Nuclear Propulsion Technical Interchange Meeting

Volume I

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

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Unclas
Nuclear Propulsion Technical Interchange Meeting
Volume I

Proceedings of a meeting sponsored and hosted by
NASA Lewis Research Center
Plum Brook Station
October 20–23, 1992
NUCLEAR PROPULSION TECHNICAL INTERCHANGE MEETING

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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.
Mars Exploration Program

Nuclear Propulsion Technical Interchange Meeting

Sandusky, OH
October 26, 1992

Dwayne Weary
Exploration Programs Office (EXPO)
NASA Johnson Space Flight Center

Space Exploration
Missions to the Moon, Mars, and Beyond ...

America wants a NASA of explorers, pioneers, and innovators to boldly expand the frontiers of air and space for the benefit of all.

Introduction: Requirements 2
Assure America's Leadership in the Next Millennium

**Mars Exploration Program**

| Activities          | 1 | 2 | 3 | 4 | 5 | 6 | 7 | 8 | 9 | 10 | 11 | 12 | 13 | 14 | 15 | 16 | 17 | 18 |
|---------------------|---|---|---|---|---|---|---|---|---|----|----|----|----|----|----|----|----|
| **Technology & Advanced Development** |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |
| Laser Infrared      |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |
| Technology Advan     |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |
| Air & Spacecraft     |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |
| **Lunar Precursors** |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |
| In-situ            |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |
| **Lunar Unmanned**  |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |
| **Lunar Piloted**   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |
| Earth-like Modules  |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |
| **Mars Precursor**  |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |
| Mars Surface        |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |
| Mars Lander         |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |
| **Mars Piloted**    |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |
| Earth-like Modules  |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |
| Mars Surface        |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |
| Mars Lander         |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |

**Office of Exploration**

**EXPLORATION PROGRAMS OFFICE**

**Introduction: Requirements**
Mars Exploration Program

Program Goals

**Technical Goal**
Verify the ability of people to inhabit the planet Mars

**Management Goal**
Demonstrate effective global cooperation in a high-technology initiative

**Societal Goal**
Demonstrate improvements in economic vitality and the quality of life for all participating nations

Mars Exploration Program

Surface Mission Objectives

- Demonstrate substantial self-sufficiency in life-support consumables and in fuel on a local scale
- Determine the potential for expansion of the initial outpost
- Explore Mars – Understand Similarity to, and Differences from, Earth
  - Life – Past and Present
  - History of Atmosphere/Climate
  - Geologic Evolution and Present State
    - Crew is assumed fit on Mars arrival
    - Back contamination assumed resolved by precursor missions

Introduction: Requirements
Mars Exploration Program
- Mission Class Considerations -

Abort Strategies

Zero-g Considerations

Galactic Cosmic Radiation Exposure

Mars Exploration Program
- Reference Mission Groundrules -

- Split Mission Strategy:
  - Basic approach - humans on fast, moderately energetic transfers; cargo and all other assets delivered to Mars via minimum-energy trajectories
  - Eliminate LEO Assembly
  - Human missions employ long duration stay (~550 days at Mars) mission profiles with fast (<160 days) Earth-Mars and Mars-Earth transit legs
  - Abort strategy:
    - Aborts for human missions post-TMI are to the surface of Mars
    - Program assets directed toward the focus of the mission

- First human mission in 2010
  - Most challenging opportunity in the 15-year Earth-Mars cycle
  - Performance margin exists for other opportunities
  - Achievable development schedule
- Cargo mission in 2008
  - Cargo requirement of ~150 t to the surface of Mars
- Crew of 3
  - Based on past studies of skills mix and threshold psychological group dynamics
  - Reasonable starting point
**Mars Exploration Program**  
- Split Mission Strategy -  

**Objectives:**
- Eliminate LEO Assembly for [specific cargo and piloted missions]
- Use the FLO HLV, or a FLO-evolved HLV (shroud)
- Reduce number of HLV launches
  - Send all surface and orbital assets to Mars on minimum energy trajectories
  - Crew-only use medium-energy, fast transit trajectories
- Provide mission flexibility to recover from contingencies
- Reduce engine testing requirements, if NTR is employed
- Provide launch window flexibility
  - 3/4 launches within the Earth-Mars window

**Results:**
- First human mission to Mars reasonably achievable in 6 total launches of a FLO HLV. Potentially achievable in 4.
- Significant mission content. 150 t of usable payload delivered to the surface of Mars
- No LEO assembly, rendezvous, or litter needed
- Significant mission flexibility with this type of strategy
Mars Exploration Program
- Operations Concept -

Office of Exploration

- Vehicle Prime
  - Realtime Systems Management
- Crew Prime
  - Realtime Exploration Functions
  - Crew Health Maintenance
  - Holly Planning and Resource Mgmt.
  - Preventive/Unscheduled Maintenance
- Crew Backup
  - Realtime Systems Management
- Ground Prime
  - Supplying Mission Objectives
  - Sustaining Engineering
  - Mission Critical Software Reconfigure
  - Crew Training
  - Procedures Development/Verification
  - Uncrewed Operations
- Ground Backup
  - Contingency Support
- Ground and Crew Share
  - Exploration

Mars Transfer Stage System Groundrules

- NTR Propulsion (2 Engines-50 klbs. Thrust Each)
- Transit Habitat for 6 Crew / 360 Days
- Lunar HLV Derivative
- No Radiation Disk Shield for Cargo Missions
- Mars Orbit - 250 km x 1 Sol Elliptical
- Separate Power Generation for Transfer Hub
- Automated Rendezvous for Mars Orbital Ops
- Storable RCS System for Vehicle Elements
- MEV for 6 Crew and 5 mt to Surface
- Zero-g Mars Transit
- Direct Entry Capability at Earth Return

NP-TIM-92
7

Introduction: Requirements
Mars Exploration Program
- Launch Vehicle Considerations -

HLLV Requirements

- HLLV Derivatives - FLO or Early HLLV
- 200 mt Class IMLEO Launch Capability
- Shroud Size Options (Cylindrical Section): 14m x 30m or 10m x 50m
- Launch Window - ~90 days: 2-4 Launches per Mars Opportunity

Mars Exploration Program
- Mars Program Schedule (2007 Cargo Launch) -

Introduction: Requirements
Mars Exploration Program
- Study Plan -

- Continue ExPO development of the Reference Mission
- Consider, compare, and contrast alternative reference mission concepts defined by non-ExPO teams
- Study system and subsystem implementation concepts to improve database
Solar System Exploration Division: Requirements for Space Nuclear Propulsion

Nuclear Propulsion Technical Interchange Meeting

Sandusky, OH
October 20, 1992

Douglas Stetson
NASA Headquarters

- Solar System Exploration Goals and Missions
- Nuclear Electric Propulsion Rationale
- Nuclear Electric Propulsion Requirements
- Low-Power Missions
- Summary
- Solar System Origins

- Planetary Evolution and State
  - Obtain an In-Depth Understanding of the Planetary Bodies in Our Solar System and Their Evolution Over the Age of the Solar System.

- Evidence of Life
  - Search for Evidence of Life in Our Own and Other Planetary Systems, and Understand the Origin and Evolution of Life on Earth and Other Planets.

- Robotic and Human Exploration
  - Conduct Scientific Exploration of the Moon and Mars, and Utilize the Moon as a Base of Scientific Study in Participation with NASA's Mission from Planet Earth.

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**SSED REQUIREMENTS FOR SPACE NUCLEAR PROPULSION**

**Solar System Exploration Goals**

- Solar System Origins

- Planetary Evolution and State
  - Obtain an In-Depth Understanding of the Planetary Bodies in Our Solar System and Their Evolution Over the Age of the Solar System.

- Evidence of Life
  - Search for Evidence of Life in Our Own and Other Planetary Systems, and Understand the Origin and Evolution of Life on Earth and Other Planets.

- Robotic and Human Exploration
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**SSED REQUIREMENTS FOR SPACE NUCLEAR PROPULSION**

**Next Mission Phases: Outer Planets**

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**Reconnaissance**

- Jupiter: Pioneer 10 Flyby
- Saturn: Pioneer 11 Flyby, Voyager 1 Flyby, Voyager 2 Flyby
- Uranus: Voyager 2 Flyby
- Neptune: Voyager 2 Flyby
- Pluto: Flyby

**Exploration**

- Jupiter: Galileo Orbiter, Jupiter Probe
- Saturn: Cassini Orbiter, Titan Probe
- Uranus: Orbiter/Probe
- Neptune: Orbiter/Probe

**Intensive Study**

- Jupiter: Grand Tour
• Nuclear Reactor Heat Source, Ion Propulsion System
  - Much More Efficient than Chemical Propulsion

• NEP Required for Next-Generation Outer Solar System Missions
  - Provides Payload Capability Unobtainable With Conventional Propulsion
  - Reduces Flight Time, Launch Vehicle Requirements
  - Also Enables High-Power Science Experiments

• SP100 Technology Baseline
  - Capable of 100KW for Outer Planet Missions
  - Lifetimes Up to 10 Years (Full Power)
  - Compatible With Active Power Conversion Technologies

PLUTO MISSION PERFORMANCE
### Science Objectives

**Thorough Characterization of Galilean Satellites**
- Geology, Morphology, Elemental Composition
- Gravitational and Magnetic Properties
- Interactions with Jupiter's Magnetosphere

**Follow-On to Galileo Study of Jupiter**
- Atmosphere, Inner Magnetosphere, Ring System

### NEP Mission Capabilities

**Sequential Orbiting of All 4 Galilean Satellites**
- Comprehensive Imaging and Spectroscopy
- Radar Sounding, Altimetry, Other Active Experiments

**Possible Addition of Jupiter Polar Orbiter or Satellite Landers**

**Large Science Payload, = 10 Year Mission Duration**
- Conventional Propulsion: 4 Separate Launches

---

### Science Objectives

**Comprehensive Study of Asteroid Physical Characteristics**
- Size, Shape, Density, Spin Properties
- Surface Composition, Solar Wind Interactions

**Variations With Solar Distance**

**Meaningful Sample Size, Variety of Spectral Types**

### NEP Mission Capabilities

**Rendezvous With 4-6 Main Belt Asteroids**
- Approximately 60 Days at Each Target
- Possible Intervening Slow Flybys
- Unlimited Orbit-Change Capability

**Large Science Payload**
- Imaging, Spectroscopy, Radiometry
- Multiple Penetrators

**Total Mission Duration = 10 Years**
- Conventional Propulsion: Max. 2 Targets, > 8 Years Duration
SSED REQUIREMENTS FOR SPACE NUCLEAR PROPULSION

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<td>&lt; 35</td>
<td>≈ 2003</td>
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- Initial NEP System Will Address Reduced Requirements
  - Simplifies Development, Reduces Cost
  - Still Capable of Excellent Planetary Missions

- Mission/System Studies Ongoing
  - Joint NASA/DOE Report Issued
  - JPL/LeRC Study Focussing on Low-Power Missions

- Preliminary Mission Options Include:
  - Mars Orbiter, Phobos-Deimos Rendezvous (SEI Focus)
  - Main-Belt Asteroid Missions
  - Jupiter Satellite Mission
  - Solar Probe

- System Requirements (Preliminary):
  - Minimum 20 kW NEP System
  - Minimum 3 Year Lifetime (Full Power)
  - Growth Potential to 100 kW, 10 Years Lifetime
- Nuclear Electric Propulsion Enables Next-Generation Outer Solar System Mission

- Requirements
  - 100 kWe, 10 -yr. Lifetime (Full-Power), < 35 kg/kWe
  - Initial System: > 20 kWe, > 3 Yr. Full-Power Lifetime
  - Full-Power System Launch ~2005
DOD REQUIREMENTS FOR SPACE NUCLEAR THERMAL PROPULSION

PRESENTATION TO

NUCLEAR PROPULSION TECHNICAL INTERCHANGE MEETING

BY

LT COL GARY A. BLEEKER
SNTP PROGRAM MANAGER
PHILLIPS LABORATORY

20 OCTOBER 1992

POTENTIAL DOD APPLICATIONS OF NUCLEAR THERMAL PROPULSION

- UPPER STAGES ON EXISTING AND/OR NEW LAUNCH SYSTEMS
- ORBIT TRANSFER VEHICLES (OTVs)
- REUSABLE OTVs
- ORBIT MANEUVERING VEHICLES
Typical Upper Stage Applications

- PBR Second Stage Typically Offers 2x - 4x Payload Improvement
- Exo-Atmospheric Operation (> 50 n.m.)

*Design Baseline 83 klbf Engine

PBR Stage Thrust
- 40 klbf
- 40 klbf
- 45 klbf
- 170 klbf
- 75 klbf
- 500 klbf

Payload to LEO (Klbs)

Complement National Launch System

Elminate NLS 1, 2

NLS Requirements

USAF/NASA Requirements:
- 20 Klbs to LEO - NLS 3
- 50 Klbs to LEO - NLS 2
- 15 Klbs to GSO - NLS 2

Add'l NASA Requirements:
- SSF: 80 Klbs - NLS 1
- SEI Heavy Lift - TBD

NLS 3 + SNTP + GEMS
NLS 3 + SNTP + GEMS

LEO
- 20
- 41
- 52
- 40

GSO
- 3.5
- 12
- 16
- (SSF)
DOD APPLICATIONS NO LONGER UNDER CONSIDERATION

- BALLISTIC MISSILE INTERCEPTOR SECOND STAGE
- ICBM SECOND STAGE

DOD/AIR FORCE NTP REQUIREMENTS

- DOD AND AIR FORCE DO NOT SPECIFICALLY CALL OUT NEED FOR NTP
  - CALL OUT MISSION REQUIREMENTS, NOT TECHNOLOGY
  - NTP COULD ENABLE MISSION ACCOMPLISHMENT (LAUNCH UPPER STAGE) AT LESS EXPENSE AND WITH GREATER RELIABILITY
SNTP PERFORMANCE
GOALS

SNTP HAS THE FOLLOWING PERFORMANCE GOALS IN DEVELOPING AN ENGINE TECHNOLOGY WITH TWICE THE SPECIFIC IMPULSE OF H2/O2 ENGINES WITH COMPARABLE THRUST TO WEIGHT

THRUST: 20,000 to 80,000 LBF
THRUST TO WEIGHT RATIO: UP TO 35 TO 1
SPECIFIC IMPULSE, ISP: 1,000 SEC
GAS CHAMBER TEMPERATURE: 3,000K
RUN TIME DURATION: 1,000 SEC
ENGINE CYCLES: 3 TO 10
ENGINE STARTUP TIME: UNDER 10 SEC

Potential Cost Benefits

Assumed $1000/Lb Launch Cost to LEO (Past Year 2000)

<table>
<thead>
<tr>
<th>Mission</th>
<th>Impact of SNTP</th>
<th>$/Mission</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td>*Non-Recurring</td>
</tr>
<tr>
<td>National Launch</td>
<td>Eliminate Large Core</td>
<td>$25 M + $2 B*</td>
</tr>
<tr>
<td>System</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Atlas Upgrade</td>
<td>Titan IV Payload</td>
<td>$130 M</td>
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<tr>
<td></td>
<td>Capability</td>
<td></td>
</tr>
<tr>
<td>Orbital Maneuvering</td>
<td>Retrieve/Repair High</td>
<td>$500 M</td>
</tr>
<tr>
<td>Vehicle</td>
<td>Value Satellites</td>
<td></td>
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</table>
Focused Technology: Nuclear Propulsion

Nuclear Thermal Propulsion  Nuclear Electric Propulsion

Presentation to SSTAC/ARTS
Thomas J. Miller
10/21/92

NASA LEWIS RESEARCH CENTER

OBJECTIVE

DEVELOP AND DEMONSTRATE TECHNOLOGY FOR NUCLEAR PROPULSION SYSTEMS TO SATISFY USER CODE MISSION REQUIREMENTS
- BALANCE TECHNOLOGY AND PERFORMANCE WITH SOUND SAFETY AND ENVIRONMENTAL POLICIES

SCOPE

- NUCLEAR THERMAL
- NUCLEAR ELECTRIC

CUSTOMER

- LUNAR/MARS EXPLORATION (OEX)
- ROBOTIC SCIENCE (OSSA)

ELEMENTS

- CONCEPT DEVELOPMENT AND SYSTEMS ENGINEERING
- INNOVATIVE TECHNOLOGY
- ENABLING TECHNOLOGY (NEP & NTP)
- FACILITIES
- SAFETY, QA AND ENVIRONMENT

NP-TIM-92 21

Introduction: Executive Summary
MISSIONS CONSIDERATIONS

- SAFETY
- PERFORMANCE
- COST
- SCHEDULE FOR DEVELOPMENT
- OPERATIONAL FLEXIBILITY
  - APPLICATION TO RANGE OF MISSIONS
  - EVOLUTIONARY GROWTH POTENTIAL

NUCLEAR PROPULSION SUMMARY

<table>
<thead>
<tr>
<th>Nuclear Thermal Propulsion</th>
<th>Nuclear Electric Propulsion</th>
</tr>
</thead>
<tbody>
<tr>
<td>Specific Impulse: 850 - 950 sec</td>
<td>Specific Impulse: 4000 - 8000 sec</td>
</tr>
<tr>
<td>Thrust to Weight: 6 - 10</td>
<td>Specific Mass:</td>
</tr>
<tr>
<td></td>
<td>Robotic Science</td>
</tr>
<tr>
<td></td>
<td>Piloted Mars</td>
</tr>
</tbody>
</table>

$\dot{I}_{sp} = \frac{T}{m}$

CHEMICAL PROPULSION (H/O): 460 sec Specific Impulse
OVERVIEW
OF
DOE SPACE NUCLEAR PROPULSION PROGRAMS

BY
ALAN R. NEWHOUSE
DEPUTY ASSISTANT SECRETARY
FOR SPACE AND DEFENSE POWER SYSTEMS
U.S. DEPARTMENT OF ENERGY

PRESENTED AT THE
NUCLEAR PROPULSION INTERCHANGE MEETING
NASA LEWIS RESEARCH CENTER, PLUM BROOK STATION
SANDUSKY, OHIO
OCTOBER 20, 1992
TYPES OF SPACE NUCLEAR PROPULSION

- **NUCLEAR THERMAL ROCKETS (NTR)**
  Develop thrust by using a nuclear reactor to heat a propellant gas and expel it through a nozzle
  - High Specific Impulse* (*Twice better than best chemical systems)
  - High thrust (fast acceleration)
  - Short lifetime (minutes to hours)
  Primary NASA option for cargo and piloted Mars mission; also, promising choice for many DOD applications

- **NUCLEAR ELECTRIC PROPULSION (NEP)**
  Develop thrust by using electricity produced from heat in a nuclear reactor to ionize a propellant and accelerate the charged particles through a thruster
  - Very high Specific Impulse* (*10 times better than best chemical systems)
  - Low thrust (continuous acceleration)
  - Long lifetime (months to years)
  Enabling or significantly enhancing for several NASA solar system robotic missions; near term NEP reactor systems will be based on space reactors currently under development

* Thrust produced per rate of propellant consumption

NUCLEAR THERMAL PROPULSION HISTORICAL SUMMARY (ROVER//NERVA)

- 18 year development program (1955-1973) ($1.4 billion expended in then-year dollars)
- 20 reactors built and tested
  - Reactor design and development - LANL
  - Design and manufacture of rocket engine systems - Westinghouse and Aerojet
  - Testing of all reactors - Nevada Test Site
- Performance demonstrated
  - Power level 4100 MWt
  - Peak fuel temperature 2750K
  - Specific impulse (isp) 850 sec
  - Start/stop cycles 28
  - Continuous operation 62 minutes
DOE'S CHARTER FOR THE DEVELOPMENT OF NUCLEAR POWER

- DOE'S CHARTER, ARISING FROM AUTHORITY IN THE ATOMIC ENERGY ACT OF 1954, AS AMENDED, IS TO SUPPORT FEDERAL AGENCIES (DOD AND NASA) IN MEETING THEIR SPECIAL POWER NEEDS FOR BOTH TERRESTRIAL AND SPACE APPLICATIONS
DOE ROLE IN SPACE NUCLEAR PROPULSION

- TRADITIONAL DOE ROLE OF DESIGNING, DEVELOPING, TESTING, AND PROVIDING NUCLEAR SYSTEMS, INCLUDING ENVIRONMENTAL, SAFETY, AND HEALTH ASPECTS

- NASA/DOE MEMORANDUM OF UNDERSTANDING (MOU) FOR ENERGY-RELATED CIVIL SPACE ACTIVITIES
  - SPECIFIC PROVISIONS FOR NUCLEAR PROPULSION

- PROJECT SPECIFIC MOU FOR NUCLEAR PROPULSION
  - DRAFT PREPARED FOR BOTH NASA AND USAF PROJECTS

- NATIONAL SPACE POLICY DIRECTIVE ON SPACE EXPLORATION INITIATIVE
  - NASA, DOD, AND DOE DIRECTED TO CONTINUE TECHNOLOGY DEVELOPMENT FOR SPACE NUCLEAR POWER AND PROPULSION

DOE SAFETY ROLE IN SPACE NUCLEAR PROPULSION ACTIVITIES

- ENVIRONMENT, HEALTH, AND SAFETY
  - OVERALL POLICY, ALARA
  - OVERSIGHT
  - NEPA PROCESS
  - SAFETY ANALYSIS, REPORTS/APPROVALS
  - PUBLIC SAFETY
  - SAFEGUARDS
  - SITE MONITORING

- NUCLEAR SYSTEM DESIGN, MANUFACTURE, ASSEMBLY, CHECKOUT AND OPERATION

- GROUND TEST FACILITY DESIGN, ACQUISITION, CONSTRUCTION AND OPERATION
SAFETY APPROACH

● SAFETY IS AN OVERRIDING CONSIDERATION:
  - FOR PROTECTION AGAINST ACCIDENTS
  - FOR PUBLIC ACCEPTANCE
  - FOR BOTH GROUND TESTING AND SPACE OPERATIONS

● ULTIMATE SAFETY OBJECTIVE:
  - MINIMIZE RISK TO PUBLIC AND CREW IN NORMAL AND
    ABNORMAL OPERATIONS

● NUCLEAR POWER SOURCE LAUNCH APPROVAL PROCESS:
  - BASED ON RIGOROUS SAFETY REQUIREMENTS, EVALUATION
    AND TESTING
  - CONSIDERS MISSION OBJECTIVES/BENEFITS VERSUS RISKS
  - BASED ON SUCCESSFUL HISTORY OF ISOTOPE AND REACTOR
    APPLICATIONS BY NASA AND DOD

NUCLEAR SYSTEM DESIGN, MANUFACTURE,
ASSEMBLY, CHECKOUT, AND OPERATION

● NUCLEAR SYSTEM DESIGN STANDARDS, REQUIREMENTS, AND CODES
  - DESIGN SAFETY FEATURES
  - SAFETY TEST REQUIREMENTS AND ANALYSIS
  - PREOPERATIONAL CHECKS AND TESTS

● MANAGEMENT OF FACTORY, SHIPPING, SITE, AND POST TEST OPERATIONS
  FOR NUCLEAR COMPONENTS AND SYSTEMS
  - FACTORY, SUB-ASSEMBLY TESTING, CRITICALS, ETC.
  - SHIPPING AND ASSEMBLY
  - FACILITY CONTROLS
  - EMERGENCY PLANNING
  - EMERGENCY ACTIONS
  - RECOVERY, CLEANUP, AND DISPOSAL ACTIONS

● NUCLEAR FLIGHT SYSTEM OVERSIGHT (WITH USER AGENCIES)
  - OVERALL POLICY DEFINITION
  - SAFETY REVIEW AND APPROVAL PROCESSES
  - FLIGHT OPERATIONS MONITORING
  - SUPPORT IN POSSIBLE EMERGENCIES
  - NORMAL AND ABNORMAL DISPOSAL OPERATIONS
  - POTENTIAL GROUND RECOVERY OPERATIONS
GROUND TEST FACILITY DESIGN
ACQUISITION, CONSTRUCTION, AND OPERATION

- SITE SELECTION AND MANAGEMENT OVERSIGHT
  - STANDARDS AND CRITERIA
  - SITE PREPARATION AND MAINTENANCE
  - SITE MONITORING
  - SHIPPING/HANDLING OF RADIOACTIVE/HAZARDOUS MATERIALS
  - DECOMMISSIONING AND DISPOSAL
- FACILITY DESIGN AND CONSTRUCTION OVERSIGHT
  - STANDARDS AND DESIGN CRITERIA
  - FUNCTIONAL REQUIREMENTS
  - SAFETY REQUIREMENTS, ANALYSES, AND APPROVALS
  - PREOPERATIONAL CHECKS AND TESTING OF EQUIPMENT AND SYSTEMS
- OVERSIGHT OF OPERATIONS
  - CONDUCT OF OPERATIONS
  - TRAINING REQUIREMENTS
  - TEST PROCEDURE APPROVAL
  - SPECIFIC TEST APPROVAL
  - POST IRRADIATION EXAMINATION
- QUALITY ASSURANCE PROGRAM OVERSIGHT
- SAFEGUARDS AND SECURITY OVERSIGHT

GROUND TESTING ISSUES

- MAJOR FACILITIES REQUIRED
  - EITHER NEW FACILITIES OR EXTENSIVE MODIFICATIONS TO EXISTING FACILITIES
  - MUST MEET CURRENT ENVIRONMENTAL AND SAFETY REQUIREMENTS (EFFLUENT CONTROL)
  - FACILITY STUDY ESTIMATES $0.5 TO OVER $1B AND 7-10 YEARS EACH
  - DOE SITES WILL BE USED
- TYPES OF FACILITIES
  - FUEL BUNDLE QUALIFICATION
  - ENGINE SYSTEM
- ISSUES
  - SAFETY AND PUBLIC ACCEPTANCE
  - LARGE COST
  - SINGLE NATIONAL TEST COMPLEX VERSUS MULTIPLE COMPLEXES
- EXPERIENCE
  - ROVER/NERVA DESIGN
  - NUCLEAR FURNACE* - HAS SHOWN GROUND TESTING CAN BE ACCOMPLISHED THROUGH USE OF A SCRUBBER SYSTEM

*A SMALL (50 MW) HIGH TEMPERATURE REACTOR USED FOR TESTING NUCLEAR THERMAL ROCKET FUEL ELEMENTS.

Introduction: Executive Summary
POTENTIAL USE OF CIS FACILITIES

- Two DOE delegations recently returned from a fact finding trip to Russia and Kazakhstan
  - Visited several nuclear propulsion facilities which could possibly be used
  - Reviewed space power and propulsion capabilities
  - Still compiling information obtained and drafting reports
  - Must carefully review and verify capabilities

- Exact DOE role in using or making use of foreign nuclear facilities or technologies still needs to be defined
  - International agreements may be needed
  - Role of U.S. industry needs to be further explored

- Topic being worked

NUCLEAR PROPULSION DEVELOPMENT NEEDS

- Past program performance not adequate for today's needs

- Performance improvements required
  - Higher specific impulse (900-1000 sec.)
  - Higher thrust/weight (25 - 35 to 1)
  - Differing requirements for civilian and military applications (e.g., run time, restarts)

- New development program needed
  - Reestablish old technology and consider new concepts
KEY NEAR-TERM NUCLEAR PROPULSION ACTIVITIES FOR SPACE EXPLORATION

- DEVELOPING AND TESTING OF CANDIDATE NTP FUELS

- EARLY STUDY AND SELECTION OF EFFLUENT TREATMENT SYSTEMS
  - TEST AND QUALIFY PROTOTYPE COMPONENTS AND SUBSYSTEMS

- NUCLEAR FACILITY PRECONSTRUCTION ACTIVITIES
  - INITIATE ENVIRONMENTAL, SAFETY, AND PRELIMINARY DESIGN ACTIVITIES
  - PROCEED TOWARD A SINGLE NATIONAL NUCLEAR PROPULSION TEST COMPLEX
    - MEETS BOTH NASA AND DOD REQUIREMENTS

- NTP CONCEPTS ASSESSMENTS AND DEFINITION
  - NERVA DERIVATIVE
  - PARTICLE BED
  - CERMET
  - CIS TWISTED RIBBON

RECENT DOE NUCLEAR PROPULSION ACTIVITIES FOR SEI

- LIMITED ASSESSMENTS OF NUCLEAR PROPULSION CONCEPTS AND ASSOCIATED TECHNOLOGIES

- NUCLEAR FUEL DEVELOPMENT

- FACILITIES EVALUATIONS AND ASSESSMENTS
  - DOE/NASA/USAF NUCLEAR FACILITIES REVIEW
    - INITIATED STUDIES FOR COMMON FACILITIES
  - PROVIDED FACILITIES INPUT FOR SNTP DRAFT EIS EFFORT
  - CIS FACILITIES VISITS

- PLANNING AND PROGRAMMATIC ACTIVITIES
SPACE NUCLEAR THERMAL PROPULSION (SNTP) PROGRAM

- USAF TECHNOLOGY DEVELOPMENT PROGRAM TO DEMONSTRATE THE FEASIBILITY AND HIGH PERFORMANCE CAPABILITIES OF A NUCLEAR PROPULSION SYSTEM USING PARTICLE BED REACTOR TECHNOLOGY FOR POSSIBLE U.S. AIR FORCE (USAF) SPACE PROPULSION NEEDS

- USAF PHILLIPS LABORATORY IS PROGRAM MANAGER FOR THE SNTP PROGRAM

- DOE IS RESPONSIBLE FOR THE NUCLEAR DEVELOPMENT PORTION OF THE PROGRAM, INCLUDING NUCLEAR SAFETY OVERSIGHT AND NUCLEAR GROUND TESTING

- SANDIA NATIONAL LABORATORIES ALBUQUERQUE (SNLA) AND BROOKHAVEN NATIONAL LABORATORY (BNL) ARE PRINCIPAL DOE LABORATORIES PARTICIPATING ON THE PROGRAM

SNTP NUCLEAR PROPULSION EIS ACTIVITIES

- DRAFT EIS
  - ISSUED FOR PUBLIC REVIEW
  - FINAL EIS EXPECTED IN NOVEMBER 1992
  - TWO SITES UNDER CONSIDERATION
    - NEVADA TEST SITE
    - IDAHO NATIONAL ENGINEERING LABORATORY TEST SITE
  - SITE SELECTION ANTICIPATED IN JANUARY 1993
SUMMARY

- LONG HISTORY OF SUCCESSFUL USE OF NUCLEAR POWER IN SPACE (AND DOE SUPPORT OF THESE SYSTEMS)

- SPACE NUCLEAR THERMAL PROPULSION IS A LONG LEAD DEVELOPMENT ACTIVITY. CONSOLIDATION OF U.S. MILITARY AND CIVILIAN EFFORTS TO THE GREATEST DEGREE POSSIBLE WOULD BE BENEFICIAL.

- DOE WILL HAVE A LEAD ROLE IN DIRECTING THE NUCLEAR ASPECTS OF SPACE NUCLEAR THERMAL PROPULSION PROGRAMS; ACQUIRING AND OPERATING THE GROUND NUCLEAR TEST FACILITIES; AND ASSURING THE SAFETY OF ALL DESIGN, DEVELOPMENT, FABRICATION, TEST, AND OPERATIONS ACTIVITIES
SPACE NUCLEAR THERMAL PROPULSION (SNTP) PROGRAM

PRESENTATION TO

NUCLEAR PROPULSION TECHNICAL INTERCHANGE MEETING

BY

LT COL GARY A. BLEEKER
PROGRAM MANAGER
PHILLIPS LABORATORY

20 OCTOBER 1992

SPACE NUCLEAR THERMAL PROPULSION PROGRAM

NUCLEAR ROCKET PROGRAM

- TECHNOLOGY CHALLENGE
  - DEVELOP ADVANCED NUCLEAR ROCKET ENGINE WITH 2X THE ISP OF BEST LIQUID ENGINES AND THRUST TO WEIGHT COMPARABLE TO H2O2
  - PROGRAM PRIORITIES ARE SAFETY, RELIABILITY, OPERABILITY, PERFORMANCE, AND AFFORDABILITY

- PAYOFF
  - WIDE VARIETY OF POTENTIAL APPLICATION FOR UPPERSTAGES, OTV's AND PLANETARY MISSIONS
  - 60-80% COST SAVINGS PER LAUNCH

Introduction: Executive Summary
Introduction: Executive Summary
SAFETY, ENVIRONMENTAL, HEALTH

- TOP PRIORITY FROM INCEPTION
  - PROGRAM SAFE: Y POLICY ESTABLISHED AND BEING FOLLOWED
  - PSAR COMPLETE AND UNDER REVIEW
  - MEETING ALL FEDERAL/STATE REGULATORY REQUIREMENTS
  - SUBSTANTIAL INTERNAL AND EXTERNAL REVIEW (DSB, DOE, NAS)
  - FOLLOWING ALARA (AS LOW AS REASONABLY ACHIEVABLE) APPROACH
SUMMARY

- **NUCLEAR WILL BE THE PROPULSION SYSTEM OF THE 21st CENTURY**
  - ESSENTIAL TO MAINTAIN U.S. COMPETITIVENESS AND SUPREMACY IN SPACE

- **SNTP CONFORMS TO NATIONAL POLICY**
  - HIGH PAYOFF R&D: MANY APPLICATIONS/MISSEIONS
  - LEVERAGE DOD, DOE, AND NASA TECHNOLOGY BASE

- BASED ON CURRENT PROGRESS, PROGRAM HAS A HIGH PROBABILITY OF SUCCESS

- **ALL APPLICABLE NUCLEAR SAFETY AND ENVIRONMENTAL OBJECTIVES WILL BE MET**
NUCLEAR THERMAL PROPULSION

SYSTEM CONCEPTS
Systems Overview

Nuclear Propulsion Technical Interchange Meeting

Sandusky, OH
October 2, 1992

Robert Corban
Nuclear Propulsion Office
NASA Lewis Research Center
Systems Overview
Requirements and Public Acceptance

- **OBJECTIVE**
  - Provide NASA with Requirements management expertise
    - Requirement definition
    - Change management and control
    - Requirements document maintenance
  - Provide public acceptance planning
- **Analytical Engineering Corporation**
  - Awarded competitive contract (small business set-aside)
  - On-going 5 year contract
  - Provide key functional analysis
  - Initial requirements document developed and controlled

Systems Overview
Requirements and Public Acceptance

The following charts provide a brief synopsis of the contracted efforts for FY92 in assessing Nuclear Thermal Propulsion requirements, concepts, and associated issues.

**Requirements and Public Acceptance**

**Objective**

This effort is to provide NASA LeRC with assistance in space nuclear propulsion system requirements management and public acceptance planning. Requirements management will include requirement definition, requirement change management and control, and requirement document maintenance. Specific objectives are to: 1) provide assistance in defining clear, concise, verifiable nuclear propulsion system requirements, 2) provide full traceability of requirements with reference, analysis, design, and historical data with the ability to assess the impact of requirement changes, 3) produce documentation of the nuclear propulsion system requirements and specifications that can easily accommodate changes, 4) provide assistance in public acceptance planning, and 5) include the resultant system requirements for a publicly acceptable SEI nuclear propulsion system.

**Analytical Engineering Corporation**

Analytical Engineering Corporation (AEC) was awarded a five year contract in FY92 to meet the objectives defined above. AEC's approach will utilize detailed functional analysis to ensure that system functional requirements are accurately interpreted and flow down to system specifications. An initial requirements document has been developed and continuous improvements are on-going.
The objective of these studies was to determine the feasibility of a nuclear thermal propulsion system based on a particular fuel element form for the nuclear reactor. The studies evaluated "state-of-the-art" concept feasibility, thrust level range implications, test facility requirements, manned mission impacts, and key component technologies required. Shown in the chart are the study teams and their associated fuel element that was the basis for their concept analysis.
Systems Overview
Concept Feasibility Assessments (continued)

Concept Definition
The Contractors were requested to define a nuclear thermal propulsion concept based on their particular reactor concept in sufficient detail to permit reasonable judgements on feasibility, weight, performance, safety features, operations, and key technology requirements. An overall assessment of the NTP engine would include the reactor assembly, nozzle, propellant feed system, thrust vector control, instrumentation and control, and propellant pressurization. The concepts were defined to meet, as a minimum, the basic performance requirements defined below. The NTP engine concepts were assessed at one specific thrust level point with sensitivities determined for two others.

Baseline Design Requirements

<table>
<thead>
<tr>
<th>PARAMETER</th>
<th>REQUIREMENT</th>
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<tbody>
<tr>
<td>Thrust</td>
<td>25K -75K</td>
</tr>
<tr>
<td>Thrust/Weight (w/ Internal Shield)*</td>
<td>≥4</td>
</tr>
<tr>
<td>Specific Impulse</td>
<td>≥850 seconds</td>
</tr>
<tr>
<td>Throttling</td>
<td>25% Thrust @ Rated Temperature</td>
</tr>
<tr>
<td>Reuse</td>
<td>Multiple (Mission Dependent ≥ 10 Restarts)</td>
</tr>
<tr>
<td>Single Burn Duration</td>
<td>60 minutes (Maximum)</td>
</tr>
<tr>
<td>Engine Life</td>
<td>&gt;270 minutes at Rated Thrust (3X Required)</td>
</tr>
<tr>
<td>Reliability</td>
<td>Manned Systems</td>
</tr>
<tr>
<td>Propellant</td>
<td>Hydrogen</td>
</tr>
</tbody>
</table>

Key Technologies
The Contractors were to determine from the defined concept the key enabling technologies that need to be addressed before the system could be developed. Technologies that would have a significant impact on the overall system performance, safety, or reliability would also be identified. For each enhancing technologies, its system impact were to be identified along with the risks associated with its development.
The objective of this study requested by NASA JSC's Exploration Project Office (EXPO) was to develop propulsion system designs that could be integrated with the provided reference vehicle. Four propulsion system options were developed using two and three engines with either boost pumps or run tanks for engine start up. The systems issues addressed consisted of TVC requirements, engine out possibilities, propulsion system failure modes and technology development requirements.

Lunar NTR Vehicle Design & Operations Study

The objective of this study was to identify and characterize the features of NTR propulsion stages for “near-term” lunar transfer vehicle missions. The study assessed NTR stage design features, performance, and operational benefits. Programmatic (schedule and cost) issues are also addressed. Comparison of various options for lunar transfer vehicles based on past studies on “all-propulsive” and “aerobraked” chemical were also addressed.
Systems Overview

- **ENABLER I & ENABLER II**
  - Based on NERVA fuel element (scaled fuel for ENABLER II)
  - Parametric weight and size analysis
    - Thrust, Chamber Temperature, Chamber Pressure, Nozzle Area Ratio
  - Continued development of Nuclear Engine System Simulation (NESS) design program
  - Contracted with SAIC

Enabler I & II

The major objective of this task was to upgrade the Nuclear Engine System Simulation (NESS) analysis code to include the NERVA solid core engine (ENABLER I) and an advanced solid-core reactor module (ENABLER II) that utilizes scaled NERVA fuel elements. Additional objectives include the parametric characterization of the ENABLER I & II engine system concepts, and to examine on the “top-level” NTP engine design risk/reliability issues and their impact on the system.
Requirements Management and Control

Presented by: Red Robbins

Discussion

- Nuclear Thermal Propulsion Systems Engineering
- Systems Requirements Status
- Functions That Need To Be Performed
- Attributes Associated With The Functions
The systems engineering process for requirements and configuration definition, shown in the figure above, includes roles and responsibilities of NASA and the Systems Contractor. The overall program requirements are derived from the mission to be performed. The program requirements, in turn, are utilized as the basis for the definition of all lower level documents that contain increasing detail concerning design requirements associated with performance, safety, operations and environment. The lower level documents are dynamic in nature. Many studies are conducted involving trades between lower level requirements and engineering system definition. The propulsion system requirements are highlighted because they are the focus of this discussion. Stage/Engine requirements have been generated, baselined by NASA and are under formal change control and propulsion system definition is underway. The Functional Analysis activity shown is simply the process of systematically identifying the generic functions to be performed at all levels and leads ultimately to the definition of the System Architecture. Since Program and Vehicle Requirements are not currently available, the propulsion system requirements were generated on the basis of representative manned lunar and Mars missions.
The Mars Mission functional analysis shown defines the mission functions to be performed in successive levels of detail. The highlighted functions at each level are those that are directly related to the propulsion system. Those functions that are not highlighted are the principal interfaces with the propulsion system. The three functions of the propulsion module are to provide thrust, provide specific impulse and control of the engine operations. The three charts which follow are simply functional breakdowns of the main engine functions to successively lower levels of detail. On the basis of this analysis, the system architecture or system definition can be completed. It should be noted that these functions are generic and are required for any nuclear rocket. No design solutions have been assumed. It is the role of the systems definition contractors to perform the system trades which result in the definition of the propulsion system.
The completion of the functional analysis previously described permits the definition propulsion system attributes that can be utilized to assess the relative merits of competing systems and to establish criteria for technology thrusts to enhance performance, safety and reliability. The attributes associated with the functions are outlined in the four charts that follow. These attributes, in addition to the system requirements, can be utilized for the evaluation of any nuclear rocket system and can provide the system contractors guidance about system characteristics that are considered to be important.
SUMMARY REMARKS

- A Consistent Set Of Propulsion System Requirements Has Been Developed By NASA
  - Traceable To Mission Needs
  - Under Formal Change Control
- Propulsion System Functions Traceable To System Requirements Have Also Been Generated
- Preliminary Propulsion System Attributes Traceable To Functions Have Been Derived To:
  - Assess The Relative Merits Of Competing Systems/Elements
  - Establish Criteria For The Definition Of Technology Thrusts
- We Request Feedback From The Program Participants - Particularly The System Definition Contractors
- Future Work Will Be Directed To The Development Of An Integrated Propulsion Systems Model Coupled With Propulsion Systems Requirements Traceable From The Lowest Level To Mission Needs

NP-TIM-92
NUCLEAR SYSTEMS IN SPACE?
DOES/WILL THE PUBLIC ACCEPT THEM?

Harold B. Finger

Public Acceptance is always raised as an obstacle to the use of nuclear energy for any purpose, in any way. It is always cited as an issue that must be resolved before nuclear energy can be used for:

- Nuclear energy plants to generate more electricity,
- Nuclear medical diagnosis and treatment,
- Food irradiation to destroy harmful bacteria.

So it is not surprising that the assumption is generally made that there is public opposition to using nuclear energy in space that could preclude its use even for missions that it makes realistically feasible. Yes, there is a broad assumption that the public generally opposes nuclear energy.

Let me start right off by telling you that assumption is **WRONG**. (Figure 1) Here are some of the attitude data that indicate the public's attitudes on nuclear energy. They are positive, not negative. Most of the public believes nuclear energy will play an important role in our energy supply, that it should play an important role, and that the need for nuclear energy to supply our electricity will increase. Only 15% would favor closing our nuclear electric plant.

In spite of those data, you are not alone in thinking the public opposes nuclear energy. When (Figure 2) opinion leaders are asked how important a role they think nuclear energy should play in meeting our future energy needs, 72% answered Very or Somewhat important. But, then, when they were asked how important they thought the public feels about the reliance on nuclear energy, only 25% thought the public felt nuclear energy should play an important role, while 63% felt the public did not believe it should be important. As Figures 1 and 2 show, 73% of the public, the same number as the opinion leaders, believe nuclear energy should play an important role. A similar perception gap exits between Congressional staff views supporting the importance of nuclear energy and what they think the public believes.
So, (Figure 3) we all do have a job to get opinion leaders and our policy makers and many other influentials in our society to understand that the public accepts and even supports the use of nuclear energy. Doing that will certainly help get favorable policy action related to nuclear energy. But it won't be easy to get that point across. It won't be easy, at least partly because the small number of committed anti-nukes are vocal and because -- as about two thirds of those who call news about nuclear energy describe those news reports as negative -- the press does generally emphasize the negative. It appears that good news is not considered newsworthy.

As the USCEA has determined, based on broad attitude research (Figure 4), there should be no expectation that the public will accept or support the use of nuclear energy unless it meets special needs and offers special and significant benefits. That is why the USCEA's public information program emphasis (Figure 5) is on gaining recognition for the growing need for electricity in a growing economy and on nuclear energy's benefits in cutting imported oil dependence, reducing pollutant emissions and preserving scarce resources.

In transferring that lesson to our space use of nuclear energy (Figure 6), it means getting recognition and support for the space program broadly and for the missions that benefit substantially from or realistically require nuclear energy for their accomplishment.

This is what a group of aerospace and other companies are now trying to organize -- a program to do just that. If any of you here, whose organizations have not yet been involved in this effort want to become part of it, please let me or Red Robbins know of your interest. We'll welcome your participation.

Developing an effective public communication program (Figure 7) requires a solid base of attitude research. We must understand the views of the public and of our policy makers. We must determine those benefits of the space program and of the missions that are realistically enabled by nuclear energy that would be effective in gaining support for the space program and those missions. In fact, we know almost nothing about the public's attitudes and knowledge on using nuclear energy in space. I doubt that the public knows that we have already used nuclear -- radioisotope- power units in space to get data from the Moon in Apollo, to get pictures of Saturn and Jupiter, and other uses whose results were broadly and proudly discussed. We need to get such information known as part of our developing program.
We do have a fairly good feel for what the public thinks about the space program; thanks largely to the excellent work supported mainly by Rockwell International and from several others. So let me review some of those research results with you.

Here (Figure 8) are the generally highly positive views of the space program. Over 80% support the space program overall; believe it is important to the United States; approves of it; and, at least back in 1988, believed that a U.S. lead in the program was important. Figure 9 shows further data. There is less, though still strong, sense of a personal benefit than a national benefit, but it is certainly encouraging that relatively few—only 25 to 30 percent—considered space exploration a luxury at those times. I’ll address that further later.

It is also important and encouraging to see the overwhelmingly positive responses when various benefits are suggested as reasons for supporting the space program (figure 10). However, all of these attributes are suggested in the interviews; there are no open-ended questions that would ask the interviewee what he or she knows and believes is most important about the space program. Of course, that will require further attitude research. In the meantime, the data of Figure 10 are very positive.

Here (Figure 11) are the responses when various goals are suggested for the space program. You’ll notice that the support for all the proposed missions dropped from 1990 to 1992. We don’t really know the reason for that drop, but it may also indicate that we have not adequately explained the economic, job, nor technology benefits of the space program. Even some Congressmen, who should know better, say we should not spend our budget IN space, that we need the work here on the ground. That’s actually an argument we faced and addressed back in the 1960’s. The response is obvious, I believe.

Although Figure 11 shows the significant downturn in support of manned lunar and Mars missions, let me turn to broader public views concerning the manned Mars mission, which we would all agree is certainly one of the primary missions for nuclear thermal propulsion. That mission is realistically enabled by nuclear propulsion.

For our Russian friends who are here, Figure 12 shows the obvious feelings of Americans that think we should do the Mars mission together with the republics of the former Soviet Union. Americans felt that way back in 1988 when we were strong
competitors. I expect the numbers would be much higher in favor of that joint effort today.

In essence, the various data here indicate that Mars and planetary investigation rates high among the alternatives suggested for future missions. Support for the President's SEI missions also shows high figures. However, it is significant that only a little over a third of those interviewed were aware of his proposals. That is only another manifestation of the fact that his initiatives were not broadly discussed and that they were not seized within the space community nor developed and pushed as dynamic goals that could provide significant benefits for the country. There was very little discussion of those goals and proposals outside the space and science community.

The question of the importance of the U.S. being first to get to Mars drew a response that, not surprisingly, change significantly after the demise of the Soviet Union and its replacement by the Commonwealth of Independent States. In 1989, there was a small margin feeling it was important that we be first, but after the Soviet coup attempt, there was a significant reversal with only 35 percent feeling it was important that we be first. The competition with the Soviet Union was no longer considered significant as a justification for an urgent effort to be first in that difficult Mars goal. As I indicated earlier, the idea of a joint effort may be viewed as an even greater opportunity than was the case in the data of the late 1980's.

Now let me turn to the telling data on putting our money where our mouth is -- how much should we be spending on the space program? In general (Figure 13), a majority of people seem to favor investment in the space program; especially when we combine those who favor an increase with those who believe it should be continued at its current levels. Not until the choice between "investment in space or...on domestic programs" do we see a significant switch in 1990 in favor of the domestic programs. I maintain that choice is not a real one. We obviously do not spend the money in space; it is actually spent in this country and it is a benefit to our domestic economy, to our technological development and to our competitiveness and job base. I feel strongly that the space effort is the peaceful alternative to the cutback in our defense effort. That may, in fact, turn out to be an effective message and a persuasive one in getting recognition for the importance, benefits and need for such a mission and such a space program. However, determining whether that is the case will require meaningful message research and evaluation.
What are the conclusions that can be drawn from all this attitude research on the space program? Here (Figure 14) are my conclusions. The attitudes concerning the space program are generally favorable, especially when we consider the economic problems our nation faces. However, many of the comments made are in response to suggested goals, benefits, etc. There is very little research that is open-ended and seeks out the level of understanding that the public actually has about the space program and the extent that they actually think about it themselves. We need such greater searching research.

It is significant that there is no research into the attitudes of the public concerning the use of nuclear systems in space nor in determining what they would think about all the nuclear systems that have already been used in space. We need greater understanding of those views.

My next three conclusions all relate to the need for an effective program that can communicate to the public and to policy makers the benefits and importance of and the need for the space program. We must determine what messages are truly effective and then devise a broad array of approaches to communicate those messages to the public and to decision and policy makers. We have no such program now. In fact, I would have expected the President's SEI goals to have become the basis for a comprehensive program planning and communication effort. But I certainly did not see that develop and I do not see it available or being developed to the level required.

Therefore, my major conclusion, punch line and appeal to all those informed on and involved in this country's space program is that we establish a strong, effective communications program that will convey the benefits of the program and rebuild the enthusiasm for space activities we used to have. LET'S GET ON WITH THAT JOB.
ATTITUDES TOWARD NUCLEAR ENERGY

**Nuclear Energy to Play Important Role** 80%

**Nuclear Energy Should Play Important Role** 73%

**Need for Nuclear Energy to Increase** 76%

**Close Down Nuclear Plants** 15%

---

**Big Perception Gap**

**Real and Perceived Public Opinion About Nuclear Energy**

<table>
<thead>
<tr>
<th>What Opinion Leaders Think...</th>
<th>Very Important</th>
<th>Somewhat Important</th>
<th>Not too Important</th>
<th>Not important at all</th>
<th>Don't know</th>
</tr>
</thead>
<tbody>
<tr>
<td>Practically speaking, how important a role do you think nuclear energy should play in meeting America's future energy needs?</td>
<td>72%</td>
<td>40%</td>
<td>15%</td>
<td>12%</td>
<td>7%</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>What Opinion Leaders Think the Public Thinks...</th>
<th>Important role</th>
<th>Not important role</th>
<th>Don't know</th>
</tr>
</thead>
<tbody>
<tr>
<td>What about the American public: Do you think the majority of Americans would say that nuclear energy should play an important role in meeting America's future energy needs, or do you think that the majority would say that nuclear energy should not play an important role?</td>
<td>29%</td>
<td>63%</td>
<td>12%</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>What the Public REALLY Thinks...</th>
<th>Very Important</th>
<th>Somewhat Important</th>
<th>Not too Important</th>
<th>Not important at all</th>
<th>Don't know</th>
</tr>
</thead>
<tbody>
<tr>
<td>Practically speaking, how important a role do you think nuclear energy should play in meeting America's future energy needs?</td>
<td>73%</td>
<td>38%</td>
<td>10%</td>
<td>17%</td>
<td>8%</td>
</tr>
</tbody>
</table>

Prepared by the U.S. Council for Energy Awareness
April 1992

NTP: System Concepts
GAINING PUBLIC ACCEPTANCE, APPROVAL, AND SUPPORT FOR USING NUCLEAR SYSTEMS IN SPACE MISSIONS

IT'S TIME TO ORGANIZE A PROGRAM TO DO THAT
Gaining that acceptance, approval, and support requires first gaining recognition of the need for and the benefits of using those nuclear systems in space.

We do not use nuclear energy in space unless the benefit and need are clear.

**Therefore, the objective is first to gain public recognition, acceptance, approval and political support for the space program broadly; and for missions that benefit substantially from or realistically require nuclear systems for their accomplishment.**
DEVELOPMENT OF AN EFFECTIVE PUBLIC COMMUNICATION PROGRAM REQUIRES A SOLID BASE OF ATTITUDE RESEARCH

- Public attitude tracking
- Strategy and message testing
- Testing communication vehicles
- Evaluation of communication effects

ATTITUDES TOWARD SPACE PROGRAM

<table>
<thead>
<tr>
<th>Issue</th>
<th>Percentage</th>
<th>Date</th>
</tr>
</thead>
<tbody>
<tr>
<td>Support space program overall</td>
<td>80%</td>
<td>Mar. 90</td>
</tr>
<tr>
<td>Space program is important to U. S.</td>
<td>88%</td>
<td>June 88</td>
</tr>
<tr>
<td>Approve of America’s civilian space program</td>
<td>80%</td>
<td>July 88 &amp; Feb. 90</td>
</tr>
<tr>
<td>U.S. lead in space technology important</td>
<td>82%</td>
<td>Feb. 88</td>
</tr>
</tbody>
</table>

Data provided by Roper Center, University of Connecticut; from Rockwell - Market Opinion Research; and Yankelovich - Time Magazine sources.
FIGURE 9

IMPORTANCE OF THE SPACE PROGRAM

<table>
<thead>
<tr>
<th>Reason</th>
<th>JULY 1988</th>
<th>FEB. 1990</th>
</tr>
</thead>
<tbody>
<tr>
<td>To our country</td>
<td>88%</td>
<td>82%</td>
</tr>
<tr>
<td>To you personally</td>
<td>71%</td>
<td>68%</td>
</tr>
<tr>
<td>Space exploration very important to the U.S. and the world</td>
<td>71%</td>
<td>67%</td>
</tr>
<tr>
<td>Space exploration is a luxury with all the problems here on Earth</td>
<td>25%</td>
<td>29%</td>
</tr>
<tr>
<td>Benefits of space program will be more important 10 years from now*</td>
<td>72%</td>
<td></td>
</tr>
<tr>
<td>Looking back 20 years; time, effort and money to land men on the moon was worth it</td>
<td>77%</td>
<td></td>
</tr>
</tbody>
</table>

Data from Rockwell - Market Opinion Research Surveys
Date noted by * from Gordon S. Black Corporation, taken from U.S.A. Today

FIGURE 10

IMPORTANCE OF REASONS FOR SUPPORTING THE U.S. SPACE PROGRAM

<table>
<thead>
<tr>
<th>Reason</th>
<th>JULY 1983</th>
<th>FEB. 1990</th>
<th>FEB. 1992</th>
</tr>
</thead>
<tbody>
<tr>
<td>Makes possible new and important scientific and medical discoveries</td>
<td>90%</td>
<td>89%</td>
<td>92%</td>
</tr>
<tr>
<td>Provides new and improved consumer products and services</td>
<td>76%</td>
<td>76%</td>
<td>74%</td>
</tr>
<tr>
<td>Develops new technology to improve U.S. productivity and economic competitiveness</td>
<td>87%</td>
<td>87%</td>
<td>88%</td>
</tr>
<tr>
<td>Helps military defend country</td>
<td>80%</td>
<td>79%</td>
<td>80%</td>
</tr>
<tr>
<td>New frontier, important to pioneering and exploration heritage</td>
<td>82%</td>
<td>79%</td>
<td></td>
</tr>
<tr>
<td>Space leadership strengthens America's worldwide prestige</td>
<td>81%</td>
<td>69%</td>
<td></td>
</tr>
<tr>
<td>Helps us understand weather, climate, environment</td>
<td></td>
<td></td>
<td>92%</td>
</tr>
<tr>
<td>Helps interest young people in science and engineering studies</td>
<td></td>
<td>88%</td>
<td>88%</td>
</tr>
</tbody>
</table>

Data from Rockwell - Market Opinion Research and Yankelovich
FIGURE 11

U.S./NASA SPACE GOALS

<table>
<thead>
<tr>
<th>Goal</th>
<th>JULY 1988</th>
<th>FEB. 1990</th>
<th>FEB. 1992</th>
</tr>
</thead>
<tbody>
<tr>
<td>Improve understanding of climate, weather, atmosphere - start new satellite and Space Station program with international participation</td>
<td>86%</td>
<td>81%</td>
<td></td>
</tr>
<tr>
<td>Explore solar system with unmanned flights</td>
<td>82%</td>
<td>85%</td>
<td>71%</td>
</tr>
<tr>
<td>Permanent manned U.S. Space Station with international participation</td>
<td>78%</td>
<td>74%</td>
<td>65%</td>
</tr>
<tr>
<td>Back to the Moon — Base for scientific research and mining lunar materials</td>
<td>70%</td>
<td>64%</td>
<td>57%</td>
</tr>
<tr>
<td>Manned mission to Mars — Science outpost and exploration</td>
<td>66%</td>
<td>62%</td>
<td>49%</td>
</tr>
</tbody>
</table>

Data from Rockwell - Market Opinion Research and Yankelovich Surveys

FIGURE 12

ATTITUDES ON MANNED MARS MISSION

1988:
- Good idea to cooperate with Soviet Union on Mars Mission: 71%
  Yankelovich-Time Survey

1988:
- Increase NASA budget to permit manned Mars mission: 64%
  Rockwell Opinion Research

1988:
- If you favor manned Mars mission:
  Should U.S. go independently? 31%
  or equal partners with Russians? 54%
  Rockwell Opinion Research

1989:
- Where should astronauts go next?
  Permanent Space Stations? 40%
  Planet Mars? 14%
  Back to the moon? 7%
  Somewhere else? 9%
  Don't send anywhere 20%
  Gordon Black Corporation
**FIGURE 12 (continued)**

**ATTITUDES ON MANNED MARS MISSION**

continued

1989: What should be the top priority of the Space Program?

