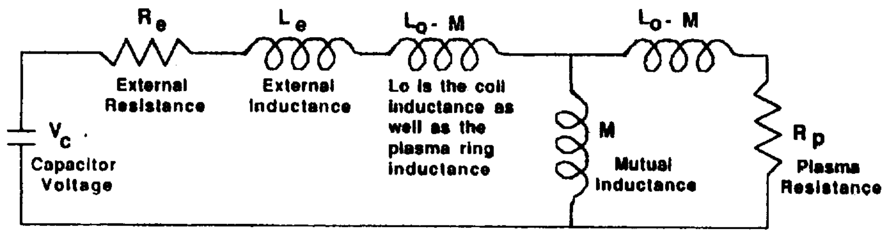


PULSED INDUCTIVE THRUSTER MODELING

JPL

PIT MODEL DISCRIPTION

- PIT analysis based on a model originally developed by TRW
  - Thruster modelled as a transformer

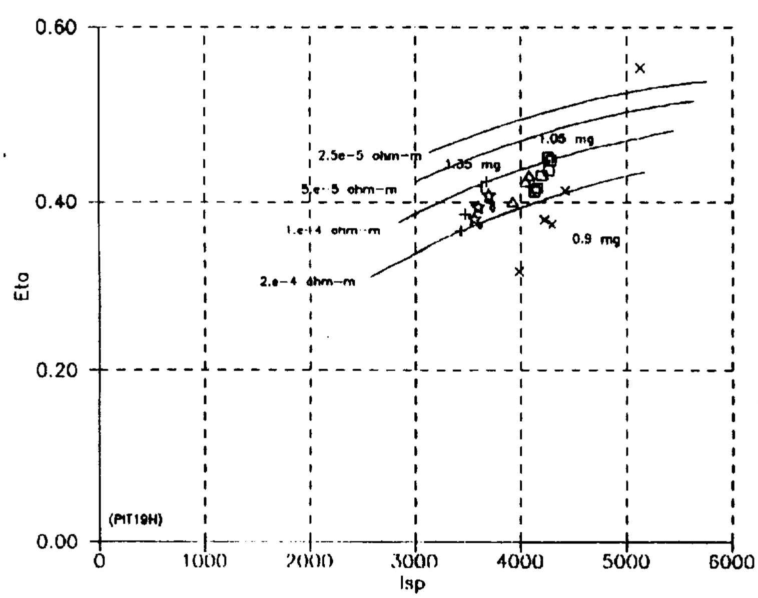


- A system of coupled differential equations discribing the model is solved to estimate the specific impulse and efficiency
- Thruster paramaters input to the model are based on the TRWMark V design:
  - Mass = 150 kg
  - Coil diameter = 1 m
  - Total Vc = 30 kV DC
  - Applied Voltage (from PPU) = Vc / 2
- Plasma resistivity (related to Rp) is propellant dependent

TRW Federal Systems Division
Space & Technology Group

TRW

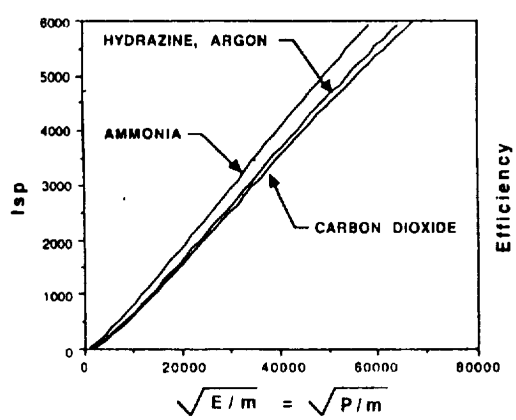
Comparison of N_2H_4 Data with Analytical Model

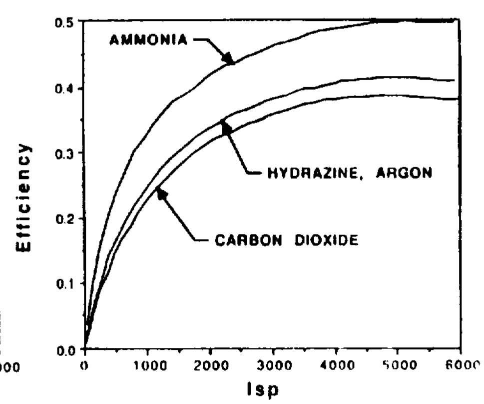


JPL PIT MASS AND POWER CONDITIONING

- Thruster Mass
  - Thruster mass is proportional to energy-per-shot (about twice capacitor mass)
  - To obtain a specific mass of 1 kg/kW requires rep-rate on the order of 100 Hz
- Power Conditioning
  - Switches needed to isolate power system from thruster circuit during shots
  - May need a dedicated Power Processing Unit (PPU) to charge capacitors between shots (supply ~15 kV DC)
  - It may be possible to use sychronous switching to charge capacitors directly from a dynamic nuclear electric power supply bus (typically 7-10 kV AC)