- Basic research - solar system and planets: 30%
- Zero-G and commercial technologies: 18%
- Space based defense shield: 14%
- Mining resources on Moon and planets: 23%

Gallup

1989: How important for the U.S. to be first on Mars?

Gallup

1991 How important for the U.S. to be first on Mars?

Gallup

1990: Manned missions to Moon and Mars will encourage science and engineering studies

Rockwell Opinion Research

1990: Favor President Bush's SEI missions

Rockwell Opinion Research

*38% of the people are aware; 61% are not aware of SEI proposals

**FIGURE 13**

**AMOUNT OF EFFORT ON THE SPACE PROGRAM**

<table>
<thead>
<tr>
<th>(Rockwell Supported Research)</th>
<th>JULY 1988</th>
<th>FEB. 1990</th>
<th>FEB. 1992</th>
</tr>
</thead>
<tbody>
<tr>
<td>Space program should be expanded</td>
<td>65%</td>
<td>53%</td>
<td>58%</td>
</tr>
<tr>
<td>Space program should continue as is</td>
<td>53%</td>
<td>66%</td>
<td>67%</td>
</tr>
<tr>
<td>Expenditures should be cut back</td>
<td>36%</td>
<td>40%</td>
<td>42%</td>
</tr>
<tr>
<td>U.S. should spend whatever necessary to maintain leadership in space</td>
<td>61%</td>
<td>56%</td>
<td>63%</td>
</tr>
</tbody>
</table>
FIGURE 13 (CONTINUED)

AMOUNT OF EFFORT ON THE SPACE PROGRAM

Amount of money being spent on U.S. space program should be:

<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Increased</td>
<td>26%</td>
<td>27%</td>
<td>17%</td>
<td>19%</td>
</tr>
<tr>
<td>Kept the same</td>
<td>41%</td>
<td>42%</td>
<td>37%</td>
<td>40%</td>
</tr>
<tr>
<td>Reduced/eliminated</td>
<td>24%</td>
<td>22%</td>
<td>32%</td>
<td>38%</td>
</tr>
</tbody>
</table>

Gallup Survey (Marist Inst. Survey)

Is investment in space worthwhile or better spent on domestic programs?

<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Worthwhile</td>
<td>43%</td>
<td>39%</td>
<td></td>
</tr>
<tr>
<td>Domestic programs</td>
<td>52%</td>
<td>57%</td>
<td></td>
</tr>
</tbody>
</table>

Gallup Survey

FIGURE 14

CONCLUSIONS

- Generally, favorable attitudes on space program
- Much of the comment was based on suggestions with very little open-ended, volunteered comment
- No data on using nuclear energy in space or on contributions already made by nuclear energy
- No significant, coordinated communications program exists
- No system for communicating with influentials and the public by constituents, scientists, etc.
- No actual message testing to define effective ones
- President Bush’s SEI was not grabbed, pushed, nor run with as the basis for building public and political support
- No clear long-term program laid out with clear short and intermediate term milestones as the basis for developing and demonstrating SEI technologies.
CONCLUSIONS

CONTINUED

A STRONG, EFFECTIVE COMMUNICATIONS PROGRAM IS REQUIRED TO REBUILD ENTHUSIASM FOR SPACE ACTIVITIES AND TO HOLD IT. THE BENEFITS TO THE NATION AND TO AMERICANS JUSTIFIES IT.

Let's start with one that will feed into the existing communications of various companies, associations, research organizations and government.
Rover/NERVA-Derived Near-Term Nuclear Propulsion

FY92 Final Review

at

NASA-LeRC
October 22, 1992
Agenda

- Introduction
- Reactor Concept Development
- Engine Conceptual Design
- Key Technology and Streamline Development Plan Assessment

Introduction

FY92 accomplishments centered on conceptual design and analyses for 75K, 50K, and 75K engines, with emphasis on the 50K engine, to NASA requirements.

During the first period of performance, flow and energy balances were prepared for each engine size with single and dual turbopumps. Plan, elevation, and isometric drawings were prepared for each of these configurations, and thrust-to-weight were estimated. A review of fuel technology and key data from the Rover/NERVA program, established a baseline for proven reactor performance and areas of enhancement to meet near-term goals. Studies were performed of the criticality and temperature profiles for probable fuel and moderator loadings for the three engine sizes, with a more detailed analysis of the 50K size.

During the second period of performance, analyses of the 50K engine continued. A chamber/nozzle contour was selected and heat transfer and fatigue analyses were performed for likely materials of construction. Reactor analyses were performed to determine component radiation heating rates, reactor radiation fields, water immersion poisoning requirements, temperature limits for restartability, and a tie tube thermal analysis. In addition, reactor safety and reliability were assessed.

Finally, a brief assessment of key enabling technologies was made, with a view toward identifying development issues and identification of the critical path toward achieving engine qualification within 10 years. Our initial appraisal suggests that critical path for the program will be the design, construction, and acceptance testing of engine test facilities.
Requirements

- Rover/NERVA-derived technology
- "Near-Term" man-rated mission
- 4.5 hours qualification test at rated conditions to validate 1.5 hours at rated conditions for manned missions
- Restartable, at least 10 starts
- Launch envelope, 30 m (length) x 10 m (diameter)
- $I_{sp} > 850$ seconds
- Thrust
  - A. Initially--25K, 50K, and 75K
  - B. Continued effort--50K
- Thrust/Weight (with internal shield) $\geq 4$

Requirements

Requirements for the FY92 NASA-funded effort derive from the Statement of Work. The basic objective was the assessment of the near-term feasibility of Rover/NERVA-derived nuclear thermal rocket engine technology for meeting piloted missions to Mars. The basic requirements for the engine provided by NASA included size limits, target specific impulse, number of restarts, operating life, and thrust-to-weight lower limit. Initial analyses were to be performed for three engine thrust sizes: 25K, 50K, and 75K. Final concept development was to be performed for the 50K thrust size engine.
Additional Ground Rules

- Pewee fuel element, temperature, and ZrH moderator
  - Chamber temperature 2,550 K
  - Power density 1.18 MW/element
- Tie tubes with expander cycle
- Dual turbopumps/loss of both pumps
- Nozzle expansion ratio, 200/1
- Radiation leakage limits from NERVA
- System requirements of NASA N.P. 002

Additional Ground Rules

Communication with NASA subsequent to issuance of the Statement of Work provided additional guidance:
- Pewee operating parameters for chamber temperature and power density, use of tie tubes with the expander cycle,
- Incorporation of dual turbopumps with consideration of pump outages, a nozzle expansion area ratio (200),
- Radiation limits from the NERVA design, and additional system requirements found in NASA N.P. 002, "Nuclear Thermal Rocket Engine Requirements."
NERVA-Derived 50K Engine Schematic

Chamber pressure = 764 psia
Chamber temperature = 2,550 K
Specific impulse = 870 seconds
Nozzle expansion ratio = 200:1
Nozzle bell = 110% length

The 50K engine features dual turbopumps supplying liquid hydrogen to the tie tubes, and the chamber and nozzle. Approximately 70% of the flow goes to cool the tie tubes and moderator; the heat pickup provides the energy for the turbines. The propellant flow used to cool the chamber and nozzle also cools the reflector and pressure vessel. The total flow is mixed together, flows through the fuel elements where the temperature is increased to 2,550 K, and is exhausted from the nozzle to produce thrust. The engine is sized and packaged to fit within given geometrical constraints; consequently, chamber pressure and bell nozzle length are selected to maximize specific impulse and thrust-to-weight.
Key Features/Attributes

- Proven technology, low risk approach
  - Nozzle technology flying with Space Shuttle
  - Existing turbopump designs applicable
  - Rover/NERVA-derived reactor
  - Minimum development time/money
  - Supports 10-year qualification goal
- $I_{sp} > 150$ seconds better than NERVA-XE'
- MCNP permits fuel loading for flat profile
- Tie-tube support approach facilitates
  - Expander cycle turbine for improved $I_{sp}$
  - Incorporation of ZrH to minimize reactor size
- Optimized packaging and flow balancing
- Can accept evolutionary improvements

The Rocketdyne-Westinghouse nuclear thermal rocket engine benefits from a combination of the technology proven in the Rover/NERVA program and modern rocket engine man-rated components. The goal of producing a qualified engine within 10-years can be achieved with minimal development, based on the current state of the art. Cooled chamber and nozzle technology from the SSME is directly applicable, and turbopumps from the J-2S, Rover, and SSME bracket current requirements. Studies were initiated to examine pump-out performance with boost pumps and multiple turbopumps; however, meaningful results were not achieved within the allocable funding limitations.

Easily achievable enhancements provide improvements in $I_{sp}$ over the last NERVA engine tested, NERVA-XE'. Incorporation of tie tubes and the expander cycle, increase of the expansion ratio from 10 to 200, regenerative cooling throughout, and increase of the chamber temperature to the Pewee conditions adds over 150 seconds of specific impulse. A further increase of chamber temperature to 2,700 K by use of composite elements would add another 30 seconds to bring the total to 900 seconds.

The preliminary configuration has the turbopumps at the side of the chamber to shorten the overall length of the engine assembly. Within that configuration flow and energy balances are optimized to minimize pressure which directly affects ducting wall thickness.
NERVA-Derived 50K Engine Isometric

Reactor exit temperature = 2,550 K
Dual turbopumps
Split-flow expander cycle
Nozzle expansion ratio = 200:1
Nozzle bell = 110% length
Specific impulse = 870 seconds
Thrust/weight = 5.3 (with shield)
Engine length = 7.6 m
Exit diameter = 2.4 m

NERVA-Derived 50K Engine Isometric

A key feature of the engine includes compact packaging, with turbopumps mounted to the side of the reactor vessel to reduce the overall height and permit a higher expansion within the geometrical constraints. Another feature has the tubular nozzle attaching to the chamber at a low expansion ratio to save weight and to facilitate ground testing. An area for evolutionary change in this design would be the substitution of uncooled composite ceramic materials for the tubular nozzle for a potential weight saving and some increase in specific impulse.
Technology Assessment Results

- Technology available for most issues
  
  Rover/NERVA, SSME, Rocketdyne state of the art, SP-100, terrestrial advanced reactors, state-of-the-art electronics and computers

- Unresolved system design issues
  
  Loss of turbopumps, lifetime, intact reentry—water subcriticality (or total dispersal), decay heat removal, engine-out cooling during operations, fuel midband corrosion

- Critical path is engine test facility

Technology Assessment Results

The assessment of key technologies led to the conclusions that (1) existing technology in reactors and engine systems is applicable to most design areas, (2) there are issues requiring attention early in the program to assure satisfactory resolution, and (3) the assured early availability of an engine/reactor test facility is critical to meet, successfully meet the 10-year engine qualification goal.
NTR Streamline Development Logic

A development logic diagram can include many layers of detail and be organized in many different ways. This high-level diagram shows many necessary tasks in setting requirements, recapturing technology, resolution of key design issues, facility design, construction, and activation, and testing of components and systems. The most important message is that the program must start with well-defined requirements and design criteria, and that the availability of key test facilities will drive the rate of achievement of the 10-year goals. Near-term activities of conceptual design, technology recovery, and resolution of design issues will provide a sound basis for proceeding quickly as substantial funding becomes available.
**NTR Streamline Development Plan Summary**

The time-phasing of key groups of activities from the development logic diagram shows that several tasks should be emphasized at the start: setting requirements, technology recapture, and establishing design criteria. Test facility design, construction and activation must also begin promptly to assure that the 10-year schedule can be met.

<table>
<thead>
<tr>
<th>Activity</th>
<th>Fiscal Year</th>
</tr>
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<tbody>
<tr>
<td>Establish Safety and Performance Requirements</td>
<td>'93 '94 '95 '96 '97 '98 '99 '00 '01 '02</td>
</tr>
<tr>
<td>Generate Design Criteria</td>
<td></td>
</tr>
<tr>
<td>Recapture Technology</td>
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<tr>
<td>Engine Conceptual Design</td>
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<tr>
<td>Resolve Design Issues</td>
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<tr>
<td>Design</td>
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</tr>
<tr>
<td>Design and Safety Review</td>
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<tr>
<td>Component Fabrication and Testing</td>
<td></td>
</tr>
<tr>
<td>Engine and Fuel Test Facilities Design and Activation</td>
<td></td>
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<tr>
<td>Design Verification Testing</td>
<td></td>
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<tr>
<td>Qualification Testing</td>
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</tr>
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</table>
Development of a nuclear thermal rocket design concept for Fast Track studies is based on the NERVA/Rover technology database. Design analyses to provide NTR designs to meet program requirements are developed with current design methodologies benchmarked to NERVA/Rover technology. The NERVA derivative reactor concept design is based on NERVA R-1 reactor design with design features upgraded to include the demonstrated capabilities of the NERVA/Rover program. A historical summary of the completed tests of the NERVA/Rover program and the NTR performance demonstrated by test results are summarized in the following pages.

Based on a set of NASA directives, parametric analyses of the size and performance characteristics of NTR reactors which provide performance consistent with 25K, 50K, and 75K lb engines was completed. Later discussions show the results of more detailed studies on the reactor design for the 50K lb engine.

Based on a review of the NERVA/Rover technology database, a current assessment of the fuel technology and nuclear safety issues for the application of the NERVA derivative reactor in the NTR program is discussed.

In summary, the lessons learned during the conduct of the work tasks are discussed.
NERVA/Rover REACTOR SYSTEM TEST SEQUENCE

The fast track engine draws upon the existing 1.4 billion dollar technology base developed by Los Alamos National Laboratory and Westinghouse during the NERVA/Rover Nuclear Rocket Engine Program.

The extent of the NERVA/Rover technology is demonstrated by the number of reactor and engine tests completed over the 1959-1972 time frame. The reactor tests completed in the KIWI/PHOEBUS/PEEWEE series demonstrated the wide range in reactor size and power capability provided by the technology. The NERVA test series culminating in the NRX-A6 and XE-Prime tests demonstrated lifetime and performance capabilities of the NERVA/Rover-based NTR's. The NERVA program successfully completed the preliminary design of the R-1 reactor design and the Fast Track reactor designs developed in the current work tasks are derived from the extensive technology database of the NERVA/Rover programs.
The demonstrated capabilities of NERVA/Rover based NTR's is summarized in the following table. The performance levels reached in each of the key tests completed as shown. As shown, the NERVA/Rover technology provides reactor performance capabilities similar to the requirements of the Fast Track program and later discussions show the capability of NERVA/Rover based design concepts to meet the Fast Track program needs.

<table>
<thead>
<tr>
<th>Reactor ID</th>
<th>Chamber Temp. (K)</th>
<th>Fuel Exit Temp. (K)</th>
<th>Space Equiv. ISP (sec)</th>
<th>Time at Full Power (min)</th>
<th>Fuel Type</th>
<th>MW/Thrust (kN)</th>
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<tbody>
<tr>
<td>K-19AB</td>
<td>1600-2130</td>
<td>2222</td>
<td>780</td>
<td>1</td>
<td>UC3/Graphite</td>
<td>914/204</td>
</tr>
<tr>
<td>K-19BC</td>
<td>1600-2100</td>
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<td>914/204</td>
</tr>
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</tr>
<tr>
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<td>820</td>
<td>16.3</td>
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<td>1100/245</td>
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</tr>
<tr>
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<td>834</td>
<td>18.5</td>
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<td>1340/298</td>
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<td>28.5</td>
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<td>1100/245</td>
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</tr>
<tr>
<td>PHOEBUS-2A</td>
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<td>820</td>
<td>30.0</td>
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<tr>
<td>PEK-1</td>
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<td>62.7</td>
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<tr>
<td>PRIMES</td>
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<td>500/111</td>
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<tr>
<td>NF-1</td>
<td>--</td>
<td>830</td>
<td>109</td>
<td>Composite/Carbon</td>
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<tr>
<td>PHOEBUS-2A</td>
<td>2560</td>
<td>805</td>
<td>12.3</td>
<td>UC3/Graphite</td>
<td>4100/913</td>
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</tr>
<tr>
<td>TESTED</td>
<td>2500</td>
<td>840</td>
<td>805</td>
<td>UC3/Graphite</td>
<td>8000/1113</td>
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</tr>
</tbody>
</table>

NP-TIM-92 79 NTP: System Concepts
NERVA Derivative Reactor Concept Design
for 50K lbf Thrust Engine

- Layout drawing
- Solid models
The 50K lb, engine reactor layout is based on the R-1 NERVA flight reactor design. The R-1 successfully completed an Air Force Preliminary Design Review before the termination of the NERVA project in 1972. The key dimensions of the reactor for the 50 k lb engine are shown. These were established based on the required engine thrust (core size), and the neutronic requirements (reflector and shield).
Solid models of the major reactor components and assembly were generated using Pro-Engineer. The graphics show the assembly model of the reactor, where the major components, including the core, can be seen. Since the solid model is parametric in nature, it can be used for a series of reactor sizes of the same type. This makes trade studies easy to perform and weight estimates for these types of reactors can be established fast and accurately. The second major reason for utilizing the Pro-Engineer solid modeling approach is that it provides a seamless interface to analytical tools. This integrated approach to design and modeling will be fully utilized in the development of the HI-19A advanced reactor design.
NTR Nuclear Parameter Study

- Neutronics Model
- 25K lbf Engine Results
- 50K lbf Engine Results
- 75K lbf Engine Results
- Heterogeneity Evaluation

Studies of the neutronics design of the NTR were based on three dimensional models derived from the NERVA design. The methodology selected for use in the parametric analyses was the MCNP Monte Carlo radiation transport method. Model parameters of the reactor system were derived from the R-Z model information of the NERVA R-1 reactor system. An automated model generation technique was used to define reactor system models for parametric analyses to size and predict performance characteristics for the various sizes of the NTR system. An R-Z annular ring model of the NTR core configuration was used in parametric analyses in a similar manner to the models in the NERVA database. Three dimensional model details were limited to the reflector control drums and used the geometric modelling capability of the MCNP method. The automated modelling technique and MCNP (Version 3B) were used to define the core and reflector sizes, fuel loading profiles, reactivity worths, and control drum worths and span for three NTR engine sizes; 25, 50, and 75 Kbf thrust levels. In addition, a limited study of the impact of heterogeneous versus homogeneous modelling of the prismatic fuel elements and REF-tubes within the NTR core was performed on a unit cell basis.
Study Guidelines

- NERVA (Prismatic) Fuel Elements
  52" Long
  0.753" Hex
  19 Coolant Channels
  600 mg/cm³ Maximum Fuel Loading

- ZrHx Moderated Tie-Tube
  SNRC (PeeWee) Maximum ZrHx
  2:1, 3:1, 6:1 Fuel Element to Tie-Tube Ratios

- Performance
  1.18 Mw/element
  2550K Chamber Temperature (Point Design)
  784 psia Chamber Pressure

---

STUDY GUIDELINES/ASSUMPTIONS

- Reactor Sizes 25K, 50K and 75K lbf Thrust Engines
- Critical Drum Angle of 80°
- NERVA/R-1 Reactor Design Configuration
- R-Z Geometry with Explicit Control Drums
- Neutronics Calculations: MCNP-3B
## Technology Base

<table>
<thead>
<tr>
<th>Design Features</th>
<th>Reactors</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>NERVA</td>
</tr>
<tr>
<td>Tie-Tubes</td>
<td>Xe-Prime</td>
</tr>
<tr>
<td>Ratios, Fe/TT</td>
<td>No</td>
</tr>
<tr>
<td>ZrH Loading (Relative)</td>
<td>0.0</td>
</tr>
<tr>
<td>Power, Mw</td>
<td></td>
</tr>
<tr>
<td>Core Diameter, in.</td>
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</tr>
<tr>
<td>Power Density, MW/FE</td>
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</tr>
<tr>
<td>Internal Shield</td>
<td>A1</td>
</tr>
<tr>
<td>Fuel Type</td>
<td>Graphite</td>
</tr>
<tr>
<td>*Not Determined</td>
<td></td>
</tr>
</tbody>
</table>

*Not Determined*
The neutronics model for the NTR system was derived from modelling information in the NERVA database and is shown as an elevation view to illustrate the modelling detail of MCNP models. The MCNP analyses used the ENDF/B V nuclear data library and were performed in the coupled neutron and photon solution mode to predict region power and required fuel loading to meet target objectives for key neutronics parameters. An actual NERVA system design configuration drawing is depicted to illustrate the modelling approach used.
The MCNP model for the 25 Klbf NTR engine and the predicted key parameters are shown in the table on the right. The design bases selected for the small NTR engine size were derived from PEWEE engine design information with a fuel-to-support tie tube ratio of 3:1, a 52 inch high active core, and 9 control drums of a fixed diameter located at the outer periphery of the Be reflector region. The peak fuel element uranium loading was limited to 600 milligrams/cm$^2$ and a maximum ZH1 loading in the tie-tubes. An iterative process based on MCNP was used to size the reactor core and predict the fuel loading profile to meet the target objectives of an excess reactivity of 0.05 and a flat radial power distribution.
A normalized fuel loading profile predicted for the 25 Klbf NTR engine is shown as a function of normalized area parameter, $R^2$. The normalized radial power distribution as predicted in the final iteration of the analysis is shown to illustrate convergence to the target objective of a flat or uniform power profile. The MCH tally method provides the cell or ring average value and more detailed tallying techniques would be required to predict the variation within each fuel annulus.
Predicted neutronics parameters for a 50 Klb thrust NTR engine are shown in the table. Key differences in the design bases selected for this size of engine were a fuel-to-support tie-tube ratio of 6:1 and reflector thickness and number of control drums. The effective core diameter required to meet target objectives is 25.18 inches.
The normalized fuel loading profile predicted for the 50 Klbf NTR engine is shown as a function of normalized area parameter, $R^2$. As shown, the fuel loading profile differs from the 25 Klbf engine data due to the larger size and the change to a 6:1 fuel-support tie-tube ratio. The lower fuel loading required in the center of the core is related to the change in the moderation of the core and the increase in median fission energy and the effect of radial leakage.
Predicted neutronics parameters for a 75 Klbf NTR engine are shown in the table on the right. The 75 Klbf engine size is similar to the NRX-A6 or R-1 size and the predicted parameters are comparable to the NERVA data. Key differences in the design bases selected for this size of engine were a decrease in the ZrH loading in the support tubes of 0.4 with respect to the SNRE loading. The reflector thickness and number of control drums for the 75K engine are the R-1 dimensions. The effective core diameter required to meet target objectives is 30.66 inches which is similar to the NERVA design.
The normalized fuel loading profile predicted for the 75 Klbf NTR engine is shown as a function of normalized area parameter, \( R^2 \). As shown, the fuel loading profile is similar to the 50K engine data and is comparable to NERVA loading profiles.
A summary of the results of the preliminary sizing of NTR engines in the 25Kbf-to-75Kbf size range is shown in the table. The design bases used in the parametric analyses are listed on the left. The prismatic fuel element length of 52 inches was adapted from NERVA and fuel performance limits defined based on the PEEDWEE data. The predicted masses for the reactor system without and without shielding illustrate the effect of engine size on the engine performance characteristics and sizes. The use of ZrH in the 75K engine size differs from the NERVA design and the impact on a reduced reactor size and mass is shown. The shield masses included in the summary table are based on the same thickness of shield with the mass differences only showing the change in shield diameter.
A limited study of the homogeneous region modeling technique for the prismatic fuel element core lattice with ZrH moderated support tie-tubes was carried out using the MCNP method. A unit cell model of a 6:1 fuel-to-support tie-tube configuration includes an annular model of the ZrH moderated tie-tube and the 19 coolant hole prismatic fuel element. A series of unit cell MCNP calculations were run to predict the effect of the ZrH tie-tube on local power distributions and to predict material or material interchange reactivity worths on a unit cell basis.
Comparisons of the effect of heterogeneous versus homogeneous modelling on the power distribution in the prismatic fuel assembly is shown in the left figure. The homogeneous model in a unit cell was derived by volume weighting of the prismatic fuel element, tie-tube materials, and hydrogen coolant of the tie-tube and fuel element. The comparison shows a peak to average local channel power of 9-10% for the explicit model of the unit cell. The smear modelling of each fuel element or tie-tube provides similar peak-to-average values. Shown in the right figure is the effect of a decrease in ZrH volume fraction or the introduction of cold (50K) hydrogen in the upward pass of the tie-tube. The maximum effect on local power occurs when the ZrH tie-tube is flooded with H_2 coolant at 50K.
ANALYSES OF 50K ENGINE DESIGN CONCEPT

- Reactivity Coefficients
- Component Nuclear Heating Rates
- Reactor Radiation Fields
  - Shielded (R-1)
  - Unshielded
  - Reduced Shield
- Material Temperature Limit Assessment
- Tie-Tube Thermal Analysis

Neutronics analyses of the NTR 50K engine configuration defined earlier were expanded to provide more detailed core performance data. The limited analyses were performed with a more detailed MCNP model to predict the design data for key design parameters as listed on the facing page. Included in the more detailed analyses was: 1) the prediction of reflector control drum worths and span, and 2) reactivity change due to water immersion of the nuclear system. In addition, component nuclear heating rates and radiation fields external to the reactor system are predicted and shown in later pages.

In addition, evaluations of the component temperature limits needed for restartability studies and analysis of tie-tube thermal performance are shown in later pages.
The predicted reflector control drum reactivity relative to the critical condition is shown on the facing page. The results of the individual MCNP calculations with the explicit modelling of the control drums in MCNP method provide results in agreement with NERVA predictions and illustrate the drum span available for control and shutdown of the 50K engine.
# Reactivity Coefficients

Case: 50 Klb, Thrust Engine

<table>
<thead>
<tr>
<th>PARAMETER CHANGED</th>
<th>REACTIVITY CHANGED</th>
</tr>
</thead>
<tbody>
<tr>
<td>Drum Worth (@ 80°)</td>
<td>7.3°C Rotation</td>
</tr>
<tr>
<td>Core Volume</td>
<td>38.9°C/%</td>
</tr>
<tr>
<td>Fuel Loading</td>
<td>15.6°C/%</td>
</tr>
<tr>
<td>ZrH Loading</td>
<td>19.2°C/%</td>
</tr>
<tr>
<td>Reflector Thickness</td>
<td>18.7°C/%</td>
</tr>
<tr>
<td>18 Drums (7.34$ $ span vs. 5.83$ $ for 12 Drums)</td>
<td>-$2.1</td>
</tr>
</tbody>
</table>

The predicted reactivity coefficients or worths for key design parameters are listed on the facing table. The predicted drum worth is based on the 80 degree position. The value of 7.3 cents/degree is in close agreement with the NERVA predicted value. Reactivity coefficients for changes in the reactor configuration, fuel loading, ZrH loading in the tie-tubes, and reflector thickness provide data for evaluating design configuration changes. The largest value is the core volume coefficient which is attributed to the change in neutron leakage from the core. Shown also is the effect of changing the number of reflector control drums from 12 to 18 drums.
Predictions of the effect of water immersion of the entire reactor system was modelled in MCNP by replacing the H₂ coolant modelled in each region with water and surrounding the entire system with water. The reflector control drums were parked and a boron-containing material was substituted for a fraction of the fuel element coolant channel volume. The reactivity change from the base case is shown as a function of the volume percent of coolant channel displaced by the boron-containing material. A value of five (5) percent by volume of the coolant channel is a 62 mil boron wire in 7 out of 19 coolant channels in each prismatic fuel element of the core. The reactivity insertion provided by the 5% by volume of boron wires is approximately -74$ with the water immersion of the system resulting in a +50$ reactivity insertion.
A summary of the nuclear heating of the major components of the 50K engine is shown on the facing page. The MCNP cell tally method was used to predict the component heating rates.
The prediction of the radiation environment external to the S5K engine were performed using the MCNP cell tally methods. Three engine models were analyzed: 1) the conceptual design sized using MCNP in the neutronics design tasks described earlier, 2) all internal shield materials removed, and 3) a modified design with a reduced mass of internal shielding. Each of these models only include the reactor system and the engine components external to the reactor vessel, e.g., tanks, piping, nozzle, are not included in the model. The engine components external to the reactor vessel can contribute to the environment within the internal shield shadow cone and should be included in future studies. The MCNP modeling used a series of annular ring cells imposed external to the MCNP R-Z model of the NTR reactor system for purposes of tallying the desired radiation environments. The facing page summarizes the type of radiation field tallies used in MCNP and either the neutron energy range of the neutron flux tally or the units of heating or neutron or gamma dose rates.
The radiation environment of the original 50K engine design is shown on the next two facing pages for three key tallies. The first 50K engine design used for this analysis included the standard NERVA R-1 Internal shield configuration of 12.3 inches (31.25 cm) of BATH shield material and 1.3 inches (3.3 cm) of lead (Pb) shielding. The second page is for an engine design with the internal shields removed. The predicted radiation environments for the shielded case are lower than the design requirements.
RADIATION FIELDS FOR A 50K UNSHIELDED REACTOR

Log Gamma Rad(C)/hr

Log Fast Neutron Flux n/cm²-sec

Zero Added Shielding

No Bath or Lead

No Shield Support Plates
RADIATION FIELDS FOR A 50K REACTOR
WITH A REDUCED INTERNAL SHIELD

- Radiation Field Criteria (in Shield Shadow Cone)
  - Gamma Dose < $1.8 \times 10^3$ Rad($\Gamma$)/hr
  - Fast Neutron Flux < $2.0 \times 10^{12}$ n/cm$^2$-sec
  - Intermediate Neutron Flux < $3.0 \times 10^{12}$ n/cm$^2$-sec
  - Thermal Neutron Flux < $6.0 \times 10^{11}$ n/cm$^2$-sec

- Reduced Shield Concept:
  - Eliminates Lead Gamma Shield
  - Reduces BATH Thickness from 12.3" to 9"

- Reduced Shield Performance ~ 900 lb. Reactor Weight over Standard Shield
  - 900 lb Mass Savings versus R-1 Type
  - Meets Above Criteria (Design Margin > 2.0)

- Per NASA Directive

Based on the design requirements imposed on the internal shield design of the NTR engine, a reduced internal shield with nine (9) inches of BATH shield material and no lead (Pb) shielding was modeled and the resulting radiation environments compared to the standard design described earlier. The facing page lists the radiation field design requirements specified for the NTR engine. The reduced shielding configuration meets design requirements with a design margin in the shadow cone of the internal shield of a factor of 2. The design change results in a reduction in shield mass of approximately 900 pounds.
The contour plots on the facing page provide data on the performance of the modified shield configuration for the 50K engine relative to the design requirements. The contour data is the ratio of the predicted radiation environment level to the design requirement discussed before. As shown by the data, the reduced shield configuration meets the design requirements within the shadow cone of the internal shield. The design margin in the shadow cone is a factor of 2 or greater in the shadow cone. As discussed before the mass savings of the reduced internal shield design is 900 pounds.
# MATERIAL TEMPERATURE LIMITS

<table>
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<th>REGION</th>
<th>MATERIAL</th>
<th>REUSE TEMP (K)</th>
<th>ALTERNATE MATERIAL**</th>
<th>TEMP (K)</th>
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<tr>
<td>Fuel Element</td>
<td>Graphite</td>
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<tr>
<td>Other Core Materials</td>
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</tr>
<tr>
<td></td>
<td>I-718</td>
<td>900</td>
<td>HD-Moly</td>
<td>2000</td>
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<td>Superalloys</td>
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<td></td>
<td>SS-304</td>
<td>750</td>
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<tr>
<td></td>
<td>Be</td>
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<td>Vessel Materials</td>
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<td></td>
<td>Ti</td>
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</tr>
<tr>
<td>Shield Materials</td>
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<td>Tungsten</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Lead</td>
<td>-550</td>
<td></td>
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</tbody>
</table>

*Must be pressurized with hydrogen (> 10 TORR)

**No materials identified which provide a capability without significant mass, performance or design penalty
The tie tube assembly serves two purposes: provides the lateral support for the fuel elements, and heats hydrogen propellant used to drive the turbopump.

The thermal analysis of the tie tube assembly was performed to establish the adequacy of the design in terms of component temperature and to determine the energy transferred to the hydrogen. The thermal model used for the analysis employed axially dependent heat generation and boundary temperature conditions, temperature and flow dependent hydrogen heat transfer coefficient, and temperature dependent material properties. The thermal model will be used to perform parametric steady-state analysis, as well as transient analysis of throttling conditions.

The radial temperature distribution at three locations (top, middle, and bottom) of the tube assembly is shown on the facing chart for full power conditions.

The temperatures of the ZrH are critical since it has the lowest temperature capability of the materials used in the tie tube assembly. As shown, the maximum calculated temperature for the conditions used exceed 1000 K by a small amount at an internal node in the ZrH cylinder. The calculated heat transferred to the tie tube is 0.18 MW.

The thermal model has been verified against the small engine in the Nuclear Engine Definition Study. The analysis demonstrates that the thermal conductivity of the ZrC insulation is the largest factor in achieving the goal of 0.31 MW per tie tube.
ASSESSMENT OF FUEL TECHNOLOGY

- Review of ROVER/NERVA Test Experience
- Evaluate the Corrosion Mechanisms Affecting Fuel Performance
- Define Problem Areas Needing Near-Term Solution
- Establish Near-Term Fuel Performance Limits
- Compare Near-Term Performance to Fast Track Needs

An assessment of the NERVA/Rover fuel technology of 1972 is needed to establish expected performance parameters for the Fast Track engine. The fuel life for the Nerva graphite type prismatic fuel element is determined by the amount of graphite weight loss which can be tolerated before the neutronic margin has been lost. The weight loss from the fuel element is due to the corrosive effect of hydrogen on the graphite, which is categorized as either "mid-band corrosion," basically results in a chemical reaction of hydrogen and carbon in intimate contact, or "hot end corrosion," carbon diffusion through a protective coating on the graphite surface.

Great strides were made near the end of the NERVA/Rover program in understanding and eliminating the mid-band corrosion, and it is a basic premise that this corrosion mechanism be suppressed in order to support the needs of the Fast Track program.

Based on the reactor-engine testing program, and the non-nuclear corrosion testing of fuel elements using the improved GEM coatings, the performance limits of "near term" fuel elements were established. The expected fuel element performance was then compared against the needs of the Fast Track program.
Fuel Elements

- Sustain controlled nuclear heat generation
  
  Pyrocarbon coated UC\textsubscript{2} fuel beads dispersed through AXM graphite matrix (630 mg/cc maximum fuel loading)
  
  UC\textsubscript{2}-ZrC in composite with graphite
  
  (700 mg/cc maximum fuel loading)

- Limit total reactivity loss to $1.00$ at end of life

  Carbide coating of flow channels

- Promote heat transfer from fuel element to H\textsubscript{2} propellant

  19 flow channels in each 3/4 in. HEX
  
  52 in. long fuel element

The NERVA/Rover prismatic graphite fuel element is 0.75 inch across the flats, and 52 inches long. It contains 19 flow holes (approximately 0.1 inch in diameter). All graphite surfaces have a protective ZrC or NbC layer to protect it from the hydrogen.

UC\textsubscript{2} fuel beads coated with pyrocarbon are dispersed through the matrix at a maximum fuel loading of 930 mg/cc. For the more recent composite type fuel element a maximum fuel loading of 700 mg/cc is achievable.

Nuclear design of the NERVA reactor limits the reactivity loss to approximately $1\dollar$ at the end of fuel life. Since the reactivity loss is mostly a result of loss of carbon due to the hydrogen corrosion, protective coatings are used to reduce the rate of carbon loss.
For all the NRX reactor and engine tests, the graphite-type fuel was used. However, toward the end of the NERVA/Rover program composite fuel emerged as the most promising candidate in reducing the hydrogen corrosion and in increasing the temperature capability of the prismatic fuel element.

The composite fuel element consisted for a dispersion of UC-ZrC web in the graphite substrate. Since this web is continuous, and essentially unaffected by hydrogen, it acts as a barrier and limits the carbon loss from the fuel.
Major Milestones in Fuel Development

- Graphite Fuel Element/HED NbC Coating (NRX-A2/A5)
- Graphite Fuel Element/HED NbC + Molybdenum Coatings (NRX-A6/XE)
- Graphite Fuel Element/GEM NbC/ZrC Coatings (PEWEE)
- High CTE Graphite Composite Fuel Element/GEM ZrC Coating (Nuclear Furnace -1)
- Carbide Fuel Element (Nuclear Furnace -1)

The standard graphite fuel element with a HED NbC coating was used on NRX-2A/5A reactor series. The HED coating process resulted in a coating with a significant number of cracks, which seemed to have an adverse effect on the mid-band corrosion protection. In order to improve the mid-band corrosion performance of these elements, a molybdenum overcoat was applied to the fuel for NRX-A6/XE prime reactors.

The next improvement in the coating technology came with the lower temperature coating process, GEM, whereby ZrC or NbC coating could be applied without cracks in the coating. Fuel elements with this coating process were run in Pewee, but resulted in significant mid-band corrosion.

The fuel elements for the Nuclear Furnace-1 (NF-1) were of the high CTE graphite composite type with GEM ZrC coating, which were predicted to have eliminated the mid-band corrosion based on non-nuclear corrosion testing. Pure (U,Zr)C fuel elements were also tested in the nuclear furnace. These were manufactured as small hexagonal rods with a single cooling channel in the center. The carbide fuel elements were projected to have very low corrosion rates and very much higher temperature capability than both the graphite and the composite fuel elements.
As a result of the improvements in the corrosion resistance of the fuel elements, the NERVA/Rover reactor tests showed a gradual increase in temperature capability and time at maximum temperature. The fuel life is dependent on the weight loss for the elements, and the resulting reactivity loss. Based on a reactivity margin of 1$ for corrosion from the fuel, the NERVA/Rover fuel life corresponds to a 15 to 20 g fuel element weight loss.
The corrosion behavior along the length of the fuel element showed two different characteristics. From an axial position of 200mm to approximately 650mm from the cold end, an enhanced corrosion (called the mid-band corrosion) dominated. The temperature regime for this mechanism is 1000 to 2000 K, significantly below the maximum fuel temperature. In the progression of coating and fuel element improvements, there seemed to be negligible improvement in mid-band corrosion except for the demonstrated benefit of the molybdenum overcoat. From approximately 650mm to the hot end of the fuel element (called the hot end corrosion), the corrosion rate seemed to temperature related, and a significant decrease in the corrosion rate was observed as the coatings were improved. Electrically heated fuel element corrosion tests performed after the NF-1 testing demonstrated further improvements in the hot end corrosion rate, including a 10-hour life of a fuel element demonstrated by Westinghouse.
Key Reference Points For Fuel Experience

The most successful graphite fuel elements were those tested in NRX-AS, which were also used in NRX-XE prime engine configuration. These fuel elements utilized the HE0 NbC coating with molybdenum overcoat, and demonstrated a significant reduction in the mid-band corrosion compared to earlier NRX series tests.

The alternative fuel element technology is the composite, which was tested in NF-1. These elements, which were called the "replacement elements," were high CTE graphite coated with a superior ZrC coating (free of initial cracks) applied by GEM process.

The weight loss results for the A-6 and the XE prime fuels indicate that the A-6 vintage fuel has a significant sensitivity to thermal cycling. The NF-1 composite fuel elements demonstrated better hot end corrosion than the A-6 graphite fuel; however, a surprising degree of mid-band corrosion was still present.

<table>
<thead>
<tr>
<th>REACTOR TEST</th>
<th>TEMP FUEL EXIT (K)</th>
<th>TIME (MIN)</th>
<th>CYCLES</th>
<th>TOTAL LOSS (G)</th>
<th>MIDBAND (G)</th>
<th>HOT END (G)</th>
</tr>
</thead>
<tbody>
<tr>
<td>NF-1 *</td>
<td>2444</td>
<td>108.8</td>
<td>4</td>
<td>13.7</td>
<td>8.5</td>
<td>5.1</td>
</tr>
<tr>
<td>NRX-AS **</td>
<td>2556</td>
<td>62.7</td>
<td>1</td>
<td>12.8</td>
<td>2.3</td>
<td>10.5</td>
</tr>
<tr>
<td>NRX-XE **</td>
<td>~2450</td>
<td>10.3</td>
<td>28</td>
<td>7.3</td>
<td>0.6</td>
<td>6.7</td>
</tr>
</tbody>
</table>

* Replacement composite fuel elements with crack free ZrC coating (GEM)
** Graphite fuel elements with NbC coating and molybdenum overcoat
Mid-band corrosion did not occur in the electrical testing of the composite fuel elements for NF-1, but caused the most significant weight loss during the reactor testing. Mid-band corrosion is believed to be a result of decreased thermal conductivity, possibly caused by fission fragment damage to the graphite matrix. The reduced thermal conductivity results in higher thermal gradients and increased thermal stresses, which causes cracking of the protective coatings and allows hydrogen to react with the graphite substrate. Mid-band corrosion must be fully understood and suppressed to meet performance requirements of the Fast Track program. Use of a molybdenum overcoat on composite fuel elements, or improved fuel particle coating in the graphite fuel to trap the fission fragments, are potential design solutions to mid-band corrosion.
Composite fuel element testing provided a good correlation between the hot end corrosion measured in electrical testing and that observed in the NF-1. Hot end corrosion is caused by carbon diffusion through a protective coating and, therefore, is sensitive to the integrity of the coating, the coating thickness, and the temperature of the coating and fuel substrate.
Based on the assumption that the mid-band corrosion will be suppressed in near-term fuel elements, and the hot end corrosion rates measured in electrical testing and NF-1 testing, performance limits for near-term composite fuel can be calculated. Similar performance data can also be generated for the NRX-A6 type graphite fuel.

Comparing the projected near-term graphite fuel performance NRX-6A type with the composite fuel (NF-1 type) shows a 100-120 K temperature advantage for the composite fuel.

The improved performance of composite fuel is attributed to either the projected improvements in corrosion due to the composite fuel form or improved coatings used for NF-1 fuel elements. The improved coatings of NF-1 fuel elements are considered the most likely contributor to improved fuel performance.
Summary and Conclusions

- 1972 Fuel Technology was making progress toward meeting life/performance specifications consistent with current Fast Track requirements
  - understanding of midband corrosion was being developed
  - excellent hot end corrosion protection (ZrC on high CTE graphite) was demonstrated
- Corrosion limit for fuel elements was established based on $1$ reactivity loss
  - for NERVA type reactors, this translates into 15 to 20 grams corrosion loss per element
- Near term fuel development must resolve midband corrosion problem
  - fission fragment damage to graphite may be reduced by beaded fuel in graphite and composite matrix
  - molybdenum overcoat may suppress midband corrosion
  - improved graphite matrix may reduce or eliminate problem
- Near term composite fuel will have 4.5 hours life at 2470K to 2520K fuel outlet temperature
  - near term graphite fuel based on GEM ZrC/high CTE graphite is expected to perform similarly to near term composite fuel
- Near term fuel elements are expected to provide ISP ~850 seconds
Assessment of Nuclear Safety Issues

- Nuclear Safety Policy Working Group (NSPWG) Recommendations
- Accidental Criticality Sources for NERVA Derivative Reactor Design
- NERVA Safety Approach
- SP-100 Safety Approach
- NERVA Derivative Safety Approach

An assessment of the nuclear safety issues for a nuclear thermal propulsion system must be made based on the current regulatory guidelines, and the recommendation from the Nuclear Safety Policy Group (NSPWG). Starting with the accidental criticality sources for the NERWA derivative reactor design, the safety approach developed for the NERVA flight engine and the current SP-100 reactor safety approach, and the planned NERVA derivative safety approach will be discussed.
Assessment of Nuclear Safety Issues
From NSPWG & NP002 Safety Recommendations

- No inadvertent reactor startup
  - Zero power testing on ground
  - Startup after achieving planned orbit

- No inadvertent criticality
  - Subcritical under all credible accident conditions
  - Highly reliable control system

- No significant radiological release or exposure
  - Only zero power testing prior to achieving planned orbit
  - Insignificant impact on Earth and space environment
  - Spacecraft not rendered unusable when crew survives accident
  - Radiological release not impair use of spacecraft

Assessment of Nuclear Safety Issues
NSPWG Safety Recommendations (Cont'd)

- Minimize probability of inadvertent reentry
- Minimize consequences of inadvertent reentry
  - (high alt. disposal or intact reentry)
  - Subcritical at all times
- Minimize impact dispersion

- Minimize hazardous materials release

- Ensure safe disposal
  - Part of mission planning
  - Adequate and reliable cooling, control and protection
  - Ensure non-premature final shutdown

- Safeguard nuclear material
  - Positive measures to prevent theft, diversion, loss or sabotage
  - Features to enhance safeguards and permit proven methods to be employed
  - Positive measures for recovery including listen and tracking

The NSPWG recommendations for safety requirements and guidelines addresses the protection of the public, the crew, the environment (both Earth and space environment), and includes recommendations for the safe disposal of the spent reactor system.
Assessment of Nuclear Safety Issues

Accidental Criticality Sources and Potential Countermeasures

<table>
<thead>
<tr>
<th>Source</th>
<th>Maximum Reactivity Insertion</th>
</tr>
</thead>
<tbody>
<tr>
<td>Core Compaction</td>
<td>-6 (80% Theoretical Density)</td>
</tr>
<tr>
<td>External Neutron Reflection</td>
<td>-3</td>
</tr>
<tr>
<td>Control Drum Roll-Out</td>
<td>$3 ($4.50 Drum Span)</td>
</tr>
<tr>
<td>Hydrogen Insertion</td>
<td>-$85</td>
</tr>
<tr>
<td>Water Immersion</td>
<td>-$76</td>
</tr>
</tbody>
</table>

| Potential Countermeasures:     | Negative Reactivity Worth     |
| Central + Peripheral Poison Wires | -$10                       |
| Central Poison Wires Only      | -$100                        |
| Control Drums *Locked* Full-in | $1.50 at Ambient Temperature |
| Safety rods                    | -$90                         |

*Limiting reactivity addition if the core could be completely flooded with high density LH₂.

The sources for accidental criticality of a NERVA derivative reactor are core compaction, external neutron reflection (from water or soil), control drum rollout, hydrogen flooding, and water immersion. The countermeasures for reactivity events must assure a subcritical condition with a negative reactivity margin of $3.

For the NERVA reactors the criticality margin was assured using poison wires (7 for each fuel element). Other reactivity control means for NERVA-type reactors are the control drums or the possible introduction of safety rods within the core.
Assessment of Nuclear Safety Issues

NERVA Safety Approach

- Poison wires in core after assembly for shipping
  - Boron/aluminum wires/elements
  - Wires would be removed before launch
- Redundant safety features to preclude drum roll out
  - Permanent magnet stepping motor used in control drum drive actuator
  - Drum roll out requires erroneous command signal and closing electrical power circuit
- Anticriticality Destruct System (ACDS)
  - To fracture reactor by use of explosives
  - No more component greater than 3 fuel element
- Prevention of hydrogen insertion
  - Closing PFS valves when flooding is detected within 300 seconds of full leakage

For the NERVA reactors the poison wires were primarily used to maintain the fully assembled and fueled reactor in a safe condition during transportation from the assembly area in Largo, Pennsylvania, to NRTS. For a flight reactor the poison wires were to be removed prior to launch. Redundant safety features were used to preclude drum roll-out prior to the planned reactor startup in a safe orbit. To preclude criticality events for a launch accident or an inadvertent reentry event, an Anticriticality Destruct System (ACDS) would be used to break up the core.

Hydrogen flooding of the core was precluded using redundant valves and hydrogen sensors.
Assessment of Nuclear Safety Issues
SP-100 Safety Approach

- Two redundant shutdown systems
  - Moveable reflector segments
  - Multiple safety rods
  
  Only planned use for ultimate shutdown

- Rhenium liner at core periphery to absorb thermal neutrons
  - Water immersion

- Inadvertent reentry and Earth Impact
  - Reactor remains intact and subcritical

The SP-100 space power reactor system (SPRS) has been subjected to more extensive safety evaluations based on current guidelines. The decisions made and planning for the SP-100 SPRS will most probably apply to the NTR.

The SP-100 safety approach employs two redundant systems, moveable reflectors, and safety rods. The safety rods are designed to provide for permanent shutdown of the SP-100 reactor system after the completed mission. However, the safety rod design allows for the retraction of the rods from an unplanned insertion.

In addition to the moveable reflectors and safety rods, the SP-100 reactor includes a rhenium liner internal to the reactor vessel to capture neutrons thermalized external to the vessel and precludes back reflection from a water or earth immersion event. The SP-100 safety approach includes reactor system design features to assure an intact inadvertent reentry and earth impact event.
Assessment of Nuclear Safety Issues
Safety Approach for NERVA Derivative Reactor

- Preliminary safety evaluations have been initiated
- Current safety guidelines appears to require dual shutdown systems
  - Control drums for normal operation
  - Safety rods for ultimate shutdown
- As part of the safety study, the design team is evaluating
  - Retractable safety rods
  - Neutron absorption at core periphery for Earth and water immersion
  - Impact of intact reentry

There has been no in-depth safety evaluation of the NERVA derivative reactor system completed to date. However, it is expected that the results of such an evaluation will be similar to SP-100 safety approach adapted to the reactor design. Based on the current safety guidelines, incorporation of dual or redundant safety shutdown systems will be needed to meet today's requirements.

As part of an ongoing safety evaluation for the NERVA derivative system, Westinghouse will evaluate the use of retractable safety rods in the core and neutron absorbing liners at the core periphery to achieve the current safety guidelines. The design impact of an intact reentry will be evaluated for the reactor design.
Reactor Development
Summary and Conclusions

- Engineering and analysis of NERVA derivative reactors were successfully benchmarked against the NERVA/Rover test reactors for:
  - Reactor size and neutronics performance
  - Design characteristics such as fuel loading, ZrH moderator requirements, and control drum span
  - Internal shielding performance
  - Thermal performance of tie tubes

- A reactor conceptual design for the NASA 50K fast track engine was validated neutronically and thermally by analysis.

- The 1972 NERVA/Rover fuels technology must be recaptured and demonstrated.

- Near term fuel technology must resolve the mid-band corrosion problem.

- Near term fuel technology will meet fast track requirements.

- NERVA/Rover safety shutdown systems appears inadequate for today's requirements.
  - A secondary shutdown system will be developed for the NERVA derivative reactor designs.

In this project we performed trade-off studies, developed a single point conceptual reactor design, and validated this design thermally and neutronically.

The engineering and analysis supporting the trade-off studies and the point design were successully benchmarked against the NERVA/Rover reactor designs. The reactor size and neutronic performance was established for a range of reactor sizes for engines producing 25 K to 75 K lbs thrust. The design characteristics such as fuel loading requirements and radial loading profile, ZrH moderator requirements, control drum worths and control span, were established.

A reactor concept for the 50 K lbf engine was developed and validated neutronically and thermally by analysis.

Internal shielding performance was established for the standard R-1 shield configuration, an unshielded, and for a reduced shield reactor.

The tie tube thermal performance was modelled, and evaluated for steady state conditions. Trade studies will establish the range of ZrC insulation properties and thermal transient performance.

The fuel technology of 1972 (the end of the NERVA program) was evaluated. This technology must be recovered and demonstrated as a baseline. Further, this technology must be advanced by eliminating or suppressing the midband corrosion problem to meet the fuel life requirements for the proposed missions. This "near term" fuel technology will meet the needs of the fast track program.

Reviewing the current requirements and recommendations for nuclear safety for the NTR, and the approach taken by other space power reactor systems, leads to the conclusion that the NERVA/Rover safety approach must be upgraded. Current plans are to evaluate a secondary shutdown system for the NERVA derivative reactor design.
50K PEWEE-DERIVED
DUAL TURBOPUMP ENGINE

\[
T_s = 2550 \text{ K}
\]

SPLIT-FLOW
EXPANDER CYCLE

\[e = 200:1\]

\[G_L = 110\%\]

\[I_t = 870 \text{ SEC}\]

\[\text{T/We} = 5.3\]

(W-SHIELD)

\[L_s = 7.66 \text{ M}\]

\[D_s = 2.44 \text{ M}\]

50K PEWEE-DERIVED
DUAL TURBOPUMP ENGINE

The 50Klb-thrust engine is based on Rover/NERVA reactor core technology. Average fuel
element exit gas temperature and core power density milestones were established by the
Pewee reactor tested at NTS to a power level of 500 MW in December 1968. The average
fuel element exit gas temperature was approximately 2550K (4600R) with an average 1.18
MW per fuel element. The Pewee reactor was also the first to incorporate zirconium
hydride moderator in the tie-rod core support elements. Regenerative-cooled tie-tubes
rather than dump-cooled tie-rods were used in the Phoebe-2 reactor tested at NTS to a
power level of 4,000 MW in July 1968. Westinghouse used these important characteristics
and results in establishing core preliminary design for the 50K reactor and the companion
25 and 75K reactors.

Dual turbopumps are used to provide an element of redundancy, as were used in the
NERVA R-1 engine design. Valves provide isolation for a shutdown turbopump. The
turbines are powered through a split-flow expander cycle where energy is derived from
cooling the tie-tubes and nozzle in parallel.

Based on envelope considerations, the nozzle expansion ratio was set at 200:1 and the bell
core length at 110% of that for a 30° conical nozzle. For the 200:1 expansion ratio, this
length provides the optimum nozzle thrust coefficient for hydrogen, resulting in a specific
impulse of 870 seconds.

Hydrogen is bled from the turbine exhaust and regulated to provide engine pneumatic
power and stage tank pressurization. Hydrogen is stored in two tanks to provide pneumatic
power for engine restart. Pressurization of the stage Liquid Hydrogen tank is not required
for starting due to 2-phase pumping capability of the pumps. If LH2 tank pressure is below
approximately 35 psia, then boost pumps are probably required.

The engine thrust-to-weight ratio is 5.3, including shielding which provides approximately
half the maximum radiation field requirements of NPO. Without shielding, the T/We is
5.8.

Provisions for thrust vector control have been made by incorporation of a gimbal bearing
at the top of the pressure vessel dome, and by two orthogonal outriggers for accepting
gimbal actuators. The outriggers are mounted to the upper portion of the pressure vessel.
Motion of the pump inlet during gimballing is accommodated by the scissor bellows, similar
to those used on J-2 engines.

The engine length from the top of the gimbal bearing to the exit plane of the nozzle is
7.66M. The exit diameter of the nozzle is 2.44M.
50K NTR, Expander Cycle, Dual T/P*

Centrifugal Pump

![Centrifugal Pump Diagram]

**DESIGN VALUES**

- **PUMP LIQUIDATE (L/DN):** 57.56 MPa (8,500 psi)
- **PUMP DISCHARGE PRESSURE:** 1,755 psi
- **NUMBER OF PUMP STAGES:** 2
- **PUMP EFFICIENCY:** 72.55%
- **TURBOPUMP RPM:** 47,500 rpm
- **TURBOPUMP POWER (TOTAL):** 3,840 kW
- **TURBINE INLET TEMP:** 293 K
- **NUMBER OF TURBINE STAGES:** 3
- **TURBINE EFFICIENCY:** 72.55%
- **TURBINE FLOW RATE (TOTAL):** 1,611 kg/s
- **HOLE (TURBINE THERMAL POWER):** 1,031 kW
- **CORE THERMAL POWER (FUEL CONSUMED + TURBINE):** 1,013 kW
- **ENGINE THRUST:** 50,000 lbf
- **NOZZLE CHAMBER TEMP./VALUE:** 2,255 K
- **CHAMBER PRESSURE (NOZZLE STATIONARY):** 784 psia
- **NOZZLE EXPANSION AREA (MAX):** 200 in
- **NOZZLE PERIOD LENGTH:** 110 in
- **VACUUM SPECIFIC IMPULSE (L/L-B/L-H):** 388.72 s

**Heat loads are as follows:**
- Nozzle-con (total): 29.44 MW
- Nozzle-div (total): 9.66 MW
- Deflector (total): 12.10 MW
- Tie-Tubes (total): 54.59 MW

**P** - PSIA
**I** - DEG K
**W** - LB/S
**H** - IT/SL
**S** - HI/SL

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

- XE-PRIME

\[
\begin{align*}
T_e & = 2770 \text{K} \\
\text{BLEED CYCLE (TURBINE 10\% FLOW)} \\
\text{NOZZLE E OF 10} \\
- \text{REGEN COOL FUEL ELEMENT SUPPORTS} & \text{+35} \\
- \text{INTEGRATE EXPANDER CYCLE} & \text{+35} \\
- \text{INCREASE E TO 200, 110\%I.} & \text{+50} \\
- \text{INCREASE T_e TO 2550K (PEWEE)} & \text{+40}\end{align*}
\]

- PEWEE-BASED, GRAPHITE ELEMENT ENGINE I

\[
- \text{INCREASE T_e TO 2700K} \quad \text{+30}\]

- COMPOSITE ELEMENT, ENGINE I

\[
\text{900 SEC}
\]

The XE-Prime is the baseline for nuclear thermal rocket engines, since it is the only engine configuration ever tested. This experimental engine was tested during much of the year in 1969, but full power and performance were not achieved in June when a chamber temperature of 2270K (4000°F) was achieved. Due to the facts that a low expansion ratio (3:1) ground test nozzle was used, and that a bleed cycle was used to power the turbine which exhausted 10\% of the engine flow at low specific impulse, an engine specific impulse of only 710 seconds was realized. Specific impulse is increased by 35 seconds by using regeneratively cooled (tie-tube) fuel elements in place of dump-cooled, tie-end fuel element supports, and by using regenerative cooling instead of dump-cooling in the core periphery where the transition is made from the irregular boundary of the hexagonal fuel elements to the circular boundary of the seal segments for sealing and handling the core.

Specific impulse is increased by 35 seconds by using the expander cycle where the turbine exhaust is combined with the balance of the engine flow and the total flow is exhausted at the high reactor exit temperature rather than using the bleed cycle where the turbine flow (10\% of the engine flow) is exhausted at low temperature and degrades engine specific impulse.

Specific impulse is increased by 50 seconds by increasing the nozzle expansion ratio from the experimental ground test engine value of 10:1 to 200:1 expansion ratio for a flight engine and using a 110\% heat contour which provides the optimum thrust coefficient for hydrogen at this expansion ratio.

Specific impulse is increased by 40 seconds by increasing reactor exit gas temperature from the 2270K (4000°F) of XE-Prime to the 2550K (4660°F) of the Pewee Reactor test.

Cumming the above advancement results in the Pewee-Based, Graphite Element, Engine Specific Impulse of 870 seconds, since the Pewee Reactor used graphite fuel elements.

Specific impulse is increased by an additional 30 seconds if composite fuel elements where a reactor exit gas temperature of 2700K (4850°F) can be achieved based on data from Nuclear Furnace, are used rather than the graphite fuel element with a reactor exit gas temperature of 2550K (4600°F) based on data from Pewee Reactor testing. This results in the Composite Element (Nuclear-Furnace-Based) Engine Specific Impulse of 900 seconds.
ENGINE LENGTH AND NOZZLE SIZING

25K ENGINE

STAGE REQUIREMENTS

ENGINE LENGTH - 6.0M
ENGINE I<sub>3</sub> - 870 SEC

CHAMBER PRESSURE INCREASED

FROM 621 PSIA (PEWEE)
TO 784 PSIA
MEETING REQUIREMENTS AND RESULTING IN:

ε 200:1
110% LENGTH

50 AND 75K ENGINES

USED 25K NOZZLE PARAMETERS

ε 200, 110%L

ENGINE LENGTH AND NOZZLE SIZING

Initial effort in the program covering 25, 50 and 75K thrust engines was directed on the 25K engine, since stage requirements were provided for this engine. Engine length was limited to 6.0 meters with a specific impulse of 870 seconds.

To meet the stage requirements, the chamber pressure of 621 psia from the Pewee test condition had to be increased to 784 psia. The higher pressure is beneficial to the reactor core with regard to heat transfer and pressure drop. The resulting nozzle has an expansion area ratio of 200:1 and a bell contour length of 110% of that for a 30° conical nozzle. With hydrogen, the 110% length provides maximum nozzle thrust coefficient for an area ratio of 200:1.

For the 50 and 75K engines, the same nozzle parameters of 200:1 expansion ratio and 110% length were used to result in a consistent family of engines from the standpoints of envelope, performance and weight.
NON-NUCLEAR COMPONENT TECHNOLOGY

- CHAMBER TECHNOLOGY
  - ROVER-KIWI, PHOEBUS
  - INCONEL-X TUBES
  - INCO 718 SHELL
  - FURNACE BRAZED ASSEMBLY
  - LIGHTWEIGHT
  - HIGH TEMP CAPABILITY

- SSME
  - SLOTTED FORGED NARLOY
  - ELECTRODEPOSITED CU/NI CLOSURE
  - INCO 718 SHELL
  - HIGH HEAT FLUX CAPABILITY
  - LOW WALL TEMPERATURE
  - SUPERIOR LIFE CAPABILITY

Rocketdyne has two technologies applicable to the NTR Chamber, the convergent and low area ratio divergent component which attaches to the bottom of the pressure vessel and ducts the reactor exit gas through the sonic region, delivering it to the high expansion ratio nozzle. One technology comes from earlier rocket engine programs, including the Rover program where tubular-wall chambers were employed, and still are today for engines such as Atlas and Delta. The other technology is the slotted, one-piece, copper-wall chamber used for the SSME.

The Rover tubular-wall chambers (as shown in the photo) were used for 7 of the 19 reactors tested in the Rover/NERVA program, including Phoebus 1B at conditions approaching 1500 MW, 750 psig chamber pressure, and throat heat flux of 50 BTU/ft²/sec. These chambers have a contraction ratio of approximately 20:1 to interface with the reactor at an inlet diameter of approximately 35 inches, and an expansion ratio of 12:1 to exhaust into the atmosphere at NTS conditions. Inconel-X tubing was used with an Inconel 718 one-piece forged shell/flange. The chamber was a furnace-brazed assembly. This technology provides a lightweight chamber with approximately 1000°F (1800°C) wall temperature capability.

The SSME slotted, one-piece, copper-wall chamber (as shown in the photo) was developed for and used on all Space Shuttle Main Engines. Three SSME's on each Space Shuttle flight have now powered 50 missions, and flight configuration engine testing exceeds 120 hours. The SSME chamber operates at a chamber pressure of approximately 3000 psia with a wall temperature at the throat of approximately 4800°F (2600°C) and heat flux of approximately 100 BTU/ft²/sec. The chamber has a contraction ratio of approximately 3:1 and an expansion ratio of 5:1 with a throat diameter of approximately 10 inches. The slotted, forged NARloy (Rocketdyne copper alloy) chamber liner is electrodeposited on the outer envelope with a thin copper and then heavier nickel closure of the coolant slots. A welded Inconel 718 shell, manifold and flange assembly complete the chamber. This technology provides high heat flux (100 BTU/ft²/sec) capability with low (100°F) wall temperature. Although the weight is somewhat higher for an NTR chamber than with the tubular-wall Rover chamber technology, the SSME chamber technology is favored due to superior Life-Cycle capability and general robustness.
NON-NUCLEAR COMPONENT TECHNOLOGY (CONT'D)

- NOZZLE TECHNOLOGY
  - SSME
  - A-286 TUBES
  - FURNACE BRAZED ASSEMBLY

- TURBOPUMP TECHNOLOGY
  - INDUCER
    - Mk 15F, Mk 25
    - 2-PHASE PUMPING CAPABILITY
    - TITANIUM
  - IMPELLERS
    - SSME HPFTP
    - TITANIUM
  - BEARINGS - HYDROSTATIC
    - Mk25, Mk29FD
  - TURBINE
    - TITANIUM, A-286, OR 718

NON-NUCLEAR COMPONENT TECHNOLOGY (CONT'D.)
NOZZLE AND TURBOPUMP TECHNOLOGY

Rocketdyne high-expansion-ratio, regeneratively-cooled, nozzle technology is exemplified by the SSME nozzle shown in the photo. The construction is tubular-wall, using A-286 tubes, furnace-brazed assembly, in order to reduce the weight of this large nozzle. The nozzle inlet is an area ratio of 5 with an exit area ratio of 75.5. The nozzle length is approximately 10 ft. with an exit diameter of approximately 7.1/2 ft. The nozzle employs approximately 1,000 tubes. As with the chamber, this SSME technology provides capability beyond the requirements of the NTR, resulting in a robust design.

Rocketdyne technology applicable to the NTR turbopump draws on elements from several programs; however, is best exemplified by the Mark 29F (shown in the photo) which was developed as the liquid hydrogen turbopump for the J-2 engine. Rocketdyne initiated design and development of large liquid-hydrogen turbopumps in 1958 under the Rover program for application to nuclear rockets. Successful testing of the first large liquid-hydrogen (Mark 9) pump in 1960 allowed commitment to the J-2 engine which used the Mark 15F (derived from Mark 9) metal liquid-hydrogen pump. The Mark 9 and evolutionary Mark 25 turbopumps were used for 11 of the 19 reactors tested in the Rover/NERVA program and in the Plum Brook B1 NTR cold-flow engine simulation teststand.

Inducer technology for liquid hydrogen pumps is exemplified by 2 phase testing of the Mark 15F and Mark 25 at inlet vapor volume fractions of up to 30%. Low flow-coefficient, larger diameter inducers were then fabricated for these pumps and tested to even higher vapor volume fractions. This capability provides for pumping of liquid hydrogen from a saturated tank without the need for pressurization to provide net positive suction head at the pump inlet. This provides weight savings to the stage in tank weight, pressurant and storage tank weight, and fueled propellant weight.

The liquid hydrogen centrifugal pump technology of the Mark 29F was advanced with the SSME High Pressure Fuel Turbopump. Significant improvement in efficiency was achieved. Hydrostatic bearings were demonstrated in the Mark 25 pump in testing in 1972 at NTS. These bearings used interior rolling-element bearings which provided the rotation at lower speeds during the slow shut-up and very slow shut-down associated with NTR's. This arrangement considerably reduces the DN requirement and life requirement for the rolling element bearing and allows use of radiation-resistant cage materials. Pure hydrostatic bearings in liquid hydrogen is an ongoing development with the Mark 29FD.

Due to the low inlet temperature (approximately 300K) and single-stage of the expander cycle turbine, the turbine technology for the NTR is simplified compared to the high temperature, multi-stage turbines developed for most rocket engines. Areas of concern are hydrogen embrittlement and hydrogen in which Rocketdyne has much of the world's applicable experience.
### 50K PEWEE-DERIVED DUAL-TURBOPUMP NTR ENGINE
### CHANGING REACTOR SUPPORT RATIO
### SIGNIFICANTLY IMPROVES T/W_e

<table>
<thead>
<tr>
<th></th>
<th>3:1 FUEL ELEMENT SUPPORT RATIO</th>
<th>∆ WEIGHT 3 TO 6:1 RATIO</th>
<th>6:1 FUEL ELEMENT SUPPORT RATIO</th>
</tr>
</thead>
<tbody>
<tr>
<td>REACTOR</td>
<td>8,200</td>
<td>-1,950</td>
<td>6,250</td>
</tr>
<tr>
<td>NOZZLE</td>
<td>1,200</td>
<td>+20</td>
<td>1,220</td>
</tr>
<tr>
<td>TURBOPUMP</td>
<td>270</td>
<td>+10</td>
<td>280</td>
</tr>
<tr>
<td>LINES AND CONTROLS</td>
<td>860</td>
<td>+60</td>
<td>920</td>
</tr>
<tr>
<td>SHIELD</td>
<td>1,100</td>
<td>-250</td>
<td>850</td>
</tr>
<tr>
<td></td>
<td>11,630 LB</td>
<td>-2,110 LB</td>
<td>9,520 LB</td>
</tr>
<tr>
<td>ENGINE T/W</td>
<td>4.3</td>
<td></td>
<td>5.3</td>
</tr>
</tbody>
</table>

Between the conceptual sizing of the 50K reactor and preliminary sizing, Westinghouse determined through nuclear analysis that a 6:1 Fuel Element to Support Element Ratio could be used rather than a 3:1 ratio, resulting in higher power density and less weight for the 50K reactor. Thus, the 50K core is more similar to the 75K core, which uses a 6:1 support ratio (as used in RVWE-84, NNX and Phoenix reactors), rather than the 25K core, which uses a 3:1 support ratio (as used in Pwee). Elimination of virtually half the supports (with their Zirconium Hydride moderator, tie-tubes and graphite parts, together with the core and reflector diameter reduction effects, results in a reactor weight reduction of approximately 2,000 lb, or approximately 25%. The shield likewise decreases approximately 25% due to the reduction in diameter.

The non-nuclear components increase slightly (approximately 5%) in weight due to increase in pump discharge pressure to provide higher pressure ratio to drive the turbine as a result of lower turbine inlet temperature because of reducing the number, and therefore, total power of the tie-tubes (one contained in each support element) by 50%.

Due to the reduction in engine weight as the result of basically cutting the number of support elements in half (going from 3 to 6:1 Fuel Element to Support Element ratio), the engine weight is reduced by approximately 20%, and therefore, the engine thrust-to-weight ratio improves by 20%.
### LIFE IMPACT ON CHAMBER TEMPERATURE AND SPECIFIC IMPULSE ADVANCEMENT

#### XE-PRIME

<table>
<thead>
<tr>
<th>T&lt;sub&gt;c&lt;/sub&gt;</th>
<th>2270K</th>
</tr>
</thead>
<tbody>
<tr>
<td>BLEED CYCLE</td>
<td>(TURBINE 10% FLOW)</td>
</tr>
<tr>
<td>NOZZLE e OF 10</td>
<td></td>
</tr>
</tbody>
</table>

- REGEN COOL FUEL ELEMENT SUPPORTS AND CORE PERIPHERY + 35 SEC
- INCORPORATE EXPANDER CYCLE + 35 SEC
- INCREASE e TO 200, 110% L + 50 SEC

- INCREASE T<sub>c</sub> WITH GRAPHITE FUEL ELEMENT TO:
  - 2550 K
  - Life: 1.5 HR
  - Δ<sub>I</sub>: + 40 SEC

- INCREASE T<sub>c</sub> WITH COMPOSITE ELEMENT TO:
  - 2700 K
  - Life: 1.5 HR
  - Δ<sub>I</sub>: + 30 SEC

<table>
<thead>
<tr>
<th>GRAPHITE ELEMENT ENGINE I&lt;sub&gt;G&lt;/sub&gt;</th>
<th>870 SEC</th>
<th>2450 K</th>
</tr>
</thead>
<tbody>
<tr>
<td>Life 1.5 HR</td>
<td>4.5 HR</td>
<td></td>
</tr>
<tr>
<td>Δ&lt;sub&gt;I&lt;/sub&gt;: + 20 SEC</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>COMPOSITE ELEMENT, ENGINE I&lt;sub&gt;C&lt;/sub&gt;</th>
<th>900 SEC</th>
<th>2550 K</th>
</tr>
</thead>
<tbody>
<tr>
<td>Life 1.5 HR</td>
<td>4.5 HR</td>
<td></td>
</tr>
<tr>
<td>Δ&lt;sub&gt;I&lt;/sub&gt;: + 20 SEC</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

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As in the prior chart, "Specific Impulse Advancement," the Xe-Prime is the baseline for specific impulse at 710 sec. Also, as in the prior chart, advancements by 1) regenerative cooling of core structure (+35 sec), 2) using the expander cycle (+35 sec), and 3) using the 200/1 expansion ratio nozzle (+50 sec), increase specific impulse by 120 seconds.

However, Westinghouse evaluation of Rover/NERVA Fuel Element Mass Loss resulted in life capability of 1.5 hours for the prior chart's Graphite Fuel Element at Power Average Exit Gas Temperature of 2550K and resulting specific impulse of 870 sec, and Composite Fuel Element Gas Temperature of 2700K with specific impulse of 900 sec. This 1.5 hour data is presented in the left hand column. "Near-Term" engine life requirements are for a life capability of 4.5 hours. To meet the 4.5 hour life with the assumed reactance loss of 18, translates into 18 to 20 grams mass loss per element and the resulting temperature and specific impulses as shown in the right hand column. For the 4.5 hour life requirement, the resulting Graphite Element Engine Specific Impulse is 850 sec, and the Composite Element Engine Specific Impulse is 870 seconds.
50K NTR, Expander Cycle, Dual T/P*

Centrifugal Pump

In conjunction with the reduction in graphite fuel element average exit gas temperature from a nominal 2550K to 2450K to meet the 4.5 hour Life requirement, a revised system balance was performed.

Compared to the balance shown on the prior chart, the average reactor exit gas temperature (nozzle chamber temperature) is reduced by approximately 4% to 2,444 K. This results in an approximate 2% reduction in specific impulse to 846.64 sec. To maintain engine thrust at 50.000 lbf requires increasing flowrate by approximately 2% to 59.01 lbf/sec. The increase in flowrate and reduction in temperature result in an approximately 3% reduction in Reactor/Engine Thermal Power to 1.002.8 Megawatts.

The reactor configuration for the 1.5 hr and 4.5 hr life would be the same. Fuel element thermal conditions and stresses actually reduce due to the 4% reduction in temperature and 3% reduction in power. Fuel element mechanical stresses stay the same since the reactor exit pressure is fixed (784 psia) and the core pressure drop is essentially the same due to the 2% reduction in velocity (3% increase in flowrate and 4% increase in density due to lower temperature) and 4% increase in density. So the reactor weight remains essentially the same between the 1.5 hr and 4.5 hr life cases.

The chamber and nozzle sizes remain the same, due to the 2% increase in flowrate and 4% reduction in temperature resulting in the same thrust area.

The pump flowrate increases by 2% and the discharge pressure increases by 4% due to the 6% increase in turbine pressure ratio required to provide the 6% higher turbopump power. This results in a 4% increase in turbopump weight which is equivalent in approximately 0.1% in engine weight. Due to the 3% increase in flowrate and the 4% increase in pump discharge pressure, the turbopump line weight increases by 6% which is equivalent to approximately 0.4% in engine weight.

So the engine weight effect in going from the 1.5 to the 4.5 hour life is an approximate 0.5% increase in weight due to the 2% increase in flow and 4% increase in pump discharge pressure with the majority of the effect being due to the pump discharge and turbine lines.

*Note: Flows indicated are for one-half of system.
The effect of thrust for the 25, 50 and 75K lb engines on engine thrust-to-weight ratio (without including radiation shielding) is shown for both Graphite and Composite Fuel elements.

As a result of discussion related to the previous chart regarding engine weight changes in going from 1.5 to 4.5 hr. Life, the engine weight increases by approximately 0.5% primarily in live weight due to the 2% increase in flowrate and the 4% increase in pump discharge pressure. This is a negligible effect to these thrust-to-weight ratio plots. Therefore, the plot for each fuel element applies for the range of Life and Specific Impulse shown.
Effect of Thrust and Use of ZrH Moderator on Engine Thrust-to-Weight Ratio

The effect of Zirconium Hydride moderation on engine thrust-to-weight ratio (without including radiation shielding) is shown over the thrust range of 25 to 100 Klb. The lower curve represents engines using a fixed 35-inch diameter, 32-inch long core containing no ZrH. The upper curve represents engines using reactors containing ZrH as necessary to minimize size and weight for the 25, 50 and 75 Klb thrust reactors as analyzed by Westinghouse. Other Westinghouse preliminary analysis indicates that ZrH does not reduce the weight of a reactor with a 35-inch diameter core for a 100 Klb thrust engine. On this basis, the dashed line was constructed between the 75 Klb ZrH moderated point and the 100 Klb point without ZrH.

So ZrH moderation provides no advantage at 100 Klb thrust, approximately 10% weight advantage at 75 Klb, approximately 35% weight advantage at 50 Klb, and approximately 75% weight advantage at 25 Klb thrust.
Effect of Shield on Engine Thrust-to-Weight Ratio

The effect of shield weight on engine thrust-to-weight ratio is shown. The upper data represents the engine thrust-to-weight ratio without including radiation shielding. The lower data represents engine thrust-to-weight ratio using the internal shield used for the NERVA R-1 engines; namely, 12 inches of BATH (Boron, Aluminum, and Titanium Hydride) and 1 1/4 inches of lead.

During the program, NPO specified neutron flux levels and gamma dose level to be met by the shielding for the "Near-Term" reactor. Due to concern about lead melting during decay heat removal, the lead was removed. The NPO-specified radiation field also allowed reduction in the BATH thickness from 12 inches down to 9 inches. The Westinghouse analysis of the resulting radiation field for the 50K engine results in neutron fluxes and gamma dose approximately half that specified by NPO, indicating that a small further reduction in BATH thickness may be made.

At the 50K thrust level, the Light Shield provides approximately 10% improvement in engine thrust-to-weight ratio over the NERVA shield. The light shield represents an approximate 9% reduction from the thrust-to-weight ratio of 5.8 for the unshielded 50K engine.

EFFECT OF SHIELD ON ENGINE THRUST-TO-WEIGHT RATIO
PEWEE-DERIVED NTR'S

Conceptual designs were performed for 25, 50 and 75K thrust engines based on Rover/NERVA reactor technology. Fuel element power was based on Pewee test results of approximately 1.2 Megawatts per fuel element average. The engine thrust-to-weight ratio requirement of >4 with shielding was not met by the 25K engine which has a value of approximately 3.6 with a shield meeting the NPO radiation field requirements. Unshielded, the 25K engine has a thrust-to-weight ratio of approximately 3.9, so the requirement of 4 is not met even without the radiation shield.

Certainly, based on engine thrust-to-weight ratio, the preferred engine thrust would be 100K or more based on the elimination of the weight penalty associated with the use of Zirconium Hydride moderator and achieving a shielded engine thrust-to-weight ratio of approximately 6.5, or approximately 7.1 without radiation shielding.

NPO selected the 50K engine for more detailed analysis and specified radiation field values to determine the shield. The radiation field requirements allowed lightening the shield by approximately 900 lb resulting in an improvement in shielded engine thrust-to-weight ratio from 4.8 to 5.3, with an unshielded thrust-to-weight ratio of 5.8.

The 50K engine has an overall length of 7.66 meters and a nozzle exit diameter of 2.44 meters.

These parameters all apply regardless of whether 1) the engine life is 1.5 hours operating at a chamber temperature of 2550K with 870 sec specific impulse, or 2) the engine life is 4.5 hours operating at a chamber temperature of 2450K with 850 sec specific impulse.
Key Technology/Streamline Development Assessment

- **Approach**
  - Identify critical design areas and technology issues
  - Assess actions and program impacts
  - Determine critical path

- **Areas addressed**
  - Safety, hydrogen pumping, nozzle, valves, instrumentation and controls, reactor, engine system and test facility

In performing this assessment, design areas of the engine system were reviewed and critical technology issues were identified, together with actions required to address these issues and their impact on the program. A critical path was inferred from this analysis. The design areas addressed were safety, hydrogen pumping, the nozzle, valves, instrumentation and controls, the reactor assembly, and the engine system and test facility. In most instances it was found that recovering or referencing existing technology provides the design basis. However, several system design issues exist where new design solutions and test verifications would be required, and effort to resolve these items should be emphasized early in the program.
### Key Technologies Assessment

Ready technology from many sources forms the foundation for the Rocketdyne-Westinghouse NERVA-derived engine concept. In reviewing key technology areas the goal was to assess both the needed actions to resolve the particular issues and the programmatic impact. In many cases technology recovery was the principal action, and there was no programmatic impact. In a few areas, issues not anchored in ready technology were identified, and their resolution should be addressed early in the program. We expect no intractable problems.

Safety issues in all phases of the program have to be adequately identified, and procedures and design solutions have to be qualified. Four safety issues are noted here: (1) water immersion criticality, (2) intact reentry or total dispersal, (3) the concern over flammability and dispersion of nuclear materials in a launch explosion and fire, and (4) the impact of the continued nuclear power generation of a shutdown engine in a cluster. The latter affects engine-cluster specific performance, but is also a safety issue because the overheating potentially can damage the stage placing the crew at risk, and ejection of the engine with its potential for generating debris may pose a threat to the stage or to future missions.

Restartability requires adequate systems for decay heat removal that do not consume excessive quantities of hydrogen. A flight-qualified decay heat removal system was never demonstrated in the Rover/NERVA program.
<table>
<thead>
<tr>
<th>AREA OF DESIGN</th>
<th>TECHNOLOGY ISSUE</th>
<th>ACTION REQUIRED</th>
<th>PROGRAM IMPACT</th>
</tr>
</thead>
<tbody>
<tr>
<td>Safety</td>
<td>Launch fire resistance of nuclear and hazardous materials</td>
<td>Analysis and design of fire retarding or resisting features such as plug in throat</td>
<td>Mockup test in simulated launch explosion/fire</td>
</tr>
<tr>
<td>Safety</td>
<td>Engine-out continued power generation; dead weight</td>
<td>Analysis of alternatives: auxiliary cooling, shielding, ejection, etc.</td>
<td>No additional impact</td>
</tr>
<tr>
<td>Restartability</td>
<td>Decay heat removal</td>
<td>Analysis and design of optimum method for conserving propellant</td>
<td>Test decay heat removal system during engine tests</td>
</tr>
</tbody>
</table>

Rockwell International
Rockwell Defense Division

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### Key Technologies Assessment

The hydrogen turbopumps, while based on mostly proven hardware, must be qualified for the radiation environment. 10 restarts in space, slow startup and shutdown transients, and 4.5 hours of accumulated full-power operation. The solutions to these issues are anchored in existing technology, but a rigorous test program will be required to demonstrate, adequately, the design integrity.

Chamber and nozzle experience with the SSME satisfies most design requirements, except those dealing with the radiation environment, such as joint seal design. Testing of seals in a radiation environment would be required.

Radiation resistance of valves--bodies, stems, guides, actuators, seals, seats--must be incorporated in the design and verified by test, and turbine bypass valve functional performance must be assured by test.

---

**KEY TECHNOLOGIES ASSESSMENT**

<table>
<thead>
<tr>
<th>AREA OF DESIGN</th>
<th>TECHNOLOGY ISSUE</th>
<th>ACTION REQUIRED</th>
<th>PROGRAM IMPACT</th>
</tr>
</thead>
<tbody>
<tr>
<td>Hydrogen pumping</td>
<td>Two-phase pumping</td>
<td>Design pumping system compatible with tank pressurization limits; recover Rover test data, Mark 25 and Mark 15 data</td>
<td>Test candidate configuration in pump test facility</td>
</tr>
<tr>
<td>Hydrogen pumping</td>
<td>Bearings</td>
<td>Design for 10 restarts, and slow start and shutdown transients; recover Mark 25 data with hybrid hydrostatic bearings, SSME experience, Mark 29FD with hydrostatic bearings</td>
<td>Demonstrate during pump qualification test</td>
</tr>
<tr>
<td>Hydrogen pumping</td>
<td>Seals</td>
<td>Select radiation-hard seal materials</td>
<td>Part of turbopump design and test</td>
</tr>
</tbody>
</table>

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*Rockwell International*  

Washington, D.C.  

*NP-TIM-92*
### KEY TECHNOLOGIES ASSESSMENT

<table>
<thead>
<tr>
<th>AREA OF DESIGN</th>
<th>TECHNOLOGY ISSUE</th>
<th>ACTION REQUIRED</th>
<th>PROGRAM IMPACT</th>
</tr>
</thead>
<tbody>
<tr>
<td>Nozzle</td>
<td>Radiation resistant joint seal</td>
<td>Test seal configuration in radiation environment</td>
<td>Need to identify test facility</td>
</tr>
<tr>
<td>Nozzle</td>
<td>High heat flux</td>
<td>Use SSME NARloy-Z slotted channel approach</td>
<td>No additional impact</td>
</tr>
<tr>
<td>Valves</td>
<td>Radiation resistance</td>
<td>Select radiation resistant materials; Recover data from Rover/NERVA, SP-100, LMFBR</td>
<td>Life-cycle test valves separately, and evaluate after engine test</td>
</tr>
</tbody>
</table>

*Rockwell International*  
*Wyer-Hilbre Corporation*
**KEY TECHNOLOGIES ASSESSMENT**

<table>
<thead>
<tr>
<th>AREA OF DESIGN</th>
<th>TECHNOLOGY ISSUE</th>
<th>ACTION REQUIRED</th>
<th>PROGRAM IMPACT</th>
</tr>
</thead>
<tbody>
<tr>
<td>Valves</td>
<td>Turbine bypass control</td>
<td>Modify SSME and J-2 valves and perform life-cycle test</td>
<td>Test in hydrogen flow facility</td>
</tr>
<tr>
<td>Reactor assembly</td>
<td>Fuel element midband corrosion</td>
<td>Develop and test coating materials and processes</td>
<td>Identify reactor test facility; test and evaluate in engine testing</td>
</tr>
<tr>
<td>Reactor assembly</td>
<td>Vessel design for intact reentry or dispersal, decay heat removal</td>
<td>Select compatible materials and configuration</td>
<td>Safety requirements drive the design</td>
</tr>
</tbody>
</table>

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### Key Technologies Assessment

In the reactor assembly, midband corrosion found in the Rover/NERVA fuel elements was being addressed when the program was canceled. Resolution of this issue will be of prime importance at the outset, with in-pile testing of fuel elements and clusters necessary to validate the solution. The reactor vessel must be designed to meet the safety requirement of intact reentry or total dispersal in the event of an inadvertent reentry from space. The control drum actuators may feature electrical or pneumatic drives, or both, based on Rover/NERVA or current design technology. The support plate must contain the tie-tube inlet and outlet flow passages, operate with minimal thermal distortion, and be structurally robust. Data from the Phoebus 2A reactor and from the NERVA design would be the bases for the new design. To achieve higher operating temperatures and performance development of composite fuel would be continued from the Rover program baseline. The instrumentation and control design area would initially address key sensors and the engine health monitoring system. Current technology would serve these areas.

Finally, the ground testing of the complete engine system is the key step in qualification for piloted operation. An operational facility will be needed with adequate engine-exhaust scrubbing to meet environmental and safety concerns, and with well-designed altitude simulation diffusers and ejectors. Because design, the environmental approval process, construction, and acceptance testing will require about 6 years to complete, embarking on this effort almost immediately is essential to meeting the desired 10 year development goal. We believe that this facility is the critical path.
## Key Technologies Assessment

<table>
<thead>
<tr>
<th>Area of Design</th>
<th>Technology Issue</th>
<th>Action Required</th>
<th>Program Impact</th>
</tr>
</thead>
<tbody>
<tr>
<td>Reactor assembly</td>
<td>Drum actuators</td>
<td>Incorporate Rover/NERVA or SP-100 designs; evaluate fluidic stepping motors</td>
<td>Testing required</td>
</tr>
<tr>
<td>Reactor assembly</td>
<td>Support plate with tie tubes</td>
<td>Evaluate Phoebus-2A and NERVA-R1 designs, fabricate and test unit</td>
<td>Test in hydrogen flow facility; parallel with pump testing</td>
</tr>
<tr>
<td>Reactor assembly</td>
<td>Higher temperature fuel</td>
<td>Develop composite fuel</td>
<td>Test reactor or nuclear furnace; required</td>
</tr>
<tr>
<td>Instrumentation and controls</td>
<td>Hydrogen flow measurement</td>
<td>Evaluate candidate flow meters, including fluidics; procure candidates and test</td>
<td>Test in hydrogen flow facility</td>
</tr>
<tr>
<td>Instrumentation and controls</td>
<td>Reactor temperature, pressure</td>
<td>Incorporate Rover/NERVA and advanced reactor sensors</td>
<td>Life test in a reactor in hydrogen; evaluate in engine test</td>
</tr>
<tr>
<td>Instrumentation and controls</td>
<td>Health monitoring system</td>
<td>Incorporate Rocketdyne state-of-the-art diagnostics</td>
<td>Evaluate in engine system test</td>
</tr>
<tr>
<td>Engine system test</td>
<td>Scrubbing engine exhaust, diffuser and ejector technology; environmental concerns</td>
<td>Proceed with site selection and facility design and construction; recover NF-1 scrubber data, state of the art Rocketdyne diffuser/ejector technology</td>
<td>Critical path to engine qualification</td>
</tr>
</tbody>
</table>

**KEY TECHNOLOGIES ASSESSMENT**

**Area of Design**

**Technology Issue**

**Action Required**

**Program Impact**
Technology Assessment Results

- Technology available for most issues
  
  Rover/NERVA, SSME, Rocketdyne state of the art, SP-100, terrestrial advanced reactors, state-of-the-art electronics and computers

- Unresolved system design issues
  
  Loss of turbopumps, lifetime, intact reentry--water subcriticality (or total dispersal), decay heat removal, engine-out cooling during operations, fuel midband corrosion

- Critical path is engine test facility

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The assessment of key technologies led to the conclusions that (1) existing technology in reactors and engine systems is applicable to most design areas, (2) there are issues requiring attention early in the program to assure satisfactory resolution, and (3) the assured early availability of an engine/reactor test facility is critical to meet, successfully meet the 10-year engine qualification goal.
Streamline Development Requires Early Agreement on Requirements

To establish a good foundation for a successful development program both NASA and the Rocketdyne/Westinghouse team must understand and accept the design requirements, program and technical interface requirements, and design criteria and testing needs. Poorly understood or shifting requirements can lead to delays and cost escalation. We believe that a QFD analysis of the program will lead to well-understood requirements and optimum design and hardware results.
A development logic diagram can include many layers of detail and be organized in many different ways. This high-level diagram shows many necessary tasks in setting requirements, recapturing technology, resolution of key design issues, facility design, construction, and activation, and testing of components and systems. The most important message is that the program must start with well-defined requirements and design criteria, and that the availability of key test facilities will drive the rate of achievement of the 10-year goals. Near-term activities of conceptual design, technology recovery, and resolution of design issues will provide a sound basis for proceeding quickly as substantial funding becomes available.
NTR Streamline Development Plan Summary

The time-phasing of key groups of activities from the development logic diagram shows that several tasks should be emphasized at the start: setting requirements, technology recapture, and establishing design criteria. Test facility design, construction and activation must also begin promptly to assure that the 10 year schedule can be met.
Advanced Propulsion Engine Assessment based on a Cermet Reactor
for the Nuclear Propulsion TIM
October 20-23, 1992
Pratt & Whitney
Randy C. Parsley
### PRATT & WHITNEY DESIGN CHOICE BASED ON FUNDAMENTAL PRIORITIES

<table>
<thead>
<tr>
<th>Priority</th>
<th>Safety</th>
<th>Reliability</th>
<th>Cost</th>
<th>Performance</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>Safety</td>
<td>Robust design/simple operation</td>
<td>Emergency operation</td>
<td>High thrust-to-weight</td>
</tr>
<tr>
<td>2</td>
<td>Reliability</td>
<td>Design life = 4X operational</td>
<td>Fundamental material compatibility</td>
<td>Ground qualification</td>
</tr>
<tr>
<td>3</td>
<td>Cost</td>
<td>Positive coolant flow management</td>
<td>Fundamental material compatibility</td>
<td>Exploration architecture</td>
</tr>
<tr>
<td>4</td>
<td>Performance</td>
<td>Retention of fuel/stoichiometry</td>
<td>Low development risk</td>
<td>High specific impulse</td>
</tr>
</tbody>
</table>

### Pratt & Whitney Design Choice Based on Fundamental Priorities

A preferred Pratt & Whitney conceptual Nuclear Thermal Rocket Engine, NTRE, has been designed based on the fundamental NASA priorities of safety, reliability, cost, and performance. The basic philosophy underlying the design of the NTRE is the utilization of the most reliable form of ultrahigh temperature nuclear fuel and development of a core configuration which is optimized for uniform power distribution, operational flexibility, power maneuverability, weight, and robustness. The NTRE system employs a fast spectrum, core-typical reactor configured in an expander cycle to ensure maximum operational safety. The cooled fuel form provides retention of fuel and fission products as well as high strength for a simplified structural design and tolerance to power and temperature cycling. System reliability has been addressed by the use of core-based fuels, moderate reactor temperatures, and a two-pass reactor flowpath. The cooled, refractory metallic fuels provide fundamental material compatibility in the expected operating environment as well as retention of fuel and stoichiometry. The two-pass reactor has been designed to a 4X life requirement and provides positive coolant flow management. A baseline 20,000 lb-thrust level is used to minimize ground qualification costs and maximize exploration mission applicability. Finally, the NTRE has been designed to provide high specific impulse at a high thrust-to-weight level.
XNR2000 System Configured As An Expander Cycle

The expander cycle was selected for the proven reliability, robustness, and high efficiency to meet NASA requirements. The XNR2000 expander cycle rocket engine uses heat pick-up in the nozzle, chamber, reflector region, and regenerative cooling of the pressure vessel and upper plenum structure to drive the pumping system and deliver hydrogen to the lower plenum of the reactor core. The reactor houses the hydrogen in a two-pass flow configuration and delivers the hydrogen propellant to the nozzles/chamber before expansion through a 300-inch area nozzle. The reactor is comprised of an outer annular core of 90% UO2 pressurized fuel elements and an inner cylindrical core of 61 W-UO2 pressurized fuel elements. An upper reflector, integral with the fuel elements, is used to provide axial neutron reflection and composition of W-25. An outer annulus of Be surrounds the reactor and serves as the radial reflector.

The baseline XNR2000 operates at propellant chamber temperature at 35-70 K and chamber pressure of 360 psi to deliver 25,000 lbs. thrust at a specific impulse of 909 seconds.
A radial cross-section of the XNR2000 NHE is shown. Looking down at the engine, hydrogen enters the outer annular ring of fuel elements (unloaded) flows up and is then directed through the inner cylinder of fuel elements (loaded) and flown down. The cross section displays the baseline control approach selected for the XNR2000. One possible option for providing redundant, reactor shutdown control would be the insertion of Bé rods inside the radial support tubes shown in the radial plane cross-section. The rods could be included in the design to prevent inadvertent reactor excursions during transportation, pre-launch, or booster transfer. The Bé rods would provide an independent back-up safety mechanism but would not be used for reactor control.
CERMET FUELS WERE PURSUED FOR BOTH PROPULSION AND POWER

Early cermet failures are most remembered not later material successes

Program refocus to power applications reinforce low temperature bias

Successful demonstrations
- 10,000 hrs at 1,950k (in reactor)
- 1,000 hrs at 2,278k
- 3 hrs at 3,178k
- Transients to 2,879k at 10,000k/sec (in reactor)
- 37 hole element fabrication

Cermet Fuels were Pursued For Both Propulsion and Power

The basic design philosophy used in the development of the XNR2000 was to employ the most reliable form of ultra-high temperature nuclear fuel. The approach used to accomplish this goal was to make use of the extensive data and lessons learned in the ROWER/Nuclear Fuel and Reactor System Development Program, Argonne National Laboratory Nuclear Reactor Program, and the General Electric Advanced Nuclear Propulsion Project 710 Program. A summary of results of cermet fuel development programs of 1960's and 80's is published in two sets of reports ANL-7236, (1968) "Nuclear Rocket Program", Terminal Report, GEMD-650, (1979), "7.0 High Temperature Gas Reactor Program Summary Report", Vol. I-91.
### Chemical stability

<table>
<thead>
<tr>
<th>Constituents</th>
<th>$T_{\text{melt}}$</th>
<th>Matrix</th>
<th>Clad</th>
<th>Hydrogen</th>
</tr>
</thead>
<tbody>
<tr>
<td>UO$_2$</td>
<td>3100k</td>
<td>Solved*</td>
<td>Total</td>
<td>Total</td>
</tr>
<tr>
<td>Tungsten</td>
<td>3650k</td>
<td></td>
<td>Total</td>
<td>Total</td>
</tr>
<tr>
<td>Tungsten – Re</td>
<td>3400k</td>
<td></td>
<td></td>
<td>Total</td>
</tr>
</tbody>
</table>

#### Element

- **Strength**: High
- **Conductivity**: High
- **Ductility (Cold)**: Adequate
- **Ductility (Hot)**: Good

Clad/matrix CET match – Good
Matrix/fuel CET match – Good

*UO$_2$ stabilized with 6% Gd or Th

---

**P&W Internal Studies Identified Cermet Approach As Superior**

Cermet fuel made of UO$_2$ dispersed in Tungsten or Molybdenum cladded with Mo or W based alloys were tested at high temperature in both nuclear and non-nuclear environments and displayed superior performance in the expected operating environment of an HTGR. Retention of fission products and fission, thermal shock resistance, hydrogen compatibility, high thermal conductivity, clad/matrix CET compatibility, and high strength are several major advantages of the cermet fuel form.
CERMET OPERATING LIMITS CAN BE ESTABLISHED FROM EXISTING DATA

A critical review of the cermet fuel development programs was used to establish operating limits for the P&W XNR2000 reactor. The XNR2000 has an endurance on the order of 100% of hours at the selected operating temperature.
CERMET FUEL SHOULD ALWAYS BE INCLUDED ON THIS CURVE

Cermets Fuel Should Always Be Included On This Curve

The predicted endurance of carbon based and cermets based fuels is shown as a function of propellant exit temperature. As shown in the Figure, the endurance of cermets fuels is independent of operating temperature up to the melt temperature of the fuel. However, the endurance of carbon based fuels is a function of propellant temperature because of stoichiometry changes due to chemical diffusion of carbon based fuels in a hot hydrogen environment. A change in the mechanical, thermal, and neutronic characteristics at carbon based fuels decreases the fuel endurance with increasing operating temperature. The cermets fuels display constant characteristics because there is no fuel/matrix diffusion and the material stoichiometry is constant with temperature.
### RESULTS OF NERVA/ROVER RESEARCH REACTOR TESTS WITH TEMPERATURE OVER 2222K

<table>
<thead>
<tr>
<th></th>
<th>Time at temperatures over 2222k (sec)</th>
<th>Maximum chamber temperature (k)</th>
<th>Time at max temperature (sec)</th>
<th>Reactivity loss (grams/element)</th>
</tr>
</thead>
<tbody>
<tr>
<td>PHOEBUS-1A, EPIV (22 June 1966)</td>
<td>651</td>
<td>2367</td>
<td>5</td>
<td>?</td>
</tr>
<tr>
<td>PHOEBUS-1B (23 Feb 1967)</td>
<td>400</td>
<td>2292</td>
<td>5</td>
<td>13.7</td>
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<tr>
<td>PEWEE-1 (Dec 1968)</td>
<td>2555 (fuel exit temp)</td>
<td>2400</td>
<td>20</td>
<td></td>
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<tr>
<td>NF-1 (June 1972)</td>
<td>2444 (fuel exit temp)</td>
<td>6528</td>
<td>13.7</td>
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</tr>
</tbody>
</table>

**Results of NERVA/ROVER Research Reactor Tests With Temperatures Over 2222K**

A short summary of often quoted NERVA/ROVER test results. It should be noted that while time at temperatures over 2222K is high, and most often quoted, the time at maximum temperature is often quite low. Additionally, fuel temperatures are often quoted rather than the lower hydrogen temperatures, adding to the confusion. Reactivity loss was proven to be a major concern in the NERVA/ROVER Program and could significantly increase the cost or even prohibit ground qualification.
### RESULTS OF NERVA/ROVER DEVELOPMENT
### REACTOR TESTS WITH TEMPERATURE OVER 2222K

<table>
<thead>
<tr>
<th>Reactor Code</th>
<th>Time at temp over 2222K (sec)</th>
<th>Max temp (k)</th>
<th>Time at max temp (sec)</th>
<th>Reactivity loss (cents)</th>
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</thead>
<tbody>
<tr>
<td>NRX-A3 (23 April 1966)</td>
<td>5</td>
<td>2244</td>
<td>3</td>
<td>22.3</td>
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<tr>
<td>NRX/EST, EPIII (2 March 1966)</td>
<td>75</td>
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<td>5</td>
<td>2.5</td>
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<tr>
<td>NRX/EST, EPIV (16 March 1966)</td>
<td>110</td>
<td>2264</td>
<td>5</td>
<td>46.7</td>
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<tr>
<td>NRX/EST, EPIVA (25 March 1966)</td>
<td>816</td>
<td>2264</td>
<td>450</td>
<td>282.4</td>
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<tr>
<td>NRX-A5, EPIII (8 June 1966)</td>
<td>473</td>
<td>2286</td>
<td>7</td>
<td>22.5</td>
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<tr>
<td>NRX-A5, EPIV (23 June 1966)</td>
<td>873</td>
<td>2333</td>
<td>7</td>
<td>212.3</td>
</tr>
<tr>
<td>NRX-A6, EPIIIA (15 Dec 1967)</td>
<td>3704</td>
<td>2405</td>
<td>10</td>
<td>70</td>
</tr>
<tr>
<td>XE-PRIME, EP5 (March 1969)</td>
<td>10</td>
<td>2278</td>
<td>5</td>
<td>-</td>
</tr>
</tbody>
</table>

PRATT & WHITNEY XNR2000 DESIGN TEAM

**Project director:** Randy Parsley (P&W)

**Pratt & Whitney**
- Steve Peery (P.I. systems)
- Russ Joyner (Missions)
- Alan Dixon (Mech Des)
- Samim Anghaie (Nuclear, T.H.)
- Gerald Feller (Nuclear)
- Mike Malone (Materials)
- Paul Harris (Materials consultant)

**Babcock & Wilcox**
- Kurt Westerman
- Steve Scoles (P.I.)
- Russ Jensen (Materials)
- James Rhodes (Nuclear)

NTP: System Concepts
XNR2000 Concept Configured To Address P&W Priorities

A conceptual nuclear thermal rocket (NTR), the XNR2000, has been developed for manned space exploration missions. The discriminating features of the XNR2000 that provide attractive attributes are the use of Cermet fuel, a dual-pass reactor flowpath, and a simple robust cycle. An XNR2000 system description, reactor thermal hydraulics summary, and expected operating temperature, and thrust size effects will be presented. This package presents the summary of a 6 month NASA funded study to develop and assess concept feasibility, thrust level range implications, and thermal mission impacts of an NTR system based on a planar Cermet reactor.
XNR2000 CONFIGURED TO MEET NASA REQUIREMENTS

Isp > 850 sec (at 200 area ratio)
T/W > 4
Throttling 25% at rated temperature
Single burn duration 60 min (max)
Engine life > 270 min at rated thrust
Remain subcritical upon impact and immersion
ALARA fission product release
Dual turbopump arrangement
25k, 50k & 75k Thrust size

XNR2000 Configured To Meet NASA Requirements

The XNR2000 Nuclear Thermal Rocket Engine, N15C, was configured to meet or exceed the performance requirements of a manned-rated NTR System. The propulsion requirements listed are described in detail in the "Nuclear Thermal Rocket Engine Requirements" document, version 3, February 10, 1982. The baseline thrust size was set at 25,000 lb., and thrust size effects were determined for engines of 50,000 and 75,000 lb. of thrust. Safety and reliability are key NTR propulsion requirements for the manned-manned Space exploration applications and were considered in the conceptual design of the XNR2000. The reactor fuel and isotope selection was specifically dictated by the ALARA fission product release and reactor subcriticality requirements.
XNR2000 System Configured As An Expander Cycle

The expander cycle was selected for the proven reliability, robustness, and high efficiency to meet NASA requirements. The XNR2000 expander cycle rocket engine uses heat pick up in the nozzle, chamber, reflector regions, and regenerative cooling of the pressure vessel and upper plenum structure to drive the pumping system and deliver hydrogen to the lower plenum of the reactor core. The reactor bears the hydrogen in a two-pass flow configuration and delivers the hydrogen propellant to the nozzle chamber before expansion through a 200 area ratio nozzle. The reactor is comprised of an outer annular core of UO2 prismatic fuel elements and an inner cylindrical core of 90 Mo-UO2 prismatic fuel elements. An upper reflector, integral with the fuel elements, is used to provide axial neutron reflection and is comprised of 160. An outer annulus of He surrounds the reactor and serves as the radial reflector.

The baseline XNR2000 operates at a propellant chamber temperature of 6500 K and chamber pressure of 750,000 psa to deliver 25,000 lbf. thrust at a specific impulse of 500 seconds.
XNH2000 Reactor is Configured From Prismatic Fuel Elements

A medial plane radial cross-section of the XNH2000 NTIS is shown. Looking down at the engine, hydrogen enters the outer annular region of fuel elements (unshaded) flows up and is then directed through the inner cylinder of fuel elements (shaded) and flows down. The cross section displays the baseline control approach selected for the XNH2000. One possible option for providing redundant reactor shutdown control would be the insertion of In rods inside the radial support tubes shown in the medial plane cross-section. The rods could be included in the design to prevent inadvertent reactor excursions during transportation, pre-launch, or booster transfer. The In rods would provide an independent back-up safety mechanism but would not be used for reactor control.
The calculated values of system $K_{eff}$ by Pratt & Whitney agree with calculations of $K_{eff}$ conducted independently by Babcock & Wilcox (B&W) and Argonne National Lab (ANL). Pratt & Whitney used both a 1G energy group boundary/continous-energy code analysis and MCNP statistical code analysis to calculate $K_{eff}$ and ANL used MCNP statistical code analysis procedures to calculate $K_{eff}$.

The plot of Be radial reflector thickness vs. $K_{eff}$ for the baseline configuration, displays the large worth of reactivity for the reflector under approximately 30 cm. This curve indicates that the system can be controlled with neutron reflection. The baseline system employs an 18 cm initial Be reflector and a 20 cm BeO axial reflector.
XNR2000 System Can be Controlled by Raising Reflector Sections

The baseline control approach was designed to provide robust reactor control with minimum complexity and weight. The control of the reactor is accomplished by varying the neutron-leakage rate by means of 108 SYMBOL \ Symbol movable annular segments of the radial reflector. The lower half of each segment is stationary while the upper half translates axially to provide reactor control through the "opening of windows". Negative control is provided by 1 bank of 3 segments while fast-shutdown capability is provided by the other, independent, bank of 3 segments. The selected control approach provides the most reactivity worth for the selected reflector size, thus maximizing thrust to weight. The relevant segments are driven by pneumatic piston-type drive mechanisms which provide linear actuation.
Estimated worth

\[
\frac{\Delta K}{K} \text{ TOTAL } = -4\%
\]

\[
\frac{\Delta K}{K} \text{ TOTAL } = -5\%
\]

Alternate Control Options Are Viable

The baseline control approach was selected for simplicity and reduced weight; however, other control options are viable for the XN92000. Shown here are two such approaches with a preliminary calculation of reactivity worth. Contemporary control drums consisting of the with partial segments of the poison material could provide sufficient negative reactivity insertion for control. Additionally, the use of rotating drums with segments of void could be used to provide control through neutron leakage in a rotating drum configuration. The optimum control of the XN92000 could be achieved through the combination of any of the three approaches presented, providing nuclear control with maximum redundancy.
Baseline Propellant System Configured With Dual Turbopumps

A flow schematic of the baseline engine is shown. Dual turbopumps are employed with quad valve arrangements to maximize system reliability. Each turbopump delivers 50% of the total reactor flow and can be isolated with block valves in case of a pump-out condition. The quad valves consist of 2 block valves followed by 2 control valves arranged in parallel. The 2 pumps pressurize and deliver hydrogen to the nozzle coolant tubes and reflector. The heated hydrogen is then expanded through the turbines and delivered to the reactor. Preliminary investigations indicate that the system could operate at 70% thrust during an engine out scenario. After engine operation pulse cooling of the reactor is provided with pressurized or tank head hydrogen through the pulse cooling quad valve to remove residual heat generation. An emergency pressurized hydrogen tank would provide pressurized hydrogen to the reactor under a 2 pump out, reactor critical condition.
T = 1659 k
P = 956 psi

J = 34.7 k
T = 103 k
P = 1933 psi

PRATT & WHITNEY XNR2000 CERMET NTRE

25448
CERMET ADVANTAGES ESTABLISHED IN GE710/ANL PROGRAMS

<table>
<thead>
<tr>
<th>Characteristic</th>
<th>Payoff</th>
</tr>
</thead>
<tbody>
<tr>
<td>Demonstrated fabrication</td>
<td>Reduced risk</td>
</tr>
<tr>
<td>Fuel matrix / cladding / hydrogen</td>
<td>Life, FFP retention</td>
</tr>
<tr>
<td>compatibility</td>
<td></td>
</tr>
<tr>
<td>High strength and conductivity</td>
<td>Thrust–to–weight, robustness</td>
</tr>
<tr>
<td>High temperature operating capability</td>
<td>Specific impulse</td>
</tr>
</tbody>
</table>

Characteristics confirmed by B&W

Cermet Advantages Established In GE710/ANL Programs

The XNIG2000 builds upon the experience and database of Cermet fuels obtained in the GE710 and ANL programs. The fuel spectrum Cermet fuel form was selected to meet the engine requirements of ALARA fuel and fission product release, multiple restart capability and authenticity under credible accident scenarios. During the GE710/ANL programs the Cermet fuel form displayed tolerance to excessive temperature/power ramps due to the high strength and conductivity of the refractory metal matrix. Additionally Cermet fuel display complete compatibility in the expected hot H2 operating environment as well as cladding and fuel matrix CTE compatibility. Finally the XNIG2000 is based upon a fuel form that was successfully fabricated and tested.
PROPOSED FUEL ELEMENT CONFIGURATION HAS BEEN UPDATED

Current baseline
37 holes

Previous baseline
169 holes

Subsequent to the Mid-term reallocation of this study, the baseline concept has been updated to incorporate a fuel element based on demonstrated technologies. The baseline fuel form incorporates 37 large diameter coolant channels compared to 169 small diameter coolant channels initially considered for this concept. The max operating fuel temperature was maintained at 2800K, well within the experimental database. Because of the increased thermal path, fuel centerline to coolant channel surface, between the fuel form the reactor exit propellant temperature was reduced to 2600K from 2900K. This channel temperature provides an lap level of 900 seconds with life greatly in excess of the NASA requirements.
## Baseline Fuel Element Within Fabrication Experience

The selected baseline prismatic Cermet fuel element is based on demonstrated technology. The outer core fuel elements consist of 60 vol% UO2-Mo fuel matrix contained within Mo-50% Be external core. The inner core fuel elements consists of 60 vol% U233-W fuel matrix contained within a W-20% Be external core. Iteration has been incorporated into the external can design to decrease the ductile-to-brittle transition temperature and provide adequate ductility for cycle life requirements. All fuel elements have a hexagonal cross section with a 1.4 inch flat-to-flat distance and contain 37 coolant channels. 14 inch in diameter.

The coolant channels are coated with the refractory metal contained within the matrix. U233 is stabilized with 6% U3O8 in both cases to provide fuel stabilization and prevent fuel migration. Fuels elements of this type were successfully fabricated and tested in the early 70s with technology that can be easily recovered and enhanced with a core recent fabrication techniques.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Inner core element</th>
<th>Outer core element</th>
</tr>
</thead>
<tbody>
<tr>
<td>Number of fuel elements</td>
<td>61</td>
<td>60</td>
</tr>
<tr>
<td>Clearance across flats [in]</td>
<td>1.40</td>
<td>1.40</td>
</tr>
<tr>
<td>Diameter of outer hole [in]</td>
<td>14</td>
<td>14</td>
</tr>
<tr>
<td>Distance of outer holes [in]</td>
<td>0.215</td>
<td>0.215</td>
</tr>
<tr>
<td>Thickness of outer tube [in]</td>
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<td>0.057</td>
</tr>
<tr>
<td>Holes of outer tube [in]</td>
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<td>0.02</td>
</tr>
<tr>
<td>Number of core elements</td>
<td>37</td>
<td>37</td>
</tr>
<tr>
<td>Fuel material</td>
<td>UO2-W-Ce2O3</td>
<td>UO2-Mo-Ce2O3</td>
</tr>
<tr>
<td>Fuel core material</td>
<td>W</td>
<td>Mo</td>
</tr>
<tr>
<td>Flow tube material</td>
<td>99.4%</td>
<td>99.4%</td>
</tr>
<tr>
<td>Uranium enrichment (w%)</td>
<td>92.0</td>
<td>92.0</td>
</tr>
<tr>
<td>Vol. fraction of UO2 in fuel</td>
<td>0.9</td>
<td>0.9</td>
</tr>
<tr>
<td>Vol. fraction of Ce2O3 in fuel</td>
<td>0.06</td>
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<tr>
<td>Vol. fraction of Mo in fuel</td>
<td>0.34</td>
<td>0.34</td>
</tr>
<tr>
<td>Vol. fraction of Be in external</td>
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<td>0.55</td>
</tr>
<tr>
<td>Vol. fraction of wall in external</td>
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<td>Flow tube wall thickness</td>
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<td>Total core power [MW]</td>
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<td>Total core volume [L]</td>
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<td>Active core volume [L]</td>
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<td>Heat transfer per unit flow area</td>
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<tr>
<td>Fuel element height</td>
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<tr>
<td>Fuel element height</td>
<td>24.1</td>
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<tr>
<td>Coolant reflector [in]</td>
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<tr>
<td>Axial length of element [in]</td>
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<td></td>
</tr>
<tr>
<td>W wall thickness on inner core</td>
<td>1</td>
<td></td>
</tr>
</tbody>
</table>
Axial Reflectors Integral With Fuel Elements

The baseline prismatic Cerenk fuel element for the inner core is shown. The axial He-O reflector section is integral with the fuel element, contained within the same structural support external can. The attachment section is used in a support system. The loaded section of the fuel element is 24 inches (61 cm) in length and the axial reflector is 7.9 inches (20 cm).
HIGH STRENGTH FUEL ELEMENTS ALLOW SIMPLIFIED CORE SUPPORT

High Strength Fuel Elements Allow Simplified Core Support

The XM12000 is not acceptable to material neutron poisoning because of the fast spectrum operation of the reactor. Therefore, high strength refractory materials can be used to build the fuel matrix and support structure to eliminate the need for the coils. The baseline conceptual core support design is shown below. The fuel elements are simply supported, at the hydrogen inlet end, in the support plate with a threaded fuel element retainer. The fuel elements are placed in trimax because of propellant (nitrogen and gaseous) pressure drop which acts to increase the natural frequency of the fuel element and reduce the propensity for flow induced vibration.
CONCEPTUAL CORE ASSEMBLY APPROACH

Conceived Core Assembly Approach

Shown below is the conceptual inner and outer core fuel element support approach. The outer core elements are simply supported at their cold end by an inner grid plate which is bolted to the inner pressure vessel. The outer core elements are allowed to translate through the upper support plate to allow for axial thermal growth. The inner core fuel elements are rigidly attached at their cold end by the upper grid plate. The upper grid support plate is bolted to the inner pressure vessel with additional support provided by axial struts attached to the upper plenum head.

A tungsten shroud will be used between the two cores to act as a thermal buffer and provide a compressive spring preload against radial inner core fuel element growth. The tungsten shroud will conform to the hexagonal cross-section of the fuel elements and extend from the upper support plate to the nozzle channels. The shroud will transition from a hexagonal cross-section to a circular cross-section in the chamber region. The nozzle coolant tubes will run behind this shroud in a circular pattern to provide chamber cooling.
Axial Power Distribution Predicted For Thermal Hydraulic Analysis

The predicted average axial power shape factors for the inner and outer cores are shown. These power profiles were determined using the 3-D diffusion theory code, BOLD VENTURIS, and benchmarked with MATH-style statistical codes. The inner core power profile decreases at the exit of the reactor where the temperatures are the highest. The sharp increase in power at the inner core inlet is caused by the BeO axial reflector located directly above the reactor. These power profiles were determined to conduct a coupled neutronic/thermal hydraulic analysis of the XMG2000 reactor.
PEAK FUEL TEMPERATURE IS MAINTAINED BELOW 2900K

![Graph showing temperature distribution]

Peak Fuel Temperature is Maintained Below 2900K.