JPL PIT MODELING RESULTS





- For a given thruster (e.g., Mark V) and propellant type, efficiency and specific impulse are both functions of the square root of energy per shot divided by mass per shot (or square root of average power divided by average mass flow rate)

PULSED INDUCTIVE THRUSTER MODELING

JPL

PIT SUMMARY

- Thruster efficiency varies from about 20 to 50 % at specific impulses between 2,000 and 6,000 lbf-s/lbm, respectively
- Thruster mass is proportional to energy per shot
- Specific mass is proportional to shot repetition rate
  - Shot rep rate ~ 100 Hz needed for ~ 1 kg/kWe
- Thruster has been operated on a variety of gases
  - Potential to utilize extraterrestrial propellants
- May have significant PPU needs for SEP or static-conversion NEP (~100 V DC source)
  - Dynamic-conversion NEP more attractive (~ 8 kV AC source)
- Propellant valve and capacitor switch lifetimes an issue

JPL

LITHIUM MAGNETOPLASMADYNAMIC (MPD) THRUSTER MODELING

- Self-field steady-state MPD thruster analysis based on a model originally developed by Blandino
  - Propellants:
    - Lithium
    - Argon
    - Hydrogen
  - Axially-uniform radial current distribution, coaxially-uniform diameter tungsten electrodes
  - Geometry ratios fixed: Ra/Rc = 5, Lc/Rc = 9
  - Maximum cathode current density = 15 kA/cm^2 (to limit erosion)
  - Lithium heat pipe technology used for annular radiator
    - Max heat flux technology-limited to < 1000 W/cm^2
    - Max heat flux calculated < 500 W/cm^2
  - Losses considered:
    - Ohmic heating of plasma & electrodes
    - Sheath voltage drops
    - Anode heating

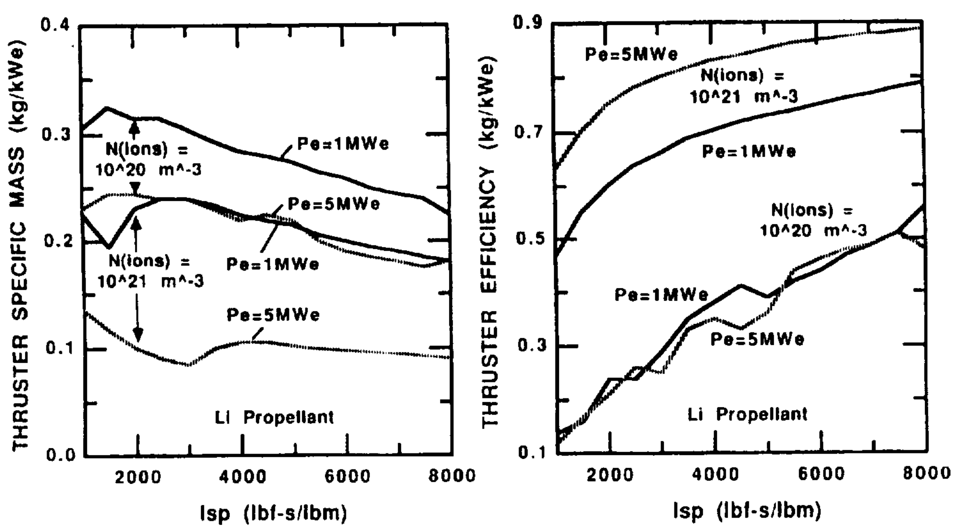
JPL SAMPLE INPUT DATA

Propellant	Argon	Lithium
Ion Mass	39.9 amu	6.9 amu
Ionization Potential	15.76 eV	5.39 eV
T electrons	2 eV	2 eV
T ions	6 eV	2 eV
N ions	10^20 m^-3	10^20 m^-3

- Modeling still in early stages
- Results shown following are preliminary only
  - Still in process of de-bugging model
    - Example - output sensitive to assumed ion number density (N ions)

JPL SPECIFIC MASS & EFFICIENCY

- Thruster power, Isp, and N (ions) used as inputs to model



- Onset limits Isp to 7000 lbf-s/lbm for I/M-DOT < 300 kA/(g/s)

LITHIUM MPD THRUSTER MODELING

JPL Li-MPD SUMMARY

- Model still being tested / verified
- In general, correct trends observed
  - Specific mass decreases and efficiency increases as Isp, power, and N(ions) increase
- But - - -
  - Efficiency & specific mass a strong function of N(ions)
    - Experimental values of N(ions) ~ 10^20 - 10^21 m^-3 for megawatt-class MPDs
    - Possible solution - convert N(ions) to a dependant variable using the Saha equation

$$\frac{\text{N(ions)}}{(\text{ N(total) - N(ions) })} = \frac{3.0\text{x}10\text{\^{}}27 \cdot \text{T(ions)\^{}3/2} \cdot \exp(\text{ I.P. / T(ions) })}{\text{N(ions)}}$$

N = m^-3, T and I.P = eV, and I.P. = Ionization Potential

JPL OTHER EP CONCEPTS

- Numerous electric propulsion thrusters and subsystems have been modeled in past and current studies:
  - Rail Guns and Mass Drivers
  - Variable-Isp Plasma Thruster (MIT)
  - Electron-Cyclotron Resonance (ECR) Plasma Engine
  - Power Processor Units (PPUs)
  - Refrigerators for Active Thermal Control of Cryogenic Propellants

OTHER EP CONCEPTS

JPL THRUSTERS MODELED IN PREVIOUS STUDIES

- Rail Guns and Mass Drivers
  - Medium-Isp (1200 lbf-s/lbm) ideal for cis-lunar orbit raising
  - Can use extraterrestrial-produced propellants (e.g., O2)

Rail Gun Mass = 126.2 MT, η_{total} = Pjet / Pe = 0.45

Mass Driver Specific Mass (total) = 2-20 kg/kWe (=MT/MWe), η_{total} = 0.80

Refrigerator (for liquid-O2 propellant storage) [MT] = 0.022 · (Mp [MT])^2/3

Freezer (for solid-O2 pellet production) [MT] = 4.18 · η_{total} · Pe [MWe]

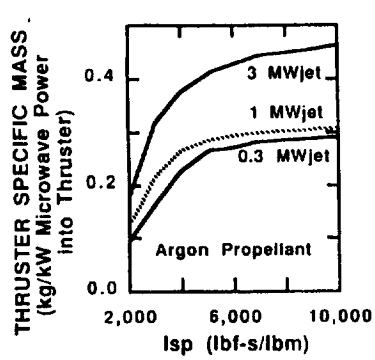
- ICRF-Heated Variable-Isp Plasma Thruster
  - NASA-supported on-going research program at MIT
  - Vary Isp (800-35,000 lbf-s/lbm) in flight to optimize trajectory
    - Potential 10-20 % savings in mass, and trip time
  - Preliminary estimates by MIT of specific mass and efficiency

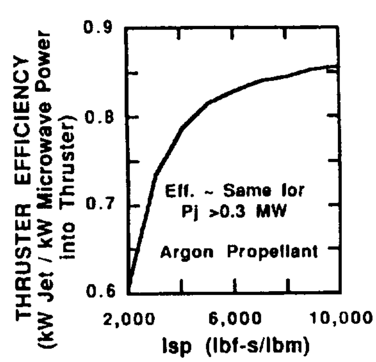
Specific Mass (total) = 4.04 kg/kWe, η_{total} = Pjet / Pe = 0.5-0.7

OTHER EP CONCEPTS

JPL THRUSTERS MODELED IN PREVIOUS STUDIES - CONT'D

- Electron-Cyclotron Resonance (ECR) Plasma Engine
  - Use on-board or remotely-transmitted microwave power
  - Electrodeless thruster (potential long life)
  - Can use extraterrestrial-produced propellants





Remote Beamed Microwave Power Source:
1-km Diameter Inflatable Optics & Waveguides = 23.6 MT
On-Board Microwave Power Source:
Magnetron Specific Mass = 0.2 kg/kW Microwave Power, η = Pmicrowave / Pe = 0.9

OTHER EP CONCEPTS

JPL POWER PROCESSOR UNITS (PPUs) MODELED IN PREVIOUS STUDIES

- Power Processing Unit (PPU) design depends on :
  - Power source output (high-voltage AC for NEP w/ dynamic conversion vs low-voltage DC for SEP or NEP w/ static conversion)
  - Thruster input (high-voltage DC for ion/PIT vs low-voltage DC for MPD, and power-per-thruster)
  - PPU system topology (switching, redundancy, devices)