The calculated propellant, fuel surface, and fuel centerline temperature distribution within the XNR2000 reactor at full power operating condition is shown. The temperature distribution is plotted against the normalized reactor coolant flow path position, where 0.0 corresponds to the outer core inlet and 1.0 corresponds to the inner core exit. This temperature distribution was calculated using a one-dimensional complex thermal-hydraulic/thermal-mechanical analysis benchmarked with detailed 3-dimensional computational fluid dynamics (CFD) procedures. As shown in the figure, the maximum fuel temperature reached in the inner core is 2680K and 2000K in the inner core. These maximum fuel temperatures were selected for design operation to exceed life requirements and ensure positive fission product and fuel retention. A propellant chamber temperature of 2660K was calculated using a 2680K max fuel temperature on the upper limit.
DUAL PASS REDUCES FUEL TEMPERATURES AND AXIAL THERMAL GRADIENTS

The figure displays several benefits of the dual pass reactor flow configuration. The dual pass provides reduced axial thermal gradients in the fuel elements. As shown in the figure, a temperature gradient of 1500 K appears across the outer core elements and a gradient of 1100 K appears across the inner core elements, in the dual pass configuration. However, in a single pass configuration a temperature gradient of 2500 K appears across each fuel element. The dual pass flowpath reduces the axial thermal gradients of the elements by approximately 30%, reducing thermal stresses and increasing fuel tolerance to power cycling. Additionally, in a dual pass reactor the fuel temperatures are reduced by approximately 1500 K for equal propellant chamber temperatures and power density. This is due to increased heat flux and decreased convective heat transfer in the single pass configuration, for equivalent reactor power density levels.
Preliminary Upper Plenum CFD Results

A Computational Fluid Dynamic (CFD) analysis was conducted to evaluate the flow distribution and heat transfer in the XN92006 reactor coolant channels and upper plenum region. The predicted flow distribution in the upper plenum is shown below. The results of the CFD analysis were used in the upper plenum design and to benchmark the one-dimensional thermal hydraulic reactor analyses.
XNR2000 RADIAL POWER DISTRIBUTION CONFIRMED BY B&W

The calculated realistic normalized power distribution within a segment of symmetry of the XNR2000 reactor is shown. Close agreement between the calculated results of Pratt & Whitney and Babcock & Wilcox is shown. As expected, the maximum power peaks of the inner core appear at the center of the reactor while the power peak of the outer core appears closest to the radial reflector. These results were used in the thermal hydraulic analysis to conduct power/flow matching evaluations. As shown in the diagram, the maximum peak-to-average fuel element power level was calculated to be 1.00 for both the inner and outer cores.
UPPER PLENUM MIXING FLATTENS
RADIAL POWER PROFILE

The calculated propellant and fuel centerline temperature distribution within the XRG2000 reactor for fuel elements having the average and maximum peak-to-average power levels is shown. The calculated radial power distribution, shown in the previous chart, was used to conduct a thermal hydraulic evaluation of the reactor to determine the impact of peak power levels on reactor temperatures. The analysis displays the upper plenum mixing advantage of the dual pass core. Any outer core hot bypassing effect due to uneven power profiles is removed from the inner core because of thermal momentum fluid mixing in the upper plenum. This mixing reduces the propellant and therefore reactor temperatures in the inner core. The energy and momentum mixing allows for up to 10% power peaking in the outer core without overfiring. As shown, the maximum temperature is approximately 2000 kelvin for the inner core and 2200 kelvin for the outer core in the fuel elements having the maximum power levels. This analysis displays the worst case scenario in which an attempt is made to flatten the power profile.
**PROFILE CAN BE ADDRESSED BY VARIABLE ENRICHMENT OR ORIFICING**

Two methods of addressing the power profile were evaluated and both were found to be acceptable. The first approach of handling the variable power profile was utilizing the propellant flow in the inner core to provide a constant 2670K reactor exit temperature. By ensuring the flow at the inlet of each fuel element the proper flow rate can be delivered to each element depending on the element power level. Shown below is the fuel element flowrate, as a function of inner core radius, required to provide a constant reactor exit temperature.

The second possible approach to flatten the power profile evaluated was variable radial Uranium enrichment. The enrichment within both the inner and outer cores was varied to determine the impact on radial power distribution. As shown in the figure a nearly constant power profile was obtained by varying the enrichment by approximately 4% across the reactor radius.
DUAL PASS CONFIGURATION HAS SIGNIFICANT ADVANTAGES

- Flat radial power profile
- Positive flow/power matching
- Upper plenum mixing reduces peak temperature
- High temperature inner core isolation
- Reduced element axial thermal gradient

* Dual Pass Configuration Has Significant Advantages

The primary attractive features provided by the dual pass reactor core are summarized. A flat radial power profile is provided by the dual-pass reactor due to the averaging of power distributions relative to two distinct regions. Positive flow/power matching is achievable because of the separation of the inner and outer cores. The maximum fuel element power shape factors appear in the outer core region because of the proximity of the radial reflector. However, because the outer core serves as the first pass, the core's hydrogen propellant passes through the outer core and eliminates fuel temperature concerns. Additionally, upper plenum mixing of the hydrogen serves to eliminate the outer core power peaks from the inner core fuel elements. The dual pass configuration isolates the hot inner core fuel elements from the rest of the engine system. This isolation provides material flexibility allowing the use of lighter weight beryllium based fuel elements in the outer core and a beryllium radial reflector which provides the most reactivity worth for the weight. The most obvious benefit of the dual pass core is the reduced axial thermal gradients and consequently thermal stress loads placed on the fuel elements.
COMPPLICATIONS OF H₂ MODERATION ELIMINATED DURING STARTUP, SHUTDOWN, AND THROTTLING

There is no impact of hydrogen moderation on the fast spectrum X5R2000 reactor. The calculated effect of hydrogen density on system $K_{eff}$ is shown. The complications of reactivity feedback from the hydrogen propellant and potential for thermal instability is eliminated during transient and steady state operation in the X5R2000.
PEAK FUEL TEMPERATURE DECREASES AT THROTTLED CONDITIONS

Peak Fuel Temperature Decreases At Throttled Conditions

The calculated propellant and fuel centerline temperatures are shown for the baseline XRG2000 at full thrust, 25% thrust, and 10% thrust throttled conditions. As displayed in the chart, the peak fuel temperatures within the reactor decrease as the engine is throttled. The reduced reactor temperatures result from the reduced power flux required to deliver the throttled mass flow rate to the design point temperature levels. This steadiness analysis was simplified because of the negligible effect of HX moderation on the reactivity of the core.
STARTUP, SHUTDOWN AND THROTTLING, UNAFFECTED BY H₂ MODERATION

The pump and turbine operating map of the XNR2000 is shown for throttled and design point conditions. The 10,10 upper stage expander cycle rocket engine turbopump characteristics were assumed in this analysis. This analysis indicates that the configuration allows throttling to at least 10% thrust at design specific impulse.
Individual Turbopump Requirements Are Similar to RL10

Demonstrated characteristics of the RL10 engine turbopumps will be required of turbopumps used in an NREL for manned space exploration missions. The XNR2000 conceptual design employs a two-stage centrifugal pump that is similar in flow rate and head rise to the RL10 turbopump, driven by a turbine operating at real inlet temperature. The system requirements call for throttling to at least 25% throttle at rated temperature and operation at low NPSH levels. These requirements are similar to those of the RL10 liquid hydrogen turbopumps. The RL10 turbopumps deliver pressure-rated fittings in the RL10 engine for upper stage applications.

This pump has successfully demonstrated zero to low NPSH capability and throttling down to 25% flow. With the introduction of hydrostatic bearings, operation in a radiation environment can be achieved because of the aluminum construction. For these reasons the characteristics of the RL10 turbopump were used in the study of the XNR2000 concept, and that a scaled or derivative version of this proven pump would be employed in the design.
NOZZLE IS ACTIVELY COOLED COPPER WITH AN UNCOOLED SKIRT

Regen section
- Coolant configuration: Two pass
- Number of tubes: 300
- Tube material: Glidcop
- Max heat flux: 51 Btu/in²/sec
- Max tube temperature: 811K (1452°F)
- Pressure drop: 225 psi

Skirt
- Coolant configuration: Radiation
- Skirt material: Columbium
- Max heat flux: 1 Btu/in²/sec
- Max skirt temperature: 1792K (3293°F)

Nozzle Is Actively Cooled Copper With An Uncooled Skirt

The XN92100 employs a regeneratively cooled chamber and nozzle and radially cooled nozzle skirt. The nozzle and chamber is cooled to an area ratio of 33 with 300 copper tubes in a two pass configuration with 30% of the total engine flow. The chamber pressure vessel consists of a 347 Stainless Steel jacket surrounding the copper tubes. The system employs a Columbium nozzle skirt from an area ratio of 33 to 200 which is radiatively cooled.
XN2000 Pressure Vessel is Similar to ANL Approach

The XN2000 employs an outer uncooled pressure vessel which surrounds the radial reflector and a regeneratively cooled inner pressure vessel which surrounds the reactor. The pressure vessel material considered is Inconel 718. Because the inner pressure vessel is subjected to a collapsing pressure of approximately 900 psi, longitudinal radial support ribs would be employed to transmit this load in the outer vessel. The radial support ribs would serve to separate and house the annular reflector segments. The two pressure vessels are capped at the top of the reactor by hemispherical heads. Hydrogen exits the reflector region and flows between the primary and secondary heads to cool the primary head covering the inner pressure vessel and provide additional heat input to the turbine.
**XNR2000 Baseline Design Exceeds NASA Requirements**

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Baseline</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Thrust (lb)</td>
<td></td>
<td>25,000</td>
</tr>
<tr>
<td>Isp (sec)</td>
<td></td>
<td>900</td>
</tr>
<tr>
<td>T/W</td>
<td></td>
<td>5.3</td>
</tr>
<tr>
<td>Reactor power (MW)</td>
<td></td>
<td>5.10</td>
</tr>
<tr>
<td>Power density (MW/m^3)</td>
<td></td>
<td>9.4</td>
</tr>
<tr>
<td>Max fuel temp (K)</td>
<td></td>
<td>2,660</td>
</tr>
<tr>
<td>Chamber temp (K)</td>
<td></td>
<td>766</td>
</tr>
<tr>
<td>Chamber pressure (psia)</td>
<td></td>
<td>27.8</td>
</tr>
<tr>
<td>Total flow (lb/sec)</td>
<td></td>
<td>1,460</td>
</tr>
<tr>
<td>Pump tip speed (ft/sec)</td>
<td></td>
<td>22.7</td>
</tr>
<tr>
<td>Turbine inlet temp (K)</td>
<td></td>
<td>222</td>
</tr>
<tr>
<td>Nozzle area ratio</td>
<td></td>
<td>200</td>
</tr>
<tr>
<td>Nozzle exit dia (in)</td>
<td></td>
<td>5.8</td>
</tr>
<tr>
<td>Max engine length (ft)</td>
<td></td>
<td>15.3</td>
</tr>
<tr>
<td>Stowed engine length (ft)</td>
<td></td>
<td>11.0</td>
</tr>
<tr>
<td>No. of inner fuel elements</td>
<td></td>
<td>61</td>
</tr>
<tr>
<td>No. of outer fuel elements</td>
<td></td>
<td>90</td>
</tr>
<tr>
<td>Throttling at design Isp (%)</td>
<td></td>
<td>10</td>
</tr>
</tbody>
</table>

**Pratt & Whitney XNR2000:**

The table displays the cycle performance information of the baseline XNR2000. The baseline XNR2000 delivers 25,000 lb of thrust at a specific impulse of 900 sec, with a thrust-to-weight ratio of 5.3. This power balance information was generated using the Marshall Space Flight Center/JSC/W Rocket Engine Transient Simulation (MOTRAN) System.
## XNR2000 ENGINE PERFORMANCE

**Thrust** = 25,000 lbf  
**T/W** = 5.3  
**Isp** = 900.0 sec

### PROPELLANT FLOW ENGINE STATION CONDITIONS

<table>
<thead>
<tr>
<th>Station Location</th>
<th>Pressure (psia)</th>
<th>Temperature (Deg K)</th>
<th>Flow (lbm/s)</th>
<th>Enthalpy (Btu/lbm)</th>
<th>Density (lbm/ft**3)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Engine Inlet</td>
<td>26.7</td>
<td>20.6</td>
<td>14.0</td>
<td>-108.0</td>
<td>4.38</td>
</tr>
<tr>
<td>Pump Inlet</td>
<td>25.7</td>
<td>20.6</td>
<td>14.0</td>
<td>-108.0</td>
<td>4.38</td>
</tr>
<tr>
<td>Pump Exit</td>
<td>2179.3</td>
<td>34.7</td>
<td>14.0</td>
<td>13.0</td>
<td>4.56</td>
</tr>
<tr>
<td>Nozzle Coolant Inlet</td>
<td>2157.6</td>
<td>34.8</td>
<td>8.4</td>
<td>13.0</td>
<td>4.55</td>
</tr>
<tr>
<td>Reflector Coolant Inlet</td>
<td>1932.6</td>
<td>103.1</td>
<td>28.1</td>
<td>-440.9</td>
<td>1.77</td>
</tr>
<tr>
<td>Turbine Inlet</td>
<td>1901.6</td>
<td>226.9</td>
<td>11.8</td>
<td>1343.7</td>
<td>0.80</td>
</tr>
<tr>
<td>Turbine Exit</td>
<td>1218.2</td>
<td>207.2</td>
<td>11.8</td>
<td>1199.9</td>
<td>0.58</td>
</tr>
<tr>
<td>Outer Core Inlet</td>
<td>1108.9</td>
<td>210.4</td>
<td>27.8</td>
<td>1221.6</td>
<td>0.52</td>
</tr>
<tr>
<td>Inner Core Inlet</td>
<td>956.3</td>
<td>1659.4</td>
<td>27.8</td>
<td>88650.0</td>
<td>0.06</td>
</tr>
<tr>
<td>Chamber</td>
<td>765.9</td>
<td>2668.7</td>
<td>27.8</td>
<td>181883.3</td>
<td>0.03</td>
</tr>
</tbody>
</table>

### 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
- Inner Core Fuel Mt'l: W-UO2,61
- Inner Core Fuel Mt'l: Mo-UO2,90
- 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: Ch 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
- DHT Actual: 143.8 Btu/lb
- AN**2(E-08): 193
- Mean Dia.: 4.66 in
**Operation at 2500K Can Be Accommodated Within Baseline Configuration**

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Baseline</th>
<th>25,000</th>
</tr>
</thead>
<tbody>
<tr>
<td>Thrust (lb)</td>
<td>25,000</td>
<td>25,000</td>
</tr>
<tr>
<td>Isp (sec)</td>
<td>965</td>
<td>965</td>
</tr>
<tr>
<td>T/W</td>
<td>5.3</td>
<td>5.3</td>
</tr>
<tr>
<td>Reactor power (Mw)</td>
<td>540</td>
<td>492</td>
</tr>
<tr>
<td>Power density (Mw/L)</td>
<td>9.4</td>
<td>9.1</td>
</tr>
<tr>
<td>Max fuel temp (K)</td>
<td>2,800</td>
<td>2,740</td>
</tr>
<tr>
<td>Chamber temp (K)</td>
<td>2,600</td>
<td>2,500</td>
</tr>
<tr>
<td>Chamber pressure (psia)</td>
<td>1.16</td>
<td>758</td>
</tr>
<tr>
<td>Total flow (lb/sec)</td>
<td>27.8</td>
<td>28.9</td>
</tr>
<tr>
<td>Pump tip speed (ft/sec)</td>
<td>1,400</td>
<td>1,482</td>
</tr>
<tr>
<td>Turbine inlet temp (K)</td>
<td>227</td>
<td>216</td>
</tr>
<tr>
<td>Nozzle area ratio</td>
<td>200</td>
<td>200</td>
</tr>
<tr>
<td>Nozzle exit dia (ft)</td>
<td>5.8</td>
<td>5.8</td>
</tr>
<tr>
<td>Max engine length (ft)</td>
<td>15.3</td>
<td>15.3</td>
</tr>
<tr>
<td>Skewed engine length (ft)</td>
<td>11.0</td>
<td>11.0</td>
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<tr>
<td>No. of inner fuel elements</td>
<td>61</td>
<td>61</td>
</tr>
<tr>
<td>No. of outer fuel elements</td>
<td>90</td>
<td>90</td>
</tr>
<tr>
<td>Ithrottling at design Isp (%)</td>
<td>10</td>
<td>10</td>
</tr>
</tbody>
</table>

**Operation at 2500K Can Be Accommodated Within Baseline Configuration**

The baseline cycle information is displayed and compared to the XNG2000 engine operating at a chamber temperature of 2500K. The power balance for both cycle points was preserved by requiring the reactor exit Mach numbers to equal 0.3 and deliver 25,000 lb. of thrust.
PRELIMINARY ENGINE CLUSTERING STUDY INDICATES LIMITED NEUTRONIC INTERACTION

A conservative engine clustering model was developed and the reactivity was evaluated for a cluster of three XNR2660 baseline engines as a function of separation distance. The separation distance is defined as shown in the figure. As displayed in the chart, core neutronic coupling was found to have no effect in clustering engines for distances required to account for nozzle skirts.
Reentry & Worst Case Accident Scenario Criticality Analysis

A 16-group diffusion code (VENTURE/THIMBLE) analysis was conducted to determine worst case accident scenario criticality. The negative reactivity insertion is shown for several accident Scenario Core conditions. The KINF000 would go subcritical for all normal conditions evaluated. The largest negative reactivity insertion occurred for water immersion. The impact of chisena core in tubes surrounding the water core denizens was also evaluated and found to provide adequate negative reactivity insertion to be utilized as a potential back-up safety mechanism. The blown reflector analysis was conducted assuming that 10% of the total reflector was removed from the system.
Design Allows Thrust Flexibility With Common Fuel Elements

The XN12000 was configured to provide thrust flexibility. The system can provide thrust ranging from approximately 20,000 lb to 50,000 using the same fuel element design, core configuration, and support technology by simply varying the number of inner and outer core fuel elements.
Reactor Neutronics Behavior Similar Over Thrust Range

Radial fission density (power) for various thrust size

- 75k lb (93% enrichment)
- 50k lb (99% enrichment)
- 75k lb (99% enrichment)

Reactor Neutronics Behavior Similar Over Thrust Range

Radial power profiles for three XNR2000 core sizes are shown. The 50,000 lb and 75,000 lb can be made critical with 70% enriched fuel at the fuel-metal volume ratio of 60/40.
**XNR2000 Cycle Parameters Are Similar for Various Thrust Sizes**

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Thrust (lb)</th>
<th>25,000</th>
<th>50,000</th>
<th>75,000</th>
</tr>
</thead>
<tbody>
<tr>
<td>isp (sec)</td>
<td>900</td>
<td>901</td>
<td>897</td>
<td></td>
</tr>
<tr>
<td>T/W</td>
<td>5.3</td>
<td>6.6</td>
<td>7.9</td>
<td></td>
</tr>
<tr>
<td>Reactor power (Mw)</td>
<td>511</td>
<td>1,022</td>
<td>1,513</td>
<td></td>
</tr>
<tr>
<td>Power density (MW/L)</td>
<td>9.4</td>
<td>9.1</td>
<td>11.1</td>
<td></td>
</tr>
<tr>
<td>Max fuel temp (K)</td>
<td>2,880</td>
<td>2,880</td>
<td>2,880</td>
<td></td>
</tr>
<tr>
<td>Chamber temp (K)</td>
<td>2,669</td>
<td>2,676</td>
<td>2,657</td>
<td></td>
</tr>
<tr>
<td>Chamber pressure (psia)</td>
<td>766</td>
<td>735</td>
<td>836</td>
<td></td>
</tr>
<tr>
<td>Total flow (lb/sec)</td>
<td>27.8</td>
<td>55.5</td>
<td>83.6</td>
<td></td>
</tr>
<tr>
<td>Pump tip speed (lb/sec)</td>
<td>1,469</td>
<td>1,527</td>
<td>1,728</td>
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</tr>
<tr>
<td>Turbine inlet temp (K)</td>
<td>227</td>
<td>230</td>
<td>257</td>
<td></td>
</tr>
<tr>
<td>Nozzle area ratio</td>
<td>200</td>
<td>200</td>
<td>200</td>
<td></td>
</tr>
<tr>
<td>Nozzle exit dia (in)</td>
<td>5.8</td>
<td>8.3</td>
<td>9.5</td>
<td></td>
</tr>
<tr>
<td>Max engine length (ft)</td>
<td>15.3</td>
<td>20.3</td>
<td>22.7</td>
<td></td>
</tr>
<tr>
<td>Stowed engine length (ft)</td>
<td>11.0</td>
<td>12.4</td>
<td>12.0</td>
<td></td>
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<tr>
<td>No. of inner fuel elements</td>
<td>61</td>
<td>127</td>
<td>169</td>
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<tr>
<td>No. of outer fuel elements</td>
<td>90</td>
<td>186</td>
<td>210</td>
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<tr>
<td>Thrusting at design isp (%)</td>
<td>10</td>
<td>TBD</td>
<td>TBD</td>
<td></td>
</tr>
</tbody>
</table>

**Reactor Thermal Hydraulics at 50K Are Similar to Baseline**

![Graph showing reactor thermal hydraulics](image)
REACTOR THERMAL HYDRAULICS AT
75K THRUST ARE SIMILAR TO BASELINE
## CERMET ENGINE WEIGHT
### SUMMARY VS THRUST SIZE

<table>
<thead>
<tr>
<th>Thrust level</th>
<th>25,000 lb</th>
<th>50,000 lb</th>
<th>75,000 lb</th>
</tr>
</thead>
<tbody>
<tr>
<td>Inner core</td>
<td>940</td>
<td>1,882</td>
<td>2,212</td>
</tr>
<tr>
<td>Outer core</td>
<td>937</td>
<td>1,944</td>
<td>1,856</td>
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<tr>
<td>Support structure</td>
<td>115</td>
<td>250</td>
<td>425</td>
</tr>
<tr>
<td>Internal shield</td>
<td>250</td>
<td>300</td>
<td>310</td>
</tr>
<tr>
<td>Axial reflector</td>
<td>50</td>
<td>80</td>
<td>100</td>
</tr>
<tr>
<td>Radial reflector and control</td>
<td>500</td>
<td>800</td>
<td>1,000</td>
</tr>
<tr>
<td>Valves and controller</td>
<td>425</td>
<td>525</td>
<td>590</td>
</tr>
<tr>
<td>Pressure vessel</td>
<td>550</td>
<td>800</td>
<td>1,000</td>
</tr>
<tr>
<td>Upper core assembly</td>
<td>220</td>
<td>300</td>
<td>400</td>
</tr>
<tr>
<td>Nozzle skirt</td>
<td>250</td>
<td>500</td>
<td>750</td>
</tr>
<tr>
<td>Turbopump</td>
<td>75</td>
<td>125</td>
<td>175</td>
</tr>
<tr>
<td>Thrust structure</td>
<td>440</td>
<td>600</td>
<td>700</td>
</tr>
<tr>
<td>Total engine (lb)</td>
<td>4,752</td>
<td>7,586</td>
<td>9,518</td>
</tr>
<tr>
<td>T/W</td>
<td>5.26</td>
<td>6.59</td>
<td>7.88</td>
</tr>
</tbody>
</table>

A Weight Summary of the XNR2000 for the thrust levels evaluated is shown. The thrust-to-weight ratios for the XNR2000 are high relative to other conventional RTHE's. Several features contribute to the high thrust-to-weight of the XNR2000. The XNR2000 can operate at a high power density because of the high conductivity of the Cermet fuel and the thermal fluid mixing in the upper plenum. The laser spectrum provides a compact core with no moderator material and the high strength refractory metal fuel elements allow a lightweight support structure. The use of refractory methods and the compact core design reduces required shielding weight. Additionally, the separation of the reactor into two engines allows the use of a lightweight Molybdenum based matrix in the upper core. These features of the XNR2000 RTHE contribute to the high thrust-to-weight.
CERMET APPROACH PROVIDES HIGH PERFORMANCE AND LOW RISK

Thrust = 25,000 lb
Isp = 900 sec
T/W = 5.3
Diameter = 5.8 ft
Stowed length = 11.0 ft
Deployed length = 15.3 ft

Thrust = 50,000 lb
Isp = 901 sec
T/W = 6.6
Diameter = 8.3 ft
Stowed length = 12.4 ft
Deployed length = 20.3 ft

Thrust = 75,000 lb
Isp = 697 sec
T/W = 7.9
Diameter = 9.5 ft
Stowed length = 12.0 ft
Deployed length = 22.7 ft

PRATT & WHITNEY XNR2000 CERMET NTRE

Cermet Approach Provides High Performance and Low Risk

A conceptual NTRE, the XNR2000, has been presented that is powered by a fuel spectrum, cermet fueled reactor core. The baseline XNR2000 system delivers 25,000 lb of thrust at a specific impulse of 500 seconds and thrust to weight of 5.3. The distinguishing features of this system are the dual present reactor configuration and fuel spectrum, cermet fueled reactor. These features have been incorporated into the design, as well as knowledge gained from the MSFC/NEI/VA, T9710 and ANL programs, to develop a safe and robust Nuclear Thermal Rocket Engine for manned space exploration missions.

NTP: System Concepts 200 NP-TIM-92
XNR2000 NEUTRONICS ARE BENCHMARKED AND CONFIRMED

- Design analysis methodology
- Benchmark analysis and criticality summary
- Power profiles
- Reactivity and control system
- Neutron and gamma-ray fluence
- Inherent safety features
MODELS DEVELOPED TO ACCURATELY PREDICT REACTOR NEUTRONICS

PRATT & WHITNEY XNR2000 CLIMATE NTNF

MODELS DEVELOPED TO ACCURATELY PREDICT REACTOR NEUTRONICS

A three dimensional model of XNR2000 core is developed. Thirty radial and axial mesh regions and 6 axial zones are used to model the details of the inner core, the outer core, the inter axial core reflector and the lateral support structure are included in the model. Six axial zones are used to address the axial temperature gradient in the inner and the outer cores. Group average neutron cross-sections for all 186 regions are generated at their average operating temperatures. Each region is divided into tens of finite volumes for the calculation of flux and effective multiplication factors.
ANALYSIS METHODOLOGY
TAILORED TO FAST SPECTRUM

• Multigroup cross-sections generated by COMBINE (ENDFB–V)

• MCNP (4.2) used for complex geometries

• BOLD VENTURE (3-D diffusion) used for power profile and reactivity

• ANISN (1-D, Sn) used for analysis of heterogeneous boundaries

• Results benchmarked with GE 710 testing

• Results independently confirmed by B&W and ANL
Material list

1 Core U, W, Ta, Al, O
2 Tube Sheet 303 SS
3 Tube Sheet and Mo 303 SS, Mo
4 Mo Transition Mo
5 Mo Plug Mo, Ta, W
6 Cladding Ta, W
7 Be (.85), Al
8 Shell 303 SS
9 Transition Al
10 Inner Reflector Zone Be (.85), Al
11 Outer Reflector Zone Be (.90), Al
12 Gap

Venture Model Generated For GE710 Mockup 1A Decodework

GE710 program Mockup 1A critical configuration was used to benchmark the 3-D, 16-group Combine/VENTURE and continuous energy MCNP 4.2 models. Mockup 1A features core physics characteristics comparable with 25000 lb XNR2000 engine design. The materials concentrations and core dimensions are taken directly from the GEMP 442 report.
OUTSTANDING PREDICTION ACCURACY IS HARD TO BELIEVE

Outstanding Prediction Accuracy

VENTURE/COMBINE calculated values of the normalized radial power profile compare well with the OS710 experimental results. Both experimental and calculated power profiles are normalized to the power level at the radial distance of 5 cm from the centerline. The calculated values at the radial power density beyond the last measurement point are not shown. The measured value of the relative power density is 4.2 where the COMBINE/VENTURE calculated maximum radial power density is 8.3. The maximum power density close to the reflector is very sensitive to the position.
VENTURE/COMBINE Calculation of GE 710 Mockup 1A

COMBINE/VENTURE calculated values of the average axial power profile compares well with GE710 Mockup 1A experimental results. Two experimental points at the top and bottom of the reactor are excluded. Large uncertainty in experimental data at the unreflected end of the Mockup 1A reactor. Additionally, the VENTURE/COMBINE calculated value of $K_{eq}$, 0.991, compares well with the measured value of 1.000.
### XNR2000 Baseline Core Criticality Independently Confirmed

<table>
<thead>
<tr>
<th>Venture/Combine MCNP (P&amp;W)</th>
<th>MCNP (P&amp;W)</th>
<th>MCNP (B&amp;W)</th>
<th>MCNP (ANL)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Keff</td>
<td>1.0183 (24 groups)</td>
<td>1.021</td>
<td>1.025</td>
</tr>
<tr>
<td></td>
<td>1.0183 (16 groups)</td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>1.0210 (12 groups)</td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>1.0601 (8 groups)</td>
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<tr>
<td></td>
<td>1.0559 (4 groups)</td>
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</tr>
</tbody>
</table>

- Good agreement between 2-D, 16 groups diffusion calculation and MCNP
- Good agreement between independently performed MCNP calculations

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XNR2000 Baseline Core Criticality Independently Confirmed

The 16 group COMMON/VENTURE $K_{eff}$ calculation of the XNR2000 core shows good agreement with MCNP calculated values of $K_{eff}$. Pratt & Whitney MCNP calculations are for a minimum of 500,000 histories. BNL and ANL Laboratory calculations of XNR2000 core $K_{eff}$ are based on a minimum of 100,000 histories. The small differences between MCNP calculated results are due to slightly different number densities and cross-section libraries used.
16 Group Accurately Models Spectrum

The selection of neutron energy groups is influenced by the location of isolated and non-isolated resonances of uranium, tungsten, molybdenum, and the energy threshold for the 14 MeV neutron. With a carefully selected energy partition, 12 group calculation proved to be adequate. The optimum choice of energy partition for 16 group calculation is shown and was used in all reactor studies.
**XNR2000 Radial Flux Profiles**

Radial flux profiles for the XNR2000 baseline core are presented. Both transport and diffusion theory are used to calculate total neutron flux for energy groups 1, 4, 10 and 15. The difference in the calculated value of flux at the vicinity of the reflector-core boundary is due to the use of the approximation used in the diffusion theory. There is also a noticeable difference between predicted values of flux for intermediate and low energy neutron groups in the reflector.
<table>
<thead>
<tr>
<th>Reactor location</th>
<th>Normalized peak power</th>
<th>Reflector raised</th>
<th>$K_{eff} = 1.000$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Top</td>
<td>.72</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Middle</td>
<td>1.0</td>
<td>1.016 cm</td>
<td></td>
</tr>
<tr>
<td>Bottom</td>
<td>.82</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

Constant Enrichment 3-D Power Profile At Criticality

Radial power profile at different axial location of X1R25000. Ten degree reflector segments are raised for 1.016 cm to achieve $K_{eff} = 1.000$. Uniform radial power profile at the axial location of 50 cm is due to the cussed reflector.
HIGH REFLECTOR WORTH ENABLES ROBUST BASELINE CONTROL APPROACH

The 25,000 lb XNR2000 baseline engine is powered by a compact fast reactor. The neutron leakage from the core region to the reflector is very significant. One of the options to control the reactor is the axial movement of the 10" reflector segments. Large reflector segments (5") can be used for large insertion of negative reactivity and reactor shutdown. The reflector worth calculations were conducted as a function of distance using for a bank of six 5" reflector segments. The shutdown calculations were conducted for six 10" reflector segments.
### Reactor Design Provides Robust Reactivity and Control Margin

#### Reactivity effect

<table>
<thead>
<tr>
<th>Effect</th>
<th>Reactivity $\Delta k$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Temperature effect (30% 3000k)</td>
<td>$-0.6 \pm 0.3$</td>
</tr>
<tr>
<td>Fuel burnup (6000 mw-hr)</td>
<td>$-0.1 \pm 0.03$</td>
</tr>
<tr>
<td>Required excess Reactivity (maximum)</td>
<td>$+1.0$</td>
</tr>
<tr>
<td>Design excess reactivity</td>
<td>$2.0 \pm 0.5$</td>
</tr>
</tbody>
</table>

#### Control system requirements

<table>
<thead>
<tr>
<th>Requirement</th>
<th>Value (k)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Installed reactivity (maximum)</td>
<td>2.5</td>
</tr>
<tr>
<td>Minimum scram requirements</td>
<td>2.5</td>
</tr>
<tr>
<td>Minimum required control system worth</td>
<td>5.0</td>
</tr>
<tr>
<td>Design control system worth</td>
<td>10.0</td>
</tr>
</tbody>
</table>
CONSTANT ENRICHMENT 3-D POWER PROFILE WITH RAISED REFLECTORS

Small reflectors raised 5 cm

Constant Enrichment 3-D Power Profile With Raised Reflectors

Power lift due to the axial displacement of ten degree reflector segments is shown. The 10 degree reflector segments are raised by 5 cm. The COMSOL MODELLER computer code system is used to calculate 3-D power distribution in the mid-core regime. The power peaking of the core central rods is increased due to the significant leakage loss of neutrons through the opening in the radial reflector.
**REACTOR HAS DESIRABLE NEGATIVE TEMPERATURE COEFFICIENT**

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**Inner Core**

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**Outer Core**

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**Reactor Has Desirable Negative Temperature Coefficient**

- Temperature coefficient of reactivity is very small.
- Inner core fuel temperature coefficient is one order of magnitude smaller than the outer core fuel temperature coefficient.
- Outer core fuel temperature is comparable with GRT10 Markups 1A fuel temperature coefficient.

**XNR2000**

- **Inner Core**: $\nu (\theta 2250)$ = -8.0 x $10^{-6}$ AR/K
- **Outer Core**: $\nu (\theta 2250)$ = -6.8 x $10^{-7}$ AR/K

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**NTP: System Concepts** 214 **NP-TIM-92**
# Adequate Internal Shielding Included in Design

<table>
<thead>
<tr>
<th>Neutrons</th>
<th>XNR2000</th>
<th>NASA limits</th>
</tr>
</thead>
<tbody>
<tr>
<td>Fast neutron flux (E&gt;1.0 mev)</td>
<td>(8.0±2.0) x 10^10</td>
<td>2.0 x 10^{12}</td>
</tr>
<tr>
<td>Intermediate energy neutron flux</td>
<td>(2.4±.6) x 10^{12}</td>
<td>3.0 x 10^{12}</td>
</tr>
<tr>
<td>Thermal neutron flux</td>
<td>(3.6±.9) x 10^{11}</td>
<td>6.0 x 10^{11}</td>
</tr>
</tbody>
</table>

**Gamma - rays**

Model results indicate gamma-ray fluence is very sensitive to system geometry. A refined estimation of gamma-ray fluence will require further definition of configuration and constraints.

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**Adequate Internal Shielding Included in Design**

MCNP is used to calculate the fast, intermediate and thermal neutron fluxes at the upper part of the core. Fast and thermal neutron fluxes are significantly lower than the limits specified for the baseline design. Accurate estimation of the gamma-ray flux at the upper part of the shielded core require more detailed information on the upper core structural materials.
• State of the art analysis techniques employed to ensure design criticality, controllability and safety

• High confidence provided by benchmark analysis and independent evaluations by B&W and ANL

• Evaluation of all major reactor issues confirm advantages and flexibility of baseline approach
Babcock & Wilcox
Assessment of the
Pratt & Whitney XNR2000

Babcock & Wilcox
Space Systems Engineering


October 1992
Babcock & Wilcox Assessment of the Pratt & Whitney XNR2000

Babcock & Wilcox
Space Systems Engineering

Pratt & Whitney contracted with Babcock & Wilcox Advanced Systems Engineering/Space Systems Engineering to provide engineering support services for their NASA SEI Task Order Contract. Among other things, B&W is a reactor system vendor with physics, thermal hydraulics, materials, systems, mechanical engineering and manufacturing capabilities. B&W is also the operator of the only commercial facility licensed to manufacture large quantities of highly enriched reactor fuel.
Introduction

**Scope of B&W Efforts**
- Fuel Element Fabricability Assessment
- Mechanical Design Review
- Neutronics Analysis Review
- Safety Assessment

**Results of Mechanical and Physics Reviews Included in P&W and U of F Presentations**

B&W performed four subtasks for P&W as follows:

1) Fuel Element Fabricability Assessment - An assessment of the fabricability and manufacturability of CERMET fuel elements and the recoverability of the applicable technology.

2) Mechanical Design Review - An overall review of the reactor system from a mechanical engineering standpoint.

3) Neutronics Analysis Review - A review of the neutronics calculations performed for P&W by the University of Florida.

4) Safety Assessment - An overall assessment of the reactor system from a safety point of view.

The results of the mechanical and physics reviews have been integrated into the design and previously presented. The results of the fuel and safety assessments are presented here.
CERMET Fuel Fabricability Assessment

The fabricability of CERMET fuel elements is a major issue in the design of the XNR2000 reactor. The reactor uses both tungsten and molybdenum based UO$_2$ CERMET fuel elements. Most work on CERMET fabricability has focussed on tungsten based fuel elements. Since tungsten based CERMET fuel elements are more difficult to fabricate, it is reasonable to assume that if they are fabricable, then molybdenum based CERMET fuel elements will also be fabricable. The same issues and considerations that apply to tungsten based fuel elements will also apply to molybdenum based fuel elements.
CERMET Experience

CERMET based fuel elements have been proposed for several programs in the past. The Aircraft Nuclear Propulsion program provided early experience with CERMETs. Some fuel was made and tested, but it was not a prismatic form. In the General Electric 710 program, prismatic 37 coolant channel CERMET fuel elements, similar to those proposed for use in the XNR2000, were constructed using at least two different techniques. Extensive testing was performed and documentation of these efforts is good. The Multimegawatt program demonstrated recovery of the ANL Nuclear Rocket Program technology. During the course of the 710 program and the Nuclear Rocket Program, significant testing of CERMET fuels was performed including: high temperature ex-core testing, high temperature in-core testing, hot hydrogen flow testing and thermal shock testing. The test results were positive. Little swelling or leakage was observed and microstructural integrity was maintained. CERMET fuel cladding integrity and its ability to retain the fuel was also verified.

The CERMET technology development that has been performed forms a good basis for the necessary follow-on work. The past work should be integrated with current technology, where appropriate, and a demonstration fuel element should be fabricated using depleted uranium or a surrogate fuel material.
CERMET Fuel Testing

- Over 100 Partial and Full Length 7, 19, 37 and 91 Channel Fuel Elements Fabricated and Tested (plus hundreds of additional test samples)
- Greater Than 300,000 Sample Test Hours Accumulated (Fuel Element Qualification Program, >120000 in-core and >180000 ex-core)
- Thermal Cycling Tests (up to 2444 K, 100 thermal cycles and 100 hours at temperature)
- Thermal Shock Tests (in-core, up to 16000 K/sec, 2870 K maximum temperature)

**Selected Test Results**

<table>
<thead>
<tr>
<th>Temperature (K)</th>
<th>In Core</th>
<th>In Core</th>
<th>In Core (4 samples)</th>
<th>Ex Core</th>
<th>Ex Core</th>
<th>Ex Core</th>
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<tr>
<td>(LTF 1)</td>
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<td>1923</td>
<td>1673 to 2273</td>
<td>2773</td>
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<td>(LTF 111)</td>
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<tr>
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<td>(LTF 11111111111)</td>
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<tr>
<td>(LTF 111111111111)</td>
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<td>1923</td>
<td>1673 to 2273</td>
<td>2773</td>
<td>2773</td>
<td>2773</td>
</tr>
</tbody>
</table>

- Time (hours): 1015, 285 1.0 to 2.1
- Power Density (MW): 20, 10
- Burnup (atom %): 90, 50
- Results: No Swelling, No Swelling, Some Leak & Swelling, No Failures, No Failures, No Failures

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Exclusively for Pratt & Whitney

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ORIGINAL PAGE IS OF POOR QUALITY
The B&W fuel fabricability assessment is based on written accounts of previous work, discussions with people who performed some of that work and discussions with our own manufacturing experts. The General Electric 710 program and the Multimegawatt program both left good documentation of their efforts. A trip to Argonne National Laboratory was made to talk to some of the people involved in the manufacture of CERMET fuel. B&W manufacturing personnel are experienced with refractory metals and UO₂. Discussions with B&W fuel manufacturing experts solidified confidence that CERMET fuel manufacturing technology is easily recoverable.
Tungsten/UC2 CERMET Fabrication Considerations

- Tungsten/UC2 Homogeneity
- Camcoat Process
- UC2 Stoichiometry Can Be Controlled During Processing
- Proposed Fabrication Techniques
  - Machining of Monolithic Subsections
  - Stacking of Wavy Plates to Form Subsections
  - Forming of Near Net Shape Subsections

The CERMETs used for the XNR2000 are formed by consolidation and densification of UC2 and tungsten (or molybdenum) powders. The UC2 in CERMET fuel elements produced using any process must be distributed uniformly in the element. Uniformity as used here has two different meanings. First, the UC2 loading must be locally uniform throughout the element subsection. Second, and perhaps more important, the UC2 fuel particles must not cluster in the CERMET, but rather must be individually isolated by tungsten matrix material. This ensures that each fuel particle will be cooled adequately. Because of the differences between the behavior of tungsten and UC2 powders, these powders must be pre-processed to ensure blending before they can be consolidated in a powder based process. It is assumed in the discussion of each consolidation process that a suitable blending process has been used prior to the actual fabrication of the element. One possible blending process is the Camcoat process developed in the General Electric 710 program.

A number of possible consolidation processes may be employed. These include, but are not limited to, pressing and sintering, high energy rate forming (HERF) and heat treating, hot pressing and hot isostatic pressing (HIP). All of these processes should be capable of providing solid element subsections with little porosity in the matrix. Some of the processes, such as extrusion, will impart an axial texture to the element material. The acceptability of such texture must be evaluated prior to selection of a texture producing process.

In all of the above consolidation processes, the stoichiometry of the UC2 fuel can be controlled. This is done by performing the consolidation operation in an atmosphere where the oxygen partial pressure is controlled. Typically, this involves consolidating in a hydrogen based atmosphere. Control of UC2 stoichiometry is critical because deleterious effects occur if the fuel is either hyper- or hypo-stoichiometric.

Three CERMET fuel element subsection fabrication techniques were evaluated: machining of monolithic subsections, stacking of wavy plates to form subsections and forming of near net shape subsections. Each technique is described and a preferred fabrication technique is recommended. Machining of monolithic subsections was used in the General Electric 710 program and forming of near net shape subsections was used in the ANL Nuclear Rocket Program to fabricate fuel elements.
Fuel Element Assembly

- Components
  - CERMET Fuel Subsections
  - Tungsten/Rhenium Structural Can
  - Tungsten Coolant Channels

- Assembly Options
  - Stack Subsections in Can, Insert Tubes For Coolant Channels, HIP To Bond Can and Tubes To Fuel
  - Diffusion Bond Subsections To Each Other, HIP To Bond Fuel To Can, Form Coolant Channels By CVD Coating or Insertion of Tubes

Fuel Element Assembly

A complete CERMET fuel element consists of the CERMET fuel subsections, a tungsten/rhenium structural can and tungsten lined coolant channels. Final assembly of these components into full length fuel elements can be accomplished in a number of different ways. One option, used in the 710 Program, is to stack the subsections, insert tungsten flow tubes in the aligned channel holes and place the stack in the tungsten/rhenium can. A HIP operation is performed to bond the can and flow tubes to the subsection stack. With this option, the can is structural and is the sole load bearing component in the element. Another option is to coat the external surfaces of the element subsection and the ID of the coolant channels with tungsten prior to bonding a structural can onto the stack. This would eliminate the need for inserting full length flow tubes into the elements. A final option would be to bond the subsections together to form an integral fuel stack prior to bonding the stack into the can. Diffusion bonding is one possibility for the bonding process. Tungsten washers or standoffs could be used between subsections to create a plenum or transition section to minimize the effects of hole misalignments. The coolant channels can be formed as coatings on the subsections prior to assembly or by inserting tubes prior to can bonding.

Of the assembly options, bonding the subsections together prior to further assembly is preferred because it reduces the dependence on the fuel element can for structural integrity. It is not clear what technique is best for forming the tungsten flow tubes. Further technology evaluation is necessary in this area.
In the monolithic process, the coolant flow channels are machined into a consolidated subsection. This may be done by drilling with diamond tooling or ultrasonic machining. Electrical discharge machining (EDM) is not feasible because of the high volume fraction of insulating oxide fuel in the CERMET. The maximum element subsection length which can be processed is limited by runout and depends on hole size, fuel loading, hole pitch and manufacturing tolerances. The maximum subsection length must be determined as part of the technology development. Experience with other materials suggests that this length is between 1 and 5 cm. This process was used in the General Electric 710 program to produce 1 cm thick CERMET subsections with smaller holes than proposed for the XNR2000.
Monolithic Discussion

Advantages
- Machining of Coolant Holes Can Start from True Positions for Each Subsection
- Short Subsections Simplify Inspection

Disadvantages
- Large Amount of Waste Generated by Machining
- Machining Exposes UO2
- Fuel Loss
- Coating Difficulties
- Limited to Short Subsections by Runout
- Joining of Many Subsections Challenging

The major advantage of the Monolithic process is that the coolant channels can be machined into the element subsections starting from true positions. Runout will increase as the depth of the hole increases, ultimately limiting the length of the element subsection which can be processed. The short length of the subsections is an advantage for inspection if the subsection flow tubes are applied as coatings before element assembly, because it will be easier to verify the integrity of the flow tube.

There are a number of disadvantages associated with the Monolithic process. As previously mentioned, subsection length is limited to between 1 and 5 cm. Stacking and bonding of the many subsections necessary to make a complete fuel element would be complicated.

Another disadvantage of the Monolithic process is that a large amount of scrap tungsten/UO2 debris will be generated by the machining process. The uranium must be recovered from this debris. In the current design, 45% of the UO2 initially in the consolidated fuel element subsection ends up as debris.

A final disadvantage of the Monolithic process is that the machining process exposes UO2 fuel particles on the channel surface. This is an important effect for the following reasons. First, a significant fraction of the total amount of fuel in an element subsection will be exposed in the coolant channels. Assuming an average UO2 particle size of 100 μm, the fraction of fuel within one-half of a particle diameter of a coolant channel is 4.9% for the current design. It is not unreasonable to assume that a large portion of the exposed fuel particles would be damaged in the channel machining process and be lost in debris, especially if fuel particles are intentionally porous (to collect gaseous fission products). This would lead to a relatively rough coolant channel surface and possibly to an unacceptable loss of fuel. A second consequence of having exposed fuel on the channel surfaces is that it may be difficult to form the coolant flow tubes by CVD. Porous or damaged fuel particles may trap halide feed material or CVD byproducts and compromise flow tube adhesion during operation. In addition, UO2 may react with the CVD feed material or byproducts to an unacceptable degree.
Wavy Plate Fabrication

A fabrication technique proposed but not actually used in the General Electric 710 program was to build element subsections from flat plates. This fabrication process begins with the formation of tungsten-UO2 powder compacts in the form of plates. Each plate would be fabricated with grooves on both of its faces. The grooves would be semicircular and correspond to one-half of a coolant channel. Stacking and aligning the plates would form an element subsection with complete, circular channels. This technique can produce elements with coolant channels arranged on a square or triangular pitch, by varying the offset between the groove patterns on the opposite faces of the plates.

After the formation of the powder compact, the plate is then consolidated by sintering. It may also be possible to perform the consolidation by hot pressing, if a suitable material for the fixturing required can be identified. Following consolidation, the plate would probably need to be ground to ensure flatness. The plate could then be coated with tungsten to form a coating on the half channels. When the plates are assembled, this coating would form the flow tube. Alternatively, the tube walls could be formed after subsection or fuel element assembly by CVD coating or insertion of tubes and HFP. After being stacked, aligned and loaded, the subsection is diffusion bonded by heating in a controlled atmosphere (1950°K in a hydrogen atmosphere for 1 hour for example).
Wavy Plate Discussion

■ Advantages
  - May Not Require Machining Of Coolant Channels
  - Inspection Simplified By Thin Sections and Exposed Coolant Holes

■ Disadvantages
  - Close Tolerances May Be Difficult to Maintain (channel position +/- .02 mm, stacking alignment +/- .02 mm)
  - Minimum Amount of Machining Required Prior to Joining

One of the major advantages of the Wavy Plate process is that the coolant channels can be fabricated without having to machine them into fully consolidated CERMET. Another major advantage is that forming the tungsten flow tubes by coating the plates prior to stacking would allow easy and detailed inspection of the integrity of the flow tube.

The major disadvantage of this process is that sintering induced shrinkage will affect the final dimensions of the plate. Accordingly, it may be difficult to maintain the required tolerances in plate dimensions. Additional difficulties may be encountered in stacking and aligning plates to form element subsections. Even if plate dimensions are such that perfect alignment is achieved, it may be difficult to maintain this alignment during the plate bonding operation. Channel position and stacking alignment tolerances will both be of the order of +/- .02 mm. The tight tolerances are necessary to minimize coolant channel offsets and maximize web contact area. A final potential problem is that, if additional machining is required after consolidation, many of the disadvantages related to machining mentioned earlier may be present.
Near Net Shape Fabrication

Fuel elements were constructed using this process in the General Electric 710 program and again during the Multimegawatt program. In this process, treated tungsten/UO₂ powder is introduced into a flexible rubber mold which contains molybdenum rods or wires. After filling, the mold is cold isostatically pressed to form a fuel element subsection powder compact. The molybdenum rods or wires are used to form the coolant channels in the compact. During mold filling, the molybdenum channel formers are held rigidly in a triangular array, with the pitch between the channel formers slightly greater than that required in the consolidated subsection, to allow for shrinkage during sintering. The key to this process is that elastic strain stored in the powder compact during isostatic pressing causes the compact to expand slightly after the pressure is removed. This spring-back effect is of a magnitude sufficient to allow the channel formers to be removed easily from the compact.

After isostatic pressing, the powder compact is sintered. The tungsten matrix can be identified at essentially theoretical density at the relatively modest temperature of 1950 K. CERMETS containing up to 61 volume percent UO₂ have been fabricated.

Segments 40 to 50 cm long were made using this process during the General Electric 710 program. The useful length may ultimately be limited to 5 to 15 cm by other factors such as channel straightness tolerances and insensitivity.
Near Net Shape Discussion

Advantages
- No Machining Required
- Possible To Fabricate Longer Subsections

Disadvantages
- Sintering Shrinkage Must Be Considered
- Inspection of Longer Subsections More Difficult

This process has two major advantages. First, a fuel element subsection is produced which is truly near net shape. Some machining of the external surfaces of the subsection may be required if inter-element spacing tolerances are tight. However, the coolant channels are formed without machining and no fuel particles are exposed in the channels. The other advantage is that this process can fabricate significantly longer element subsections which drastically reduces the number of bonds required to fabricate an integral full length element.

The major disadvantage of this process is that significant process qualification and control will have been performed to ensure that dimensional tolerances in the consolidated element subsections will be met. This work is necessary to guarantee that sintering shrinkage is reproducible from run to run during production. Process optimization may be necessary for each batch of powder used. Another disadvantage of this process is that the longer length subsections produced make inspection of the coolant channel surfaces more difficult.
Inspection and Q/A Requirements

- Define Fuel Element Specifications and Tolerances
- Validate the Chosen Process Using Destructive Testing To Verify and Quantify
  - Homogeneity
  - Uranium Assay
- Use Nondestructive Testing Techniques To Check For
  - Gross Defects
  - Dimensions (Size, Shape, Straightness, Roughness)
  - Bond Integrity

**Inspection and Q/A Requirements**

Significant development effort will be required in the areas of inspection techniques and quality assurance procedures. For critical items, such as fuel elements in a man-rated system, there is no such thing as an excessive amount of inspection. Manufacturing tolerances and specifications for fuel element quality must also be determined.

One obvious area for inspection is conformance to dimensional requirements. Aside from the more obvious dimensional measurements, measurements of coolant channel straightness and surface roughness must be performed. Both of these attributes would be expected to affect thermal hydraulic behavior.

Before element subsections are assembled into full length fuel elements, it will be necessary to verify that they are structurally sound. At a minimum, the porosity of the tungsten matrix should be determined and the absence of gross defects verified. Also required is measurement of the integrity of all bonds in the fully assembled fuel element. These include the bonds between element subsections, between the coolant flow tube and the CERMET fuel and between the can and the fuel element. Ultrasonic and eddy current inspection techniques can be used for these measurements.

Where direct measurements are not possible, verification has to be performed by qualifying the process. This is accomplished by running process control samples through the element fabrication process and performing destructive evaluations on them. Two measurements for which this may have to be done are UO_2 content and homogeneity of the fuel subsections. The necessity for homogeneity has already been discussed. Fuel content in is required for SNM accountability. It is also necessary to verify the fuel loading of each element to ensure that the reactor will have sufficient reactivity.
Fuel Element Assessment
Conclusions

- Recommended Baseline Fabrication Approach - Forming of Near Net Shape Subsections
- Demonstrated in Two Previous Programs
- Well Documented Process
- Technology Recoverability Demonstrated
- Other Fabrication Approaches Still Viable As Backup Options

- No Materials Incompatibilities Noted
- Fuel Element Performance Limited by Melting Point Of UO₂
- Further Process Qualification and Development of Inspection Techniques Required

Fuel Element Assessment Conclusions

There are no insurmountable obstacles to fabrication of tungsten or molybdenum UO₂ CERMET fuel elements. XNR2000 fuel is manufacturable using demonstrated technology. There were no materials incompatibilities noted in this investigation. Fuel element performance is not limited by structural considerations but by the melting point of the UO₂. The maximum nominal fuel temperature in the XNR2000 is well below the UO₂ melting point.

The recoverability of the CERMET processing technology has been demonstrated. The development of CERMET fuel technology should not impose cost or schedule limitations.

Of the three processes considered for the fabrication of tungsten (or molybdenum) based UO₂ CERMET fuel elements, the Near Net Shape process is preferred. As discussed, this process has the potential of producing long length element subsections without having to machine the coolant channels. The process uses well known, technically simple processing steps. These steps will, of course, have to be extremely well characterized, to control sintering shrinkage and allow dimensional tolerances to be met. Finally, this process has been investigated extensively in the past and there is a significant experience base with it.

The wavy plate fabrication scheme also has potential. It should be further investigated in parallel as a backup option.
Safety Assessment

The Pratt & Whitney XNR2000 NTRE has been designed with safety as a primary consideration.

The safety of the public, mission personnel, the crew and the terrestrial and non-terrestrial environment have all been considered. The main safety characteristics of the XNR2000 are highlighted, and the effect of thrust level on safety is considered.
Basis of B&W Safety Assessment

The first step in the B&W safety assessment was to determine what the requirements and safety concerns for SEI NTRE systems are. Some of the documents used are listed here:


Kniskern; "Nuclear Thermal Rocket Engine Requirements, Revision 3; NASA N.P. #002; 1992.


The XNR2000 design was then evaluated with respect to these requirements and concerns based on information supplied by Pratt & Whitney and the University of Florida.
Reactor System Safety Characteristics

- **Flow Path**
  - Dual Pass Scheme Ensures Low Temperatures In Outer Core
  - No Moderator To Cool

- **Thermal Margins**

  Reactor Temperatures (K)

<table>
<thead>
<tr>
<th></th>
<th>Outlet Hydrogen</th>
<th>Peak Fuel</th>
<th>UO2 Melt</th>
<th>Refractory Melt</th>
</tr>
</thead>
<tbody>
<tr>
<td>Outer Core</td>
<td>1659</td>
<td>2007</td>
<td>~3150</td>
<td>2900</td>
</tr>
<tr>
<td>Inner Core</td>
<td>2669</td>
<td>2880</td>
<td>~3150</td>
<td>3700</td>
</tr>
</tbody>
</table>

- **Control/Shutdown Redundancy**

Reactor System Safety Characteristics

Safety considerations must be an integral part of the design process for any man-rated space system. Inherent safety is the preferred goal, and passive systems are preferred over active systems. The dual pass flow scheme ensures low temperatures in the outer core in a simple way. The fact that there is no moderator to cool also simplifies the flow path. The peak fuel temperature in the inner core is 2880°C (outlet hydrogen temperature of 2669°C); this compares to tungsten and UO2 melting points of 3700°C and 3150°C respectively. In the outer core, the peak fuel temperature is 2007°C (outlet hydrogen temperature of 1659°C); the melting point of molybdenum is 2900°C. These large thermal margins provide advantages in transient and accident situations. The reflector windowing leakage control scheme proposed for the XNR2000 is simple and robust. Redundant control and shutdown systems are provided.
CERMET Fuel Form Safety Characteristics

- Thermal Shock Resistance
- Long Term Stability
  - Compatible With Hot Hydrogen
  - Low Swelling
- High Thermal Conductivity
  - Tungsten - ~100 W/m*K
  - UO₂ - ~2 W/m*K
  - Bulk CERMET - ~33 W/m*K
- Tungsten Used As Primary Barrier For Fission Product Retention
  - Demonstrated Performance In Hot Hydrogen
  - Low Tungsten Diffusion Coefficient
  - Low Volatilization To Vacuum

The XNR2000 design benefits from the inherent stability, robustness and transient tolerance of the CERMET fuel form. This may be a particular asset in the long dormant phase and in assuring operability until disposal. Tungsten and molybdenum CERMET fuels are more resistant to hydrogen erosion than carbide fuels, and therefore, lifetime is not limited by coatings technology. They also exhibit low swelling and are effective at retaining fission products. CERMET fuels exhibit excellent thermal shock resistance over a wide range of conditions. The high thermal conductivity of the fuel is advantageous in undercooling scenarios and decay heat removal. Bulk tungsten/UO₂ CERMET thermal conductivity is in the range of 33 W/m*K, based on tungsten and UO₂ thermal conductivities of 100 and 2 W/m*K respectively. When damage thresholds are exceeded, the CERMET fuel form can handle significant degradation before failing catastrophically.

Tungsten is effective at retaining fission products. Its inherent stability, performance in hot hydrogen, low volatility to vacuum and low diffusion coefficient all combine to make it one of the best materials for this purpose. The General Electric 710 program demonstrated that the CERMET matrix alone will retain 85% of the fission products it contains. With intact tungsten cladding retention approaches 100%.
Fast Spectrum Safety Characteristics

- Negligible Hydrogen Worth (< .2 %ρ)
- Less Excess Reactivity Required
  - Lower Delayed Neutron Fraction
    Makes Given Reactivity Insertion Worth More
  - Negligible Xenon Reactivity Effects

The fast spectrum of the XNR2000 has several positive safety effects. The worth of the hydrogen in the XNR2000 is negligible (<.2 % ρ as calculated by the University of Florida and verified by Babcock & Wilcox). This is helpful at reactor startup, since there will be no large reactivity insertion due to the cold hydrogen. Little excess reactivity is required in the XNR2000. The lower delayed neutron fraction in the fast spectrum makes a given reactivity insertion worth more in terms of reactor response. Also, there a negligible xenon reactivity effect due to the fast spectrum, so there is no need to provide excess reactivity to overcome a large xenon transient.

The fast spectrum of the XNR2000 may also affect ground testing. Ground test facilities will have to be designed to handle the fast spectrum leakage from the NTRE.
XNR2000 Emergency Safety Characteristics

- Flow Blockage
  - High Thermal Conductivity
  - Thermal Margin
- Reactor Power Limitation
- Turbopump Deep Throttling (to 10% Available)
- Loss of Turbopumps
  - Pressure Fed Cooling Available
- Inadvertent Reentry
  - Reactor Subcritical For All Compaction and Immersion Accidents

The high thermal conductivity and structural stability of the CERMET fuel is a benefit in accident situations. For partial or full flow blockage in a coolant channel, the high thermal conductivity mitigates the temperature rise. Temperatures around a blocked channel may approach or exceed the melting point of UO₂. This will not be a problem in the localized area involved. The tungsten can easily contain molten UO₂.

A loss of turbopumps accident is handled by pressure fed cooling. A high pressure reservoir will provide the hydrogen flow necessary for the critical period immediately following shutdown. After the first minute or so, the reactor can be cooled by feed tank pressure.

No NTRE currently under consideration can survive a full-power total loss of coolant accident. A low-power total loss of coolant would cause rapid shutdown of the reactor due to the negative reflector temperature coefficient. A total loss of coolant at very low power or during decay heat removal might not be catastrophic for the XNR2000 due to the robustness of the CERMET fuel.

The XNR2000 turbopumps can be throttled to 10% of full flow. This is a safety advantage for cases where the reactor power is limited for some reason. This feature allows the NTRE to provide reduced thrust at nearly full L. This enables mission completion or full abort capability for a limited reactor power scenario.

It has been shown by the University of Florida, and verified by Babcock & Wilcox, that for all the accidents of concern in an inadvertent reentry (compaction and submersion) the reactor remains subcritical by a significant margin.
Safety Characteristics of Small Engines

- Ground Testing
  - Lower Throughput Results in a Smaller and Less Costly Facility
  - Lower Fissile Inventory Mitigates Consequences of Accidental Release

- Early Design Tradeoffs Are Needed To Define the Optimum Engine Size From a Safety Standpoint
  - Tradeoffs Should Include Mission Parameters (total thrust/length of burn)
  - Effect of Engine Size/Number of Engines on Redundancy and Reliability

The thrust levels required for SEI missions can be achieved using a few large engines or multiple small engines. It is not obvious what conclusion would be drawn from the tradeoffs for small versus large engines from an operational safety point of view. Redundancy goes up for small engines, but overall system reliability may go down.

One area where small engines have a clear safety advantage is in ground testing. Small engines will be easier and less costly to test than large engines. The test support requirements and effluent throughput will both be lower, resulting in a smaller and less costly facility. Accidents consequences will also be mitigated due to the lower fissile inventory in a small NTRE.

A detailed fault tree failure analysis will be required to determine the optimum arrangement from an overall safety point of view. This analysis should be performed as soon as possible. The tradeoffs in a safety evaluation should include certain mission parameters. For example, a lower thrust level for a longer time may result in a safety advantage. This would probably also result in a higher initial mass in low earth orbit, but this might be a worthwhile trade for increased safety.
Overall Conclusions

- The XNR2000 Uses a Demonstrated Fuel Technology Which Has Been Shown To Be Recoverable
- CERMET Fuel Has Demonstrated High Fuel Integrity and Safety Features
- At NASA SEI Conditions, Superior Fission Product Retention Expected
- There Are Ground Testing Safety Benefits To Use of Small Engines
- No Obvious Roadblocks To the Development of the XNR2000 For NASA SEI Applications Were Identified In Any of the B&W Tasks

B&W’s overall opinion of the XNR2000 is positive. It uses a demonstrated fuel technology which has been shown to be recoverable. The CERMET fuel form has been demonstrated to have high fuel integrity and important safety features. At NASA SEI operating conditions, superior fission product retention is expected.

Ground testing considerations point to a safety advantage for small engines.

None of the B&W tasks have identified any roadblocks to the development of the XNR2000 as a viable NASA SEI NTRE.
### Composite NTR ISP is Limited By Burn Time

<table>
<thead>
<tr>
<th>Thrust size</th>
<th>Mission burn (hr)</th>
<th>Design life</th>
<th>Composite ISP</th>
<th>Cermet ISP</th>
</tr>
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<tbody>
<tr>
<td>25k</td>
<td>4.5</td>
<td>14</td>
<td>825</td>
<td>900</td>
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<tr>
<td>50k</td>
<td>2</td>
<td>8</td>
<td>845</td>
<td>900</td>
</tr>
<tr>
<td>75k</td>
<td>1.5</td>
<td>6</td>
<td>855</td>
<td>900</td>
</tr>
</tbody>
</table>

*Composite NTR ISP is Limited By Burn Time*

The performance level, specific impulse and propellant exit temperature, of the cermet based NTRG systems is not limited by burn time. However, the operating temperatures and consequently burn time of the composite NTRG systems is limited by burn time due to the inherent chemical instability between the hot hydrogen environment and the carbon-based fuel form.
ADVANTAGES OF DUAL PASS REACTOR

- Power/Flow Matching
- Material Temperature Matching
- Flat Radial Profiles (Mixing)
- Isolated Hot Core

900 seconds $I_p$
Superior Fn/Wt

Advantages of Dual Pass Reactor

The primary attractive features provided by the dual pass reactor core are summarized. A flat radial power profile is provided by the dual-pass reactor due to the averaging of power distributions relative to two distinct regions. Positive flow/power matching is achievable because of the separation of the inner and outer cores. The maximum fuel element power shape factors appear in the outer core region because of the proximity of the radial reflector. However, because the outer core serves as the first pass, the coolant hydrogen propellant passes through the outer core and eliminates fuel temperature concern. Additionally, upper plenum mixing of the hydrogen serves to eliminate the outer core power peaks from the inner core fuel elements. The dual pass configuration isolates the hot inner core fuel elements from the rest of the engine system. This isolation provides material flexibility allowing the use of lighter weight Moly based fuel elements in the outer core and a Be radial reflector which provides the useful reactivity worth for the weight. The most obvious benefit of the dual pass core is the reduced axial thermal gradients and consequently thermal stress loads placed on the fuel elements.
ADVANTAGES OF FAST SPECTRUM CERMET REACTOR

- Safety
  - Positive Fuel and Fission Product Retention
  - Compaction/Immersion
  - Long Life

- Simple Design
  - No H₂ Reactivity Feedback
  - Simple Support (No Tie Tube Complexity)
  - Control Flexibility

- Strong High Conductivity Fuel
  - Self Supporting
  - Dimensional Stability
  - Resistance to Thermal/Physical Shock

Adapted from NTP: System Concepts

The XN1000 builds upon the experience and database of CERMET fuels obtained by the G770 and ANL programs. The fast spectrum CERMET fuel form was selected to meet the unique requirements of ALAVI fuel and fission product release, multiple contact capability and survivability under credible accident scenarios. During the CETF/ANL program the CERMET fuel form displayed tolerance to excessive temperature/power ramps due to the high strength and conductivity of the refractory metal matrix. Additionally, CERMET fuel display complete compatibility to the expected hot test operating environment as well as cladding and fuel matrix CTE compatibility. Finally the XN1000 is based upon a fuel form that was successfully fabricated and tested.

The selected CERMET fuel form provides a more robust, simple system design because of the elimination reactivity feedback from hydrogen moderation. A simplified support structure is possible due to the high strength of the refractory metal based fuel form.
• 37 hole fuel element fabrication trial
  – Near net shape
  – Wavy plate
• Refine XNR2000 baseline design
  – Mechanical design
  – Transient, off nominal
  – Reliability analysis
  – Manufacturing study
  – Health monitor/control definition
• Ensure "fast spectrum" testability in PIPET
NTRE Extended Life Feasibility Assessment

Final Report
23 OCT 92

Presented to
NASA LeRC

By
Aerojet Propulsion Division
Energopool, Babcock & Wilcox
We Have an Effective NTRE Team

Aerojet Propulsion Division
Sacramento, California

Babcock & Wilcox Advanced Systems
Engineering
Lynchburg, Virginia

Energopool – Moscow, Russia

NASA LeRC TOC
Program Objectives

• Assess Feasibility of a Long Life, Reusable Nuclear Thermal Rocket

• Two Reactor Concepts
  – Particle Bed Reactor (PBR)
  – Commonwealth of Independent States (CIS)

• Tasks
  – Conceptual Layouts (75K lbf)
  – Thermodynamic Cycle Balance
  – Preliminary Neutronic and Thermal – Hydraulic Analysis
  – System Mass Estimates
  – Preliminary Life and Reliability Assessment
  – Safety Assessment
  – Scaling to 25 and 40K lbf (PBR Only)
The NASA LeRC TOC Addresses the Emerging NTRE Requirements

- Thrust: 25K, 40K, 75K
- Thrust/Wt (With Internal Shield): > 4
- Isp: > 850 sec
- Length: 30 Meters
- Diameter: 10 Meters
- Throttling: 25%
- Restarts: > 10
- Single Burn Duration: 60 Min (Max)
- Life: > 270 Min at Rated Thrust
- Reliability: Manned

GenCorp • Energopool • Babcock & Wilcox

NASA LeRC TOC Final Report
Agenda

- Introduction: Wayne Dahl
- Technical Overview: Mel Bulman
- Concept Definition: Don Culver
- Engine Design: Roy Squires
- Integrated Engine: Mel Bulman
- Engine Reliability and Safety: Mel Bulman
- PBR Engine Sensitivity Study: Mel Bulman
- PBR Reactor System: Richard Rochow
- CIS Engine: Don Culver
- CIS Reactor System: Richard Rochow
- Technology Road Maps: Mel Bulman
- Summary: Mel Bulman

GenCorp • Energopool • Babcock & Wilcox
The NTRE is a highly integrated machine. As we will show, interactions between reactor and engine level operations are significant. Our systems approach to NTRE design reveals exciting new possibilities for improving the reliability and performance of spacecraft.

The NTR Engine Is a Highly Integrated Machine (Not Just a Reactor Between a Pump and Nozzle)
Our basic NTREs meet all current NASA requirements.

The thrust to weight ratio of the PBR engine is 6.3. The CIS engine is somewhat heavier with a F/W of 4.7. The PBR Isp is 65 seconds higher than the requirement at 915 sec. The higher temperature of the CIS engine produces an Isp of 959. Both engines fit well within the space allowed for in the SOW.

Our advanced pump design and engine management system permits throttling 20:1 compared to the requirement of 4:1.

Our preliminary life evaluation indicates the engines will be able to operate longer than currently required. Our preliminary reliability and hazards analysis indicate man rating of these engines is achievable within the scope of the engine development.

**Our Basic Engine Meets All Current NASA Requirements With a Recuperated Topping Cycle**

<table>
<thead>
<tr>
<th>Requirement</th>
<th>PBR Value</th>
<th>CIS Value</th>
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</thead>
<tbody>
<tr>
<td>Thrust</td>
<td>75 Kibf</td>
<td>75 Kibf</td>
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<tr>
<td>Thrust/Weight With Shield</td>
<td>&gt; 4</td>
<td>8.3</td>
</tr>
<tr>
<td>Isp</td>
<td>≥ 850</td>
<td>915</td>
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<tr>
<td>Length</td>
<td>≤ 30M</td>
<td>7.7M</td>
</tr>
<tr>
<td>Diameter</td>
<td>≤ 10M</td>
<td>2.2M</td>
</tr>
<tr>
<td>Throttling</td>
<td>≥ 4:1</td>
<td>20:1</td>
</tr>
<tr>
<td>Reuse</td>
<td>≥ 10 Restarts &gt; 10</td>
<td>&gt; 10</td>
</tr>
<tr>
<td>Single Burn</td>
<td>60 min</td>
<td>&gt; 60 min</td>
</tr>
<tr>
<td>Engine Life</td>
<td>&gt; 270 min</td>
<td>&gt; 270 min</td>
</tr>
<tr>
<td>Reliability</td>
<td>Manned Stage</td>
<td></td>
</tr>
</tbody>
</table>

*GENCORP - Energopool - Babcock & Wilcox*
With our recuperated cycle, we avoid complex core designs that produce heat to drive the turbopumps, yet we have increased the engine operating pressure to reduce its size and weight and to increase its performance.

We have studied two NTRs with heterogeneous reactors. One employs the particle bed reactor concept developed in the U.S. The other is based on 20+ years of development in the CIS. The CIS reactor utilizes a twisted ribbon type fuel and has been tested at over 3000K for over 1 hour.

- PBR
  In order to meet the NASA life requirement we have changed the fuel stoichiometry and lowered the operating temperature. We have arranged for deep throttling and closed loop decay heat removal.

- CIS
  We have modified our engine drive cycle and structure slightly to best make use of CIS fuel assembly technologies.

Technical Approach: Apply Our Recommended Engine Cycles to Two Heterogeneous Reactor Types

- Engine Cycle
  - Delete Gas Heater Fuel Assemblies
  - Raise Operating Pressure
  - Integrated Engine Option

- Particle Bed Reactor
  - Increase Design Life
  - Provide Deep Throttling/Decay Heat Removal
  - Integrated Engine Option

- CIS Reactor
  - Fuel Developed
  - High Operating Temperature
  - Integrated Engine Option
There are several ways in which heterogeneous reactors are superior to homogeneous ones, and all result from physically separating fuel and moderator, the characteristic of the heterogeneous concept. Moderator and fuel have different requirements, and separating them allows selection of optimum solid materials for each function.

High temperature carbides are suitable for fuel, because, when used correctly, they can deliver high reactor gas outlet temperatures, which enables high engine specific impulse. High gas temperatures are available, because carbide fuel can operate at high temperatures and because it formed into thin elements, internally generated heat need pass only a small distance to the coolant. Thus, it need not pass through moderator material to reach a cooled surface, as in the homogeneous reactor concept. Propellant gas can attain a temperature very close to the fuel’s maximum internal temperature. The PBR attains this advantage by using small diameter spheres in fuel particle beds, while the CIS reactor uses bundles of thin, twisted ribbons of fuel.

Efficient neutron moderators are hydrogenous, and no solid materials of this type can withstand temperatures in the range of fuel or desired outlet gas temperatures. Efficient neutron moderators are important, because uranium fission cross-sections are very low at fission neutron energy levels, and without good moderator material a larger amount of fissile material is needed in the reactor. Several negative features occur simultaneously when large amounts of highly enriched U235 are used in a reactor. Primarily, the safeguards problem is worsened. Secondly, launch safety is inherently less. Third, fast reactors need a more rapid control system, which exacerbates development and safety risks, and, fourth, fuel cost is much greater than that of the moderator which may replace it in a heterogeneous reactor. The PBR moderator is hexagonal blocks containing cavities filled with LHI that surround each fuel bed, while the CIS moderator is ZrH2 rods close-packed between the fuel assemblies.

For Mars mission NITRE we need a specific-impulse-loss-free turbopump power cycle to minimize total mission costs, including Earth-to-orbit launch. Thus, topping cycles are used, which have turbine life advantages over bleed cycles. In a heterogeneous engine the lower temperature moderator and reflector materials are cooled with a separate hydrogen loop prior to final heating by the fuel elements. This moderator and reflector heat is automatically the major portion of the topping heat needed for turbine inlet gas heating – for turbine drive power. In a homogeneous engine, at least the moderator heat is lost for turbine drive use. Lower engine operating pressure results, all other things being equal, and this leads to large, heavy engines with inferior Mars mission performance. Further, the moderator cooling loop also enables integration of a closed engine cooling and electric power generating system that can reduce Mars mission IMLEO by about 100 tons.

### Heterogeneous Reactors Superior to Homogeneous Types (NERVA)

<table>
<thead>
<tr>
<th>Features</th>
<th>Benefits</th>
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</thead>
<tbody>
<tr>
<td>Fuel Separated from Moderator</td>
<td>Moderator Cooling Powers</td>
</tr>
<tr>
<td></td>
<td>Turbine and Enables Closed Loop Cooling</td>
</tr>
<tr>
<td>PBR</td>
<td>(Reliability and Weight)</td>
</tr>
<tr>
<td>CIS</td>
<td></td>
</tr>
<tr>
<td>Fuel More Efficient</td>
<td>Higher Gas Outlet Temperature</td>
</tr>
<tr>
<td>Spheres</td>
<td>(High ISP)</td>
</tr>
<tr>
<td>Twisted Ribbon</td>
<td></td>
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<tr>
<td>Moderator More Efficient</td>
<td>Lower Fissile Inventory</td>
</tr>
<tr>
<td>Be Hex</td>
<td>(Safety and Weight)</td>
</tr>
<tr>
<td>with ZrH2 Rods</td>
<td></td>
</tr>
<tr>
<td>Cavities</td>
<td></td>
</tr>
</tbody>
</table>

**GENCORP**

*Energopool • Babcock & Wilcox*

NTP: System Concepts 252  NP-TM-92
Our basic engine meets or exceeds all NASA requirements. It provides for robust operation and takes up little room in the launch vehicle.

As we studied these engines we recognized some significant and beneficial differences between these heterogeneously moderated reactors and the homogeneously moderated NERVA type reactors. The separate moderator allows us to extract significant heat from the core without the need to flow hydrogen through the fuel elements. The full utilization of this in our Integrated engine provides many benefits including: (1) reliable, efficient NRE start up, (2) reduced decay cooling losses; (3) RCS and OMS at high ISP, (4) electrical power up to 100 kW (E) per engine.

Technical Approach: Two Engine Options Are Presented

- **Basic Engine**
  - Meets or Exceeds All Current NASA Specs
  - Robust Operation
  - Reliable, Efficient Engine Starting
  - Small Size

- **Integrated Engine**
  - Builds on Basic Engine
  - Reduces Decay Cooling Losses
  - Improves Mission Reliability and Performance by:
    - Integrating Stage and Engine Subsystem
    - Main Propulsion
    - RCS
    - OMS
    - Option for Electric Power (~ 100 kWe)
Our platelet technology enables us to turn the requirement to cool the internal gamma shield into a cycle-enhancing recuperator without mass penalty. This allows us to operate the engine at higher chamber pressures than otherwise possible, resulting in a smaller and lighter weight engine. In addition, the recuperator provides the bulk of the energy for the engine start. Sufficient energy is stored in the recuperator to accelerate the turbopumps to full power without additional heat. With this magnitude of stored energy, it would take over 10 aborted starts to significantly reduce the starting power of our cycle.

In addition to providing power, the recuperator provides thermal and hydraulic stability during all modes of engine operation. The reactor and feed system are effectively decoupled during high reactor transients.

Recuperated Cycle Provides Superior Engine Operation

- Provides Cooled, Internal Gamma Shield
- Enables High Chamber Pressure
- Provides Thermal Energy for Turbopump Start
  - Energy Available for Many Starts
- Provides Safe, Controllable Reactor Start
  - Prevents Liquid Hydrogen Entry Into the Core
  - Decouples and Damps System Oscillations

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Aerojet
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Engine Concept Definition

Don Culver

Our engine and major component design concepts are selected to meet all current NASA requirements. Any concepts that cannot meet these safety and reliability, performance, and operational requirements in the near term were discarded. In addition, many NASA goals that impact safety and reliability, mission benefits, development cost and technical risk were used to guide system configuration selections and design and operating parameter optimization studies.

NASA Goals and Requirements
Impact APD NTRE Selection

Requirements
- Safety
  - Radiation Protection
  - Manrate, Verify, Automate
- Performance
  - 850 sec Isp
  - 4:1 Thrust Weight
  - Throttling @ Tmax
  - 15-75K Ibf Thrust
- Operation
  - Reusable, Long Life
  - Bootstrap Start w/o Power
  - Degraded/Failed Tolerance

Goals
- Safety
  - Minimize Radioactive Materials
  - Hazard Mitigation and Reliability
- Mission Benefit
  - IMLEO/Trip Time (Isp and F/W)
  - Mission Commonality
  - 2006 Availability
  - Simplicity (Inherent Reliability)
- Technical Risk and Development Cost
  - Technology Readiness and Tests Needed
  - Propulsion System Integration
  - Facility Requirements

Energopool  Babcock & Wilcox
We begin our concept definition studies with trade studies. Of course, we perform trade
studies in each important area of requirements and goals.

When trade studies are completed, optimum design parameters are known, and engine
layout and component design studies can be finalized. When the point design is known,
sensitivity studies are made to check the impact of important design and operating parameters
on engine characteristics.

Trade Studies Define Engine Concept and Design Point

- Safety and Reliability
  - Nuclear
  - Non Nuclear

- Criticality Trades (B&W)
- Feed System Reliability

- Performance and Mission Benefit
  - Mission Payload
  - Power Cycle
  - Control System

- Versus Cycle Type, Pc, Nozzle Design
- Definition (Shield Integration)
- Architecture Study

- Operation and Technical Risk
  - System Operation
  - Propulsion System Integration
  - Technology Readiness

- Modes and Procedures Identified
- Shield, Decay Heat, Deep Throttling
- Major Component Status

GenCorp
Aerojet

Energopool ▪ Babcock & Wilcox

NTP: System Concepts 256

NP-TIM-92
Our Reliability Plan Is Tailored to Project Phase

• Concept Phase (TOC)
  – Reliability Block Diagrams With Typical Component Failure Rates
  – Preliminary FMEA to Component Level
  – Hazards Analysis (Crew, Ground Support and Populace)

• Design Phase
  – FMEA
  – Fault Tree Analysis
  – Safety Studies

*GenCorp\nAerojet

• Energopool • Babcock & Wilcox
A result of our feed system reliability block diagram study is that use of twin turbopumps on NTRE should improve mission reliability by reducing the probability of total failure and engine loss to about 1/4 of that of single turbopump fed engine. However, the twin turbopump engine has nearly twice the probability of failing to a degraded mode of performance. This usually means that one turbopump fails and the other continues to operate the engine at nearly 3/4 thrust. This is of little consequence at any time except a TMI (or TLI) burn.

Twin Turbopumps Improve Mission Reliability*

- Single TPA System Has ~ 4 Times the Probability of Total Failure vs 2 TPAs

- Twin TPA System Has ~ 1.7 Times the Probability of Failure to Degraded Mode (~ 70% Thrust) vs 1 TPA

*Industry Standard Component Failure Rates Applied to Feed Systems
Mission performance depends on rocket engine thrust/weight and mission average specific impulse (Isp). Engine thrust/weight depends largely on reactor type and power density, engine configuration, and operating conditions. Mission average Isp depends mainly on engine Isp and on operational Isp losses, and they depend on mission type, engine design details, and operating conditions. We will discuss our trade study results for each of these factors in the following charts and in the reactor design sections.

**Mission Isp Depends on Engine Isp and Operational H₂ Losses**

- Engine Isp = f (Tout)\(^{1/2}\)
  - Theoretical Isp (Tout and \(c\))
  - Tout max – Tout Mixed Mean
  - Nozzle Losses (Cooling, Divergence)
  - Power Cycle Bleed Losses

- Operational H₂ Losses
  - Open Loop Cooldown @ T < Tmax
  - Boiloff and Leaks
  - Start-up Bleed
We have studied three fundamental engine configurations:

1. DeLaval nozzle behind thermal reactor
2. Forward flow thermal reactor within an expansion-deflection (E-D) nozzle
3. Forward flow thermal reactor within a plug nozzle

The E-D nozzled engine appears to have the best mission performance potential, but it needs further study, and at this time it is recommended for a second-generation engine. However, we recommend this concept be studied in more detail soon, because it is rapidly developing into a more practical concept than was believed possible earlier.

The plug nozzled engine does not seem competitive, because of its large nozzle surface area in the high heat flux region of the throat and its consequent low Isp and high weight potentials.

The DeLaval nozzled design is, thereby, recommended for a near-term engine.
A trade study evaluated both engine weight and specific impulse by estimating the Mars mission payload delivery capability of identical vehicles powered by similar engines of conventional geometry having different power cycles, operating pressures, and nozzle area ratios. Both hot bleed cycle engines and topping cycle engines were evaluated over reactor outlet pressures (Pc) from 1000 to 3000 psia, with nozzle area ratios from 150 to 500, and with nozzle lengths from 80 to 120 percent bell. In each case a cooled, copper and steel nozzle was used with a carbon-carbon nozzle extension from area ratio 10 to the exit.

Results showed that bleed cycle engines are not competitive, based on their lower delivered specific impulse. Their payload carrying capabilities were consistently low by about 20 percent. Nozzle contours of 110 percent bell length were found to be best for nearly all engine variants. Engines with high nozzle area ratios benefited most from high engine pressure, because their nozzles are smaller and lighter in weight, better offsetting the increased turbopump weight required of high pressure feed systems. Conversely, engines with low nozzle area ratios are relatively insensitive to engine design pressure. (Both reactor design teams agreed that reactor, vessel, and shield weight totals are not greatly affected by design pressure in the range of our study.)

The design point selected was area ratio 300 with pressure of 2000 psia, because it appeared to be the lowest pressure – lowest area ratio combination to attain high mission performance. At 200 nozzle area ratio about five percent payload is lost, regardless of engine pressure selection.

High Pressure Topping Provides Maximum Mission* Performance

Study Results

<table>
<thead>
<tr>
<th>Hot Bleed Cycle</th>
<th>Topping Cycles</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
</tr>
<tr>
<td>Normalized Payload Weight Returned to Earth</td>
<td>Normalized Payload Weight Returned to Earth</td>
</tr>
<tr>
<td>0.5</td>
<td>1.0</td>
</tr>
<tr>
<td>Pc, kpsia (110% Bell Nozzle Length)</td>
<td>Pc, kpsia (110% Bell Nozzle Length)</td>
</tr>
<tr>
<td>1.0</td>
<td>1.0</td>
</tr>
<tr>
<td>0.75</td>
<td>0.75</td>
</tr>
<tr>
<td>0.5</td>
<td>0.5</td>
</tr>
</tbody>
</table>

Engine Selection

- Topping Cycle
  - Pc = 2,000 psia
  - r = 300 (De = 92 in.)
  - L-noz = 110% Bell (210)
  - Tc = 2,700K

* 4 Burn, All-Up, Manned Mars Mission With C2 = 16 km2/s2 and IMLEO = 775 Tonnes

Energopool - Babcock & Wilcox
We examined all reasonable turbopump drive power cycles, based on examination of heat sources for turbine inlet gas and destinations for turbine exhaust gases. Those cycles which bleed turbine exhaust gas overboard rather than through the throat of the engine's nozzle lose specific impulse, because they cannot expand low pressure and temperature turbine exhaust gas to high velocities. We found that these engines are not in contention for Mars missions on a performance basis.

Three topping cycles were analyzed. The simple expander cycle cannot provide enough heat to power the engine reliably to pressures of 1000 psia or above. We have seen the Mars mission performance decrement of low pressure engines. Therefore, extra heat must be added to the topping heat that can be recovered indirectly from the reactor core by cooling engine components, such as moderator, reflector, pressure vessel, shields, etc. One way to do this is to devote a portion of the fuel assemblies to turbine drive heat. This requires additional manifolding in the reactor core, an unwanted complexity which may reduce neutronic efficiency and cost engine size and weight. Another way is to use a high heat rate heat exchanger to transfer turbine exhaust heat to the pump discharge to augment the topping cycle heat. This is the scheme we selected, in spite of the fact that engine designers usually feel that highly effective recuperators are large and heavy.

**Recuperated Expander Cycle Selected**

<table>
<thead>
<tr>
<th>Turbine Exhaust Destination</th>
<th>Heat Sources for Turbine</th>
<th>System (Expander)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Overboard = Bleed Cycles</td>
<td>Hot Bleed</td>
<td>Recup. Bleed</td>
</tr>
<tr>
<td>Reactor = Topping Cycles</td>
<td>Augmented</td>
<td>Similar to Cold Bleed</td>
</tr>
<tr>
<td>* More Valves</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

- Hot Turbine
- Nozzle Port
- Mixer Fatigue
- ~20% P/L Loss
- Core Complexity
- Reactor Size and Weight
- Recuperators Typically Large and Heavy
- Max. P/L
- Larger Toploss or
- Hot Turbine
- ≥14% P/L Loss
- Limited Power
- Restart Heat?
- 7% P/L Loss

*GenCorp*  •  *Energopool*  •  *Babcock & Wilcox*
The Impacts of the recuperator on our engines' size and weight is nil, because:

(1) We have demonstrated our ability to fabricate large heat rate heat exchangers of very compact dimensions with our platelet technology, for example in the SSME heat exchanger program.

(2) The steel recuperator can function as the gamma shield for the NTRE and the forward closure of the reactor vessel. We have shown that the sum of these two weights in a conventionally designed engine are greater than the required recuperator weight. Thus, we incur no weight penalty for the heat exchanger itself.

(3) Low density material, such as steel, may be used efficiently for a gamma shadow shield, because it is located close to the large diameter reactor, and the radiation tends to be planar to all surfaces, because of the self shielding provided at all other angles.

Recuperator Weight Impact Is Nil

Problem — Large Recuperator Size and Weight

Solution — Compact Stainless Steel Platelet HEX Doubles as Cooled Internal Gamma Shield and Forward Pressure Vessel Head

Distributed Source Shield Weight Is Not Dependent on Material Density
Recuperated Cycle Provides Superior Engine Operation

- Provides Cooled, Internal Gamma Shield
- Enables High Chamber Pressure
- Provides Thermal Energy for Turbopump Start
  - Energy Available for Many Starts
- Provides Safe, Controllable Reactor Start
  - Prevents Liquid Hydrogen Entry Into the Core
  - Decouples and Damps System Oscillations

GENCORP Aerojet
- Energopool
- Babcock & Wilcox
The power cycle we have selected for both engines uses a recuperator to transfer heat from turbine exhaust to pump discharge, and each cool the nozzle with a small side stream of liquid hydrogen from the pump. The PBR cycle variation is shown here; it requires a pump discharge pressure of 4750 psia to deliver an engine Pc of 2,000 psia. It does this with a low turbine inlet temperature of 847 R (470 K) and a low turbine pressure ratio of less than 1.5 to 1. Pump stage pressure ratios are low, too, because four stages of pumping are used. However, four stage rotating assemblies are not needed with our concept, because our turbopumps emulate a quadruplicate system. We use two turbopumps in parallel to provide the total engine flow, and each turbopump consists of two identical rotating assemblies operating in series. Each rotating assembly is the simplest configuration possible, two pumps and a turbine on the shaft with two bearings between the three rotors. Reliability, performance, risk, and cost benefits result from this subcritical speed design.

The recuperator heat rate is about 125,000 Btu/sec, which is larger than the sum of the topping heat. If more power is needed this cycle has two main design variables, turbine pressure ratio and recuperator heat rate. The latter controls the turbine inlet temperature. The power balance shown has ten percent excess turbine power for turbine bypass control authority.

The flow scheme through the engine is as follows: through the pumps in parallel, with a 5-1 flow split after their flows join; the small flow cools the nozzle and pressure vessel; the large flow gets heated in the high pressure side of the radial outflow recuperator, where it enters the moderator and reflector cooling flow at the front of the core; the full flow passes through the turbines in parallel to rejoin and cool in the low pressure, radial inflow passages of the recuperator; cooled flow is manifolded to the inlets of each fuel assembly for heating to full outlet temperature and passage through the rocket nozzle.

Flow control elements include a low power electric feed pump, pump and turbine inlet and outlet valves, a turbine bypass control valve, and a pulse cooling valve. Reflector drive motor shafts penetrate the recuperator at its periphery, outside of the heat exchanger region, and a launch poison rod penetrates it at its center.

A High-Power, Loss Free Engine Power Cycle is Selected
The basic engine has five operating states in addition to two additional control states, checkout and emergency. Before starting the engine, the pumps are chilled with tank head or feed pump flow, and GH2 is vented overboard as required. However, much of the GH2 is pumped under pressure into the engine power loop by the feed pump. The engine can be held in this stage of chilled and pressurized readiness for long periods; with occasional chill down flows until impulse is needed.

Starting the engine consists of opening the turbine inlet valves and blowing down the loop, spin starting the turbines. The large, available amount of sensible heat in the recuperator bootstraps the feed system until reactor heat is available. The recuperator prevents liquid hydrogen from ever entering the reactor.

During engine operation at high power the engine thrust is controlled with turbine bypass valve position and specific impulse with reactor control drum position. The propellant tank is pressurized by a bleed from the low pressure recuperator outlet manifold.

Following reactor shutdown with control drum rotations, the 10-1 throttling turbopumps are throttled to maintain outlet temperature by their bypass control valves. When they have reached their minimum flow, one turbopump is shut down and the other (throttled up 2-1) will follow the reactor power down to about five percent and then begin to overcool the core, reducing specific impulse at this low thrust level. The electromechanical feed pump is started, and propellant is pumped under pressure at low flow rate into the cooling loop. When the loop pressure is high and the core cool, the second turbopump is shut down and the pulse cooling valve actuated. The core heats during pump shutdown and overcools during the cooling pulse. The pulse valve shuts, the feed pump pressurizes the loop while the core heats, and the valve cycles again, holding the average core outlet temperature and tsp above what it would be without pulse cooling.

While the core power decays the duty cycle of the pulse cooling valve changes continually, and eventually it stays closed. This happens when the pressure vessel is able to radiate the residual core afterheat to space.

**Operation Features Robust Start and Efficient Cooldown**

- **Readiness**
  - Pressurize Loop With Feed Pump
  - Chill Pumps and Vent GH2

- **Start**
  - Blowdown Start TPAs With Start Valves
  - Bootstrap on Recuperator and Reactor Heat

- **Run**
  - Control Valve Throttling
  - Bleed-Pressurize Tank Ullage

- **Cooldown**
  - Shutdown Reactor
  - Throttle on Decay Heat
  - Shutdown 1 TPA and Throttle to 5%
  - Overcool Fuel and Start Feed Pump
  - Shutdown 2nd TPA and Pulse Cool

- **Soakout**
  - Stop Pulse Cooling and Radiate

- **Energopool • Babcock & Wilcox**
Recuperated Cycle Provides Superior Engine Operation

- Provides Cooled, Internal Gamma Shield
- Enables High Chamber Pressure
- Provides Thermal Energy for Turbopump Start
  - Energy Available for Many Starts
- Provides Safe, Controllable Reactor Start
  - Prevents Liquid Hydrogen Entry Into the Core
  - Decouples and Damps System Oscillations

Engine Design

Roy Squires
Aerojet NTRE Is Small and Lightweight

Aerojet NTRE

<table>
<thead>
<tr>
<th>Component</th>
<th>PRR</th>
</tr>
</thead>
<tbody>
<tr>
<td>Thrust, lbf</td>
<td>75000</td>
</tr>
<tr>
<td>Chamber Pressure, psia</td>
<td>2000</td>
</tr>
<tr>
<td>Nozzle Area Ratio, A_r/A_t</td>
<td>300</td>
</tr>
<tr>
<td>Engine Specific Impulse, sec</td>
<td>915</td>
</tr>
<tr>
<td>Mars Mission Specific Impulse, sec</td>
<td>867</td>
</tr>
<tr>
<td>Engine Total Weight, lbf</td>
<td>11879</td>
</tr>
<tr>
<td>Thrust/Weight</td>
<td>6.1</td>
</tr>
<tr>
<td>Engines per Vehicle</td>
<td>2</td>
</tr>
<tr>
<td>Payload Returned to Earth, lbf</td>
<td>44900</td>
</tr>
</tbody>
</table>

Engine Weight Breakdown

- Uncooled Nozzle: 240 lbf
- Cooled Nozzle: 1000 lbf
- Pressure Vessel, Reactor Manifolds & CSS: 1501 lbf
- Reactor, Reactor H&C: 3932 lbf
- Turbopump Assemblies (2): 410 lbf
- Recuperator / Shield: 2168 lbf
- Secondary Shield: 521 lbf
- Plumbing/Valves: 1339 lbf
- Controls and Shielding: 737 lbf

<table>
<thead>
<tr>
<th>NTRE w/ Stage Power/Heat Removal</th>
<th>PRR</th>
</tr>
</thead>
<tbody>
<tr>
<td>Stage Power &amp; Heat Removal Sys, lbf</td>
<td>2000</td>
</tr>
<tr>
<td>Engine with Power Sys, lbf</td>
<td>11879</td>
</tr>
<tr>
<td>Mars Mission Specific Impulse, sec</td>
<td>867</td>
</tr>
<tr>
<td>Payload Returned to Earth, lbf</td>
<td>44900</td>
</tr>
</tbody>
</table>

NTP System Concepts
Aerojel NTRE

CAD TPA Dual Spool 3-D Color Exterior View
The XLR-134 program addressed the need for a relatively low thrust engine to move large fragile structures from low earth orbit into higher orbits. It was an Air Force program originating from the Phillips Lab at Edwards Air Force Base and spanned 1986 through 1990.

The program included initial studies to define the requirements and the engine size/cycle. From these requirements the engine and component designs were derived. The selected engine was a 500 lbft. LOX/LH\textsubscript{2} single expander cycle engine (gaseous hydrogen turbine drive). The turbopumps for both the LOX and LH\textsubscript{2} were designed and fabricated at Aerojet. The general arrangement consists of two shafts with 3 pump stages and one turbine stage on each mounted “end to end.” In this configuration the turbines are counter-rotating. The LOX TPA is basically a two stage single spool machine of a similar design as the LH\textsubscript{2} TPA with appropriate material and tolerance changes.

The LH\textsubscript{2} TPA was tested both as a single spool (3 stage) TPA and finally as the complete dual spool TPA. No development problems were encountered, due to the robust design and subcritical shaft speed. Of significant merit during dual spool testing was the start up and steady state operation of the two pump spools. This highly successful testing demonstrated over 4200 seconds of run time in LH\textsubscript{2} with full speed TPA operation, speed tracking of the two spools, successful bearing performance and subcritical shaft speed.

### Aerojet TPA Technology

**Increases Life and Reliability**

<table>
<thead>
<tr>
<th>Features</th>
<th>Benefits</th>
</tr>
</thead>
<tbody>
<tr>
<td>• Dual Spool</td>
<td>• High Turbine Efficiency</td>
</tr>
<tr>
<td></td>
<td>• Low Weight</td>
</tr>
<tr>
<td></td>
<td>• Commonality of Parts</td>
</tr>
<tr>
<td>– Short Shafts With 3 Rotors per</td>
<td>• Subcritical Shaft Speed Operation for Deep Throttling</td>
</tr>
<tr>
<td>• Operate Below Design Speed</td>
<td>• Increased Life and Reliability</td>
</tr>
<tr>
<td>• Hydrostatic Bearings</td>
<td>• Increased Life</td>
</tr>
</tbody>
</table>

![XLR-134 Fuel TPA](image)

- Energopool
- Babcock & Wilcox

NIP: System Concepts 270
Radiation transmission through manifolds includes 36 2.5 cm holes for gas flow. Shield penetrations for drum control rods and the central "poison" rod were ignored.

Heating in the LiH/Pb dedicated shield will be of the order 40-60 kW and may require some cooling during extended operation at full power.

"Internal Shield" Concept for NTRE Provides Significant Reduction in Accountable Shielding Mass
Source strength and shield attenuation calculated by B&W using MCNP (Monte Carlo Neutron Photon transport code).

NASA radiation specification met or exceeded at a point 1 meter above the top reflector on the core axis.

Shields for electronics and controls assumes optimum placement and 100K-rad hardened electronics.

Engine Components and Dedicated Shielding
Attenuate Radiation to Meet NASA Requirements
and Protect Electronics and Controls

<table>
<thead>
<tr>
<th>Components</th>
<th>Gamma Factor</th>
<th>Fast Neutron Factor</th>
<th>Mass (Kg)</th>
<th>Comments</th>
</tr>
</thead>
<tbody>
<tr>
<td>Gas Manifolds and Recuperator</td>
<td>101</td>
<td>21</td>
<td>1178</td>
<td>Dual Function: Cools and Shields</td>
</tr>
<tr>
<td>Dedicated Shield</td>
<td>4.4</td>
<td>80</td>
<td>236</td>
<td>Additional Shield Necessary to Meet NASA Spec</td>
</tr>
<tr>
<td>Distributed Electronics and Controls Shield</td>
<td>6.3 x 10⁴</td>
<td>1.75 x 10³</td>
<td>130</td>
<td>Required Beyond NASA Spec for 3.5 Hours at Full Power</td>
</tr>
</tbody>
</table>

Energopool - Babcock & Wilcox
Recuperator

Function

- Internal gamma shield cooled by LH$_2$
- Provides thermal energy for starting and operating TPA
- Enables high chamber pressure for lightweight, compact NTRE

Design and Performance Parameters

<table>
<thead>
<tr>
<th>Propellant</th>
<th>H$_2$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Cold-Side Inlet Pressure</td>
<td>4750 psia</td>
</tr>
<tr>
<td>Cold-Side Inlet Temperature</td>
<td>67°F</td>
</tr>
<tr>
<td>Cold-Side Flow Rate</td>
<td>68 lbm/sec</td>
</tr>
<tr>
<td>Cold-Side Pressure Drop</td>
<td>150 psid</td>
</tr>
<tr>
<td>Hot-Side Inlet Pressure</td>
<td>2650 psia</td>
</tr>
<tr>
<td>Hot-Side Inlet Temperature</td>
<td>775°F</td>
</tr>
<tr>
<td>Hot-Side Pressure Drop</td>
<td>150 psid</td>
</tr>
<tr>
<td>Thermal Load</td>
<td>126,000 Btu/sec</td>
</tr>
<tr>
<td>Envelope</td>
<td>40 in. dia x 7 in. height</td>
</tr>
<tr>
<td>Weight</td>
<td>2500 lbm</td>
</tr>
<tr>
<td>Material</td>
<td>CRES SS (A-286)</td>
</tr>
</tbody>
</table>

Characteristics

The 300 series stainless steel, platelet design, countercflow HEX accepts 83% of the LH$_2$ flow from the TPAs and heats the hydrogen to 57°F gas in the high pressure circuit of the HEX. The outflow cools the reflector and moderator, ensuring that LH$_2$ does not enter these components. This gas is combined with the 17% flow, which bypassed the HEX to cool the nozzle and pressure vessel and was gaseified in the process, to provide 100% flow at 84°F to drive the turbine. The turbine effluent then passes through the low pressure circuit of the HEX giving up much of its heat to the high pressure circuit before delivery to the reactors many fuel elements.

NTRE Recuperator Is Based on Aerojet SSME HEX Technology

SSME HEX

Energopool • Babcock & Wilcox
Cooled Nozzle

Function
- Provides DeLaval nozzle entrance section and exit to area ratio 10:1

Design and Performance Parameters

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Propellant</td>
<td>H2</td>
</tr>
<tr>
<td>Coolant Inlet Temperature</td>
<td>67°F</td>
</tr>
<tr>
<td>Coolant Inlet Pressure</td>
<td>4750 psi</td>
</tr>
<tr>
<td>Coolant Flow Rate</td>
<td>14 lbm/sec</td>
</tr>
<tr>
<td>Coolant Pressure Drop</td>
<td>700 psi</td>
</tr>
<tr>
<td>Throat Diameter</td>
<td>5.121 in.</td>
</tr>
<tr>
<td>Exit Area Ratio</td>
<td>10:1</td>
</tr>
<tr>
<td>Chamber Pressure</td>
<td>2000 psi</td>
</tr>
<tr>
<td>Gas Temperature</td>
<td>4859°F</td>
</tr>
<tr>
<td>Flowrate</td>
<td>82 lbm/sec</td>
</tr>
<tr>
<td>Material</td>
<td>ZrCu Liner/A286 SS Structure</td>
</tr>
<tr>
<td>Envelope</td>
<td>40 in. dia x 32 in. long</td>
</tr>
<tr>
<td>Weight</td>
<td>1000 lbm</td>
</tr>
<tr>
<td>Cylindrical Length</td>
<td>5.00 in.</td>
</tr>
<tr>
<td>Wall Temperature</td>
<td>600°F</td>
</tr>
</tbody>
</table>

Characteristics

The cooled nozzle uses a zirconium/copper, formed platelet liner to maintain wall temperature below the life limit. The liner will consist of 8 to 10 panels and include an approximate total of 400 coolant channels. It is bonded to a two-piece, A-286 jacket by a hot isostatic press (HIP) process. A two-piece, formed platelet A-286 throat stiffening shell provides structural support against bending moments. Its construction and cooling approach is similar to that of the pressure vessel shell. Coolant enters a manifold at the 10:1 area ratio and flows forward through the liner and shell wall as shown. It exits into the aft closure ring manifold of the pressure vessel.

Cooled Nozzle Concept Is Based on Current Technology

Formed Platelet Liner 40Klbf Chamber
Cooled Nozzle Concept

- Studies of the SSME main combustion chamber show wall temperature reductions of up to 200 F using platelet liner technology.
- Cooled hot gas wall
  - Formed platelet liner
  - ZrCu platelets
  - -400 channels
  - 8-10 panels
- A-286 structure
- Cooled throat support ring
  - Platelet A-286 structure with internal coolant channels formed into conical shape

Common Manifolding Provides Coolant for Nozzle and Throat Support Ring

- Energopool - Babcock & Wilcox
Pressure Vessel Function / Concept

- Pressure Containment
- Plenums / Manifolding for:
  - Moderator coolant
  - Control drum coolant
  - Core flow
  - Pressure vessel wall
- Interfaces
  - Recuperator
  - Cooled nozzle
  - Reactor
  - Core support

Pressure Vessel Provides Pressure Containment, Core Support and Manifolding

- Cooled Hot Gas Wall
- Formed Platelet Design
- A-286 Stainless
- 3-4 Sections
Pressure Vessel (PV)

Function
- Contains pressure and supports reactor
- Provides manifolding for H2
- Directs recuperator cold flow to control drum and moderator/reflector outflow
- Combines nozzle/PV coolant outflow with moderator/reflector outflow for delivery to turbine
- Delivers recuperator warm flow to reactor heating elements

Design & Performance Parameters

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Propellant</td>
<td>H2</td>
</tr>
<tr>
<td>Coolant Inlet Temperature</td>
<td>651°F</td>
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<tr>
<td>Coolant Inlet Pressure</td>
<td>4050 psia</td>
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<tr>
<td>Coolant Outlet Temperature</td>
<td>847°F</td>
</tr>
<tr>
<td>Coolant Flowrate</td>
<td>14 lbm/sec</td>
</tr>
<tr>
<td>Coolant Pressure Drop</td>
<td>150 psid</td>
</tr>
<tr>
<td>Core Propellant Temperature</td>
<td>357°F</td>
</tr>
<tr>
<td>Core Propellant Pressure</td>
<td>2500 psia</td>
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<tr>
<td>Core Propellant Flowrate</td>
<td>62 lbm/sec</td>
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<tr>
<td>Moderator Coolant Temperature</td>
<td>572°F</td>
</tr>
<tr>
<td>Moderator Coolant Pressure</td>
<td>4600 psia</td>
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<tr>
<td>Moderator Coolant Flowrate</td>
<td>50.5 lbm/sec</td>
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<tr>
<td>Reflector Coolant Temperature</td>
<td>572°F</td>
</tr>
<tr>
<td>Reflector Coolant Pressure</td>
<td>4600 psia</td>
</tr>
<tr>
<td>Reflector Coolant Flowrate</td>
<td>14.5 lbm/sec</td>
</tr>
<tr>
<td>Envelope Weight</td>
<td>2610 lbm</td>
</tr>
<tr>
<td>Envelope Material A286 SS</td>
<td></td>
</tr>
</tbody>
</table>

Characteristics

Formed, A-286, diffusion bonded platelet wall sections are welded together to make the right circular shell of the pressure vessel. The forward end of the shell is welded to a manifold assembly, which is welded to the recuperator. A coolant ring manifold is welded to the aft end of the shell. Annular closure rings are welded fore and aft as shown to combine flows per the engine system schematic.

Hydrogen gas enters the aft end manifold from the cooled nozzle. It flows through the shell coolant passages, etched into the wall platelets, and into a forward closure ring, where it mixes with the moderator/reflector coolant outflow for delivery to the TPA turbines. The foremost closure ring delivers recuperator flow to the moderator/reflector coolant passages.

Core Support Structure Provides Reactor Manifolding
A carbon-carbon nozzle extension for NTR is a cost-effective, low weight component. Carbon-carbon has good mechanical properties for temperatures in excess of 5000°F and will only suffer a total recession of less than 0.005 inch due to hydrogen chemical attack (assuming a temperature of 2500°F, pressure of 20 psia, and total duration of 4.5 hrs). Carbon-carbon is noted for radiation resistance and was baselined as the nozzle extension for the NERVA rocket engine at Aerojet.

Carbon-carbon structures can be fabricated in many different ways, but only several are appropriate for thin wall nozzle extensions. Involute, 3-D cylindrical, braided, and Novotex™ preforming are the four most realistic techniques to provide carbon-carbon nozzle extensions. None of these techniques can provide a single piece nozzle the size required without facility capitalization and development.

A one-piece carbon-carbon nozzle extension is estimated to weigh about 170 lbs and 240 lbs for area ratios nozzles of 200:1 and 300:1, respectively. The thicknesses reflect minimum wall thicknesses of approximately 0.5 in. and 0.2 inch for the entry and exit regions.

<table>
<thead>
<tr>
<th>Propellant</th>
<th>H2</th>
</tr>
</thead>
<tbody>
<tr>
<td>Temperature</td>
<td>4860°F</td>
</tr>
<tr>
<td>Flowrate</td>
<td>82 lbm/sec</td>
</tr>
<tr>
<td>Attach Area Ratio</td>
<td>10:1</td>
</tr>
<tr>
<td>Exit Area Ratio</td>
<td>200:1</td>
</tr>
<tr>
<td>Nozzle Shape</td>
<td>110% Bell</td>
</tr>
<tr>
<td>Material</td>
<td>Carbon/Carbon</td>
</tr>
<tr>
<td>Envelope</td>
<td>88.7 in. dia x 160 in. long</td>
</tr>
<tr>
<td>Weight</td>
<td>450 lbm</td>
</tr>
</tbody>
</table>

A Full Size One Piece Carbon-Carbon Nozzle Extension Will Weigh Less Than 450 lbs

However, Facilities Must Be Upgraded for Size and Nozzle Fabrication Must Be Validated

GenCorp
AERCUR
NTP: System Concepts

Energonull Babcock & Wilcox

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Instead of fabricating a one-piece carbon-carbon nozzle extension, fabricating carbon-carbon segments is an option which will not require facilitization nor extensive validation. A defect in a large one-piece carbon-carbon nozzle would cause the rejection of the whole nozzle or acceptance of materials of lower mechanical properties, while an unacceptable nozzle segment will only require the rejection of that one segment. The segmented carbon-carbon nozzle extension concept shortens the design and fabrication cycle by at least one year.

Aerojet has pursued the segmented nozzle approach under IRAD and has validated the mechanical approach via demonstration aluminum and fiberglass epoxy segments. The main drawback corresponds to the thickened sections in the nozzle to effect the segments attachment.

The segmentation approach is estimated to increase the weight 30%.

A Segmented Carbon-Carbon Nozzle, Though Heavier, Is Robust and Cost Effective

- **Fabricability**
  - Present Facilities Are Large Enough to Produce Required Segmented Pieces
  - Lower Rejectable (Only Bad Segments Need to Be Replaced, Not Whole Nozzle)

- **Compactness**
  - Disassembled Pieces Are Easy to Store, Ship, and Reassemble

- **Robustness**
  - Smaller Pieces Are Easier to Fabricate and Inspect ➡️ Stronger and Less Flaw Sensitivity

- **Cost Effectiveness**
  - Facility Upgrade Is Not Required
  - Shorter Schedule (Start With Design Plus Fab)

- **Penalties Are Acceptable**
  - Attachment Will Add Only 150 lbs

- **Energopool • Babcock & Wilcox**
COMPONENT: Quad Channel Fault Tolerant Controller (Three Channel Version Depicted)

FUNCTION: The Engine Controller is responsible for closed loop control of the NTRE engine and auxiliary power generation components.

The engine controller performs a complete engine system checkout and calibration prior to engine operation. This includes calibration of the individual instruments and control system components. During the start sequence the controller controls reactor reactivity, pump chill, turbopump ramp up, and power ramp up.

The engine controller actively controls engine steady state operation to maximize engine Isp. Engine power down and post firing cool down is actively controlled to minimize propellant usage and maximize total engine Isp.

The engine controller performs periodic engine system health monitoring and life prediction throughout engine operation.

Brayton cycle power generation is actively controlled throughout the mission. The controller is capable of adjusting the power output level over a 5:1 range to meet varying mission demands.

ARCHITECTURE: The NTRE engine controller is a 32-bit full voting four channel fault tolerant processor (FTP) that is derived from the Charles Stark Draper FTP architecture. The four channel controller provides full Fail Op/Fall Safe operation (higher levels of fault tolerance are available with degraded fault coverage).

Additionally, the four channels are electrically and mechanically isolated from each other. This prevents a catastrophic electrical failure from propagating from one channel to the next.

The 32-bit Intel 80960 microprocessor provides the processing power for the engine controller. The 80960 is optimized to efficiently execute the Ada language. This central processor provides many advanced enhancements such as automatic exception handling and memory management that facilitate the efficient processing of Ada language. Over 2Mb of memory is provided on the digital computer unit module. This complement of memory is more than sufficient for both engine system control code and advanced health monitoring and life prediction algorithms.


SIZE: 8 in. x 16 in. x 10 in.

WEIGHT: 59 lbs (46 lbs Electronics + 13 lbs Shield)

TOTAL DOSE: 100 K RADs (SI)

Advanced Fault Tolerant Controller
Improves Mission Reliability
COMPONENT: Dual Channel Electro-Mechanical Actuator

FUNCTION: Provide actuation for modulating valves over a -1 to 90 deg arc. The EMA feature load insensitivity and high positional accuracy/repeatability.

ARCHITECTURE: The EMA is a fully redundant actuator featuring dual channel redundant bus interfaces, dual redundant power interfaces, dual channel redundant control electronics, dual electric redundant motors on a common shaft, and dual resolvers on a common shaft. The two EMA channels contain no electrical cross strapping and are mechanically isolated from each other. This prevents a catastrophic electronics failure in one channel from propagating to the other channel.

INTERFACES: 2 MIL-STD-1553B Valve Command Channels, 2 Power Buses

SIZE: 4 in. x 6 in. x 10 in.

WEIGHT: 31 lbs (9 lbs Electronics/Mechanics + 32 lbs Shield)

TOTAL DOSE: 100K RADs(Si)

TORQUE: 600 In.-lb

SLEW RATE: 360 deg/sec

POSITIONAL ACCURACY/REPEATABILITY: ± 0.5 deg

Advanced Electro-Mechanical Actuator Combines High Torque and Small Size

Energopool • Babcock & Wilcox

NP-TIM-92 281 NTP: System Concepts
COMPONENT: Quad Channel Control System

FUNCTION: Provide full Fail Op/Fail Safe engine and auxiliary power generation control.

ARCHITECTURE: The control system is designed with a high degree of symmetry and redundancy. The symmetry of the control system greatly reduces the complexity of the redundancy management software and improves system reliability and verifiability. Critical control valves such as the engine isolation valves are fully quadded thus allowing them to tolerate one stuck open or one stuck shut failure. Other valves are either simple or dual (serial or parallel) depending upon the function of the valve.

There are two interfaces to the electro-mechanical actuators. These two interfaces are referred to as the Active Effector Control Bus and the Passive Effector Control Bus. Each control bus is actually a redundant 1553B implementation. This provides a total of four data paths to each actuator thus providing full Fail Op/Fail Safe capabilities of the control system.

Each solenoid actuator has dual coils. This provides fully redundant interfaces to the engine controller. Like the effector control buses the solenoid interfaces are organized as active and passive interfaces. Additionally solenoids contain a mechanical preload that forces the solenoid into a safe position in the event of a total interface failure.

Critical engine parameters, such as chamber pressure, are fully quad redundant. Other parameters such as moderator temperature are simplex or dual per moderator element as called for by FEMA reliability analysis.