Mass of SEP/NEP(Static)-Ion Thruster PPU (kg) = { 138.36 · (Pe [kWe] / 62)^0.71 · (K+M) + 1.02 · (2·(K+L) + 3·(K+M)) } · { 1 + 0.025 · (Max. Voltage - 3 kV) } and η = 0.955

Mass of NEP(Dynamic)-Ion Thruster PPU (kg) = 1.0867 · { 617 · (K · Pe [MWe] / 4.97)^0.75 + (16.86 + 10.57 + 14.29) · (K+M) · (Pe/0.71) + 3.5 · ((K+L) + (1+K) · (K+M)) } · { 1 + 0.025 · (Max. Voltage - 6 kV) } and η = 0.992

where Pe = power (electric) per thruster (but PPU limited by transformer to 5 MWe per PPU)
K = number of operating thrusters = number of operating PPUs
L = number spare thrusters
M = number of spare PPUs
and Thruster redundancy typically ≥25 %, PPU redundancy ≥12.5 %

- SEP-Ion PPU significantly heavier, less eff. than dynamic-NEP-ion PPU
  - DC-to-AC inverter required for SEP or static-NEP PPU
  - Economy-of-scale for common transformer in dynamic-NEP PPU
  - Lower eff. of SEP PPU contributes significantly to waste-heat rejection requirements (4.5 % vs 0.8 % of Pe as waste heat)

OTHER EP CONCEPTS

JPL REFRIGERATORS FOR ACTIVE THERMAL CONTROL OF CRYOGENIC PROPELLANTS MODELED IN PREVIOUS STUDIES

- Active thermal control may be needed for long missions
  - Trade Refrigerator mass against boiloff

PROPELLANT	PROPELLANT TEMP. (K)	TANK COOLING LOAD (Wcool)	REFRIGERATOR MASS (kg)
Xe	165	0.005 · Mp^2/3	0 + 13 · Wcool
Kr	121	0.008 · Mp^2/3	15 + 16 · Wcool
Ar	88	0.011 · Mp^2/3	31 + 18 · Wcool
O2	90	0.012 · Mp^2/3	
N2	77	0.016 · Mp^2/3	
H2	21	0.083 · Mp^2/3	46 + 21 · Wcool

Mp = PROPELLANT MASS (kg)

STATUS & PLANS

- Status
  - C60 EB ion thruster modeling complete
  - Completion of C60 Radio Frequency Ion Thruster (RIT) modeling (mass breakdown) awaiting reply from Prof. Loeb, University of Giessen, Germany
  - PIT modeling complete
  - Li-MPD modeling underway
- Plans
  - Complete C60-RIT ion thruster modeling
  - Complete Li-MPD thruster modeling
  - Complete final report (including summary of high-power ion, MPD, ECR, Variable-Isp, and Rail-Gun/Mass-Driver thrusters, and MET thruster modeling under APC RTOP)

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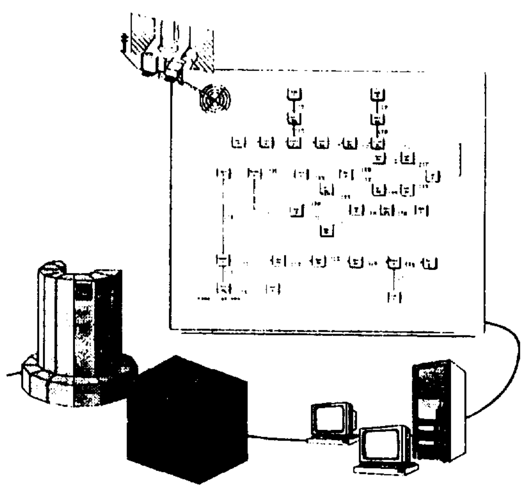
GPS Sytem Simulation Methodology

Thomas F. Ewing

Argonne National Laboratory
Engineering Physics Division



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Talk Outline

Background

GPS Methodology Overview

Graphical User Interface

Current models

Application to Space Nuclear Power/Propulsion

Interfacing requirements

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History

- SALT (system analysis language translator) - Early 80's
  - PL/I code for IBM mainframes
  - Moved to multiple platforms and languages (C, C++)
  - Batch oriented - translate, compile, run
  - Used model and property libraries
  - Optimizations and system analysis