Heavy use will be made of sensor analytyical redundancy techniques. These techniques allow a failed parameter to be substituted by using a physical model and related measurements.

INTERFACES: 2 MIL-STD-1553B Valve Command Channels, 2 Power Buses

SIZE: 8 in. x 16 in. x 10 in.

WEIGHT: 31 lbs (9 lbs Electronics/Mechanics + 32 lbs Shield)

TOTAL DOSE: 100K RADs(Si)

TORQUE: 600 In-lb

Quad Channel Control System
Improves Mission Reliability
COMPONENT: Operational Flight Program

FUNCTION: The Operational Flight Program (OFP) provides the control, health monitoring and life prediction capabilities seen in the control system. All of the dynamic engine control laws are found in the OFP. Engine system health monitoring and life prediction algorithms are also resident within the OFP. Additionally the OFP manages the interface hardware within the engine controller.

The OFP implements the required vehicle communications protocols and validates commands. Additionally status and data packets are sent back to the vehicle.

ARCHITECTURE: The OFP design is based on a highly modular, structured, functional decomposition of the required functionality. Related functions are combined into modules. Thus all the engine control functions are grouped into the Engine Control Module; all the Health Monitoring functions are grouped within the Health Monitoring Module. Modules have rigid functional, interface, protocol, and temporal specifications. These specifications minimize the interactions between modules, increasing software reliability and reducing verification and validation efforts.

The modular architecture allows individual modules to be upgraded throughout the life of the NTRE program while preserving the software investment. Modules are designed to be "plugged in" in a manner similar to mechanical components thus reducing the costs associated with software verification and validation.

INTERFACES: Controller/IO Devices
Integrated Engine

Mel Bulman

Integrated NTRE Improves Mission Performance

The SEI stage will require many subsystems in addition to the engine and tanks. Our integrated NTRE includes a number of systems normally assigned to the stage. It can provide reaction control and orbit maneuvering thrust during coast. During the main burn, the engine can provide autogenous tank pressurization and electrical power. After shutdown, the reactor can be used as a heat source for generating up to 100 kW (e) per engine. All of this is accomplished at lower weight than if separate systems are employed to achieve these functions.
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 Pressurized $H_2$ From Moderator Loop
  - TPA Acceleration Dominated by Engine Thermal Mass (Power For Approximately 10 Starts In Recuperator Alone)

Integrated Engine Increases Start Reliability and Safety

- Pre-Chill Pumps
- Fast Start Reduces Isp Loss, Improves Navigational Accuracy
- Immediate Restart Capability ($\times 10$ times)
  Enhanced Multiple Engine Start

* Original Page is of Poor Quality

NP-TIM-92 285 NTP: System Concepts
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 Optimum Path to Full Power, Balancing:
  - Isp Loss
  - Fuel Element Thermal Shock

Our Integrated Engine Starts More Reliably
And With Less Impulse Loss than Nerva Type Engines
Tapping into the hot H₂ in the moderator loop of our integrated NTRE during operation allows us to generate up to 100 kW(e), attitude control impulse, and tank pressurization at lower cost than if provided by separate systems.

**Integrated Engine Provides RCS and Tank Pressurization During “Burns”**

- Two TPAs Allow 20:1 Throttling

*Energopool* • *Babcock & Wilcox*
The decay heat build up in the engine during the main burn must be removed from the reactor or it will overheat. Decay heat persists for days even after a short fifteen minute burn. Our integrated engine can reject a significant portion of the heat through its radiators, greatly reducing the expenditure of propellant to cool the engine. This reduces vehicle mass (IMLEO).

Integrated Engine Saves Over 100,000 lbm LH₂

- Shutdown Fission Power and Throttle on Decay Heat
- Shutdown One TPA, Throttle to 5% Thrust
- Use Pulse Cool Valve With Radiator Rejecting Heat, Feed Pump Provides Make-up
- Electrical Power and RCS Are Available Throughout Cooldown

GenCorp - Energopool - Babcock & Wilcox
Our Closed Cycle Cooling System Saves Over 7500 lbm During Perigee Pulsing

### Conventional Cooling System

<table>
<thead>
<tr>
<th>Phase 1</th>
<th>Cool 1</th>
<th>Phase 2</th>
<th>Cool 2</th>
<th>Phase 3</th>
<th>Cool 3</th>
</tr>
</thead>
<tbody>
<tr>
<td>Initial Mass (lbm)</td>
<td>1000000</td>
<td>850842</td>
<td>840942</td>
<td>95357</td>
<td>93276</td>
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<tr>
<td>Burn Time (Sec)</td>
<td>1600</td>
<td>968</td>
<td>968</td>
<td>1006</td>
<td>932</td>
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<tr>
<td>Propellant Consumed (lbm)</td>
<td>149158</td>
<td>9990</td>
<td>145586</td>
<td>10692</td>
<td>141694</td>
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<tr>
<td>Effective isp (Sec)</td>
<td>915</td>
<td>640</td>
<td>815</td>
<td>600</td>
<td>915</td>
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<tr>
<td>Final Mass (lbm)</td>
<td>850842</td>
<td>840942</td>
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<td>684655</td>
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<tr>
<td>AV (ft/sec)</td>
<td>4755</td>
<td>796</td>
<td>299</td>
<td>6825</td>
<td>384</td>
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<tr>
<td>Mission Velocity (ft/sec)</td>
<td>4755</td>
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<td>10591</td>
<td>10890</td>
<td>17716</td>
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<tr>
<td>Mission Effective isp (Sec)</td>
<td>896</td>
<td>900</td>
<td>694</td>
<td>902</td>
<td>892</td>
</tr>
</tbody>
</table>

### Closed Cycle Cooling System

<table>
<thead>
<tr>
<th>Phase 1</th>
<th>Cool 1</th>
<th>Phase 2</th>
<th>Cool 2</th>
<th>Phase 3</th>
<th>Cool 3</th>
</tr>
</thead>
<tbody>
<tr>
<td>Initial Mass (lbm)</td>
<td>1000000</td>
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<td>843520</td>
<td>692280</td>
<td>690080</td>
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<td>Burn Time (Sec)</td>
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<td>174 Min</td>
<td>995</td>
<td>1006 Min</td>
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<td>2200</td>
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<td>148094</td>
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<tr>
<td>Effective isp (Sec)</td>
<td>915</td>
<td>760</td>
<td>915</td>
<td>760</td>
<td>915</td>
</tr>
<tr>
<td>Final Mass (lbm)</td>
<td>845720</td>
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<td>692280</td>
<td>690080</td>
<td>541986</td>
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<td>AV (ft/sec)</td>
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<td>5816</td>
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<td>Mission Velocity (ft/sec)</td>
<td>4832</td>
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<td>10812</td>
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<td>18001</td>
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<tr>
<td>Mission Effective isp (Sec)</td>
<td>913</td>
<td>914</td>
<td>913</td>
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<td>913</td>
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<tr>
<td>Propellant savings (lbm)</td>
<td>2578</td>
<td>5415</td>
<td>5415</td>
<td>7505</td>
<td></td>
</tr>
</tbody>
</table>

Mission Average Isp Improved 21 Seconds With Closed Cycle Cooling

### Closed Cycle Cooling Reduces IMLEO Over 8%

950 Sec. Burn @ 150,000 lbf (2-3 Engines)

Closed Cycle Benefits
- Reduced Coolant Ejection
- Higher Ejection Isp (cut off low Temp. Tail)
- 75% Reduction Mission Isp Loss f(Δisp & ΔMass)
- Reduced g Loss (Cooldown Impulse Delivered Faster)
- Closed Cycle Has All Brayton Power Cycle Components (Except Generator)

GenCorp - Energopool - Babcock & Wilcox

NP-TIM-92 289 NTP: System Concepts
Our Integrated NTRE Can Reduce IMLEQ 100-200+ Klbm

<table>
<thead>
<tr>
<th>Conventional Cooling System (H2/O2 OME System)</th>
<th>Payload Returned</th>
<th>Burn 1</th>
<th>OMS 1</th>
<th>Burn 2</th>
<th>OMS 2</th>
<th>Burn 3</th>
<th>OMS 2</th>
<th>Burn 4</th>
</tr>
</thead>
<tbody>
<tr>
<td>Initial Mass (Lbm)</td>
<td></td>
<td>1803176</td>
<td>871599</td>
<td>812424</td>
<td>397457</td>
<td>243063</td>
<td>228305</td>
<td>228305</td>
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<tr>
<td>Propellant Consumed (Lbm)</td>
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<td>846852</td>
<td>58175</td>
<td>2812622</td>
<td>140333</td>
<td>142264</td>
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<tr>
<td>Effective Isp (Sec)</td>
<td></td>
<td>892</td>
<td>450</td>
<td>892</td>
<td>892</td>
<td>892</td>
<td>892</td>
<td></td>
</tr>
<tr>
<td>Final Mass (Lbm)</td>
<td></td>
<td>953288</td>
<td>813424</td>
<td>5261872</td>
<td>257104</td>
<td>226845</td>
<td>123854</td>
<td></td>
</tr>
<tr>
<td>AV (ft/sec.)</td>
<td></td>
<td>18200</td>
<td>1000</td>
<td>12500</td>
<td>12500</td>
<td>12500</td>
<td>12500</td>
<td>12500</td>
</tr>
<tr>
<td>Mission Velocity (Fe/sec.)</td>
<td></td>
<td>18200</td>
<td>19200</td>
<td>31700</td>
<td>44200</td>
<td>45200</td>
<td>63400</td>
<td></td>
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<tr>
<td>Δ IMLEO (Klbm%)</td>
<td></td>
<td>257</td>
<td>16.64</td>
<td>1684</td>
<td>50000</td>
<td></td>
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</tr>
</tbody>
</table>

| Over Integ. NTRE                                |                  |        |       |       |       |       |       |       |

<table>
<thead>
<tr>
<th>Conventional Cooling System With Main Engine</th>
<th>Restart for Plane Change Maneuver</th>
<th>Burn 1</th>
<th>OMS 1</th>
<th>Burn 2</th>
<th>OMS 2</th>
<th>Burn 3</th>
<th>OMS 2</th>
<th>Burn 4</th>
</tr>
</thead>
<tbody>
<tr>
<td>Initial Mass (Lbm)</td>
<td></td>
<td>1678028</td>
<td>841106</td>
<td>275875</td>
<td>374444</td>
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<td>Propellant Consumed (Lbm)</td>
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<td>35231</td>
<td>273683</td>
<td>132744</td>
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<td>Effective Isp (Sec)</td>
<td></td>
<td>892</td>
<td>700</td>
<td>892</td>
<td>892</td>
<td>700</td>
<td>892</td>
<td></td>
</tr>
<tr>
<td>Final Mass (Lbm)</td>
<td></td>
<td>9532916</td>
<td>71883</td>
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<tr>
<td>AV (ft/sec.)</td>
<td></td>
<td>18200</td>
<td>1000</td>
<td>12500</td>
<td>12500</td>
<td>12500</td>
<td>12500</td>
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</tr>
<tr>
<td>Mission Velocity (Fe/sec.)</td>
<td></td>
<td>18200</td>
<td>19200</td>
<td>31700</td>
<td>44200</td>
<td>45200</td>
<td>63400</td>
<td></td>
</tr>
<tr>
<td>Δ IMLEO (Klbm%)</td>
<td></td>
<td>132</td>
<td>8.55</td>
<td>8.55</td>
<td>50000</td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

| Over Integ. NTRE                                |                                  |        |       |       |       |       |       |       |

<table>
<thead>
<tr>
<th>Integrated NTRE Closed Cycle Cooling System</th>
<th>Nuclear OMS (without restart)</th>
<th>Burn 1</th>
<th>OMS 1</th>
<th>Burn 2</th>
<th>OMS 2</th>
<th>Burn 3</th>
<th>OMS 2</th>
<th>Burn 4</th>
</tr>
</thead>
<tbody>
<tr>
<td>Initial Mass (Lbm)</td>
<td></td>
<td>1545680</td>
<td>760445</td>
<td>727453</td>
<td>350094</td>
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<tr>
<td>Propellant Consumed (Lbm)</td>
<td></td>
<td>713539</td>
<td>33032</td>
<td>252415</td>
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<td>94959</td>
<td>99761</td>
<td></td>
</tr>
<tr>
<td>Effective Isp (Sec)</td>
<td></td>
<td>913</td>
<td>760</td>
<td>913</td>
<td>913</td>
<td>760</td>
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<td></td>
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<tr>
<td>Final Mass (Lbm)</td>
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<td>475308</td>
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<td>227603</td>
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<tr>
<td>AV (ft/sec.)</td>
<td></td>
<td>18200</td>
<td>1000</td>
<td>12500</td>
<td>12500</td>
<td>12500</td>
<td>12500</td>
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</tr>
<tr>
<td>Mission Velocity (Fe/sec.)</td>
<td></td>
<td>18200</td>
<td>19200</td>
<td>31700</td>
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<tr>
<td>Δ IMLEO (Klbm%)</td>
<td></td>
<td>132</td>
<td>8.55</td>
<td>8.55</td>
<td>50000</td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

* Four Burn Full Up Mars Mission

10/18/92

Closed Cycle Cooling Can Reduce IMLEO Over 100K lb*

* Four Burn Full Up Mars Mission

Gulfstream - Energopool - Babcock & Wilcox

Aerojet

NTP: System Concepts 290

NP-70M-92
During coast, our integrated engine is kept warm while generating up to 100 kW(e). The mean electrical power will be less than 100 kW(e). At 20 kW(e) the Brayton cycle efficiency is approximately 30% requiring a thermal power of approximately 60 kW(t), which causes only a small additional burn up of the reactor fuel.

Integrated Engine Eliminates Additional Stage Components and Improves Stage Performance

- Main Tank Refrigeration is an Option

- Energopool - Babcock & Wilcox
The Integrated engine adds life to the fuel, because it allows fuel to be kept warm during coast (-1800 K). Therefore, the fuel will see only -1000 K AT's during startup and shutdown. 1800K was chosen to balance the effects of vaporization rate, thermal cycling, and power cycle efficiency.

Engine Does Not Cool Fully Between Major Burns

Aerojet Cycle Benefits
- Reduced Thermal Shock
- Integrated Power Supply
- ACS and OMS Power
- Full Thrust Available on Short Notice

Energopool • Babcock & Wilcox
Reliability and Safety
Mel Bulman

Our Turbopump System Improves Mission Reliability

- Single TPA System Has ~ 4 Times the Probability of Total Failure vs 2 TPAs
- Twin Spool 4 Stage Pump Is More Reliable Than Single Shaft TPAs at the Same Discharge Pressure

*Industry Standard Component Failure Rates Applied to Feed Systems
Dual Spool TPA Provides High Margin for NTRE

- Impellers Stressed Less for Same Weight and Performance – Less ΔP Per Impeller
- Four Bearings to Share Loads Rather Than Two
- Unshrouded, Machined Impellers Have Higher Strength Than Casting
- Runs Subcritical for Less Dynamic Stress

Reliability Increased With Lower Stressed Parts

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NTRE – Concept Design

Failure Modes and Effects Analysis

A Failure Modes and Effects Analysis was completed for the Particle Bed Reactor concept. The failure modes of the major components were identified. The criticality and effect of each mode was determined and possible ways to minimize the occurrence of each mode were also identified. This analysis will be expanded and updated during the preliminary design phase to reflect design maturation. The FMEA will be the basis for developing a Reliability Fault Tree, which will show system interactions graphically. Quantitative evaluation of the base events of the Fault Tree will allow reliability predictions to be made of the system. The Fault Tree will be developed using CAFTA, a Computer Aided Fault Tree Analysis code, which facilitates reliability and system safety analysis of complex systems.

NTRE – Concept Design

Failure Modes and Effects Analysis

- Preliminary FMEA Has Been Completed for Engine Concept
  - Component Failure Modes Identified
  - Actions to Minimize Occurrence Are Incorporated
    Redundancy
    Robust Design
    Adequate Testing and Inspection
- FMEA to Be Updated During Design Phase to Reflect Maturing Design
NTRE – Concept Design

Reliability

Methodology to Evaluate the Effect of Redundant Components on the System

Reliability block diagrams and industry standard failure rates of like components will be used to assist in the evaluation of the effect of redundancies of various components on the reliability of the system.

A system level Fault Tree will be developed which will be used to analyze the reliability of the system during the various operating phases of the proposed mission.

A system Fault Tree has the advantage of being able to see graphically the interactions of a complex system. It is difficult to model these interactions using only block diagrams. Block diagrams are useful in studying effects on the system of series redundancy or parallel redundancy of a few components. But the overall system reliability is better evaluated by doing a quantitative evaluation of a system Fault Tree.

A reliability Fault Tree differs from a system safety Fault Tree only in the definition of the top event. Process and human errors resulting in system failure are always included in the system safety Fault Tree. They can also be included in the Reliability Fault Tree. If the purpose of the Reliability Fault Tree is to assess the reliability of the design then the possible process and human errors would not be included in this Fault Tree.

• Reliability Block Diagrams and Failure Probabilities Using Industry Standard Failure Rates for Like Components Will Be Used to Evaluate Need for Redundant Components

• Reliability Fault Tree Will be Developed of Overall System to Be Used in Evaluating System Reliability

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NTRE Design Criteria

- Safety Factors

**Pressure Loads**

Yield Safety Factor $F_{Sy} = 1.25$
Ultimate Safety Factor $F_{Su} = 1.50$

**Thermal Loads**

Yield Safety Factor $F_{Sy} = 1.00$
Ultimate Safety Factor $F_{Su} = 1.00$

- Margin of Safety

$MS = \text{(Allowable Stress)} \times (FS \times \text{Applied Stress}) - 1.0$

System Safety

Design Includes Hazard Elimination and Control Provisions for Wide Range of Potential Hazards

- A Preliminary Hazard Analysis Has Been Accomplished

- This Analysis Is the Initial System Safety Task Which Included a Review of the NTRE Components, Potential Hazardous Conditions, Effect on the System if an Undesirable Condition Took Place, and Recommended Controls in the Design to Prevent an Occurrence From Happening

  - In Addition to Component Review, Natural Environment, Oxygen Rich Environment, and Aerospace Ground Equipment Hazards Were Considered

- This Study Illustrates a Combination of 17 Hazardous Conditions That Were Considered
Nuclear Safety

- Water Immersion
  - Most Reactor Voids Filled With Poison During Launch
  - Reactor Design Remains Subcritical
    - Fuel Internally Retained
  - Launch Safety/Emergency Shutdown Procedures
  - Qualification and Acceptance Tests

- Impact Compaction (Ground)
  - Collision Reduces Void Fraction
  - Reactor Design Ensures/Remains Subcritical
    - Fuel Internally Retained
  - Launch Safety/Emergency Shutdown Procedures
  - Qualification and Acceptance Tests

Nuclear Safety

Emergency Cool-down Procedure

- 1 TPA Failure
  - Fast (~ 1 sec) or Slow (~ 30 Seconds) Single Failure
    - Normal Reactor Shutdown
    - Cool-down at High Power With 2nd TPA
    - Continue Mission With 2nd TPA

- 2 TPA Failure
  - 1 Fails Fast, 1 Fails Slowly
    - Shutdown Reaction at High Power With Slowly Failing TPA
    - Employ Pulsed Cooling System Prior to 2nd TPA Failure

- 2 TPA Failure
  - 2 Fail Fast or Nearly Simultaneously
    - Shutdown Reactor
    - Cool-down at High Power With Crossover System
Nuclear Safety
Reactors Leakage Potential Is Minimized

- Use of Non-Radiation Embrittlement Materials
- Maintain Within Temperature Extremes
- Develop Approved Installation Procedures
- Post-Reactor Installation Leak Checks
- Test Area Monitored for Leakage
- Non-Nuclear Qualification and Acceptance Tests

Minimize Leakage Effect

- Radiation Hardened Material/Electronics
- Shielding
- Qualification and Acceptance Tests

Nuclear Safety
Design Controls Radiation Hazards to a Minimum

- Shielding Material – "Burn-In" Process
- Internal Shielding
  - Attenuates Levels at Propellant Tank
  - Reduces Levels to Engine and Stage Components
- External Shielding
  - Reduces Level to Crew and Stage
- Components Nuclear Hardened
- Proof-of-Concept Testing
Nuclear Safety

Reactor Risks and Hazards Are Minimized, i.e., System Design

- Controller Architecture
  - Diagnostic Instrumentation Monitors Reactor
  - Quad Channel Fault Tolerance Operation
  - Software Redundancy Design
  - Multiple Signals Required to Activate Valves
- Reactor
  - Control Drums Have Redundant Drive Motors and Couplings
  - Safety Rods Have Redundant Drive Motors and Couplings
- Shielding of Safety Rod Drives Ensures Rod In or Out Control Capability
- Non-Nuclear Vibration, Thermal and Shock Tests to Verify Structural Integrity

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Nuclear Safety

Design Controls All Identified Energy Sources

- Reactor
- Pressurized Propellant Feed Lines/Fittings/Valves
- Turbopump Assembly
- Pyrotechnic Isolation Valves
- Electronics
- Hydrogen in Tank

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Aerojet
- Energopool
- Babcock & Wilcox
System Safety

Summary

- Nuclear Safety Hazards Will Be Controlled Through Preventative Measures
  - Margins
  - Redundancy
  - Diversity
- NTRE Design Will Meet the Applicable Safety Requirements for Operation on the Eastern Test Range or Western Test Range
- The APD/B&W Team Will:
  - Ensure Design Meets Proof-of-Concept Objectives With Risk Reduced to as Low as Reasonably Achievable
  - Support the Nuclear Safety Policy Working Group Recommendations as Applicable

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PBR Engine Sensitivity Studies

Mel Bulman
CIS*  PBR  40K PBR  25K PBR  25K X PBR  10K X PBR  500K X PBR  1000 psia
Thrust, lbf 75000 75000 50000 50000 75000 75000 75000 75000 75000
Chamber Pressure, psia 75000 75000 50000 50000 75000 75000 75000 75000 75000
Engine Specific Impulse, sec 300 300 300 300 300 300 300 300 300
Mass Flow Exhaust Specific Impulse, sec 959 959 959 959 959 959 959 959 959
Engine Total Weight, lbm 51500 51500 51500 51500 51500 51500 51500 51500 51500
Thrust/Weight 6.4 6.5 6.5 6.5 6.2 6.2 6.7 6.7 6.8
Engines per Vehicle 2 2 2 2 2 2 2 2 2
Payload Returned to Earth, lbm 47687 44900 27947 98997 24682 34282 62201 44222

Engine Weight Breakdown

Component Weight, lbm
Uncoupled Nozzle 272 216 173 139 109 89 70 58 50
Cooled Nozzle 955 1056 728 577 438 329 260 214 200
Pressure Vessel, Reactor Monitors & CIS 2526 1551 701 513 352 238 172 126 106
Reactors, Reactor SIG 5413 2882 2264 1818 1375 1005 776 658 587
Thermal Groove Assemblies (5) 410 410 328 262 200 142 102 86 70
Reactor None: Shield 2424 2234 1870 1495 1117 872 660 556 480
Secondary Shield 643 643 573 573 573 573 573 573 573
Plumbing/Valves 1320 1320 1064 762 560 418 330 270 228
Controls & Plumbing 758 758 646 461 365 292 230 186 156

NTES or Stage Payoff/Removal

Stage Proper & Fuel Removal Sys Wt, lbm 3200 3200 1285 890 1285 1500 2000 2000 2000
Engine with Power Rect Wt, lbm 13720 13720 8563 7123 7123 10227 17361 13572 13572
Mass Flow Exhaust Specific Impulse, sec 959 959 959 959 959 959 959 959 959
Payload Returned to Earth, lbm 52950 52950 43249 33161 58168 39511 47518 49522

* CIS Engine Fuel Bundle Power Density is 12 MW/liter. Therefore we expect its weight to scale approximately as the PBR engine with power density.

High Thrust and Power Density Increase Engine Thrust-to-Weight

Energopool - Babcock & Wilcox

NTP: System Concepts
Our PBR engine concept is best summarized by including the rationale behind the selection of each major subsystem concept or operating parameter. 2700 K mixed mean reactor outlet gas temperature is selected by B&W fuel experts to meet the 4.5 hour life requirement with an appropriate margin by the end of fuel assembly development. The engine design/operating selection of 300:1 nozzle area ratio and nozzle inlet pressure of 2,000 psia is the result of a Mars mission payload trade study; it gives the best combination of engine specific impulse and weight. A recuperated turbopump drive cycle was selected for several reasons: (1) nozzle inlet pressures of 1000 psia and above are enabled by recycling topping heat through the turbine, and no reactor manifolding need be added to extract turbine drive heat directly from the core, (2) engine start and shutdown transients are smoothed and assisted by the large, available heat capacity of the high surface area recuperator, (3) the steel heat exchanger adds no weight to the engine, because its weight is determined by its other duties as a large part of its internal gamma shield and as the forward closure of the reactor vessel. A forward core support structure was selected, largely because it is the American experience base. The forward structure is cool, it forms part of the gamma shield, and it is used as propellant manifolding. Fuel assemblies operate in tension and are constructed of steel and beryllium, according to U.S. experience. A single DeLaval nozzle is selected that is internally cooled with hydrogen; no hydrogen bleed is necessary, because our formed platelet nozzle operates with low internal wall temperatures at high heat fluxes. A 40 Klb nozzle of similar design, material and coolant is now in test at NASA. The nozzle is small, because of the engine's high operating pressure, and we use a low weight, carbon-carbon nozzle extension. Its surface may be converted to ZrC to improve its life in hydrogen environment, using near term technology processes similar to those in work at Aerojet, however this is probably unnecessary, because total surface recession in 4.5 hours of operation is expected to be less than 0.025 in. (0.64 mm). A LiH neutron shield is encapsulated in aluminum and located external to the recuperator in order to simplify the core support structure. A secondary gamma shield is located at its forward face to provide sufficient gamma attenuation at all times during engine life. Both shields are cooled by propellant flow.

**Design Rationale for NTRE With PBR Is Clear**

<table>
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<tr>
<th>Design Parameter</th>
<th>Rationale</th>
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<td>1 H₂ Mixed Mean Outlet Temperature (2,700K)</td>
<td>Expected 4.5 Hours Fuel Life</td>
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<td>2 Pₜ (2,000 psia)</td>
<td>Best Mars Mission Performance With Reusable Engines</td>
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<td>4 Power Cycle (Recuperated)</td>
<td>• Enable High Pressure With Simple Reactor • Enhances Transient Operation • No Weight Penalty (γ Shield)</td>
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<td>5 Core Support (forward)</td>
<td>U.S. Experience Base</td>
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<td>6 Nozzle (Single DeLaval) (H₂ Cooled Without Bleed Flow) (C-C Extension)</td>
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<td>7 Neutron Shield (External)</td>
<td>Simplifies Core Support</td>
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</tbody>
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*GenCorp* - Energopool - Babcock & Wilcox
REACTOR DESIGN PHILOSOPHY

BABCOCK & WILCOX HAS APPLIED ITS KNOWLEDGE OF THE PARTICLE BED REACTOR (PBR) TO MEET THE NASA DESIGN REQUIREMENTS. THE OBJECTIVES OF THE DESIGN WERE TO STAY WITHIN THE TECHNOLOGY BASE FOR THE PBR. THIS INCLUDES THE MIXED MEAN EXHAUST GAS TEMPERATURE, ENGINE PERFORMANCE AND PRIMARILY THE SYSTEM SAFETY. THE PBR IS CAPABLE OF VERY HIGH T/W RATIOS HOWEVER, FOR MAN-FATED SYSTEMS OUR BASELINE DESIGN HAS BEEN CONSERVATIVELY DESIGNED AND INCORPORATES ROBUST AND THEREFORE RELIABLE COMPONENTS. THE PBR CONCEPT HAS THEREFORE INCURRED SOME MASS PENALTIES WHICH ARE BELIEVED TO BE PRUDENT IN TERMS OF SAFETY FOR THE CREW.

REACTOR DESIGN PHILOSOPHY:

STAY WITHIN THE TECHNOLOGY

- NASA Requirements - Isp/Gas Temperature
- Mass Penalties Accepted
- Highest Possible Performance is NOT Top Priority

- SAFETY IS -
PBR Provides a Compact High Power Engine

- Fuel Element & Moderator
- Control Drum
- Manifolds
- Recuperator/Internal Shield
- Nozzle

THE PBR IS A PARTICULARLY ATTRACTION CONCEPT DUE TO THE BURDEN AREA OF THE PARTICLE FUEL FORM. THE HEAT TRANSFER CAPABILITY IS UNSURPASSED. SMALL PARTICLES, BY DEFINITION, ALSO LIMIT THE THERMAL DISSESS WITHIN THE PARTICLES DUE TO THE SHORT CONDUCTION PATH LENGTH. THESE FEATURES ALLOW THE PBR TO OPERATE AT HIGH POWER DENSITY AND THEREFORE REQUIRE LESS CORE VOLUME. SMALLER CORE VOLUME TRANSLATES TO REDUCTIONS IN VESSEL DIAMETER AND SHIELD DIAMETER, ALL OF WHICH TRANSLATES TO LOWER MASS (OR HIGHER T/W).

THE HETEROGENEOUS CORE UTILIZES 30 FUEL ELEMENTS ON A PITCH OF 11.7 M. THE SUPPORT FOR THE CORE IS ACCOMMODATED BY THE MANIFOLD STRUCTURE. THE CORE IS "HUNG" FROM THE TOP WHERE COOL GAS HELPS KEEP THE STRUCTURE WITHIN ITS ALLOWABLE TEMPERATURE REGIME. BERYLLIUM MODERATOR HEXAGONS SURROUND EACH FUEL ELEMENT AND ACT AS ITS PRIMARY STRUCTURAL SUPPORT.

A TOTAL OF 18 ROTATING CONTROL DRUMS SURROUND THE CORE. THESE DRUMS CONTAIN A NEUTRON POISON (B,C) SEGMENT TO CONTROL THE POWER OF THE REACTOR. A SAFETY ROD IS LOCATED AT THE CENTERLINE OF THE CORE. IT IS AN AXIALLY CONTROLLED POISON ROD (B,C) AND IS A REDUNDANT SHUT DOWN SYSTEM IN THE EVENT THE CONTROL DRUMS BECOME INOPERABLE.

THE RECUPERATOR SERVES MANY FUNCTIONS AND IS NECESSARY FOR POWER BALANCE. IT "RECIRCLES" WASTE THERMAL ENERGY TO DRIVE THE TURBOPUMPS. IN ADDITION, IT PROVIDES A SIGNIFICANT PORTION OF THE INTERNAL SHIELDING FOR THE TURBOPUMPS. PLATELET TECHNOLOGY WILL BE USED TO FABRICATE THE RECUPERATOR AND THE COOLED PORTION OF THE NOZZLE.
THE RECUPERATED CYCLE BENEFITS THE ENGINE IN MANY WAYS. THE RECUPERATOR CONTRIBUTES TO THE
SHEILDING REQUIREMENTS (PARTICULARLY GAMMA). IT ENABLES HIGH CHAMBER PRESSURE BY SATISFYING THE
POWER BALANCE OF THE SYSTEM. IT ALSO HOLDS A LARGE AMOUNT OF THERMAL ENERGY. THERE IS
SUFFICIENT THERMAL ENERGY WITHIN THE RECUPERATOR FOR SEVERAL RESTARTS. FROM THE REACTOR
STANDPOINT, PERHAPS THE MOST IMPORTANT BENEFIT OF THE RECUPERATED CYCLE IS THAT IT PREVENTS
LIQUID HYDROGEN FROM ENTERING THE CORE. THIS ELIMINATES THE POSSIBILITY OF HIGH EXCESSIVE
REACTIVITY DUE TO VERY DENSE HYDROGEN IN THE CORE. IN ADDITION, THE RECUPERATOR DAMPS HYDRAULIC
OSCILLATIONS THROUGH THE USE OF SMALL PASSAGES AND FLOW DIRECTION CHANGES. IT INSURES THE
DELIVERY OF UNIFORM, STEADY FLOW TO THE FUEL ELEMENTS WITHOUT SHARP PRESSURE PULSES.

Recuperated Cycle Provides
Superior Engine Operation

- Provides Cooled, Internal Gamma Shield
- Enables High Chamber Pressure
- Provides Thermal Energy for Turbopump Start
  - Energy available for many starts
- Provides Safe, Controllable Reactor Start
  - Prevents Liquid Hydrogen entry into the core
  - Decouples and damps system oscillations

ENCORP - Energopool - Babcock & Wilcox
THERE ARE SEVERAL NOTeworthy REACTOR SPECIFICATIONS. PERHAPS THE MOST IMPORTANT IS THE AVERAGE POWER DENSITY THAT WAS CHOSEN FOR THE BASELINE. THE VALUE OF 33 MW/L WAS CHOSEN FOR THIS MISSION BECAUSE WE BELIEVE IT IS ACHIEVABLE. FURTHERMORE, THE RUSSIAN ENGINEERS HAVE DEMONSTRATED UP TO 90 MW/L (FOR MINUTES) WITH SIMILAR FUEL COMPOSITION. IT IS IMPORTANT TO NOTE THAT THE POWER DENSITY DETERMINES THE SIZE OF THE REACTOR AND THEREFORE THE ENGINE SIZE. THE CHANGE IN MASS OF THE REACTOR WITH POWER DENSITY VARIATIONS IS SHOWN SEPARATELY.

THE BASELINE FUEL COMPOSITION IS (U,Zr)C WHICH IS COATED WITH NbC. THIS COMBINATION OFFERS HIGH TEMPERATURE CAPABILITY (i.e., MELT IS APPX. 3,300-3,400K) AND THE COATING PROVIDES THE FISSION PRODUCT RETENTION AND IS SELECTED TO MATCH THE THERMAL EXPANSION OF THE BINARY FUEL BETTER THAN ZrC.

**REACTOR SUMMARY: KEY SPECS**

- Reactor Power: 1.56 GW
- Thrust (200:1 nozzle): 75,000 lbf
- Gas Outlet Temp (mixed mean): 2,700 K (4,400°F)
- Propellant Flow Rate: 82 l/sec
- Fuel Form: Binary (Zr,U)C/NbC
- Particle Diameter: 500 Micron (20 mils)
- Core Power Density (ave): 3.6 MW/I
- Fuel Volume: 47.7 liters
- Number of Elements: 36
- Safety Shutdown: Central Poison Rod
- Vessel Diameter: 1.03 meters
- Reactor Drilled Length: 92.5 cm
- Reactor Mass (no recaps/shielding): 3,400 Lb (2,420Kg)

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Enegopool  Babcock & Wilcox
REACTOR FEATURES EFFICIENT INTEGRATION OF COMPONENTS

The fuel elements are located on a hexagonal array. This allows the 36 fuel elements to be efficiently integrated into the smallest possible volume while meeting criticality limits, thermal hydraulic and structural requirements. A satisfactory pitch was found to be 11 cm. This was primarily sized for hydraulic considerations (low pressure drop through the moderator and inlet plenum). Further studies and optimization will likely result in a change in the pitch.

The control drums act as the reflector and the control system for the reactor. They are approximately 11 cm in diameter with an outer segment of 12 mm thickness and 120 degree arc containing B,C. They are placed as close to the core as possible. Originally there were 24 control drums but the "corner" six were providing little control benefit and were therefore removed. The safety rod is located in the center of the core where its worth is maximized.

Reactor Features Efficient Integration of Components

CONTROL DRUM (18)
FUEL ELEMENT (36)
SAFETY ROD
STEEL VESSEL

NTP: System Concepts
NP:TIM-92
MANIFOLDS PROVIDE CORE SUPPORT

The various flow loops of the engine system have been greatly simplified through the use of an innovative manifold/core support structure. This component is unique in that it not only provides a very rigid structure to which the fuel elements are attached but it also contains two plenums for the moderator cooling loop and a feed-through for the fuel element propellant loop.

The fabrication of the core support structure is similar to that of a honeycomb composite. The internal webs carry the shear loads while the top and bottom skins of steel carry the membrane loads. As the aircraft industry is aware, this configuration is extremely efficient in its specific load carrying capability and stiffness, therefore the thicknesses of the steel skins and web are minimized.
THE CONTROL DRUMS ARE LOCATED CLOSE TO THE CORE, WITH AS LARGE A DRUM SIZE AS POSSIBLE WITHOUT INTERFERENCE, (APPROXIMATELY 11 CM OUTSIDE DIAMETER) TO ENHANCE NEUTRON REFLECTION AND CONTROL WORTH, WHILE MINIMIZING THE SURROUNDING PRESSURE VESSEL SIZE. THE DRUM HEIGHT IS SLIGHTLY LONGER THAN THE ACTIVE FUEL BED HEIGHT OF ABOUT 92 CM. THE CONTROL DRUMS FOR THE CONCEPTUAL DESIGN ARE MADE OF BERYLLIUM WITH A B,C POISON SEGMENT 12 MM THICK OVER A 120 DEGREE ARC SEGMENT. THE CONTROL DRUM WORTH IS 0.10 DELTA-K/K WITH THE SAFETY ROD WITHDRAWN, AND 0.14 WITH THE SAFETY ROD INSERTED. TOTAL CONTROL WORTH IS 0.26 DELTA-K/K, FOR A SHUTDOWN REACTIVITY OF -0.20 DELTA-K/K (K- EFFECTIVE = 0.83). NOMINAL HYDROGEN GAS DENSITIFIES, OR WORTH IS 0.07 DELTA-K/K, WERE INCLUDED. THUS, WITHOUT HYDROGEN GAS, THE REACTOR WILL BE 0.01 DELTA-K/K SHUTDOWN EVEN IN THE MOST REACTIVE CONTROL POSITIONS. A STUDY OF INDIVIDUAL DRUM WORTHS SHOWED THAT A FIXED BERYLLIUM REFLECTOR SECTION IN THE CORNER POSITION ENHANCES REFLECTION AND CONTROL WORTH, INCREASING TOTAL DRUM WORTH BY ABOUT 0.01 DELTA-K/K, WHILE ALSO ALLOWING SMALLER PRESSURE VESSEL SIZE. A TRADE STUDY SHOWED POISON THICKNESS AS THIN AS 1 MM IS QUITE EFFECTIVE AND THAT THE MAXIMUM WORTH WAS REACHED AT APPROXIMATELY 10 TO 15 MM.
SAFETY ROD LOCATED FOR MAXIMUM WORTH

THE CENTRAL SAFETY ROD IS LOCATED IN A POSITION OF HIGH NEUTRON FLUX, RESULTING IN A CONTROL WORTH WHICH EXCEEDS THE COMBINED WORTH OF THE 18 CONTROL DRUMS. IT CONTAINS B₄C IN A CYLINDRICAL SHAPE WITH AN OUTSIDE DIAMETER OF ALMOST 11 CM. A B₄C REFLECTOR SEGMENT IS MOUNTED ON THE AFT END TO MINIMIZE NEUTRON STREAMING THROUGH THE SAFETY ROD OPENING AND TO REDUCE HEATING DURING OPERATION. THE SAFETY ROD WORTH IS 0.12 DELTA-K/K WITH THE CONTROL DRUMS' POISON OUTWARD, AND 0.16 DELTA-K/K WITH THE CONTROL DRUMS' POISON INWARD. THE TOTAL CONTROL WORTH IS 0.26 DELTA-K/K, FOR A SHUTDOWN REACTIVITY OF -0.20 DELTA-K/K (K-EFFECTIVE = 0.83). NOMINAL HYDROGEN GAS DENSITIES, WORTH 0.07 DELTA-K/K, WERE INCLUDED. THUS, WITHOUT HYDROGEN GAS, THE REACTOR WILL BE 0.01 DELTA-K/K SHUTDOWN EVEN IN THE MOST REACTIVE CONTROL POSITIONS.
THE PBf4 CORE IS HETEROGENEOUS WITH A COOLED MODERATOR JACKET SURROUNDING THE COLD FRIT, FUEL, AND HOT FRIT. THE MODERATOR CONSISTS OF MACHINED BLOCKS OF BERYLLIUM JOINED TOGETHER TO FORM A RIGID JACKET NEARLY 1 METER IN LENGTH. THE BLOCKS ARE PRE-DRILLED WITH CAREFULLY SIZED HOLES. TWELVE OF THE 36 HOLES ARE FILLED WITH ZIRCONIUM HYDRIDE, AND THE REMAINDER ARE USED FOR MODERATOR COOLING. THE RATIOS OF ZIRCONIUM HYDRIDE, BERYLLIUM AND HYDROGEN YIELD PROPER NEUTRONICS, STRUCTURAL AND THERMAL HYDRAULIC PERFORMANCE. THE BASELINE DESIGN UTILIZES A RATIO OF 62%Be, 8%ZH AND 10%H2. THE PRESSURE DROP WITHIN THE COOLANT LOOP OF THE MODERATOR IS APPROXIMATELY 300 PSI, THE THICK BERYLLIUM WALLS ARE MORE THAN ADEQUATE TO PROVIDE STRUCTURAL SUPPORT AND THE ZIRCONIUM HYDRIDE, EVEN IN SUCH SMALL QUANTITIES PROVIDES ENHANCED MODERATION FOR THE CORE. FURTHER OPTIMIZATION CAN SIGNIFICANTLY ENHANCE THE PERFORMANCE OF THE MODERATOR AND REACTOR.

THE COLD FRIT DISTRIBUTES THE FLOW TO THE FUEL BED IN PROPORTION TO THE LOCAL HEAT GENERATION. COOLANT FLOW IS DIRECTED RADIIALLY INWARD AND IS HEATED BY THE FUEL BED. THE HOT GAS PASSES THROUGH HOLES IN THE HOT FRIT WHERE IT COLLECTS IN THE HOT CHANNEL AND IS EXHAUSTED TO THE NOZZLE. THE COLD FRIT IS MADE OF STEEL AND WILL LIKELY BE OF PLATELET DESIGN. THE HOT FRIT IS MADE OF NIQUEM CARBIDE COATED GRAPHITE. GRAPHITE TECHNOLOGY HAS EVOLVED OVER THE YEARS AND IT IS NOW POSSIBLE TO OBTAIN GRAPHITE WITH A THERMAL EXPANSION COEFFICIENT THAT MATCHES THAT OF NIQUEM CARBIDE EXACTLY.

THE HOT CHANNEL (THE AREA INSIDE THE HOT FRIT) IS SIZED SUCH THAT THE MAXIMUM VELOCITY OF THE HOT HYDROGEN IS NOT GREATER THAN MACH 0.25 TO AVOID COMPRESSIBILITY EFFECTS.

FUEL ELEMENT INTEGRATED EFFICIENTLY
THE FUEL ELEMENT WAS ANALYZED USING FOTVE, A 1-D BWI PROPRIETARY COMPUTER CODE WHICH PREDICTS PRESSURE LOSSES IN THE DIFFERENT COMPONENTS OF AN ELEMENT, AS WELL AS TEMPERATURES OF THE PROPELLANT AND FUEL. EACH FUEL ELEMENT WAS INITIALLY ALLOTTED A PRESSURE DROP OF 500 PSI. RESULTS FROM FOTVE INDICATE MAXIMUM PRESSURE DROPS OF ABOUT 300 PSI. THE PROPELLANT FEED CHANNEL WAS DESIGNED TO MINIMIZE PRESSURE LOSS OVER THE LENGTH OF THE CHANNEL WITHOUT IMPACTING THE WEIGHT AND SIZE OF THE FUEL ELEMENT ASSEMBLY. RESULTS OF A STUDY TO COMPARE PRESSURE LOSSES TO THE DIFFERENT GAP SIZES BETWEEN THE MODERATOR AND COLD FRIT INDICATED THAT THE OPTIMUM GAP WAS AT 0.3 CM.

THE COLD FRIT IS DESIGNED TO BE THE PRIMARY FLOW CONTROLLER. IN FOTVE RUNS, THE MINIMUM COLD FRIT PRESSURE DROP TO THE BED PRESSURE DROP RATIO IS MAINTAINED AT 8 TO 1. THIS RATIO ENSURES THAT THE COLD FRIT HAS CONSIDERABLY MORE CONTROL OVER THE FLOW DISTRIBUTION TO THE FUEL BED THAN THE PROPELLANT FEED CHANNEL OR THE BED ITSELF.

THE MODERATOR DESIGN WAS ANALYZED USING A CHANNEL FLOW CODE CALLED FIGTH, ANOTHER BWI PROPRIETARY CODE. THE CODE CALCULATES PRESSURES, TEMPERATURES AND FILM COEFFICIENTS ALONG THE LENGTH OF A HEATED CHANNEL. THE MODERATORS WERE INITIALLY ALLOTTED A PRESSURE DROP OF 500 PSI. RESULTS INDICATE A MAXIMUM DROP OF ABOUT 300 PSI.

THIS CORE DESIGN HAS NOT BEEN OPTIMIZED. HOWEVER, A FUEL ELEMENT AND MODERATOR FROM ONE OF THE SIX ASSEMBLIES WHICH SURROUND THE SAFETY ROD WERE ANALYZED FOR THIS CORE CONFIGURATION. THESE ANALYSES DEMONSTRATE A WORKABLE DESIGN BUT DETAILED ANALYSES MUST BE PERFORMED ON A CORE SYSTEM LEVEL TO PROVIDE INSIGHT ON FLOW SPLIT CHARACTERISTICS AND ITS IMPACT ON PRESSURES AND TEMPERATURES FOR FULL POWER, THROTTLING AND DECAY HEAT CONDITIONS.
FUEL PARTICLE PROVIDES HIGH POWER DENSITY & TEMP

- 48 Liters of Fuel
- 33 MW/l (ave)
- 500 Micron Dia.

Loading:
- 127 kg Uranium
- 21 kg Carbon
- 57 kg Zirconium
- 60 kg Niobium
- 265 kg Total Bed

- 73% Enrichment
- 3,300+ K Melt Approx.
The PBR NTRE is Thermally Stable

The PBR is thermally stable. The fuel temperature (when half of the fuel reaches its melt temperature) is approximately 2800K when the mixed mean gas temperature is 2700K. The amount of local flow can be reduced by 15-22% before the fuel kernel reaches its melt temperature. Based on thermal hydraulic stability studies, a local flow disruption does not cause a propagation but rather a stable transition to a new temperature. This is due to the high Reynolds number and turbulent regime over which this PBR operates. For high power reactor operation, the PBR is quite stable (as is expected).

Thermal hydraulic instability can occur for low flow regimes (i.e., low power). The bed hydraulic resistance, which is formed by viscous and inertial forces (typified by the Ergun correlation) is dominated by inertial forces for high power, high flow operation. However, for low flow operation, the viscous term can dominate. For such cases, a perturbation in flow can cause increased local gas temperature, and thus higher pressure, which causes higher gas temperatures...and so on. B&W understands the mechanisms involved and the regimes of operation which must be avoided. It is shown that for a three-dimensional analyses it is possible to retain nearly all the PBR performance (impulse) while throttling to 5-10% of full flow. Another solution to the low flow instability can be found by reducing the change in gas temperature as it flows through the fuel bed. By increasing the inlet gas temperature to 450K, the flow stability can be maintained even for conservative analyses.

The PBR NTRE is Thermally Stable

During Main Burns

During Cool Down

Percent of Local Coolant Flow

Percent of Full Power

Percent

Fuel Temp (K)

3,400

3,300

3,200

3,100

3,000

2,900

2,800

2,700

Fuel Melt Range

Increase Inlet Temp

TL = 450 K

PBR

P = 2700 K

1-d Stability

3-d Stability
FUEL DESIGN BALANCES REACTIVITY AND CONTROL

The reactor and fuel design is flexible. The 48 liter fuel volume was based on 20 MW/LITER power density and 1900 MW total power. The 36 fuel element core with 11 cm pitch and moderator composed of 82% beryllium, 8% ZrH, and 10% hydrogen coolant passages were based primarily on mechanical and thermo-hydraulic considerations, and physics trade-offs.

For high reactivity, the fuel particle maximizes uranium loading. The baseline particle is 70% U/2% ZrC kernel surrounded by 30% NbC coating, by volume, and the kernel contains 50% UC, for average bed uranium density of 2.7 g/cc. It is expected the uranium loading will be reduced by 50% or more through optimization of the core with fully enriched fuel, and no HCI hot channel inserts, the maximum reactivity is 0.15 delta-K/K. High excess reactivity provides flexibility for optimizing the design, and allows margin for modeling uncertainties and overcoming reactivity losses due to xenon transients and fuel depletion during operation.

Design variables to balance high reactivity and control are fuel particle uranium content, uranium enrichment, installed neutron poisons, moderator materials, fuel element pitch, and core reflector and controls design. Conical HCI inserts were placed in the hot channels to provide fixed neutron poison hold down of 0.2 delta-K/K and 1% of total power in extra hydrogen gas heating. Hafnium also provides resonance neutron absorption which will improve prompt temperature feedback; however hydrogen flow distribution must compensate for the shift in axial power shape. The U enrichment was also reduced, increasing the negative prompt temperature feedback of the fuel due to the Doppler coefficient of the larger fraction of U. Increasing prompt nuclear feedback enhances the stability, and thus control and safety of the reactor. The final maximum reactivity of the conceptual design is 0.06 delta-K/K. The controls have large reactivity worths; the safety rod worth of 0.15 delta-K/K is somewhat higher than the control drums worth of 0.10 delta-K/K.

FUEL DESIGN BALANCES REACTIVITY AND CONTROL

Energoopool, Babcock & Wilcox

NTP: System Concepts 316 NP-TM-92
THE PBR IS DESIGNED FOR SAFETY

THE PBR WILL BE MAINTAINED SUBCRITICAL FOR ALL LAUNCH ACCIDENT SCENARIOS. PRELIMINARY ESTIMATES FOUND THAT, FOR THE CONCEPTUAL DESIGN, ADDITIONAL MITIGATING MEASURES ARE NEEDED IN ORDER TO MAINTAIN THE REACTOR SUBCRITICAL IF THE PRIMARY VOID REGIONS. THE COLD AND HOT CHANNELS, ARE FILLED WITH WATER AND THE REACTOR IS SURROUNDED BY WATER TO SIMULATE A WATER IMMERSION ACCIDENT. WHEN THE HOT CHANNELS ARE FILLED WITH TEMPORARY LAUNCH INSERTS MADE OF Hf, FOR EXAMPLE, THE CALCULATED REACTIVITY AFTER IMMERSION IS 0.22 DELTA-K/K (EFFECTIVE = 0.82), THE REACTIVITY PRIOR TO IMMERSION IS ABOUT 0.32. FOR THE UNCHANGED BASELINE DESIGN, THE REACTIVITY AFTER IMMERSION WAS 0.07 DELTA-K/K, WHICH IS CLEARLY UNACCEPTABLE. FOR THESE ESTIMATES, THE CHANGES IN THE VOID REGIONS OF THE HOT FRIT, COLD FRIT, FUEL BED, AND MODERATOR COOLING PASSAGES WERE NEGLECTED. IT WILL BE POSSIBLE, THROUGH ADDITIONAL STUDY, TO OBTAIN ACCEPTABLE RESULTS WITHOUT RESORTING TO TEMPORARY LAUNCH INSERTS BY MAKING OTHER MODIFICATIONS. FOR EXAMPLE, WITH CHANGES IN PITCH AND/OR MODERATOR HYDROGEN CONTENT, BALANCED BY DECREASES IN PARTICLE URANIUM CONTENT AND/OR ENRICHMENT, IT WILL BE POSSIBLE TO OBTAIN A MORE NEUTRONICALLY OPTIMUM INITIAL CORE CONDITION, SUCH THAT THE WATER IMMERSION WILL RESULT IN EITHER OVER-MODERATION AND A DECREASE IN REACTIVITY, OR AT LEAST A SMALLER INCREASE IN REACTIVITY. THESE SAME CHANGES WILL ALSO HELP MITIGATE POTENTIAL REACTIVITY INCREASES DUE TO ANY EXCESSIVE HYDROGEN GAS DENSITY IN THE CORE DURING STARTUP OR TRANSIENTS.

BASED ON PITCH TRADE STUDIES PERFORMED USING DIFFERENT FUEL BED THICKNESSES, THE IMPACT OF GEOMETRY CHANGES ASSOCIATED WITH A LAUNCH ACCIDENT (E.G. DEFORMATION OR COMPACTION) IS EXPECTED TO BE LESS SEVERE THAN THE WATER IMMERSION. THE NEUTRONIC OPTIMIZATION DISCUSSED ABOVE WILL SERVE TO MITIGATE THIS EVENT AS WELL.

THE PBR IS DESIGNED FOR SAFETY

- **NUCLEAR CRITICALITY**
  - Subcritical In Water Immersion
    - 0.62 Keff using hot frit plugs
  - Two Independent Shut-down Systems
  - Recuperator Prevents Excessive Hydrogen reactivity insertion in core (i.e. no HI2)

- **THERMAL PROTECTION**
  - Five Systems to Cool the Reactor
    - Twin Turbopumps (70 % full flow capacity ea.)
    - One Electrical Pump (Approx. 5 % capacity)
    - One Circulation Pump (Several Megawatt cap.)
    - Tank Bleed
  - Cross Feed is an Option

**Energopool** - Babcock & Wilcox

NP-TIM-92 317 NTP: System Concepts
THE PBR IS SCALABLE IN POWER DENSITY.

THE PBR CAN OPERATE OVER A WIDE RANGE OF POWER DENSITIES. FOR POWER DENSITIES ABOVE 20 MW/L THERE IS ONLY A SMALL IMPROVEMENT IN REACTOR MASS. THE KNEE IN THE CURVE APPEARS TO BE AROUND 20 MW/L. THE LOWER EXHAUST GAS TEMPERATURE (2500K) DOES NOT HAVE A SIGNIFICANT IMPACT ON REACTOR MASS BUT WILL IMPACT PERFORMANCE IN TERMS OF IMPULSE AND THEREFORE HYDROGEN TANKAGE. THE LOWER CHAMBER PRESSURE REACTOR (1000 PSI) APPEARS ATTRACTIVE FROM A REACTOR MASS STANDPOINT, HOWEVER THE REDUCTION IN REACTOR MASS WILL ALMOST CERTAINLY BE NEGATED BY THE INCREASE IN SHIELD MASS SINCE THE VESSEL IS LARGER. THE KEY FEATURES OF THE 33, 20 AND 10 MW/L REACTORS ARE:

**SCALABILITY - Power Density**

<table>
<thead>
<tr>
<th>Reactor Power Density</th>
<th>33</th>
<th>20</th>
<th>10 MW/l</th>
</tr>
</thead>
<tbody>
<tr>
<td>Reactor Size</td>
<td>1,500</td>
<td>1,500</td>
<td>1,500 MW</td>
</tr>
<tr>
<td>Weight</td>
<td>75,000</td>
<td>75,000</td>
<td>75,000 lb</td>
</tr>
<tr>
<td>Reactor Mass</td>
<td>2,350</td>
<td>2,379</td>
<td>3,687 Kg</td>
</tr>
<tr>
<td>TW (reactor mass only)</td>
<td>14</td>
<td>12</td>
<td>9</td>
</tr>
<tr>
<td>Outlet Gas Temp</td>
<td>7,796</td>
<td>7,796</td>
<td>7,799 K</td>
</tr>
<tr>
<td>Fuel Volume</td>
<td>48</td>
<td>79</td>
<td>157 liters</td>
</tr>
<tr>
<td>Propellant Flow Rate</td>
<td>37</td>
<td>37</td>
<td>37 kg/h</td>
</tr>
<tr>
<td>Number of Fuel Elements</td>
<td>36</td>
<td>40</td>
<td>60</td>
</tr>
<tr>
<td>Vessel Diameter</td>
<td>140</td>
<td>186</td>
<td>119 cm</td>
</tr>
<tr>
<td>Reactor Fused Length</td>
<td>92</td>
<td>92</td>
<td>92 cm</td>
</tr>
</tbody>
</table>

**The PBR is Scalable in Power Density**

![Graph showing Reactor Mass (kg) vs. Power Density (MW/l)]
The PBR is Scalable with Thrust

<table>
<thead>
<tr>
<th>Thrust (lb)</th>
<th>20,000</th>
<th>40,000</th>
<th>60,000</th>
<th>80,000</th>
</tr>
</thead>
<tbody>
<tr>
<td>Reactor Mass (kg)</td>
<td>2,000</td>
<td>1,400</td>
<td>1,000</td>
<td>800</td>
</tr>
</tbody>
</table>

**SCALABILITY**

- **Thrust**: 75,000, 40,000, 25,000 lb
- **Reactor Power**: 1,560, 832, 520 MW
- **Reactor Mass**: 2,420, 1,301, 1,160 Kg
- **T/W (reactor mass only)**: 14, 14, 10
- **Outlet Gas Temp**: 2,700, 2,700, 2,700 K
- **Fuel Volume**: 48, 25, 20 Liters
- **Propellant Flow Rate**: 37, 20, 12 Kg/s
- **Number of Fuel Elements**: 36, 18, 18
- **Vessel Diameter**: 100, 78, 76 cm
- **Reactor Fueled Length**: 92, 66, 66 cm
THE FUEL IS THE NTRE'S KEY TECHNOLOGY

THE PARTICLE BED REACTOR IS UNIQUE BECAUSE OF THE FUEL FORM. ITS SMALL SIZE MEANS THAT IT IS COMPLETELY "PRE-CRACKED." THE DIAMETER OF THE FUEL PARTICLES ARE NEARLY AS SMALL AS THE MANUFACTURING PROCESS WILL ALLOW. OVER 10 MILLION PARTICLES ARE CONTAINED WITHIN EACH OF 36 FUEL ELEMENTS. BECAUSE THE FUEL PARTICLES ARE SO SMALL, THE STRESSES WITHIN THE PARTICLES ARE REDUCED AND THEREFORE THE RELIABILITY IS IMPROVED. HIGHLY STRESSED FUELS FORM WILL FAIL DURING OPERATION.