  Applied to

  - Open-cycle and liquid-metal MHD systems
  - Fuel cells
  - Ocean thermal energy conversion
  - Municipal solid waste processing
  - Fusion
  - Breeder reactors
  - Geothermal and solar energy systems

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Next Generation Implementation - GPS

- Designed for modern workstation environments
- Developed in C++, moved to C for greater portability
- Steady-state & dynamic model libraries concept of SALT, but accessed as *class objects*
- Complete, extensible, object-oriented control language with numerous procedures for optimizations, equations solving, system constraints, parametric analysis
- Language interpreted, but uses compiled, fully optimized models and math procedures ==>
  - Fast prototyping cycles
  - On-the-fly creation of/interaction with simulations
  - Simulation systems can be interupted, queried and changed, then resumed

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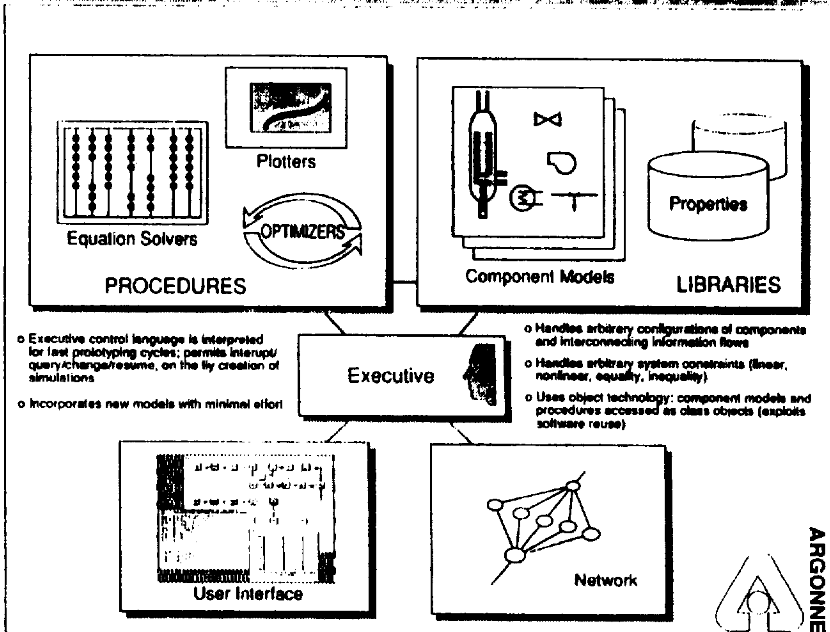
Simulation/Modeling Approach



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GPS Operators

- 86 built-in operators
- I/O functions (fopen, printf, sscanf, sprintf)
- Math functions (atan2, pow, exp, max, ln, log10)
- Numerical procedures (vary, cons, icons, mini, diff)
- Looping and flow control

  cond {...} if

  cond {...} {...} ifelse

  start inc bound {...} for

  count {...} repeat

  {...} loop

  {cond} {...} while

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Miscellaneous Operators

- Allocate new model class instance - **cdef**
  /pump1 { pump: /param1 12.0 /param2 0.495 } cdef
- Set a debug level (0 thru 5) - **debug**
- Run gps simulation from a input file - **run**
  "input.fil" run
- Interrupt simulation to permit queries/interactions
  sintrp (followed by **resume** to continue)

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GPS Steady-State Power System Models

Basic component models

gas -	gas flow initiator
sp -	gas flow splitter
mx -	gas flow mixer
ht -	gas flow heater/cooler
hx -	gas flow heat exchanger
cp -	compressor
gt -	gas turbine
pump -	pump
df -	diffuser
nz -	nozzle
power -	calculate system powers

Basic thermionic models

reac -	reactor model
ti -	thermionic converter
rad -	thermal radiator
sp -	power flow splitter
res -	electrical resistor
bc -	boost converter
bus -	electrical bus
mass -	mass calculations

More sophisticated models

therm -	thermal flow initiator
hprad -	heat pipe radiator
tds -	thermionic diode subsystem
shx -	simple, multinode heat exchanger
nhx -	multinode, general purpose HT model

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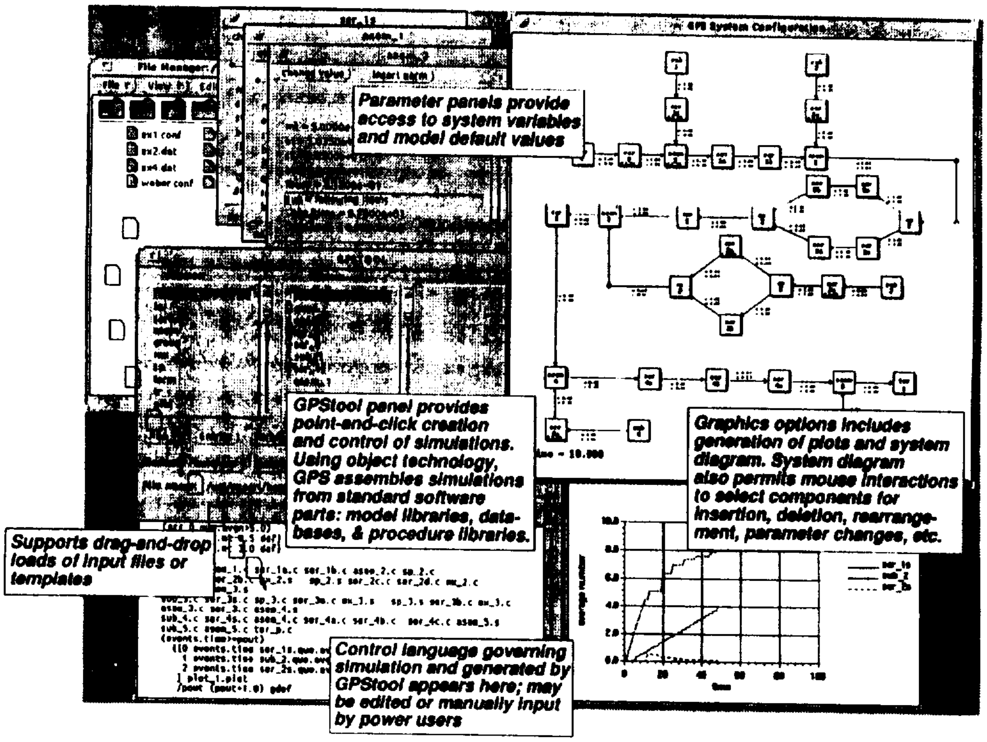
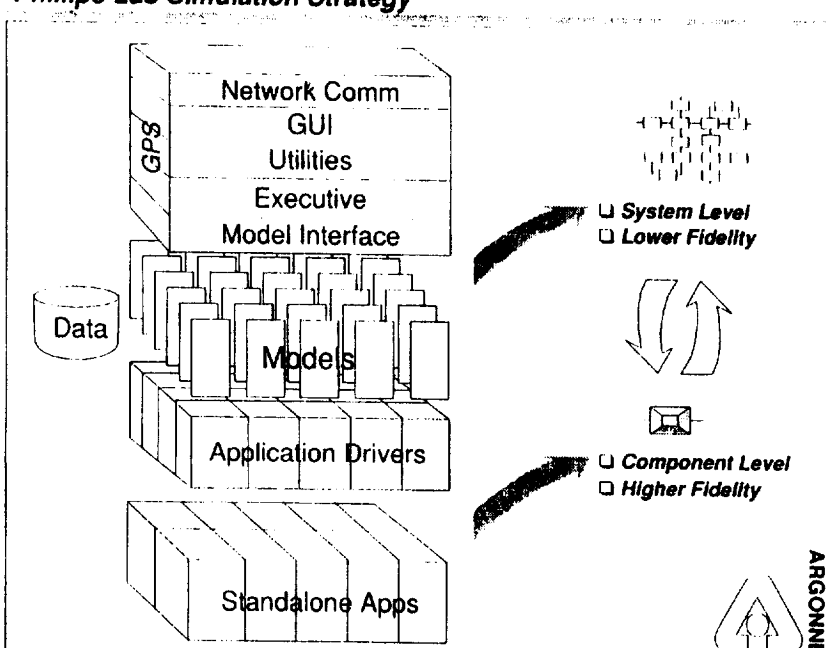
GPSTool - Graphical User Interface