FINALLY, THOUSANDS OF INDIVIDUAL PARTICLES CAN FAIL WITH LITTLE OR NO EFFECT ON REACTOR PERFORMANCE; WHEREAS MONOLITHIC FUEL FORMS MAY NOT DEGRADE SO GRACEFULLY.

THE FUEL IS THE NTRE's KEY TECHNOLOGY

FUEL FORM AND ARRANGEMENT FAVOR FISSION PRODUCT RETENTION

* PBR FUEL FORM IS ATTRACTIVE BECAUSE:
  - Design options permit reduction of fission product retention with additional coatings
  - Individual particle failures have little or no effect on reactor performance
  - Particles have low thermal gradients (small size)

GenCorp - Energopool - Babcock & Wilcox

NTP: System Concepts 320 NP-TIM-92
CONCLUSIONS

The results of this work indicates that the PBR design meets all the NASA requirements. The recuperated PBR appears to be well suited for the SEI mission. Throughout the design process, trades were performed to find appropriate blends of safety, reliability and strong robust components. Very few optimization studies were performed to exceed the performance requirements but it is believed that significant gains can be accomplished from such optimization. The PBR technology appears to be capable of very high performance.

The PBR Design has been Successfully Adapted for the SEI Mission

- High Performance (with mass penalties)
- Throttling Capability (>>4:1)
- Superior Decay Heat Removal System
- Integrated into Practical Engine Design
- High Reactivity, Control and Safety
CIS Engine Design

Don Culver

Our CIS engine concept is best summarized by including the rationale behind the selection of each major subsystem concept or operating parameter. 2900 K mixed mean reactor outlet gas temperature is selected to meet the 4.5 hour life requirement with an appropriate margin by the end of fuel assembly development. The engine design/operating selection of 300:1 nozzle area ratio and nozzle inlet pressure of 2,000 psia is the result of a Mars mission payload trade study; it gives the best combination of engine specific impulse and weight. A recuperated turbopump drive cycle was selected for several reasons: (1) nozzle inlet pressures of 1000 psia and above are enabled by recycling topping heat through the turbine, and no reactor manifolding need be added to extract turbine drive heat directly from the core, (2) engine start and shutdown transients are smoothed and assisted by the large, available heat capacity of the high surface area recuperator, (3) the steel heat exchanger adds no weight to the engine, because its weight is determined by its other duties as the forward closure of the reactor vessel. An aft core support structure was selected, because it has been shown by test that CIS fuel assembly life is superior when held in compression. A single DeLaval nozzle is selected that is internally cooled with hydrogen; no hydrogen bleed is necessary, because our formed platelet nozzle operates with low internal wall temperatures at high heat fluxes. A 40 Klb nozzle of similar design, material and coolant is now in test at NASA. The nozzle is small, because of the engine’s high operating pressure, and we use a low weight, carbon-carbon nozzle extension. Its surface may be converted to ZrC to improve its life in hydrogen environment, using near term technology processes similar to those in work at Aerojet, however this is probably unnecessary, because total surface recession in 4.5 hours of operation is expected to be less than 0.025 in. (0.64 mm). A borated ZrH and LiH neutron shield is located within the reactor vessel, between the core and recuperator/gamma shield. It is cooled by propellant flow in steel wafers located between its many transverse layers.

Design Rationale for NTRE With CIS Reactor Is Clear

<table>
<thead>
<tr>
<th>Design Parameter</th>
<th>Rationale</th>
</tr>
</thead>
<tbody>
<tr>
<td>1 H₂ Mixed Mean Outlet Temperature (2,900K)</td>
<td>Expected 4.5 Hours Fuel Life (Demoed &gt; 1 Hour at 3000K)</td>
</tr>
<tr>
<td>2 Pc (2,000 psia)</td>
<td>Best Mars Mission Performance With Reusable Engines</td>
</tr>
<tr>
<td>3 Ae/At (300)</td>
<td>• Enable High Pressure With Simple Reactor</td>
</tr>
<tr>
<td></td>
<td>• Enhances Transient Operation</td>
</tr>
<tr>
<td></td>
<td>• No Weight Penalty (γ Shield)</td>
</tr>
<tr>
<td>4 Power Cycle (Recuperated)</td>
<td>Optimum With CIS FAs (Test Data)</td>
</tr>
<tr>
<td>5 Core Support (aft)</td>
<td>Near Term Technology Use (Formation Cu Platelet)</td>
</tr>
<tr>
<td>6 Nozzle (Single DeLaval) (H₂ Cooled Without Bleed Flow) (C-C Extension)</td>
<td>Lowest Weight and Risk</td>
</tr>
<tr>
<td>7 Neutron Shield (Internal)</td>
<td></td>
</tr>
</tbody>
</table>
Reactor gas outlet temperature for our CIS engine was selected by analyzing fuel assembly in-reactor test data. Fuel assembly tests demonstrated lifetimes of 4000 hours at gas outlet temperatures averaging 2000 K and lifetimes of 4000 seconds at gas outlet temperatures between 3000 K and 3100 K. An Arrhenius law study was applied to the data to predict lifetimes at other outlet temperatures. This work showed that fuel assembly lifetimes of 4.5 hours had been demonstrated at mean outlet gas temperatures of about 2800 K. In-reactor tests did not terminate with destroyed fuel assemblies, however, and Russian scientists have estimated the lifetime demonstrable with a three to five year fuel assembly development program to be about 2.8 hours with 3000 K outlet gas temperature. Arrhenius analysis shows this corresponds to a 4.5 hour life at gas temperatures above 2900 K. Thus, 2900 K nozzle inlet temperature was selected to provide the greatest mission benefit within current NASA life requirements.

### CIS Fuel Life Is Expected to Be 4.5 Hours at 2900°K

<table>
<thead>
<tr>
<th>Tc (°K)</th>
<th>Minimum Life (Hours)</th>
<th>Maximum Life (Hours)</th>
<th>Life Expected in 3-5 Years (Hours)</th>
</tr>
</thead>
<tbody>
<tr>
<td>3200</td>
<td>0.400</td>
<td>0.700</td>
<td>1.000</td>
</tr>
<tr>
<td>3100</td>
<td>0.655</td>
<td>1.111</td>
<td>1.637</td>
</tr>
<tr>
<td>3000</td>
<td>1.111</td>
<td>1.820</td>
<td>2.778</td>
</tr>
<tr>
<td>2900</td>
<td>1.954</td>
<td>3.100</td>
<td>4.855</td>
</tr>
<tr>
<td>2800</td>
<td>3.579</td>
<td>5.470</td>
<td>8.948</td>
</tr>
<tr>
<td>2700</td>
<td>6.856</td>
<td>10.08</td>
<td>17.14</td>
</tr>
<tr>
<td>2600</td>
<td>13.80</td>
<td>19.45</td>
<td>34.50</td>
</tr>
<tr>
<td>2500</td>
<td>19.40</td>
<td>39.57</td>
<td>73.56</td>
</tr>
<tr>
<td>2400</td>
<td>66.67</td>
<td>35.40</td>
<td>166.7</td>
</tr>
<tr>
<td>2300</td>
<td>162.4</td>
<td>197.1</td>
<td>406.0</td>
</tr>
<tr>
<td>2200</td>
<td>428.7</td>
<td>400.7</td>
<td>1,072</td>
</tr>
<tr>
<td>2100</td>
<td>1,242</td>
<td>1,333</td>
<td>3,105</td>
</tr>
<tr>
<td>2000</td>
<td>4,000</td>
<td>4,000</td>
<td>10,000</td>
</tr>
<tr>
<td>Tc at 4.5 Hours, °K</td>
<td>Minimum Life (Hours)</td>
<td>Maximum Life (Hours)</td>
<td>Life Expected in 3-5 Years (Hours)</td>
</tr>
<tr>
<td>---------</td>
<td>----------------------</td>
<td>----------------------</td>
<td>-----------------------------------</td>
</tr>
<tr>
<td>4,500</td>
<td>2,764</td>
<td>2,914</td>
<td>1,242</td>
</tr>
<tr>
<td>K</td>
<td>8.574 x 10^{-8}</td>
<td>3.709 x 10^{-7}</td>
<td>8.574 x 10^{-8}</td>
</tr>
<tr>
<td>λ</td>
<td>49,132</td>
<td>46,160</td>
<td>49,132</td>
</tr>
</tbody>
</table>

\[
\text{Life} = Ke^\left(\frac{-E}{RcT}\right) \text{(Arrhenius Law)}
\]
The CIS engine’s turbopump is powered by a topping cycle so that no specific impulse is lost through turbine exhaust overboard bleed flows. In this power cycle reactor heat that is deposited in engine components (and must be removed continuously) is removed by the hydrogen propellant, heating pump discharge flow to energy levels high enough to drive the turbines. About 9% of the reactor heat is removed, directly or indirectly, from the engine’s nozzle, moderator, reflector, pressure vessel, and radiation shutdowns. However, about 12% of the reactor heat is needed to drive the engine to a nozzle inlet pressure of 2000 psia with adequate (10%) power control margin and reasonable operating conditions for all engine components.

The power cycle requires a liquid hydrogen pump discharge pressure of 4950 psia. We have elected to use two 15,000 horsepower turbopumps in parallel. About 1/6 of the total flow (13 lb/sec) cools the copper nozzle to area ratio 10. The balance of the hydrogen is heated to turbine inlet conditions in the high pressure side of the recuperator (heat exchanger). The turbine inlet temperature is about 850 °F (470 K), and the turbine pressure ratio is less than 1.6. Turbine exhaust flow internally cools the walls of the reactor vessel and joins with the nozzle coolant flow at the all end of the reactor core. Here, the 775 R (430 K) flows join and cool the moderator rods and side wall neutron reflector with parallel flows, exiting at about 1150 °F (40 K). These gases rejoin 100% of the propellant flows through the low pressure side of the recuperator. There it loses over 670 °F (370 K) temperature to the high pressure side for turbine drive power. This cool, recuperator outlet gas is distributed to the 102 fuel assembly inlets via the internal neutron shield. Fuel assemblies provide about 93% of the reactor’s 1650 MW(i) power to the propellant, so that its maximum exit temperature is 5220 °F (2900 K) at a pressure of 2000 psia. This gas flows through the DeLaval rocket nozzle to space.

CIS TOC Recuperated Topping Cycle Flow Schematic

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GenCorp - Energopool - Babcock & Wilcox

NTP: System Concepts 324

NP-TIM-92
The CIS engine flow scheme begins with two parallel turbopumps. Pump discharge flow splits into two streams; the smaller one cools the copper nozzle through internal passages as in a regeneratively cooled nozzle for bipropellant rocket engines. The nozzle support is cooled by a small portion of this flow. The larger pump discharge flow enters the center of the recuperator/gamma shield located at the forward end of the reactor vessel, distributes to thousands of parallel flow, high pressure passages, and flows radially outward to a peripheral manifold. This heated hydrogen gas flows to and through the turbopump’s turbine, the exhaust being routed to an inlet manifold on the aft section of the reactor vessel wall. This flow cools the wall internally as it moves aft to join the nozzle coolant at the aft core support structure. 100% of the propellant flows through this structure, cooling it and the aft peripheries of the 102 fuel assemblies. Propellant is metered forward from the support structure through the reactor’s moderator and reflector sections in parallel, collecting in a plenum at the forward end of the core. Here, gas flows radially outward to cooling channels in the forward section of the reactor vessel wall. Propellant flows forward to the periphery of the recuperator, where it enters low pressure, radial inflow heat exchanger passages. Cooled hydrogen leaves the aft face of the recuperator’s center and distributes axially through the two layers of the plate-type neutron shields. Propellant flows radially outward through these shields inside of metallic platelets to discharge at the inside wall of the reactor vessel. Hydrogen flows to the fuel assembly inlet plenum and into the 102 inlets, where it is heated to maximum temperature and flows aft into the rocket nozzle inlet and through the nozzle to space.

NTRE Flow Scheme With CIS Reactor
## CIS Engine Layout

### Engine Weight Breakdown

<table>
<thead>
<tr>
<th>Component</th>
<th>Weight, lbm</th>
</tr>
</thead>
<tbody>
<tr>
<td>Uncooled Nozzle</td>
<td>232</td>
</tr>
<tr>
<td>Cooled Nozzle</td>
<td>985</td>
</tr>
<tr>
<td>Pressure Vessel, Reactor Manifolds &amp; CSS</td>
<td>2536</td>
</tr>
<tr>
<td>Reactor, Reactor I&amp;C</td>
<td>6613</td>
</tr>
<tr>
<td>Turbopump Assemblies (2)</td>
<td>410</td>
</tr>
<tr>
<td>Recuperator / Shield</td>
<td>2425</td>
</tr>
<tr>
<td>Secondary Shield</td>
<td>842</td>
</tr>
<tr>
<td>Plumbing/Valves</td>
<td>1320</td>
</tr>
<tr>
<td>Controls and Shielding</td>
<td>758</td>
</tr>
<tr>
<td><strong>HTIB w/ Stage Power/Heat Removal</strong></td>
<td></td>
</tr>
<tr>
<td>Stage Power &amp; Heat Removal Sys Wt, lbm</td>
<td>2000</td>
</tr>
<tr>
<td>Engine with Power Sys Wt, lbm</td>
<td>17900</td>
</tr>
<tr>
<td>Mars Mission Specific Impulse, sec</td>
<td>949</td>
</tr>
<tr>
<td>Payload Returned to Earth, lbm</td>
<td>53299</td>
</tr>
</tbody>
</table>

---

**Table Values:**
- Thrust, lbf: 75000
- Chamber Pressure, psia: 2000
- Nozzle Area Ratio, Av/Ai: 500
- Engine Specific Impulse, sec: 959
- Mars Mission Specific Impulse, sec: 830
- Engine Total Weight, lbm: 15800
- Thrust/Weight: 4.7
- Engines per Vehicle: 2
- Payload Returned to Earth, lbm: 47057

**Diagram and Notes:**
- **GenCorp **
- **Aerojet **
- **Energopool • Babcock & Wilcox **
CIS Reactor Design

Richard Rochow

Reactor Design Summary

- Heterogeneity of the Core
- Average Specific Power Density in FE – 20 MW/1
- Fuel-Elements Twisted Rods on the Base of Solid U-Zr-Nb Carbide Solutions
- Neutron Moderator – Zirconium Hydride Rods
- Controls – 18 Drums in Reflector and 1 Rod in Core
- 7 Safety Rods in Core Against Water Filling Accident
- Reflector – Be
- Internal Shielding – ZrH(B); LiH, Recuperator Steel
### Main Characteristics of NRE Reactor

<table>
<thead>
<tr>
<th>Characteristics</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Thermal Power, MW</td>
<td>1650</td>
</tr>
<tr>
<td>Neutron Spectrum</td>
<td>Thermal</td>
</tr>
<tr>
<td>Average Exit Temperature of Propellant, K</td>
<td>2900</td>
</tr>
<tr>
<td>Propellant Pressure in Nozzle Chamber, bar</td>
<td>136</td>
</tr>
<tr>
<td>Propellant Flow Rate, kg/s</td>
<td>35.4</td>
</tr>
<tr>
<td>Reactor Dimensions, mm:</td>
<td></td>
</tr>
<tr>
<td>- Diameter</td>
<td>1050</td>
</tr>
<tr>
<td>- Height (including inner shielding)</td>
<td>2100</td>
</tr>
<tr>
<td>Mass, kg</td>
<td>5800</td>
</tr>
</tbody>
</table>

### NRE Reactor Components Mass (kg)

<table>
<thead>
<tr>
<th>Component</th>
<th>Mass, kg</th>
</tr>
</thead>
<tbody>
<tr>
<td>Core:</td>
<td></td>
</tr>
<tr>
<td>- FAs</td>
<td>1250</td>
</tr>
<tr>
<td>- Moderator</td>
<td>710</td>
</tr>
<tr>
<td>Reflector</td>
<td>600</td>
</tr>
<tr>
<td>Inner Shielding</td>
<td></td>
</tr>
<tr>
<td>- Zirconium Hydride</td>
<td>430</td>
</tr>
<tr>
<td>- Lithium Hydride</td>
<td>120</td>
</tr>
<tr>
<td>- Recuperator Material (Steel)</td>
<td>1100</td>
</tr>
<tr>
<td>Rotating Drum Drives (18)</td>
<td>360</td>
</tr>
<tr>
<td>Safety Rod Drives (7)</td>
<td>80</td>
</tr>
<tr>
<td>Supporting Structure</td>
<td>300</td>
</tr>
<tr>
<td>Pressure Vessel</td>
<td>850</td>
</tr>
<tr>
<td>Total</td>
<td>5800</td>
</tr>
</tbody>
</table>
Main Core Parameters

\[ ^{235}\text{U Loading, kg} \]
\[ ^{235}\text{U Enrichment, \%} \]
\[ \text{Average Specific Power Density In FE, MW/liter} \]
\[ \text{Non-Uniformity Power Release} \]
- With the Core Radius
- With the Core Height
\[ \text{Thermal Neutron Flux Density, cm}^{-2}\cdot\text{s}^{-1} \]
\[ \text{Core Dimensions, mm} \]
- Diameter/Height

CIS/NTRE Reactor Cross Section

NP-TIM-92 329
NTP: System Concepts

ORIGINAL PAGE IS OF POOR QUALITY
Fuel Assembly Description

Max. FA Thermal Power: Up to 22 MW
FE per FA: 356
Pressure Drop: 40 bar
Max. Mass Flow Rate: 0.42 kg/s

FA Dimensions, mm:
- Fuel Bundle Length: 100
- Fueled Length: 1000
- Bundle Diameter: 45
- Overall Length: 1500
- Overall Diameter: 55
Mass: 12 kg

Fuel Assembly Performance Exceeds Engine Requirements

Required Performance Parameters

<table>
<thead>
<tr>
<th></th>
<th>Tr0, °K</th>
<th>Ts</th>
<th>St/Stt</th>
<th>qv</th>
<th>Propellant</th>
<th>No. (Start)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Aerojet Engine</td>
<td>2900-3000</td>
<td>1000</td>
<td>25</td>
<td>H2</td>
<td>1...2</td>
<td></td>
</tr>
<tr>
<td>In-Pile Tests</td>
<td>3000-3150</td>
<td>1500</td>
<td>~400</td>
<td>25</td>
<td>H2</td>
<td>3</td>
</tr>
<tr>
<td>Max. Parameter values (In-Reactor)</td>
<td>3100</td>
<td>4000</td>
<td>Up to 1000</td>
<td>35</td>
<td>H2</td>
<td>12</td>
</tr>
</tbody>
</table>

IVG-1 Performance Parameters

<table>
<thead>
<tr>
<th>Start-up</th>
<th>Tr0, °K</th>
<th>Is</th>
<th>St/Stt</th>
<th>qv</th>
<th>Propellant</th>
<th>Power, MW</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>3170</td>
<td>~500</td>
<td>100...150</td>
<td>~22.5</td>
<td>H2</td>
<td>4.74</td>
</tr>
<tr>
<td>2</td>
<td>3110</td>
<td>~500</td>
<td>100...150</td>
<td>~22.5</td>
<td>H2</td>
<td>4.61</td>
</tr>
<tr>
<td>3</td>
<td>3030</td>
<td>~500</td>
<td>100...150</td>
<td>~22.5</td>
<td>H2</td>
<td>4.85</td>
</tr>
</tbody>
</table>

No Fuel Technology Improvements Required
Fuel Element Description

Composition: UC + ZrC + NbC
Max. Fuel Loading of U: Up to 20%
Enrichment: 90%
Max. Design Temperature (at Hot End): 3500 K
Operational Temperature (at Hot End):
  - Average: 2950 K
  - Maximum: 3100 K
Average Power Density:
  - In FE: 20 kW/cm³
  - In FA: 12 kW/cm³
Amount FA Tested at H₂ Above 2,900 K: More than 10,000

Fuel Element Heating Bundle From IVG Reactor
Reactivity and Control Characteristics

1. Maximum Reactivity Margin, % ΔK 4.2

2. Reactivity Effects, %
   - Doppler Effect and Effect of Moderator Temperature -1.0
   - Hydrogen Filling Effect +0.6
   - Fuel Burn Up Effect (Compensated by Burning Poisons) -0.4
   - Water Filling Effect +7.4

3. Control System Efficiency, %
   - 18 Drums 3.0
   - 7 Safety Rods in “dry” Reactor 18.7
   - 7 Safety Rods in “wet” Reactor 8.3
   - Central Rod (in function of regulator) 2.7

4. Poisoning Material of Controls B₄C

AEROFIN/ENERGOPOOL/B&W NTRE
NASA LeRC Final Report

Prepared by R.F. Rochow
Babcock & Wilcox, ASE
Oct 23, 1992
CIS CAPABILITIES

NUCLEAR REACTORS
- IVG-1 3100K, 240 MW
- IRTT 2650K, 60 MW
- IGR 3100K, 35 MW/t
- Critical Reactors
- Shielding Test Reactors
- Materials Test Reactors

DESIGN
- 250,000 Man-years of NRE Design/Test
- 30 Years Experience

MANUFACTURING
- Fuel Line 2 Core/yr
- Insulating Mat's ZrCNiC
- Bulk Fabrication ZrII, LiII
- Single Crystal Technology

TEST FACILITIES
- Baikal-1 IVG-1, IRGTT
- Plasmotron 100 MW
- Creep Test Rig 200 kW
- Corrosion Tester 250 kW
- Failure Mode Rig 100 kW

CIS REACTOR DESIGN PHILOSOPHY

- Heterogeneous Core
- Solid Carbide Solution "twisted ribbon" Fuel Elements
- Zirconium Hydride Moderator Rods
- ZrH(B) and LiII Internal Shielding
- Core Support at Hot End
- 12 MW/l Ave. Fuel Bundle Power Density
The CIS Reactor Utilizes Demonstrated Hardware

**REACTION SUMMARY: KEY SPECS**

- Reactor Power: 1.65 GW
- Thrust (200 l nozzle): 75,000 lbf
- Gas Outlet Temp (mean): 2,900 K (4,200 °F)
- Propellant Flow Rate: 79 l/sec
- Specific Impulse: 959 seconds
- Fuel Composition: (U,Nb,Zr)C
- Fuel Form ("Twisted Ribbon"): 100x1.6x1.0 mm - Approx
- Fuel Bundles (Power Density: 12 MW/l)
- Core Power Density: 37 MW/l
- Fuel Volume: 762 liters
- Number of Assemblies (1 element): 102
- Safety Shroud: 6 Safety Rods
- Vessel Diameter: 1.05 meters
- Reactor Fueled Length: 100 cm
- Reactor Mass (not including shielding): 4,150 kg (9,280 lb)
Fuel Element Design Has Been Refined

Numerous materials and fuel forms tested
- Carbides
- Carbonitrides
- W, Mo, Re
- Graphite Carbides
- Particles (1-8 mm Dia.)

More than 10,000 'twisted ribbons' tested above 2,900 K
3,000 K for thousands of seconds in H2 demonstrated

Fuel Element Specifications

<table>
<thead>
<tr>
<th>Composition</th>
<th>(U,Zr,Nb)C</th>
</tr>
</thead>
<tbody>
<tr>
<td>Max. Uranium Loading</td>
<td>20%</td>
</tr>
<tr>
<td>Enrichment</td>
<td>90%</td>
</tr>
<tr>
<td>Max. Design Temp.</td>
<td>3,500 K</td>
</tr>
<tr>
<td>Max. Operating Temp.</td>
<td>3,100 K</td>
</tr>
</tbody>
</table>

Power Density:
- in Fuel Element 20 MW/l
- in Fueled Volume 12 MW/l

Fuel Assembly Design Has Been Refined

Fuel Composition is Tailored
- Axially and radially
- For mechanical properties

Insulation has been developed
- Monolithic NbC and ZrC tubes
- Temp capability up to 3,100 K
- Greater than 50% porosity

Hundreds of Assemblies Tested

Fuel Assembly Specifications

<table>
<thead>
<tr>
<th>Max. Thermal Power</th>
<th>22 MW</th>
</tr>
</thead>
<tbody>
<tr>
<td>Pressure Drop Nom.</td>
<td>40 bars</td>
</tr>
<tr>
<td>Mass Flow (max.)</td>
<td>0.42 kg/s</td>
</tr>
<tr>
<td>Mass</td>
<td>12 kg</td>
</tr>
<tr>
<td>Dimensions</td>
<td></td>
</tr>
<tr>
<td>Fueled Length</td>
<td>1.0 m</td>
</tr>
<tr>
<td>Fuel Rod Diameter (outside)</td>
<td>45 mm</td>
</tr>
<tr>
<td>Overall Length</td>
<td>1.5 m</td>
</tr>
<tr>
<td>Overall Diameter</td>
<td>55 mm</td>
</tr>
</tbody>
</table>

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CORE SUPPORT STRUCTURE

INTERNAL SHIELD CONFIGURATION
Technology Roadmap

- Demonstrated Reactor Technologies
- Technology Schedules

Mel Bulman

Energopool companies have tested NTRE fuel assemblies in reactors with hydrogen outlet temperatures as high as 3100 K. At the highest temperature the assemblies have been tested successfully for 4800 seconds including 12 starts or thermal transients. Thermal transient rates of about 400 K/sec have been used, but a single start-up rate as high as 1000 K/sec was observed without incident. Fuel assemblies have been shown to resist long duration vibration and high impact loads without critical damage. Fuel power density has been demonstrated to 25 Watt/liter, or about 15 Watt/liter of fuel bundle volume.

Based on results to date, Russian scientists estimate that their fuel assemblies will be able to demonstrate better performance and life within a 3 to 5 year demonstration program. Such improvements can lead to higher performance, lower weight, longer life, rocket engines.

Russia Has Fuel Elements and Fuel Assemblies for NRE Reactor and Plans to Improve Them

<table>
<thead>
<tr>
<th>Parameters</th>
<th>Reactor</th>
<th>Achieved</th>
<th>Expected in 3-5 Years</th>
<th>Future</th>
</tr>
</thead>
<tbody>
<tr>
<td>Hydrogen Temperature, °K</td>
<td>Nerva</td>
<td>2550</td>
<td>3000</td>
<td>≥ 3200</td>
</tr>
<tr>
<td>Specific Impulse, s</td>
<td></td>
<td>850</td>
<td>975</td>
<td>&gt; 1000</td>
</tr>
<tr>
<td>Life-Time, Seconds</td>
<td></td>
<td>3600</td>
<td>4000 s in Hydrogen at T = 3000°K</td>
<td>-10,000 s in Hydrogen at T = 3000°K</td>
</tr>
<tr>
<td>Power Density, Mw/Litre</td>
<td></td>
<td>2.5</td>
<td>25</td>
<td>40</td>
</tr>
<tr>
<td>H₂ Temperature Transient, °K/sec</td>
<td></td>
<td></td>
<td>400</td>
<td></td>
</tr>
<tr>
<td>Number of Starts-Ups</td>
<td></td>
<td>12</td>
<td>12</td>
<td>20</td>
</tr>
<tr>
<td>Vibro-Strength g/Frequency (Hz)/ Testing Time (h)</td>
<td></td>
<td></td>
<td>15/ Up to 3 KHz/50 hrs</td>
<td></td>
</tr>
<tr>
<td>Impacts, g</td>
<td></td>
<td>300</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>
Russians have developed several families of high temperature nuclear fuel and have tested them in many ways in material laboratories, nuclear reactors, and in hot cells for post-irradiation properties. Carbide fuels have demonstrated the highest temperature capabilities with good material properties. U, Zr, Nb tricarbide alloys are selected for NTRE fuel, the exact composition depending on core location. Low uranium concentration is used in the highest temperature, aft fuel bundles, and in the center of the core to flatten the radial power distribution profile. The favored geometry of individual fuel elements is that of thin, twisted strips. Coatings of carbides, nitrides, carbonitriles and pyrocarbon mixtures of tungsten, molybdenum, and rhenium have been developed and tested for corrosion resistance and thermal strength characteristics.
Engeropool fuel assemblies for NTRE are about a meter long and 30 to 45 mm in diameter. The forward, inlet end runs cooler than the rest of the assembly, and it contains an adjustable flow metering orifice, neutron reflector, and thermal expansion compliance (bellows). The center of the fuel assembly contains several bundles of fuel elements placed in series, flow passing through each sequentially, so that the hottest gas exits the aft end of the last bundle. A grid plate holds the fuel element bundles in place and allows the hot gas to pass through to the outlet plenum of the fuel assembly. Each grid plate resembles fuel bundles, except that the carbide elements are fused together and contain no uranium. The outlet plenum delivers gas to either a DeLaval rocket nozzle or to a subsonic diffuser (selected for our Mars engines). The outer envelope is metallic and hydrogen cooled within the reactor, and it is insulated internally with several layers of graphitic material.

Each Fuel Assembly Is a Fully Integrated Unit

- Energopool - Babcock & Wilcox
Fuel assemblies have been tested in the steady state nuclear reactor, IVG-1, both singly and in clusters of seven in hydrogen, and under high temperature and pressure at thermal neutron fluxes above $10^{15}$ neutrons/cm$^2$-sec.

Typical Test Arrangements of Fuel Assemblies in IVG-1 Reactor

Obtained Conditions of Tests

1. Power of 1 FA ≤ 11 MW
2. Temperature of $H_2$ = 3100K
3. Power Density $q_{v}^{\text{max}}$ = 35 MW
4. Heat Flux $q_{s}^{\text{max}}$ = 13 MW/m
5. $H_2$ Temperature Transient 150°K/s
6. Reactor Starts Per Month, 2
7. Thermal Neutron Flux = $2\times10^{15}$ neutrons/cm$^2$-s

Tested Object: Module of Active Zone With 7 FA
1. $H_2$ Flow Rate: 1.6 kg/s
2. Pressure of $H_2$: 16 MPa
3. $H_2$ Flow Rate: 0.7 kg/s
4. Pressure of $H_2$: 14 MPa
Many reactor materials have been fabricated and tested by Energopool in laboratories, in reactors, and in post-irradiation hot cells. Structural, neutron moderating, neutron reflecting, and neutron absorbing materials have been tested to high fluxes, fluences, temperatures, and immersion times in hydrogen and other media.

Materials Testing Results

<table>
<thead>
<tr>
<th>Materials</th>
<th>Neutrons cm²/s</th>
<th>Neutrons cm²</th>
<th>T, °K</th>
<th>Life-Time, Hours</th>
<th>Medium</th>
<th>Notes</th>
</tr>
</thead>
<tbody>
<tr>
<td>Steel and its Alloys (16 Types)</td>
<td>$10^{13}$</td>
<td>up to $3 \times 10^{12}$</td>
<td>77...1100</td>
<td>500 - 12,000</td>
<td>$H_2$, He</td>
<td>Vacuum</td>
</tr>
<tr>
<td>Be, Be-Al, Be with Coatings</td>
<td>$10^{2}$...$10^{4}$</td>
<td>up to $2 \times 10^{4}$</td>
<td>77...1200</td>
<td>500 - 2,400</td>
<td>$H_2$, He</td>
<td>$H_2O$ Vacuum</td>
</tr>
<tr>
<td>ZrH, ZrH + B, LiH</td>
<td>$10^{2}$...$10^{4}$</td>
<td>up to $4 \times 10^{19}$ up to $2 \times 10^{21}$</td>
<td>400...1,000</td>
<td>500 - 12,000</td>
<td>$H_2$, $CO_2$</td>
<td>$H_2O$</td>
</tr>
<tr>
<td>Absorbing Elements</td>
<td>$10^{2}$...$10^{4}$</td>
<td>up to $2 \times 10^{2}$</td>
<td>600...1,300</td>
<td>500 - 10,000</td>
<td>$H_2$, He</td>
<td>Vacuum</td>
</tr>
</tbody>
</table>
The experimental capability of Energopool is rather complete. Between six companies they have experienced analysis, design, manufacturing, and test personnel. Energopool has facilities that cover the range of critical assemblies, shielding investigation, material and equipment manufacturing, reactor safety analysis, and many testing laboratories. These laboratories include facilities for material investigation in pre, in-pile, and post-irradiation environments, hot cells, and a variety of research reactors for fuels investigation.

Experimental Capabilities of CIS Test Facilities for Validation of NRE Reactor Development
(Carried Out Tests)

- EWG-1 Reactor
  - J E SIA "Luch" 300 FAs Tested
  - T = 2800–3100 K
  - r = 400 sec
  - Multiple Cycles
  - Failures Not Observed
- IGR Reactor
  - J E SIA "Luch"
  - T = 600 K
  - dT = 35 MW/1
  - T = 3100 K
- IRIGIT Reactor
  - J E SIA "Luch"
  - T = 2600 K
  - N = 50 MW
- RA Reactor
  - J E SIA "Luch"
  - T = 2300 K
  - Int. Hot Cell
  - t = 8000 hr
- Electric Thermal
  - Benchmarks
  - J E SIA "Luch"
  - T = 2400 K
  - H2, Vacuum
  - NRE Reactor for Mars Mission
  - RUV-2M Reactor
  - SB-RIPPE
  - T = 70–1000 K, $\sigma = 10^{-27}$ cm$^2$
  - Hydride Moderator and Structural Materials;
  - Absorbing Materials T = 1200 K, $\sigma = 10^{-27}$ cm$^2$
  - H2 Media
- MANUFACTURING
  - 2FE Process Line
  - SIA "Luch"
  - 1 Core/Year
  - Reactor Components
  - (Including Shielding)
  - Plant
  - SB-RIPPE
  - 90% Complete (Full Set of Equipment)
  - 1 Set/Year
- IH-50 Reactor
  - RIPPE
  - VVR Reactor
  - SC "Kurchatov Institute"
  - Shielding Investigation
- Critical Assemblies
  - SC "Kurchatov Institute"
  - and RIPPE

GenCorp • Energopool • Babcock & Wilcox
Parallel 4-Year Technology Development for an NTRE Breadboard Engine
Summary

Mel Bulman

Our Integrated Engines Provide

Safety and Reliability

- Simple Thermodynamic Cycle
- Integrated Auxiliaries Simplify Propulsion
  - Start System
  - RCS (No Igniters, O₂, Combustion)
  - Electric Power and H₂ Refrigeration
  - Four Core Cooling Systems
- Improved Engine Start (Preheat)
  - No Thermal Shocks
  - Enhances Multiple Engine Safety
  - No LH₂ in Core (Reduced Reactivity Insertion)
  - Thermal and Acoustic Damping
  - Assured Restart
- High Margins – Long Life
  - Low Fuel Temp and Stress (4600°F)
  - Low Turbine Temperature (400°F)
  - Low Nozzle Temperature (600°F)
  - Low Moderator Temperature (400°F)
  - No Deep Thermal Cycles
Our Integrated Engines Provide

Mission Benefit

- Improved Mission Average Isp
  - Heterogeneous Reactor
  - Greatly Reduced After Cool Loss, Save > 100 K lb LH2
  - LH2 Refrigeration Option
  - OMS Thrust at > 700 sec Isp (w/o Pump Start)
  - ACS Thrust at > 500 sec Isp

- Improved Engine Thrust/Weight
  - High Power Density Reactor
  - High Pc (Reduces Shield and Nozzle Size and Weight)

- Operational Benefits
  - Deep Throttling (Enables Multiple Burn TMI)
  - ~ 100 kWe Electric Power/Engine
  - Rapid Restart

Our Integrated Engines Provide

Low Life Cycle Cost

- TRL 4 to 6 for Major Components
  - CIS Fuel Developed
- Smaller, Lower Cost Components
  - High Pc
  - Nozzle
  - Pressure Vessel
  - Shield
- High Pc Enables Small ETF
  - High Pressure Storage
  - High ΔP Scrubbers
- Reduced ETO Cost
  - Reduced IMLEO
  - Smaller Payload Bay
- Design Flexibility and Growth Potential
  - Reduces Cost
    - Recuperated Cycle
    - Electrical Power System
Lunar NTR Vehicle Design & Operations Study

**Objectives**

Perform an Evaluation of the Potential Applications of a Specific Nuclear Thermal Rocket (NTR) Design to Past and Current (First Lunar Outpost) Mission Profile(s) for Piloted and Cargo Lunar Missions, and to Assess the Applicability of Utilizing Lunar Vehicle Design Concepts for Mars Missions

- **Products**
  - Define and Size the Stage/Transfer Vehicles for Lunar NTR Applications.
  - Perform a Conceptual and Programmatic Assessment.
  - Perform a Chemical/NTR Lunar Concept Comparison.

**Lunar Orbit & Direct Design Concepts**

**Key Subsystem & Operations Sensitivities**

**Mars Growth/Evolution Approach**
Functional Reqs

1.0 Preflight Proc
2.0 Launch Ops
3.0 Transfer to Lunar Surface
4.0 Surface Ops
5.0 Earth Return

System Reqs

Lunar Direct
NLS & Saturn V derived launch systems
4 Day Transit
Cargo Mission - 33 t
Manned Mission - 5 t
LOI Opn Altitude = 300 km
Post TLI Altitude = 30 m/s
Post LOI Altitude = 860 m/s
Surface Stay = 45 Days
NTR reuse altitude = 500 km
Return 200 kg to Earth
Post TLI disposal AV = 194 m/s

Lunar Orbit Rendezvous
66t & 135t launch system
Only 2 HLLV Flts per mission
2nd HLLV Flt within 3 days
3 Day Transit
LOI Ops Altitude = 300 km
Post TLI Disposal AV = 30 m/s
Post LOI Disposal AV = 860 m/s
Surface Stay = 180 Days
NTR reuse altitude = 500 km
Return 500 kg to Earth
Post TLI disposal AV = 194 m/s

Mission Case I - NTR Performs TLI
Earth Orbit Insertion
Trans Lunar Injection
Earth Orbit Injection
Trans Earth Injection

Mission Case II - NTR Performs TLI & LOI
Earth Orbit Insertion
Trans Lunar Injection
Earth Orbit Inertion
Trans Earth Injection

Mission Case III - NTR Performs TLI, LOI, & TEI
Earth Orbit Insertion
Trans Lunar Injection
Earth Orbit Insertion
Trans Earth Injection

Mission Case IV - NTR Performs TLI, LOI, TEI, & EOC
Earth Orbit Insertion
Trans Lunar Injection
Earth Orbit Insertion
Trans Earth Injection

MARTIN MARIETTA

Mission Statement
- Support Exploration & Habitation of Lunar Surface with consideration for evolvable to Mars
- Lunar IOC: 2000-2005
- Mars IOC: 2005 Cargo 2007 Piloted
LD Space Operations Overview

Key Features:
- NTR Performs TLI, LOI
- Longest single burn time 30 min
- 45 Day Lunar Surface Stay
- 1 HLLV Launch
- 160I Capability
- Post-LOI NTR Disposal (860 m/s)

LOI
ΔV = 907 m/s
& NTR Separation

Ascent
ΔV = 1810 m/s

Descent
ΔV = 1830 m/s

LOI
ΔV = 907 m/s

NTR Stage in LLO

Rendezvous & Dock in LLO

Rendezvous & Dock and Reconfigure in LEO

LOR Space Based Operations Overview

Key Features:
- NTR Performs TLI, LOI, TEI, and EOC
- Infrastructure Requirements:
  - STS delivers/retrieves crew at SSF
  - Cab and Lander refueled at SSF
  - Provide on-orbit refueling or tank exchange for Lander & NTR

LOI
ΔV = 907 m/s

Descent
ΔV = 1830 m/s

Rendezvous & Dock in LLO
System Design Considerations

- Expendable vs. Reusable
  Operations Complexity Too High For Reusability
  Performance Maximization Achieved With Expendable Mission
  Safety Concerns Lessened With Expendable Mission

- Shielding Considerations
  NERVA Disk Shield
  Modified Disk Shield Optimizes Design and Use of Propellant
  Propellant and Tankage
  Lander Propellant and Structure

- Launch Vehicle Considerations
  Lunar Direct Mission
  Smallest Launch Vehicle Necessary to
  Complete FLO Mission
  Lunar Orbit Rendezvous
  Complete Reasonable Lunar Architecture
  Using Medium Sized Launch Vehicle

- Thermal Protection Considerations
  Active Refrigeration Too Heavy For Benefit & Abort Mission If
  Failed MLI & SOFI

- Lander Considerations
  Lunar Direct Mission
  2 Stage Cryo/Storable Removes Long Term
  Hydrogen Storage on Orbit
  Lunar Orbit Rendezvous
  1 1/2 Stage Cryo/Cryo Out-performs 2 Stage
  Lander Consistently in Past STV Studies

- Material and Construction Considerations
  Aluminum Lithium Technology On Schedule For
  Flight Use By 2005
  Isogrid Construction Promising For Structural
  Considerations

- Engine Configuration
  Single vs Cluster
  25k vs 50k vs 75k

MARTIN MARIETTA
Lunar Orbit Rendezvous Configurations

The chart below shows the top seven candidates of the Lunar Orbit Rendezvous configurations. We started with eighteen possible configurations in this category, and through performance runs, design constraints and operational issues, that eighteen was narrowed to the following seven. The missions that utilized a cryogenic liquid oxygen and liquid hydrogen TLI stage were extremely close in performance to those missions using NTR to perform the TLI burn. Therefore, it was beneficial to show the elimination of an entire technology, use of a LOX/LH₂ stage, and to show that a lunar mission can be supported solely by NTR technology. Another criteria that eliminated candidates was performance at least 5.00 tonnes to the lunar surface on a piloted mission. Also eliminated in the earlier phases of the study were two HLLV candidates. We started with four HLLV candidates: 661, 1051, 1321, and 1351 launch capacities. We narrowed that field to two candidates based on past STV analysis showing the 1321 and 1351 vehicles virtually even on performance. Of the three that were left, 661, 1051, 1351, we eliminated the 1051 because of study complexity and to demonstrate that NTR can be utilized on the two extreme launch vehicles and, therefore, everything in between.

Three of the configurations shown below started with a cluster of three 25Klbf NTR engines, but further analysis their performance was greatly enhanced by going to a single engine configuration. Also, two of the configurations are two-stage NTR configurations. The first NTR stage performs the TLI burn and is then staged off to perform a lunar assist disposal burn into heliocentric space. The second NTR stage then performs the rest of the mission and is also disposed of after the TFI burn.

---

**Lunar Orbit Rendezvous Configurations**

<table>
<thead>
<tr>
<th>Configuration</th>
<th>NTR Performs TLI &amp; Dispose 129 HLLV</th>
<th>NTR Performs TLI, LOI, TEI &amp; Dispose 661 HLLV</th>
<th>NTR Performs TLI, LOI, TEI, &amp; Dispose 132 HLLV</th>
<th>NTR Performs TLI, LOI, TEI, &amp; EOC 139 HLLV</th>
<th>NTR Performs TLI, LOI, TEI &amp; Dispose 661 HLLV 2 Stage</th>
<th>NTR Performs TLI, LOI, TEI &amp; Dispose 139 HLLV 2 Stage</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Mass (tonnes)</strong></td>
<td>254.75</td>
<td>263.25</td>
<td>266.50</td>
<td>268.63</td>
<td>267.83</td>
<td>261.68</td>
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<tr>
<td><strong>Performance</strong></td>
<td>59.55</td>
<td>19.55</td>
<td>7.2</td>
<td>36.68</td>
<td>19.35</td>
<td>5.88</td>
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<tr>
<td><strong>Cargo</strong></td>
<td>47.55</td>
<td>20.62</td>
<td>25.66</td>
<td>61.53</td>
<td>44.93</td>
<td>23.09</td>
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</table>

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**NTP: System Concepts**

---

**ORIGINAL PAGE IS OF POOR QUALITY**
Lunar Orbit Rendezvous Configurations

NTR Performs TLI and Disposed
NTR Performs TLI and Reused

<table>
<thead>
<tr>
<th>Component</th>
<th>LOI/TEI Stage</th>
<th>NTR Stage</th>
</tr>
</thead>
<tbody>
<tr>
<td>Structure</td>
<td>2.46</td>
<td>1.37</td>
</tr>
<tr>
<td>Tankage</td>
<td>0.99</td>
<td>5.9</td>
</tr>
<tr>
<td>Subsystems</td>
<td>2.90</td>
<td>2.90</td>
</tr>
<tr>
<td>Engine Structure</td>
<td>0.51</td>
<td>0.63</td>
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<tr>
<td>Engines</td>
<td>0.80</td>
<td>11.50</td>
</tr>
<tr>
<td>Contingency (15%)</td>
<td>14.61</td>
<td>12.46</td>
</tr>
<tr>
<td>Total Dry</td>
<td>8.75</td>
<td>23.01</td>
</tr>
<tr>
<td>Propellant</td>
<td>49.91</td>
<td>91.86</td>
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<tr>
<td>Total Wet</td>
<td>48.56</td>
<td>114.19</td>
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<table>
<thead>
<tr>
<th>Component</th>
<th>NTR Stage &amp; Propellant</th>
<th>NTR Stage &amp; Reused</th>
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<tbody>
<tr>
<td>Structure</td>
<td>1.37</td>
<td>1.95</td>
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<tr>
<td>Tankage</td>
<td>3.71</td>
<td>6.80</td>
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<tr>
<td>Subsystems</td>
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<td>2.87</td>
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<tr>
<td>Engine Structure</td>
<td>0.41</td>
<td>0.96</td>
</tr>
<tr>
<td>Engine</td>
<td>3.73</td>
<td>6.83</td>
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<tr>
<td>Shield</td>
<td>1.50</td>
<td>4.50</td>
</tr>
<tr>
<td>Contingency (15%)</td>
<td>1.81</td>
<td>3.59</td>
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<tr>
<td>Total Dry</td>
<td>14.61</td>
<td>27.50</td>
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<tr>
<td>Propellant</td>
<td>63.73</td>
<td>123.09</td>
</tr>
<tr>
<td>Total Wet</td>
<td>78.34</td>
<td>150.59</td>
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</tbody>
</table>
Lunar Orbit Rendezvous Configurations

NTR Performs TLI, LOI, TEI and EOC - 1351 HLLV

NTR Stage

<table>
<thead>
<tr>
<th>Component</th>
<th>t</th>
</tr>
</thead>
<tbody>
<tr>
<td>Structure</td>
<td>2.91</td>
</tr>
<tr>
<td>Tankage</td>
<td>9.70</td>
</tr>
<tr>
<td>Subsystems</td>
<td>3.45</td>
</tr>
<tr>
<td>Engine Structure</td>
<td>0.96</td>
</tr>
<tr>
<td>Engine</td>
<td>6.83</td>
</tr>
<tr>
<td>Shield</td>
<td>4.50</td>
</tr>
<tr>
<td>Contingency (15%)</td>
<td>4.25</td>
</tr>
<tr>
<td>Total Dry</td>
<td>32.60</td>
</tr>
<tr>
<td>Propellant</td>
<td>160.08</td>
</tr>
<tr>
<td>Total Wet</td>
<td>192.68</td>
</tr>
</tbody>
</table>

Lunar Orbit Rendezvous Configurations

2 Stage NTR Performs TLI, LOI, TEI and Dispose - 661 HLLV

NTR Stage 1 & 2

<table>
<thead>
<tr>
<th>Component</th>
<th>t</th>
</tr>
</thead>
<tbody>
<tr>
<td>Structure</td>
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</tr>
<tr>
<td>Tankage</td>
<td>4.50</td>
</tr>
<tr>
<td>Subsystems</td>
<td>1.69</td>
</tr>
<tr>
<td>Engine Structure</td>
<td>0.76</td>
</tr>
<tr>
<td>Engine</td>
<td>14.91</td>
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<tr>
<td>Shield</td>
<td>1.50</td>
</tr>
<tr>
<td>Contingency (15%)</td>
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<tr>
<td>Total Dry</td>
<td>29.84</td>
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<tr>
<td>Propellant</td>
<td>54.42</td>
</tr>
<tr>
<td>Total Wet</td>
<td>84.26</td>
</tr>
</tbody>
</table>
NTR Reuse Examined in Study

- Developed vehicle mass properties, payload capabilities, space operations
  - 2 cases consider reuse for NTR only
  - 2 cases consider space based fully reusable vehicles

Notes: Reusable Hardware elements designed for 5 mission
NTR returns to 500 km circular orbit between missions

- Identified infrastructure needs (assumed existing)*
  - STS for crew delivery/retrieval
  - SSF for refueling
  - Capability for on-orbit refueling or tank exchange

*Infrastructure costs (elements & associated operations) were not included in cost analysis

Reuse Cost Analysis

Lunar Orbit Rendezvous

Lunar Direct

Reuse Reduces
Vehicle Recurring
Cost approximately
50%

Reuse Increases
Vehicle DDT&E
approximately 10%
NTR/Cryo Reusable System Comparison

Space Based 90-Day Study Derived LTS

Reusable Rendezvous & Docked NTR/Cryo LTS

<table>
<thead>
<tr>
<th>MASS PROPERTIES</th>
<th></th>
<th></th>
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</thead>
<tbody>
<tr>
<td>NTR</td>
<td>LTS</td>
<td>LTS</td>
</tr>
<tr>
<td>Dry Mass</td>
<td>27.56</td>
<td>27.56</td>
</tr>
<tr>
<td>NTR/LU Stage</td>
<td>(1) under</td>
<td>(1) under</td>
</tr>
<tr>
<td>Total Propellant</td>
<td>261.0</td>
<td>261.0</td>
</tr>
<tr>
<td>Pilot Cargo</td>
<td>5.0</td>
<td>5.0</td>
</tr>
<tr>
<td>TOTAL SYSTEM</td>
<td>283.6</td>
<td>283.6</td>
</tr>
</tbody>
</table>

Rendevous & Docked LTS (2 135 HLLVs)

Mars Evolution Key to Affordability

Total Cost Breakeven Analysis

Sharing Development Cost of Common Elements with Mars Program

Lowers Total Cost of Lunar Missions

Example: Splitting the engine development cost between the lunar and Mars programs shifts breakeven point from 300 t to less than 60 t.

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Conclusions

- Near term NTR Provides Feasible Alternative to Cryo system for Lunar Missions

  - Performance
    NTR LD concept offers smaller IMLEO for same payload capability as cryo system
    NTR LOR offer greater payload delivery capability for same IMLEO
  - Cost
    NTR more cost efficient ($/kg) than cryo system
    NTR Development cost greater than cryo systems
  - Ops
    LOR option requires on-orbit cryo transfer (technology risk)
  - Schedule
    1st cargo launch capability in 2002

- Near term NTR enable efficient evolution to Mars

  - Cost
    Shared development cost of common elements enhances affordability
    Hardware commonality
  - Mission
    Lunar NTR adaptable to wide range of mission architectures (Direct, MOR)
Mission Design Considerations
for
Nuclear Risk Mitigation

Mike Stancali
John Collins

- Safe Return of NTR to Earth Orbit
  Pulsed cooldown propellant can be used to lower
capture orbit to selected operations altitude

- Lunar/Mars NTR Disposal
  Modest cost, low risk disposal to heliocentric orbits for
all transfer trajectories

Aim Point Bias Offsets Targeting Errors

One of the operational safety concerns for nuclear propulsion is how to manage safe return of a nuclear powered transfer vehicle to Earth orbit. Although the mission profiles used in this study call for crew return by an Earth Crew Capture Vehicle (ECCV), capturing the transfer vehicle offers the added flexibility and possible cost savings of reuse. The return orbit must be low enough to be accessible from Earth launch at reasonable cost, yet high enough to ensure safety.

Experience shows that orbit insertion errors are caused by some or all of the following factors:

- Errors in Trajectory Correction Maneuvers (TCM), resulting from off-nominal thrust level, direction, or duration. These may be caused by the propulsion or attitude control subsystems.
- Inherent uncertainty in determining the spacecraft’s orbit
- Small errors in precise location of natural bodies

However, the resulting variations in spacecraft orbit parameters are small; orbit insertion altitudes vary by only a few kilometers. The performance of a nuclear thermal rocket should be similar to past experience with chemical systems. The critical item, then, is to select a nominal return orbit that matches lifetime characteristics with the needs of short-term storage in Earth orbit.

The aim point can be biased so as to raise the distance of closest approach, and capture into some orbit higher than the desired one. After exact position and status of the vehicle is determined, a series of smaller burns lowers the orbit to match the final size. This approach will be the basis of a proposed strategy for making effective use of pulse-mode cooldown propellant.
Aim Point Bias Offsets Targeting Errors

- Targeting errors result from off nominal TCM burns, uncertainty in orbit determination.
- Typical injection errors are small: altitude dispersions of a few kilometers.
- Damp out dispersion effects by changing the aim point to a higher orbit (elliptic or circular), then use small impulses to lower to desired final orbit.

Cooldown Propellant Characteristics

Since NTP systems must cool the reactor with flowing hydrogen after every main burn, it would be desirable to use as much as possible of this propellant for productive thrust. The continuous cooldown flow lasts for a few minutes, and is assumed to handle part of the required capture impulse. The pulsed flow lasts over several hours, with the exact profile depending on the main burn duration. Pulsed flow averages a specific impulse of about 440 seconds, but at a very low thrust level.

The table in the lower right corner opposite shows the four phases for a main burn of 600 seconds, typical of Earth orbit capture burns for return from the Moon. In this case, the pulsed flow must occur over 31.5 hours to keep the reactor within the specified temperature range.
Altitude Profile During Pulsed Cooldown

The baseline profile for a piloted mission to either the Moon or Mars is to separate the crew in an ECCV for direct reentry at Earth return. This eliminates the operational concerns of crew safety on board during the extended cooldown time interval. Modeling of the pulsed flow shows a total propellant requirement of 1,473 kg for cooldown purposes. From the LTV mass requirements, shown elsewhere in this study, this translates to a ΔV of 117 m/s, assuming the maximum effective thrust could be imparted by the propellant flow.

The problem is to match the required cooldown propellant flow with an orbit modification strategy that meets mission requirements, while deriving maximum value from the available ΔV. The approach used here is to accept the pulsed flow profile as given in the Aerojet FEP output runs referenced on the previous page (although it may be possible to modify it), and to use each low thrust impulse to lower the altitude of the initial capture orbit. We begin with a circular orbit at some altitude, and proceed to apply a sequence of many small impulses to lower to the desired 500 km orbit at the end of pulsed cooldown. Starting with a circular orbit at 716 km will produce the desired final altitude of 500 km at the end of this thrusting program.

This use represents one example of how the pulsed cooldown propellant may be used for transfer vehicle thrust. A complete characterization for the range of engine sizes considered in this study will require burn simulations for various thrust levels, burn times, and numbers of engines staged together.
Altitude Profile During PulsedCooldown

- LOR Mode, with NTR performing 4 main burns
- Crew separates from LTV on approach and returns in ECCV
- 600 sec burn for Earth capture

- Efficient use of pulse-mode cooling for thrust is the active constraint in selecting an initial capture orbit
- Biasing to counter possible orbit insertion errors is easily satisfied by this initial selection

NTR Disposal for Lunar Missions

Two broad categories of long-term disposal orbits have been examined for use with nuclear thermal propulsion in lunar and Mars transfer vehicles. These are: Earth orbits at altitudes high enough to ensure long life before reentry, and various heliocentric options. Although all of these offer real possibilities for reactor disposal, selection may depend on programmatic guidelines. As a conservative approach, we consider the heliocentric options to be preferable, so long as the propulsion requirements are reasonable. As the table opposite shows, AV to reach a particular disposal orbit is highly dependent on where the transfer vehicle is when the disposal operation commences. Since this study considers a variety of NTP use scenarios, there is no single lowest-cost solution.

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NTR Disposal for Lunar Missions

- Want permanent solution:
  - small long-term risk of Earth reentry
  - no requirement for active management

- Two classes of candidates: Earth orbits and heliocentric orbits

- Cost (AV) influenced by location when disposal sequence begins

- Final selection will depend on program's balancing of risk (real and perceived), and cost

<table>
<thead>
<tr>
<th>Disposal Starts From</th>
<th>NTR Final Disposal Location</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Earth Orbit &gt; 1000 km</td>
</tr>
<tr>
<td>Post-TLI</td>
<td>3110</td>
</tr>
<tr>
<td>Lunar Orbit</td>
<td>4210</td>
</tr>
<tr>
<td>Post-TEI</td>
<td>3110</td>
</tr>
<tr>
<td>LEO</td>
<td>200</td>
</tr>
</tbody>
</table>

Crossing Orbits

Previous work by SAIC identified a class of Earth-crossing orbits that lie just inside or just outside Earth's orbit, and that are slightly out of the ecliptic. Using a Monte Carlo simulation code to estimate the lifetimes and reentry probabilities of a body in such an orbit shows promising statistics for use as a disposal location. The Earth reencounter probability shown in the graph represents a slightly higher risk than collision with an asteroid in similar time periods.

As the table on the previous page showed, the crossing orbits are easily attained from most points in lunar mission profiles. The exception is disposal of a vehicle from LEO.
Crossing Orbits

- Graze Earth's orbit at 0.88 x 1.0 A.U. or 1.0 x 1.15 A.U.

- Are slightly inclined to minimize reentry probability: $i > 2 \text{ deg}$

- Are predicted to have: [Planetary Encounter Probability Analysis (PEPA) code]
  - mean lifetimes of the order of $10^7$ years
  - probability of Earth reencounter in $\leq 10,000$ years of 1-2%
  - probability of Earth reencounter in $\leq 1,000,000$ years of 17-18%

![Graph showing Earth's orbit and disposal orbit.]

Stable Orbits

The second category of heliocentric orbits was also identified by SAIC as a possible permanent storage location for hazardous waste in space.\(^2\) 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-Moon and Earth-Mars cases is 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 graph in the lower left corner. This graph plots heliocentric distance as a function of time (note the x-axis scale!) for the periapse and apoapse of the stable orbit. The Mars periapse 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.