Phillips Lab Simulation Strategy



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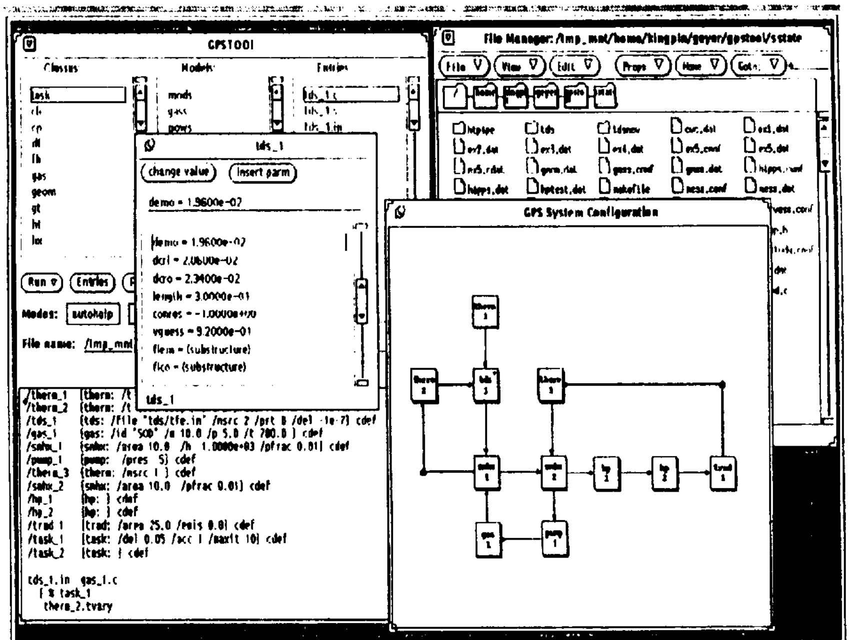

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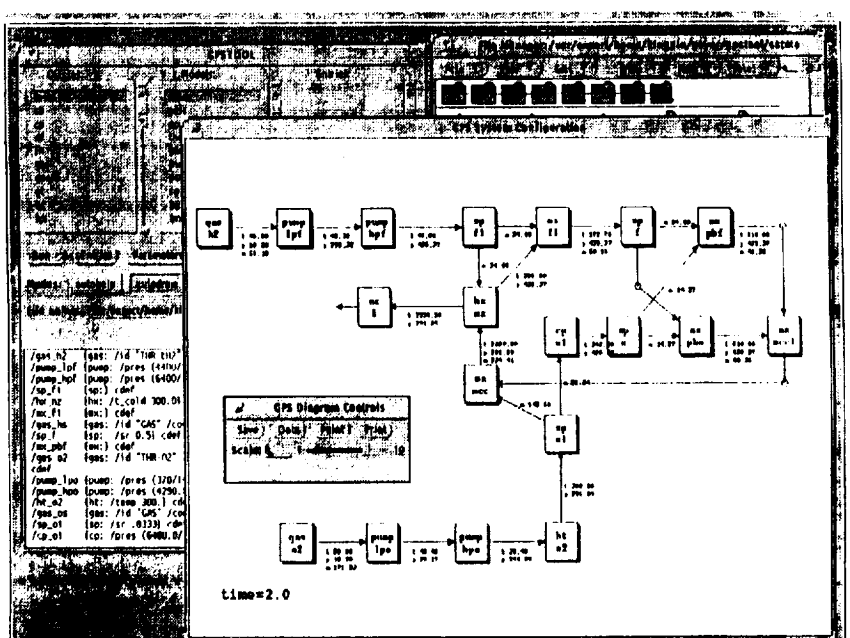

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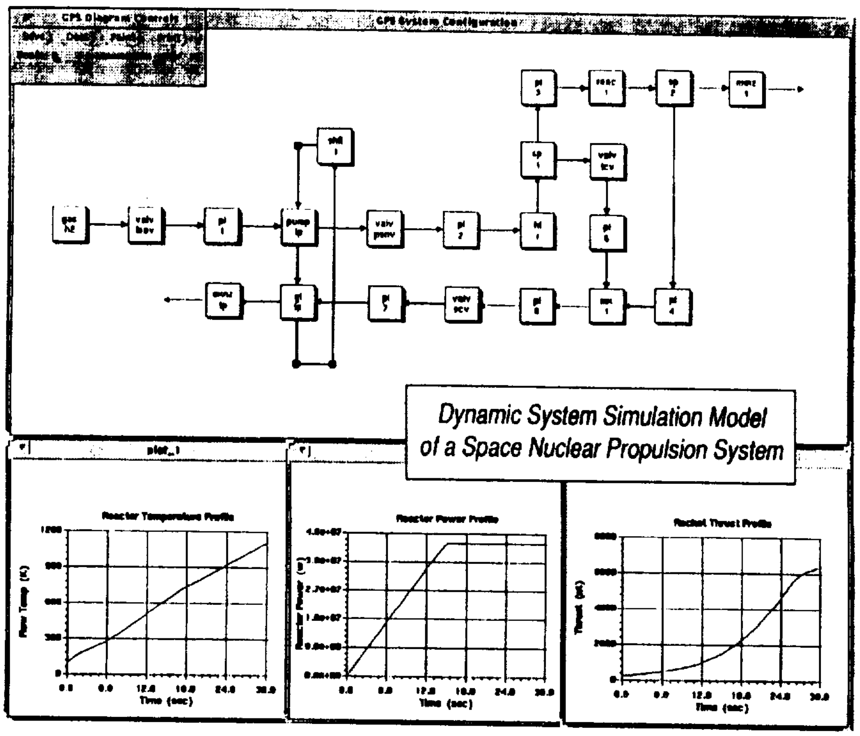
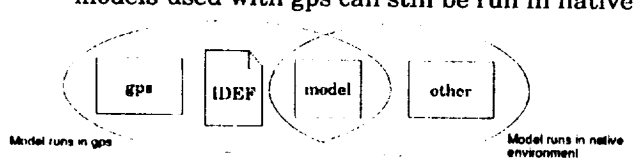


Advantages as Integrating Environment

- Consistent user interface to models
- Diverse models can be combined for use in arbitrarily complex systems
- Suite of gps system analysis capabilities (sweeps, optimizations) and numerical methods/properties available to models
- Interface definitions external to models ==>
  - can adapt models developed independent of gps
  - can use proprietary models available only as object code
  - models used with gps can still be run in native mode



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Interfacing Considerations

- Component models can be Fortran, C, or other Sun languages which generate linkable object code
- Standalone codes must be structured as subroutines with argument list of variables/parameters that must be known to GPS system
- Use of Fortran common blocks prevents (presently) having multiple instances of that model in a system
- Because models may be cycled through numerous convergence iterations with perturbed input flows

  Models must be true functions of their inputs

  Models must be reasonably robust

  I/O routines should be moved outside computation routines

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Converting a standalone code

- Two step process:

  Convert code to one or more subroutines

  Create a interface definition file (IDEF)
- GPS uses IDEF to generate small C code to handle interfaces
- Model can still be run independently of gps (standalone) by writing a main program to call subroutine

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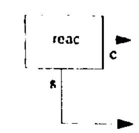
Interface Specification File Format

Interface specifications external to models

- User-prepared ASCII file used by GPS preprocessor to generate C stub code to handle gps interfacing
  - Model name
  - Variable types and initial values (arguments + gps I/O)
  - Entry procedures (name, arguments if Fortran routine, in and out flow variables)
  - Print variables (used as default gps output)

EXAMPLE
Thermionic reac model



c - mass/sizing, thermionic component of power

s - waste heat flow

```
model      reac
 char      names[16] namess[16] namers[16] nameboom[16]
 double    pow = 1e6 eff = 0.13 radius height sep = 10.0 rhoboom = 10.0
           lboom radiusrs volrs heightrs = 0.37
 flowtype  ll fls
 masstype  mcore mss mrs mboom

entry      c
 outflow   mcore mss mrs mboom ll

entry      s
 outflow   fls

print      pow eff radius height
print      radiusrs volrs heightrs sep
```

INTERFACE SPECIFICATION

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Example Conversion

Fortran Standalone code - TDS

- 8400 lines of Fortran code (includes TECMDL)
- Required 32 line interface definition file
- Conversion completed in < 2 hrs.
- Same model now runs standalone (called from main) or in gps environment
- Both open (once through) and closed systems have been run in gps
- Have successfully run problems with 250,000 nonlinear constraints in nested loops

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13. ABSTRACT ***(Maximum 200 words)***

The Nuclear Propulsion Technical Interchange Meeting (NP-TIM-92) was held at NASA Lewis Research Center's Plum Brook Station in Sandusky, Ohio on October 20–23, 1992. Over 200 people attended the meeting from government, Department of Energy's national laboratories, industry, and academia. The meeting was sponsored and hosted by the Nuclear Propulsion Office at the NASA Lewis Research Center. The purpose of the meeting was to review the work performed in fiscal year 1992 in the areas of nuclear thermal and nuclear electric propulsion technology development. These proceedings are an accumulation of the presentations provided at the meeting along with annotations provided by the authors. The proceedings cover system concepts, technology development, and systems modeling for NTP and NEP. The test facilities required for the development of the nuclear propulsion systems are also discussed.

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