As the earlier table indicated, significant impulses (1450-4550 m/s) are required from the lunar flight path to deliver the LTV to this orbit. Although there is a cost difference over the crossing orbit, the stable orbit offers a greater risk reduction potential. Whether the additional risk reduction is required will depend in large measure on program guidelines and policy.

---

Stable Orbits

- An orbit is stable over time $T$ if a body in that orbit doesn't cross a planet's path in $T$
- Starts at $1.19 \times 1.19$ A.U., becomes elliptic, but doesn't cross Mars or Earth

Two options are considered for Mars missions: the stable heliocentric orbit, or disposal along a transfer trajectory that the transfer vehicle is following. For the former, the ΔV requirements for two split/conjunction mission pairs are shown on the facing page. Cargo missions need two impulses to leave Mars and to circularize. Crew mission trajectories are modified to perform Earth gravity assist after ECCV separation, saving roughly 2 km/s impulse. The orbit plot shows the 2007 crew return profile, with Earth swingby to final capture burn at 1.19 A.U.

The second option is to leave the transfer vehicle in its flight path. In all cases, the flight path crosses at least one planet's path, setting up possible unintended gravity assists in the future. However, predicted chance of Earth reentry in one million years is generally of the same order as the likelihood of colliding with a typical near-Earth asteroid. The only exception is the near-Hohmann transfer leg from Earth to Mars for the cargo vehicle.
NTR Disposal for Mars Missions

- Consider Stable orbit, or disposal on interplanetary path

<table>
<thead>
<tr>
<th>Mission</th>
<th>Disposal Starts From</th>
<th>Required Maneuvers</th>
<th>( N )</th>
</tr>
</thead>
<tbody>
<tr>
<td>2005 Cargo</td>
<td>Mars orbit, after rendezvous</td>
<td>Depart Mars Orbit</td>
<td>0.664 km/s</td>
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<tr>
<td></td>
<td></td>
<td>Circularize at 1.19 A.U.</td>
<td>0.998</td>
</tr>
<tr>
<td>2007 Crew</td>
<td>Earth approach, after ECCV separates</td>
<td>Earth Gravity Assist</td>
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<tr>
<td></td>
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<td>Circularize at 1.19 A.U.</td>
<td>2.954</td>
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<tr>
<td>2007 Cargo</td>
<td>Mars orbit, after rendezvous</td>
<td>Depart Mars Orbit</td>
<td>0.665</td>
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<td>Circularize at 1.19 A.U.</td>
<td>1.000</td>
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<tr>
<td>2009 Crew</td>
<td>Earth approach, after ECCV separates</td>
<td>Earth Gravity Assist</td>
<td>0</td>
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<tr>
<td></td>
<td></td>
<td>Circularize at 1.19 A.U.</td>
<td>3.017</td>
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</tbody>
</table>

Chance of Earth Reentry in \( 1 \times 10^6 \) years - %

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<thead>
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<th>2007/09</th>
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<tr>
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<td>E-M</td>
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<td>2.2</td>
</tr>
<tr>
<td></td>
<td>E-D</td>
<td>1.2</td>
</tr>
</tbody>
</table>

E = Earth  M = Mars  D = Disposal

SAIC

Senior Applications Innovative Concepts

NP-TIM-92 365 NTP: System Concepts
ENABLERS I AND II ENGINE SYSTEM DESIGN
MODELING AND COMPARISONS

23 OCTOBER 1992

PRESENTED BY:
DENNIS G. PELACCO AND CHRISTINE M. SCHUH
SCIENCE APPLICATIONS INTERNATIONAL CORPORATION
ALBUQUERQUE, NM 87123

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

TOPICS

- OBJECTIVE/APPROACH
- ENGINE SYSTEM DESIGN/MODELING ASSUMPTIONS
- ENGINE SYSTEM SCALING/COMPARISONS
- CONCLUDING REMARKS
OBJECTIVE/APPROACH

- **Objective:**
  - Define a Near-Term Solid-Core NTP Engine System Scaling Database
    - Identify/Document Unified Set of Performance, Weight and Size Scaling Data
    - Results Should Be Useful to Meet Initial Mission and Concept Design Study Requirements

- **Approach:**
  - Acquire/Review Past Rover/NERVA Engine Design Work
  - Assess Current Engine System Data
  - Conduct Preliminary NTP Engine System Design Trades Using the NESS Design Program
    - Establish Operating Range of Interest and Technology Design Approach
  - Design Analysis Responsibilities
    - SAIC - Engine System
    - Westinghouse - Reactor and Internal Shield (ENABLEX Reactor)
  - Establish a Catalog of Enabler I and II Engine System Design for a Range Configurations and Operating Conditions
ENABLER I AND II ENGINE SYSTEM DESIGN
DATABASE DEVELOPMENT CLOSELY PARALLELED NESS PROGRAM DEVELOPMENT

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 CUSMIC
- PC and Vax Versions

OVERALL NTP ENGINE SYSTEM ASSESSMENT APPROACH

ESTABLISH ENGINE DESIGN
- Cycle Type
- Redundancy Issues
- Technologies and Component Design Approaches to be Employed

UPGRADE, CORRELATE NESS CYCLE ANALYSIS CODE
- Use Representative Rocketdyne 75,000lb Thrust Engine Case for Comparison

WESTINGHOUSE REACTOR AND INTERNAL SHIELD DESIGN MODULE

IDENTIFY DESIGN CONFIGURATION OPERATING RANGES OF INTEREST
- Thrust
- Reactor Type
- Chamber Pressure & Temperature
- Area Ratio

PRELIMINARY ENGINE SYSTEM DESIGN ANALYSIS

ESTABLISH NTP WEIGHT, PERFORMANCE DATA BASE
OVERALL NTP ENGINE SYSTEM ASSESSMENT APPROACH

- Establish Engine Design
  - Cycle Type
  - Reactor Base
  - Technologies and Components
  - Design Approaches to be Employed

- Upgrade Correlate Nuss Cycle Analysis Code
  - Use Representation Reactorlyse
  - 25,000lb Thrust Engine Case
  - For Comparison

- Westinghouse Reactor and Internal Shield Design Module

- Identify Design Configuration
  - Operating Ranges of Interest
  - Thrust
  - Recovery Type
  - Chamber Pressure & Temperature
  - Area Ratio

PRELIMINARY ENGINE SYSTEM DESIGN ANALYSIS

ESTABLISH NTP WEIGHT, PERFORMANCE DATA BASE

KEY TECHNICAL DOCUMENTS REVIEWED

- XE-Prime Engine - Final Report
- Small Nuclear Engine Final Report
- Experience Gained From the Space Nuclear Rocket Program (ROVER)
- History Summary Report
- Safe, Compact, Nuclear Propulsion - Final Report
- NERVA Preliminary Design Safety Report
- Nuclear Rocket Engine Optimization Program
- NASA Lewis Nuclear Thermal Propulsion Workshop, 1990
- On-Going NASA Lewis NTP System Studies
ENABLER REACTOR DESIGN AND OPERATING PARAMETERS EXAMINED

- Full Element/Chamber Temperature Range
  - Graphite: 2,200 - 2,500K
  - Composite: 2,500 - 2,900K
  - Carbide: 2,900 - 3,300K
- Thrust Level: 15,000-250,000 lbf
- Chamber Pressure: 500 and 1000 psia

ENABLER (NERVA TYPE) NUCLEAR THERMAL ROCKET ENGINE
**PRISMATIC FUEL ELEMENTS AND SUPPORTS**

**FUEL AND SUPPORT ELEMENTS PARAMETERS**

<table>
<thead>
<tr>
<th>Fuel Element Composition</th>
<th>Graphite</th>
<th>Composite</th>
<th>Carbide</th>
</tr>
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<tbody>
<tr>
<td>Temperature Range K</td>
<td>2200 - 2500</td>
<td>2500 - 2900</td>
<td>2900 - 3300</td>
</tr>
<tr>
<td>Fuel</td>
<td>Coated Particle</td>
<td>UC • ZrC Solid Solution and Carbon</td>
<td>(U, Zr) C Solid Solution</td>
</tr>
<tr>
<td>Coating</td>
<td>ZrC</td>
<td>ZrC</td>
<td>—</td>
</tr>
<tr>
<td>Unfueled Support Element Composition</td>
<td>Graphite</td>
<td>ZrC-Graphite Composite</td>
<td>ZrC</td>
</tr>
<tr>
<td>Unfueled Element Coating</td>
<td>ZrC</td>
<td>ZrC</td>
<td>—</td>
</tr>
</tbody>
</table>
REACTOR PARAMETERS/CHARACTERISTICS
AS A FUNCTION OF THRUST LEVEL

<table>
<thead>
<tr>
<th>Thrust (1000 lb)</th>
<th>15</th>
<th>25</th>
<th>&gt;50</th>
</tr>
</thead>
<tbody>
<tr>
<td>Reactor Power Range (MW)</td>
<td>275 - 400</td>
<td>460 - 670</td>
<td>920 - 6700</td>
</tr>
<tr>
<td>Fuel and Support Element Length [in (inch)]</td>
<td>0.89 (35)</td>
<td>0.89 (35)</td>
<td>1.32 (52)</td>
</tr>
<tr>
<td>Pressure Vessel Length [in (inch)]</td>
<td>2.10 (82.6)</td>
<td>2.13 (84)</td>
<td>2.58 (101.6)</td>
</tr>
<tr>
<td>Fuel Element Power (MW)</td>
<td>0.629</td>
<td>0.808</td>
<td>1.20</td>
</tr>
<tr>
<td>Relative Fuel Element Power Density</td>
<td>0.778</td>
<td>1.0</td>
<td>1.0</td>
</tr>
<tr>
<td>Ratio of Fuel Elements (N) to Support Elements</td>
<td>2:1</td>
<td>3:1</td>
<td>6:1</td>
</tr>
<tr>
<td>Pressure Vessel Material</td>
<td>Aluminum</td>
<td>Aluminum</td>
<td>Aluminum</td>
</tr>
<tr>
<td>Reflector Material</td>
<td>Beryllium</td>
<td>Beryllium</td>
<td>Beryllium</td>
</tr>
<tr>
<td>Internal Shield Material</td>
<td>BATH*/Lead</td>
<td>BATH/Lead</td>
<td>BATH/Lead</td>
</tr>
</tbody>
</table>

*RATH = Borated Aluminum Titanium Hydride

RADIATION LEAKAGE LIMITS CRITERIA ASSUMED
- AT A PLANE 160 CM (63 INCHES) FORWARD OF THE CORE CENTER -

<table>
<thead>
<tr>
<th>Type of Radiation</th>
<th>Radiation Leakage Limits Within Pressure Vessel Outside Radius</th>
</tr>
</thead>
<tbody>
<tr>
<td>Gamma Carbon KERMA Rate</td>
<td>1.8 x 10^7 rad (c)/hr</td>
</tr>
<tr>
<td>Fast Neutron Flux</td>
<td>2.0 x 10^12 n/cm² - sec, E_R &gt; 1.0 MeV</td>
</tr>
<tr>
<td>Intermediate Neutron Flux</td>
<td>3.0 x 10^12 n/cm² - sec, 0.4 eV ≤ E_R ≤ 1.0 MeV</td>
</tr>
<tr>
<td>Thermal Neutron Flux</td>
<td>6.0 x 10^11 n/cm² - sec, E_R &lt; 0.4 eV</td>
</tr>
</tbody>
</table>
ENGINE SYSTEM ANALYSIS APPROACH/ASSUMPTIONS

- Use SAIC NESS Cycle Analysis Code
  - Check/Adjust Code to a Reasonable Test Case
  - Rocketdyne ENABLER I and ENABLER II
    - 75,000 lb, 1000 psi Engine System Design Selected
- Model Engine as an Expander Cycle
- Incorporate near-Term State-of-the-Art Technologies
- Incorporate Dual Turbopump Feed system
  - Single Propellant turbopump With Dual Valving
    - per Feed Leg - 80% Thrust Level Capability per Leg
  - Centrifugal Turbopumps Used
  - Boost Pumps Assumed Only for the ENABLER II
    Engine System
- Use Westinghouse ENABLER Reactor System
  Design Model
  - Includes an Internal Shield Model
  - Examine Nozzle Ratios of 200 and 500:1

---

KEY NTP ENGINE DESIGN AND TECHNOLOGY FEATURES

<table>
<thead>
<tr>
<th>Design/Technology Feature(s)</th>
<th>Comment(s)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Overall Engine Mount</td>
<td>Based on the Space Shuttle Main Engine (SSME) Design Approach</td>
</tr>
<tr>
<td>Turbopump Assembly</td>
<td>- Based on NERVA Design Approach (Redundancy Considerations)</td>
</tr>
</tbody>
</table>
  - Dual Feed System Legs with Dual Valving |
  - 80% Pumping Capability per Feed Leg |
  - Centrifugal Turbopumps |
  - Pump Material: Inconel |
  - Turbine Material: MAR-M246 |
| Solid-Core NERVA Type Reactor Design | - Uses State-of-the-Art Reactor System Fuel/ Technologies/Materials (Westinghouse ENABLER Design) |
  - Internal Shield |
| Nozzle Assembly | - Conservative, High Performance Design, Common to the Propulsion Community |
  - Thrust Region |
  - 2.5:1 Upstream to 6:1 Downstream |
  - Slanted Hugan Wall Construction of Copper |
  - Intermediate Region |
  - 6:1 to 150:1 Downstream |
  - Hugan Coated Inconel Tube Paddles |
  - Exit Nozzle Extension |
  - 150:1 to Exit |
  - Radiation Coated Carbon-Carbon |
| Miscellaneous Hardware | - Uses State-of-the-Art Rocket Propulsion Materials Technology Base |
  - Propellant Lines |
  - Valves |
  - Electronics/Instrumentation/Processors |
<table>
<thead>
<tr>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>NTP: System Concepts</td>
<td></td>
</tr>
</tbody>
</table>

NP-TIM-92 15

Science Applications International Corporation

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**EXTENSIVE NESS PROGRAM VERIFICATION**
**CONDUCTED IN PARALLELL AS THE**
**ENBLER NTP DATABASE DEVELOPED**

SAIC NTP-ELES EARLY PROTOTYPE
VERSION OF NESS)
- Primarily Focused of Anchor/Verification
  of Non-Nuclear Components and
  Performance

NESS
- Primary Focused on Anchor/Verification
  Reactor System and Engine Integration
  Features

**INITIAL ENGINE COMPONENT WEIGHT COMPARISON**
- 75,000 lbf NTP ENGINE CASE -

<table>
<thead>
<tr>
<th>Parameter</th>
<th>NETIVA</th>
<th>Rocketdyne</th>
<th>SAIC ELES-NTP</th>
<th>Adjustments/Comments</th>
</tr>
</thead>
<tbody>
<tr>
<td>Chamber Temperature (°K)</td>
<td>2500</td>
<td>2700</td>
<td>2700</td>
<td>-</td>
</tr>
<tr>
<td>Chamber Pressures (psig)</td>
<td>450</td>
<td>1000</td>
<td>1000</td>
<td>-</td>
</tr>
<tr>
<td>Area Ratio</td>
<td>100</td>
<td>500</td>
<td>500</td>
<td>-</td>
</tr>
<tr>
<td>Specific Impulse - Vac (sec)</td>
<td>850</td>
<td>923</td>
<td>922.8</td>
<td>-</td>
</tr>
<tr>
<td>Reactor (kg)</td>
<td>5890</td>
<td>5824</td>
<td>5823</td>
<td>-</td>
</tr>
<tr>
<td>Internal Shield (kg)</td>
<td>1583</td>
<td></td>
<td>1523</td>
<td>-</td>
</tr>
</tbody>
</table>
| Nozzle Assembly (kg)           | 1051   | 440        | 421           | • ELES NTP Value Increased by 5%  
  - Rocketdyne Weight Considered a Good Baseline |
| Turbopump Assembly (kg)        | 243    | 304        | 104           | • ELES NTP Value Increased by 30%  
  - Rocketdyne Considered Conservative for SOA Designs |
| Nonnuclear Support Hardware (kg) | 2425  | 1815       | 1284          | • ELES NTP Value Increased by 40%  
  - Rocketdyne Weight Considered a Good Baseline - Scaled From Previous Design Work |

* Rocketdyne uses their Mark-25 type axial turbopump (4 stages); ELES-NTP used a single-stage centrifugal pump.
### INITIAL ENGINE COMPONENT WEIGHT COMPARISON* - 75,000 lbf NTP ENGINE CASE -

<table>
<thead>
<tr>
<th>Parameter</th>
<th>NERVA</th>
<th>Rocketdyne</th>
<th>SAIC ELES-NTP</th>
<th>Adjustments/Comments</th>
</tr>
</thead>
<tbody>
<tr>
<td>Chamber Temperature (°F)</td>
<td>2500</td>
<td>2700</td>
<td>2700</td>
<td>-</td>
</tr>
<tr>
<td>Chamber Pressures (psia)</td>
<td>450</td>
<td>1000</td>
<td>1000</td>
<td>-</td>
</tr>
<tr>
<td>Area Ratio</td>
<td>100</td>
<td>500</td>
<td>500</td>
<td>-</td>
</tr>
<tr>
<td>Specific Impulse - Vac (sec)</td>
<td>850</td>
<td>923</td>
<td>922.8</td>
<td>-</td>
</tr>
<tr>
<td>Reactor (kg)</td>
<td>5890</td>
<td>5824</td>
<td>5823</td>
<td>-</td>
</tr>
<tr>
<td>Internal Shield (kg)</td>
<td>1583</td>
<td>-</td>
<td>1523</td>
<td>-</td>
</tr>
<tr>
<td>Nozzle Assembly (kg)</td>
<td>1051</td>
<td>440</td>
<td>421</td>
<td>-</td>
</tr>
<tr>
<td>Turbopump Assembly (kg)</td>
<td>243</td>
<td>304</td>
<td>104</td>
<td>-</td>
</tr>
<tr>
<td>Nonnuclear Support Hardware (kg)</td>
<td>2425</td>
<td>1815</td>
<td>1264</td>
<td>-</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th></th>
<th>Rocketdyne</th>
<th>ELES-NTP</th>
</tr>
</thead>
<tbody>
<tr>
<td>Nozzle Chamber Temp (°K)</td>
<td>2700</td>
<td>2700</td>
</tr>
<tr>
<td>Nozzle Chamber Pres. (psia)</td>
<td>1000</td>
<td>1000</td>
</tr>
<tr>
<td>Nozzle Exit Diameter (m)</td>
<td>4.15</td>
<td>4.15</td>
</tr>
<tr>
<td>Nozzle Expansion Ratio</td>
<td>500</td>
<td>500</td>
</tr>
<tr>
<td>Specific Impulse-Vac (sec)</td>
<td>923</td>
<td>922.8</td>
</tr>
<tr>
<td>Pump Speed (rpm)</td>
<td>37,500</td>
<td>34,913</td>
</tr>
</tbody>
</table>

* Rocketdyne uses their Mark 25-type axial turbopump (4 stages); ELES-NTP used a single-stage centrifugal pump.

### INITIAL ENGINE CYCLE PARAMETER COMPARISON* - 75,000 lbf NTP ENGINE CASE -

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Rocketdyne</th>
<th>SAIC - ELES NTP</th>
</tr>
</thead>
<tbody>
<tr>
<td>Pump Flowrate (kg/s)</td>
<td>36.7</td>
<td>36.9</td>
</tr>
<tr>
<td>Pump Discharge Pres. (psia)</td>
<td>1544</td>
<td>1538.3</td>
</tr>
<tr>
<td>Turbine Flowrate, % Pump</td>
<td>50</td>
<td>50</td>
</tr>
<tr>
<td>Turbine Inlet Temp. (°K)</td>
<td>555.6</td>
<td>555.3</td>
</tr>
<tr>
<td>Turbine Inlet Pres. (psia)</td>
<td>1412</td>
<td>1416.8</td>
</tr>
<tr>
<td>Turbine Pressure Ratio</td>
<td>1.25</td>
<td>1.295</td>
</tr>
<tr>
<td>Reactor Inlet Pres. (psia)</td>
<td>1130</td>
<td>1255.4</td>
</tr>
<tr>
<td>Reactor Power, (MW)</td>
<td>1645</td>
<td>-</td>
</tr>
<tr>
<td>Reactor Core Flowrate (kg/s)</td>
<td>36.7</td>
<td>36.9</td>
</tr>
<tr>
<td>Nozzle Chamber Temp (°K)</td>
<td>2700</td>
<td>2700</td>
</tr>
<tr>
<td>Nozzle Chamber Pres. (psia)</td>
<td>1000</td>
<td>1000</td>
</tr>
<tr>
<td>Nozzle Exit Diameter (m)</td>
<td>4.15</td>
<td>4.15</td>
</tr>
<tr>
<td>Nozzle Expansion Ratio</td>
<td>500</td>
<td>500</td>
</tr>
<tr>
<td>Specific Impulse-Vac (sec)</td>
<td>923</td>
<td>922.8</td>
</tr>
<tr>
<td>Pump Speed (rpm)</td>
<td>37,500</td>
<td>34,913</td>
</tr>
</tbody>
</table>

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### INITIAL ENGINE CYCLE PARAMETER COMPARISON* - 75,000 lbf NTP ENGINE CASE -

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Rocketdyne</th>
<th>SAIC - ELES NTP</th>
</tr>
</thead>
<tbody>
<tr>
<td>Pump Flowrate (kg/s)</td>
<td>36.7</td>
<td>36.9</td>
</tr>
<tr>
<td>Pump Discharge Pres. (psia)</td>
<td>1544</td>
<td>1530.3</td>
</tr>
<tr>
<td>Turbine Flowrate, % Pump</td>
<td>50</td>
<td>50</td>
</tr>
<tr>
<td>Turbine Inlet Temp. (°K)</td>
<td>555.6</td>
<td>555.3</td>
</tr>
<tr>
<td>Turbine Inlet Pres. (psia)</td>
<td>1412</td>
<td>1416.8</td>
</tr>
<tr>
<td>Turbine Pressure Ratio</td>
<td>1.25</td>
<td>1.295</td>
</tr>
<tr>
<td>Reactor Inlet Pres. (psia)</td>
<td>1130</td>
<td>1255.4</td>
</tr>
<tr>
<td>Reactor Core Flowrate (kg/s)</td>
<td>1645</td>
<td>1650</td>
</tr>
<tr>
<td>Nozzle Chamber Temp (°K)</td>
<td>36.7</td>
<td>36.9</td>
</tr>
<tr>
<td>Nozzle Chamber Pres. (psia)</td>
<td>1000</td>
<td>1000</td>
</tr>
<tr>
<td>Nozzle Exit Diameter (m)</td>
<td>4.15</td>
<td>4.15</td>
</tr>
<tr>
<td>Nozzle Expansion Ratio</td>
<td>500</td>
<td>500</td>
</tr>
<tr>
<td>Specific Impulse-Vac (sec)</td>
<td>923</td>
<td>922.9</td>
</tr>
<tr>
<td>Pump Speed (rpm)</td>
<td>37,500</td>
<td>34,913</td>
</tr>
</tbody>
</table>

* Rocketdyne uses their Mark 25-type axial turbopump (4 stages); ELES-NTP used a single-stage centrifugal pump.

---

### CYCLE PARAMETER COMPARISON* - 75,000 lbf ENabler I, EXPANDER CYCLE -

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Rocketdyne</th>
<th>SAIC - ELES NTP</th>
<th>SAIC NESS</th>
</tr>
</thead>
<tbody>
<tr>
<td>Total Flowrate (kg/s)</td>
<td>36.7</td>
<td>36.9</td>
<td>37.27</td>
</tr>
<tr>
<td>Pump Discharge Pres. (psia)</td>
<td>1,544</td>
<td>1,538.3</td>
<td>2,298.3</td>
</tr>
<tr>
<td>Turbine Flowrate, % Pump</td>
<td>50</td>
<td>50</td>
<td>50</td>
</tr>
<tr>
<td>Turbine Inlet Temp. (°K)</td>
<td>555.6</td>
<td>555.3</td>
<td>622.3</td>
</tr>
<tr>
<td>Turbine Inlet Pres. (psia)</td>
<td>1,412</td>
<td>1,416.8</td>
<td>1,969.0</td>
</tr>
<tr>
<td>Turbine Pressure Ratio</td>
<td>1.25</td>
<td>1.295</td>
<td>1.739</td>
</tr>
<tr>
<td>Reactor Inlet Pres. (psia)</td>
<td>1,130</td>
<td>1,255.4</td>
<td>1,132.1</td>
</tr>
<tr>
<td>Reactor Power, (MW)</td>
<td>1,645</td>
<td>-</td>
<td>1,587</td>
</tr>
<tr>
<td>Reactor Core Flowrate (kg/s)</td>
<td>36.7</td>
<td>36.9</td>
<td>36.2</td>
</tr>
<tr>
<td>Nozzle Chamber Temp (°K)</td>
<td>2,700</td>
<td>2,700</td>
<td>2,700</td>
</tr>
<tr>
<td>Nozzle Chamber Pres. (psia)</td>
<td>1,000</td>
<td>1,000</td>
<td>1,000</td>
</tr>
<tr>
<td>Nozzle Exit Diameter (m)</td>
<td>4.15</td>
<td>4.15</td>
<td>4.22</td>
</tr>
<tr>
<td>Nozzle Expansion Ratio</td>
<td>500</td>
<td>500</td>
<td>500</td>
</tr>
<tr>
<td>Specific Impulse-Vac (sec)</td>
<td>923</td>
<td>922.8</td>
<td>912.9</td>
</tr>
<tr>
<td>Pump Speed (rpm)</td>
<td>37,300</td>
<td>34,913</td>
<td>40,383</td>
</tr>
</tbody>
</table>

* 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.
### CYCLE PARAMETER COMPARISON*
- 75,000 lbf ENBLER I, EXPANDER CYCLE -

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Rocketdyne</th>
<th>SAIC ELES-NTP</th>
<th>SAIC NESS</th>
</tr>
</thead>
<tbody>
<tr>
<td>Specific Impulse - Vac (sec)</td>
<td>923</td>
<td>922.8</td>
<td>912.9</td>
</tr>
<tr>
<td>Reactor (kg)</td>
<td>5,824</td>
<td>5,823</td>
<td>4,783</td>
</tr>
<tr>
<td>Internal Shield (kg)</td>
<td>—</td>
<td>1,523</td>
<td>1,108</td>
</tr>
<tr>
<td>Nozzle Assembly (kg)</td>
<td>440</td>
<td>421</td>
<td>535</td>
</tr>
<tr>
<td>Turbopump Assembly (kg)</td>
<td>304</td>
<td>104</td>
<td>221</td>
</tr>
<tr>
<td>Nonnuclear Support Hardware (kg)</td>
<td>1,815</td>
<td>1,264</td>
<td>1,493</td>
</tr>
</tbody>
</table>

* Rocketdyne used 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.

### CYCLE PARAMETER COMPARISON*
- 75,000 lbf ENBLER I, EXPANDER CYCLE -

<table>
<thead>
<tr>
<th>Wall Temperature (°F)</th>
<th>Barrier Temperature (°F)</th>
<th>Isp (Sec.)</th>
<th>Fuel Film Cooling Fraction</th>
</tr>
</thead>
<tbody>
<tr>
<td>1400</td>
<td>1630</td>
<td>912.9</td>
<td>0.03</td>
</tr>
<tr>
<td>1800</td>
<td>2106</td>
<td>915.9</td>
<td>0.03</td>
</tr>
<tr>
<td>2000</td>
<td>2429</td>
<td>917.5</td>
<td>0.02</td>
</tr>
<tr>
<td>2400</td>
<td>2892</td>
<td>919.4</td>
<td>0.02</td>
</tr>
<tr>
<td>2800</td>
<td>3418</td>
<td>921.2</td>
<td>0.02</td>
</tr>
<tr>
<td>3000</td>
<td>3651</td>
<td>921.9</td>
<td>0.02</td>
</tr>
<tr>
<td>3200</td>
<td>3864</td>
<td>922.4</td>
<td>0.02</td>
</tr>
</tbody>
</table>

* Cell Temperature = 485°F (270°C)
DESIGN CASE COMPARISON OBSERVATIONS

- NESS Design Exhibits 1% Lower Performance 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 H₂ Properties

ENGINE SYSTEM SCALING/COMPARISONS
EXTENSIVE ENABLER I AND II ENGINE SYSTEM DESIGN DATABASE HAS BEEN DEVELOPED

- Database Covers a Large Engine Design/Operating Parameters
  - Fuel Type/Chamber Temperature
  - Thrust level
  - Chamber Pressure
  - Nozzle Area Ratio

- Top-level Design Scaling Trends Produced for the ENABLER I Engine System
  - Little Design Trend Analysis Conducted to Date on the Enabler II Database

- All Engine Summary Design Data is Cataloged and is Available Through NASA Lewis

---

ENABLER ENGINE SYSTEM DESIGN TRADE SPACE ANALYZED

<table>
<thead>
<tr>
<th>Fuel Type/Chamber Temperature (oK)</th>
<th>Thrust (lbf)</th>
<th>Chamber Pressure (psia)</th>
<th>Nozzle Area Ratio</th>
</tr>
</thead>
<tbody>
<tr>
<td>Graphite/2500</td>
<td>15,000</td>
<td>500</td>
<td>200:1</td>
</tr>
<tr>
<td>Composite/2700</td>
<td>40,000</td>
<td>750</td>
<td>500:1</td>
</tr>
<tr>
<td>Carbide/3100</td>
<td>75,000</td>
<td>1,000</td>
<td></td>
</tr>
<tr>
<td></td>
<td>125,000</td>
<td>1,500</td>
<td></td>
</tr>
<tr>
<td></td>
<td>200,000</td>
<td>2,000</td>
<td></td>
</tr>
<tr>
<td></td>
<td>250,000</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>
ENABLER I DESIGN SCALING TRENDS

REACTOR AND INTERNAL SHIELD MASS SCALING
- Graphite Fuel -

CHAMBER PRESSURE
- 1000 psia
- 1000 psia

MASS (kg)

10,000
20,000
30,000
40,000

0
1000
2000
3000
4000
5000
6000
REACTOR POWER, MW

CHAMBER TEMPERATURE = 2500 °K

CHAMBER PRESSURE = 1000 psia
REACTOR AND INTERNAL SHIELD MASS SCALING
- Composite Fuel -

![Graphs showing mass scaling for composite fuel with reactor and internal shield. The graphs compare Chamber Pressure and Chamber Temperature effects on mass.](image)

CHAMBER TEMPERATURE = 2700 K
CHAMBER PRESSURE = 1000 psia

REACTOR AND INTERNAL SHIELD MASS SCALING
- Carbide Fuel -

![Graphs showing mass scaling for carbide fuel with reactor and internal shield. The graphs compare Chamber Pressure and Chamber Temperature effects on mass.](image)

CHAMBER TEMPERATURE = 3100 K
CHAMBER PRESSURE = 1000 psia

NP-TIM-92
NTP: System Concepts
REACTOR PRESSURE VESSEL DIMENSIONS AS A FUNCTION OF POWER AND CHAMBER PRESSURE

TYPICAL NTP ENGINE PERFORMANCE AS A FUNCTION OF CHAMBER PRESSURE, TEMPERATURE AND AREA RATIO
- 75,000 lbf Thrust -
NTP ENGINE WEIGHT AS A FUNCTION OF THRUST, CHAMBER PRESSURE, AND AREA RATIO
- Graphite Fuel, Chamber Temperature - 2500 K -

**Graph:**
- X-axis: THRUST (lb)
- Y-axis: TOTAL ENGINE WEIGHT (kg)
- Lines with different symbols correspond to different chamber pressures and area ratios.

**Key:**
- Symbol: M
- Chamber Pressure: 500 psi
- Area Ratio: 500

NTP ENGINE WEIGHT AS A FUNCTION OF THRUST, CHAMBER PRESSURE, AND AREA RATIO
- Composite Fuel, Chamber Temperature - 2700 K -

**Graph:**
- X-axis: THRUST (lb)
- Y-axis: TOTAL ENGINE WEIGHT (kg)
- Lines with different symbols correspond to different chamber pressures and area ratios.

**Key:**
- Symbol: M
- Chamber Pressure: 500 psi
- Area Ratio: 500

NTP ENGINE WEIGHT AS A FUNCTION OF THRUST, CHAMBER PRESSURE, AND AREA RATIO
- Carbide Fuel, Chamber Temperature - 3100 K -

NTP ENGINE SYSTEM AS A FUNCTION OF THRUST;
- Graphite and Composite Fuel Engines -

**NTP: System Concepts**
**NTP ENGINE SYSTEM AS A FUNCTION OF THRUST,**
- Carbide Fuel Engines -

![Graph showing total engine length as a function of thrust.]

**KEY**

<table>
<thead>
<tr>
<th>DIMENSION</th>
<th>SYMBOL</th>
<th>CHAMBER PRESSURE (psia)</th>
<th>AREA RATIO</th>
</tr>
</thead>
<tbody>
<tr>
<td>Length</td>
<td></td>
<td>500</td>
<td>500</td>
</tr>
<tr>
<td></td>
<td></td>
<td>1,000</td>
<td>100</td>
</tr>
<tr>
<td>Diameter</td>
<td></td>
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</tr>
<tr>
<td></td>
<td></td>
<td>1,000</td>
<td>200</td>
</tr>
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</table>

**NTP ENGINE SUBSYSTEM WEIGHT BREAKDOWN AS A FUNCTION OF THRUST,**
- Composite Fuel, $T_c = 2700$ K, $P_c - 1000$psia, $\varepsilon = 500:1$ -

![Graph showing weight breakdown as a function of thrust.]

- Reactor and Internal Shield
- Turbopump Assembly
- Nozzle
- Nonnuclear Support Hardware
**NTP ENGINE SUBSYSTEM PERCENT WEIGHT DISTRIBUTION FOR TWO THRUST LEVELS**

- Composite Fuel, $T_c = 2700 \, \text{K}$, $P_c = 1000 \, \text{psia}$, $\varepsilon = 500:1$

![Graph showing percent weight distribution for two thrust levels.](image)

**BASELINE NTP ENGINE DESCRIPTION**

- 75,000 LBF THRUST, EXPANDER CYCLE, COMPOSITE FUEL

\[ T_c = 2700 \, \text{K}, \quad P_c = 1000 \, \text{psia}, \quad \varepsilon = 500:1 \]

<table>
<thead>
<tr>
<th>Component</th>
<th>Features</th>
<th>Features</th>
<th>Number</th>
</tr>
</thead>
<tbody>
<tr>
<td>Reactor</td>
<td>- Reactor + Internal Shield Weight</td>
<td>- Fuel Type - Composite</td>
<td>1</td>
</tr>
<tr>
<td></td>
<td>- Case Material - Aluminum</td>
<td>- Reactor Diameter</td>
<td></td>
</tr>
<tr>
<td></td>
<td>- Reactor Length</td>
<td>- Fuel Mass Flow Rate</td>
<td></td>
</tr>
<tr>
<td></td>
<td>- Fuel Mass Flow Rate</td>
<td>- Reactor Exit/Nozzle Entrance</td>
<td></td>
</tr>
<tr>
<td></td>
<td>- Nozzle Weight</td>
<td>- Exit Chamber Pressure</td>
<td></td>
</tr>
<tr>
<td></td>
<td>- Nozzle Material</td>
<td>- Temperature</td>
<td></td>
</tr>
<tr>
<td></td>
<td>- Shifted Frenzen Wall Construction</td>
<td>- Conductor Tube Bundles Area Ratio of 150:1</td>
<td></td>
</tr>
<tr>
<td></td>
<td>- Copper to Area Ratio of 6:1</td>
<td>- Extension Material - Carbon</td>
<td></td>
</tr>
<tr>
<td></td>
<td>- Regeneratively Cooled by Propellant to an Area Ratio of 159:1</td>
<td>- Nozzle Length</td>
<td></td>
</tr>
<tr>
<td></td>
<td></td>
<td>- Throat Diameter</td>
<td></td>
</tr>
<tr>
<td></td>
<td></td>
<td>- Exit Diameter</td>
<td></td>
</tr>
<tr>
<td></td>
<td></td>
<td>- Area Ratio</td>
<td></td>
</tr>
<tr>
<td></td>
<td></td>
<td>- Delivered Vacuum to lbp</td>
<td></td>
</tr>
<tr>
<td></td>
<td></td>
<td>- Delivered Thrust</td>
<td></td>
</tr>
<tr>
<td>Nozzle</td>
<td>6576 kg (14,500 lbm)</td>
<td>442 kg (975 lbm)</td>
<td>1</td>
</tr>
<tr>
<td></td>
<td>1.32 m (52 in)</td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>2.59 m (102 in)</td>
<td>36.9 kgf (84 lbm)</td>
<td></td>
</tr>
<tr>
<td></td>
<td>6935 kPa (1000 psia)</td>
<td>2.7% 4 K (4,860°F)</td>
<td></td>
</tr>
<tr>
<td>Main Pump Turbine</td>
<td>623 cm (242 in)</td>
<td>19.4 cm (7.6 in)</td>
<td>2</td>
</tr>
<tr>
<td></td>
<td>415.6 cm (163.7 in)</td>
<td>500 lb</td>
<td></td>
</tr>
<tr>
<td></td>
<td>6043 N x seckg (623 sec)</td>
<td>233.8 kN (52,000 lb)</td>
<td></td>
</tr>
<tr>
<td></td>
<td>30,000 rpm</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

*Total Component Weight - Typical*
BASELINE NTP ENGINE DESCRIPTION*
- 75,000 LBF THRUST, EXPANDER CYCLE, COMPOSITE FUEL.

\[ T_c = 2700 \, K, \, P_c = 1000 \, \text{psia}, \, \varepsilon = 500:1 \]  

(Cont.)

<table>
<thead>
<tr>
<th>Component</th>
<th>Features</th>
<th>Number</th>
</tr>
</thead>
<tbody>
<tr>
<td>Main Fuel Pump</td>
<td></td>
<td></td>
</tr>
<tr>
<td>- Main Pump Weight</td>
<td>44.6 kg (98.4 lbs)</td>
<td>2</td>
</tr>
<tr>
<td>- Material - Inconel</td>
<td></td>
<td></td>
</tr>
<tr>
<td>- Single Stage Centrifugal Pump</td>
<td>10,260 kPa (1486 psi)</td>
<td></td>
</tr>
<tr>
<td>- Pump Speed</td>
<td>29,226 rpm</td>
<td></td>
</tr>
<tr>
<td>- Pump Diameter</td>
<td>25.7 cm (10.1 in)</td>
<td></td>
</tr>
<tr>
<td>- Pump Efficiency</td>
<td>0.715</td>
<td></td>
</tr>
<tr>
<td>- Pump Horsepower</td>
<td>8143 HP</td>
<td></td>
</tr>
</tbody>
</table>

| Misc. Hardware Weights     |                                 |        |
| - Thrust Mount             | 737 kg (1624 lbs)               | 1      |
| - Thrust Support Hardware  | 637 kg (1423 lbs)               | 1      |
| - Engine Lines             | 61.9 kg (139.7 lbs)             | 2      |
| - Main Valve               | 182.8 kg (400.9 lbs)            | 4      |
| - TPA Ignition             | 15.7 kg (34.7 lbs)              | 1      |
| - Control System           | 34.9 kg (78.0 lbs)              | 1      |

Subtotal
- Total Nonstructural Weight (TPA + Misc. Hardware + Nozzle): 2106 kg (4643 lbs)
- Margin (5%): 44.9 kg (99.9 lbs)
- Total Engine System: 8810 kg (19,440 lbs)

* Total Component Weight - Typical

BASELINE NTP ENGINE CYCLE, OPERATING PARAMETERS
- 75,000 lbf Thrust, Composite Fuel,

\[ T_c = 2700 \, K, \, P_c = 1000 \, \text{psia}, \, \varepsilon = 500:1 \]  

<table>
<thead>
<tr>
<th>STATION</th>
<th>PRESSURE (psi)</th>
<th>TEMP (°C)</th>
<th>m (kg/s)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>90</td>
<td>22.2</td>
<td>10.4</td>
</tr>
<tr>
<td>2</td>
<td>1538</td>
<td>40.3</td>
<td>8.3</td>
</tr>
<tr>
<td>3</td>
<td>1487</td>
<td>40.3</td>
<td>8.3</td>
</tr>
<tr>
<td>4</td>
<td>1417</td>
<td>555.5</td>
<td>58.9</td>
</tr>
<tr>
<td>5</td>
<td>1404</td>
<td>555.5</td>
<td>58.9</td>
</tr>
<tr>
<td>6</td>
<td>1255</td>
<td>533.6</td>
<td>26.9</td>
</tr>
<tr>
<td>7</td>
<td>1000</td>
<td>2700.0</td>
<td>38.9</td>
</tr>
</tbody>
</table>
### Baseline NTP Engine Cycle, Operating Parameters

- 75,000 lbf Thrust, Expander Cycle, Composite Fuel,
  \( T_c = 2700 \text{ K}, P_c = 1000 \text{ psia}, \epsilon = 500:1 \)

#### Engine Summary

<table>
<thead>
<tr>
<th>Component</th>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Chamber Cycle</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Expander Cycle</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Turbojet Engine</td>
<td></td>
<td></td>
</tr>
<tr>
<td>NTP Engine</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

#### NTP Engine Thrust-To-Weight Ratio as a Function of Thrust

- \( \epsilon = 500:1 \)

#### Thrust/Weight vs. Thrust

![Graph showing thrust-to-weight ratio vs. thrust for different fuel types: composite fuel and non-composite fuel.](image)

#### Chamber Temperature

- 2700° K

#### Composite Fuel

- Chamber Temperature = 2700° K

#### NTP: System Concepts

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CONCLUDING REMARKS

- A Near-Term (ENBLER I and II) NTP Solid-Core System Database has Been Established
  - Based on the Well Documented/ Anchored SAIC NNESS Design Program
  - Incorporates Westinghouse’s SOA Reactor System Design Correlations
    - Database is Organized, Documented and is Available Through NASA Lewis

- Future Recommendations
  - Perform a Comparative Assessment of the Database
    - Past Engineering Data Generated
    - Technology Sensitivity Studies
  - Initiate a Similar Study Activity With Engine Systems Using Different Reactor Types
Clustered Engine Study Team

Kyle Shepard  Study Manager
Paul Sager  Propulsion
Sid Kusunoki  Vehicle Design
John Porter  Systems Analysis
Al Campion  Mass Properties
Gunnar Mouritzan  Propulsion
Will Glunt  Trajectory/Performance
George Vegter  Guidance and Control
Rob Koontz  Guidance and Control

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Mission Description ......................................................................... 6
Reference Vehicle and Engine .......................................................... 8
Propulsion System Preliminary Design ............................................... 11
Run Tank Based Systems .................................................................. 12
Boost Pump Based Systems ............................................................... 21
Systems Analysis .............................................................................. 31
Summary and Conclusions ................................................................. 43
The presentation will cover several topics which together encompass this preliminary assessment of nuclear thermal rocket engine clustering. The study objectives, schedule, flow and groundrules are covered. This is followed by the NASA groundrule mission and our interpretation of the associated operational scenario. The NASA reference vehicle is illustrated, then we zoom in on the four propulsion system options examined in this study. Each propulsion system's preliminary design, fluid systems, operating characteristics, thrust structure, dimensions and mass properties are detailed as well as the associated key propulsion system/vehicle interfaces. A brief series of systems analysis will also be covered including: thrust vector control requirements, engine out possibilities, propulsion system failure modes, surviving system requirements and technology requirements. The presentation concludes with an assessment of vehicle propulsion system impacts due to the lessons learned in this study.
CLUSTERED NTR STUDY OBJECTIVE

"To develop a top level assessment of the feasibility of clustering Nuclear Thermal Rocket engines."

A NASA reference vehicle and mission scenario were given.

The approach then was to develop four propulsion system designs that could be used as reference configurations for future engineering assessments.

The Study addresses:
- Two and three engine propulsion system designs with either boost pumps or run tanks for engine start up
- Thrust Vector Control (TVC) Requirements
- Engine out possibilities
- Propulsion system Failure modes
- Technology requirements

The objective of the study was to develop propulsion system designs that could be integrated with the provided reference vehicle and fly the provided reference mission. Four propulsion system options were developed using two and three engines with either boost pumps or run tanks for engine start up. Our intent was to develop propulsion systems with a cluster of NTR engines that could be used as reference configurations for future systems optimization. In doing this we considered the following system issues: TVC requirements, Engine out possibilities, propulsion system failure modes and technology development requirements.
the study was a five week effort beginning the first week of December 1991, with a christmas holiday in the middle and ending on Jan. 15, 1992. The propulsion system preliminary designs and systems analysis were primarily completed in the first three weeks of the study. The remainder was used for analysis and design iterations as well as presentation preparation.
The study was initiated with a series of NASA LeRC provided groundrules and requirements. These were provided in appendix form and served as the point of departure for the NTP vehicle, mission, engine and propulsion system.

The primary study activity consisted of developing preliminary propulsion system designs for two and three engine propulsion systems with either run tanks or boost pumps for engine start up. As these propulsion systems were developed, several design issues arose. Design issues were addressed at LeRC-GDSS weekly telecons where issues were raised, resolved and the resulting decision(s) were applied to the design work. This iteration process continued throughout the study.

Upon completion of the design phase, mass properties were developed and a series of systems analysis took place. The systems work concentrated on issues relating to the engine out scenario. This analysis allowed us to quantify thrust vector control, reactor burn time and technology requirements as well as assess impacts to the vehicle such as mission performance penalties and failure modes.
The reference trajectory is a short opposition type trajectory. It is developed around the 2014 mission opportunity and includes a Venus swingby on Earth return. The outbound leg lasts 150 days and includes three perigee burns for Earth departure. Upon arrival at Mars, the crew performs a surface science mission lasting 90 days. The Earth return leg lasts 310 days and includes a single burn for Mars departure. Note that there is a robust Mars powered flyby abort mode available should a problem occur after Trans Mars Injection (TMI) or before Mars orbit capture (MOC).

TMI, MOC and Trans Earth Injection (TEI) burns were considered in our engine out/mission performance analysis. We consider cases for either 1 or 2 engines out for the boost pump and run tank based propulsion system options.
REFERENCE MARS TRANSFER SYSTEM

- Vehicle assessed from core tank aft

- Primary Vehicle Modifications:
  - Core Propellant Tank
    (For Boost Pump Config's Only)
  - Thrust Structures
  - Run Tanks
  - Reactor Shields
  - Reactors
  - Nozzle Extension

Our analysis concentrates on the vehicle elements from the core tank and aft. The core propellant tank is resized for the boost pump case. Sizing is based on a combination of a fully integrated propulsion system launch requirement on either STS or Titan IV. This scenario enables a more traditional intertank adapters/thrust structure. In moving from the two to three engine case, the run tanks are re-sized to take advantage of a reduced requirement for propellant volume at start. The reactor shields are modified to remove the center shield section and include a side shields. This is done to reduce shielding mass. The reactors themselves are also reduced in size due to the reduced thrust requirement on the three engine case. Lastly, an engine without a nozzle extension was groundruled.
The reference engine is a NERVA "full flow" concept developed by Alseimer of Aerojet Nuclear Systems Company, circa 1971. It is an engine typical of that era. For the two engine cases, the reference 75 klbf engine was used, the three engine cases utilized a scaled down version of this engine sized at 50 klbf.
PROPULSION SYSTEM PRELIMINARY DESIGN

RUN TANK BASED SYSTEMS
This configuration utilizes two 75 klf thrust NERVA nuclear thermal rocket engines with separate run tanks. The run tanks are used to minimize pressurization gas requirements for engine start. Gaseous helium for pressurizing the run tanks is supplied by high pressure bottles located above the run tanks. Once engine start is achieved, hydrogen gas is bled from the engines and used to pressurize the core tank. After the core tank is sufficiently pressurized, propellant from the core tank is fed through the run tanks to continue to feed the engines. As the end of each burn, the run tanks may be filled to capacity to repeat the procedure for the next engine start.

The run tank, engine, and thrust structure combine to form the propulsion module. The propulsion module is launched separately from the rest of the vehicle and is coupled to the core tank on orbit. Fluid system and electrical disconnects and structural latches are provided to allow for on orbit coupling of the propulsion module to the core tank.

An aluminum-lithium (Al-Li) tubular intertank truss structure transfers the thrust from the propulsion module to the core tank. Lateral Al-Li tubular struts stiffen the structure for gimbaled thrust vector loads at the end of the run tank aft skirt. Symmetrical Al-Li tubular truss thrust structures are used to transfer the engine thrust loads to the run tank aft skirts.

The run tanks are spaced to allow the maximum distance between engines possible without exceeding the 10 meter diameter limit. This provides a distance of 6 meters between the engine centers which is more than the 5 meter minimum required to minimize neutronic coupling impacts. This spacing allows one engine to gimbal inboard a maximum of 9 degrees with the other engine in the neutral position. The overall length of this configuration from star of intertank adapter to engine exit is 23.5 meters.
### PROPULSION SYSTEM R-2

**Mass Properties**

<table>
<thead>
<tr>
<th>ITEM</th>
<th>WEIGHT IN POUNDS</th>
</tr>
</thead>
<tbody>
<tr>
<td>CLUSTER TANKED WEIGHT</td>
<td>82490</td>
</tr>
<tr>
<td>- Structure</td>
<td></td>
</tr>
<tr>
<td>- Core Tank Lower Support Structure</td>
<td>9620</td>
</tr>
<tr>
<td>- Run Tank Upper Support Structures (2)</td>
<td>4930</td>
</tr>
<tr>
<td>- Run Tanks (2) (Includes insulation)</td>
<td>690</td>
</tr>
<tr>
<td>- Run Tank Lower Thrust Structure (2)</td>
<td>2959</td>
</tr>
<tr>
<td>- Feed System</td>
<td></td>
</tr>
<tr>
<td>- Feedlines</td>
<td>1390</td>
</tr>
<tr>
<td>- Valves</td>
<td></td>
</tr>
<tr>
<td>- Disconnects</td>
<td>82490</td>
</tr>
<tr>
<td>- Gimbal Joints</td>
<td>9620</td>
</tr>
<tr>
<td>- Line Insulation</td>
<td>300</td>
</tr>
<tr>
<td>- Pressurization System</td>
<td></td>
</tr>
<tr>
<td>- Helium Bottles</td>
<td>31290</td>
</tr>
<tr>
<td>- Supports</td>
<td>269</td>
</tr>
<tr>
<td>- Lines</td>
<td>80</td>
</tr>
<tr>
<td>- Engine Assemblies (2)</td>
<td></td>
</tr>
<tr>
<td>- Engines</td>
<td>51320</td>
</tr>
<tr>
<td>- External Shields</td>
<td>17200</td>
</tr>
<tr>
<td>- Contingency (10%)</td>
<td>360</td>
</tr>
<tr>
<td>- Helium</td>
<td></td>
</tr>
</tbody>
</table>
This configuration utilizes three 50 kibl thrust nuclear thermal rocket engines with separate run tanks. The run tanks are used to minimize pressurization gas requirements for engine start. Gaseous helium for pressurizing the run tanks is supplied by high pressure bottles located above the run tanks. Once engine start is achieved, hydrogen gas is bled from the engines and used to pressurize the core tank. After the core tank is sufficiently pressurized, propellant from the core tank is fed through the run tanks to continue to feed the engines. At the end of each burn, the run tanks may be filled to capacity to repeat the procedure for the next engine start.

The run tank, engine, and thrust structure combine to form the propulsion module. The propulsion module is launched separately from the rest of the vehicle and is coupled to the core tank on orbit. Fluid system and electrical disconnects and structural latches are provided to allow for on orbit coupling of the propulsion module to the core tank.

An aluminum-lithium tubular intertank truss structure transfers the thrust from the propulsion modules to the core tank. Lateral Al-Li tubular struts stiffen the structure for gimbaled thrust vector loads at the end of the run tank aft skirt. Symmetrical Al-Li tubular truss thrust structures are used to transfer the engine thrust loads to the run tank aft skirts.

The run tanks are spaced to allow the maximum distance between engines possible without exceeding the 10 meter diameter limit. This provides a distance of 5.2 meters between the engine centers which is more than the 5 meter minimum required to minimize neutronic coupling impacts. This spacing allows one engine to gimbal inboard a maximum of 8 degrees with the other engine in the neutral position. The overall length of this configuration from start of intertank adapter to engine exit is 21.5 meters.
**PROPULSION SYSTEM R-3**

**Mass Properties**

<table>
<thead>
<tr>
<th>ITEM</th>
<th>WEIGHT IN POUNDS</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>CLUSTER TANKED WEIGHT</strong></td>
<td>99000</td>
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<tr>
<td>Structure</td>
<td>11760</td>
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<tr>
<td>- Core Tank Lower Support Structure</td>
<td>4650</td>
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<tr>
<td>- Run Tank Upper Support Structures (3)</td>
<td>970</td>
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<tr>
<td>- Run Tanks (3) (includes insulation)</td>
<td>4440</td>
</tr>
<tr>
<td>- Run Tank Lower Thrust Structure (3)</td>
<td>1500</td>
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<tr>
<td>Feed System</td>
<td>1560</td>
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<td>- Feedlines</td>
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<tr>
<td>- Valves</td>
<td>260</td>
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<td>- Disconnects</td>
<td>290</td>
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<tr>
<td>- Gimbal Joints</td>
<td>230</td>
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<tr>
<td>- Line Insulation</td>
<td>90</td>
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<tr>
<td>Pressurization System</td>
<td>2440</td>
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<tr>
<td>- Helium Bottles</td>
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<td>- Supports</td>
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<td>- Lines</td>
<td>120</td>
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<tr>
<td>Engine Assemblies (3)</td>
<td>55780</td>
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<tr>
<td>- Engines</td>
<td>35940</td>
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<tr>
<td>- External Shells</td>
<td>19840</td>
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<tr>
<td>Contingency (10%)</td>
<td>1130</td>
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<tr>
<td>Helium</td>
<td>530</td>
</tr>
<tr>
<td>Hydrogen Capacity</td>
<td>25800</td>
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</tbody>
</table>

NASA LeRC NPO/ASAO reference weights were used for the engine, shield and run tank assemblies. Other weight estimates were developed by General Dynamics Space Systems (GDSS) and are estimated from existing Centaur system weights and/or NLS predesign weights. All structural weights were calculated using Aluminum Lithium (Al 2090, ρ = 0.092 lb/in³). The trussed adapter utilized 24 truss elements per engine. Intertank and run tank adapters were assumed to be semi-monoocoque construction. Machined isogrid adapters could be significantly lighter if no frequency/stiffness problems exist. The intertank adapter will likely have many cutouts for fuel lines and/or access. An additional 25% was added to the basic structural weight in order to account for additional localized structure needed around cutouts. An additional 10% contingency factor was added to the GDSS developed weights. The NASA LeRC NPO/ASAO weights were supplied with contingency included.
All the structural components were based on aluminum-lithium construction. Semi-monocoque cylindrical tank skirts were used to be conservative until more stress analysis can be performed. Tubular truss structures were used for part of the engine thrust structure and intertank adapter.

The intertank adapter between the core tank and the run tank consists of the core tank aft skirt, the truss structure, and the run tank forward skirt. The sizing of these structures was based on either launch or flight loads. The core tank aft skirt and the truss structure would be launched with the core tank. Launch loads from the fully loaded core tank would be transferred through the aft skirt and into a payload adapter bypassing the truss structure. Launch accelerations were assumed to be 3.0g axial and 1.5g lateral for a 300 kibh type heavy lift launch vehicle. The truss structure would only see engine thrust loads once the vehicle was fully assembled. The run tank forward skirt would be launched with the propulsion module on a Titan IV type launch vehicle. The propulsion module would be launched empty and inverted such that the launch loads would be taken through the run tank forward skirt and into the payload adapter. Launch accelerations for a Titan IV type launch vehicle were assumed to be 2.3g axial and 1.5g lateral.

The thrust structure consists of the run tank aft skirt and truss structure. Both of these structures would also need to withstand the launch loads from a Titan IV type vehicle due to the engine mass since they are all part of the propulsion module.
A propellant feed system with a run tank in addition to the core tank makes it possible to start the propulsion system without pressurizing the core tank first. The smaller volume run tank is pressurized for engine start up. When steady state operation of the engines is established, the core tank is pressurized by autogenous pressurization using hydrogen gas from the turbine outlet. The run tank is then vented enough to allow the tank to be filled with pressurized propellant from the core tank. Two independent main turbopumps were chosen for each engine to guarantee safe engine operation in case of failures in one pump system. The pumps are powered by preheated gaseous hydrogen in an expander cycle arrangement for simplicity and high reliability.

The propellant valves are generally electromechanical. However, due to the large size main propellant feed lines the tank shut-off valves are pneumatically controlled for fast shut-off. The pilot control valves for the pneumatic operated valves are solenoid operated valves. Pyrotechnic valves in the pneumatic system guarantees that the propellant feed system can not be inadvertently opened before the vehicle is ready for operation.

Helium is used for run tank pressurization but an alternative gaseous Hydrogen system could be used with a single 3.5 ft diameter low pressure (300 psia) gas storage bottle that can be continually recharged with hydrogen by feeding liquid Hydrogen from the tank through an electric heater.

Each engine in a two or three engine configuration has its own independent propellant feed system, so that with one engine system out, the mission can be completed with the remaining engine(s).
R2, R3 PROPULSION SYSTEMS
Operating Characteristics

<table>
<thead>
<tr>
<th></th>
<th>R-2</th>
<th>R-3</th>
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<tbody>
<tr>
<td><strong>START-UP</strong></td>
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<tr>
<td>THRUST, lbf</td>
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<td>GIMBAL RATE, degrees/s</td>
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<tr>
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<td>GIMBAL ACCELERATION, degrees/s</td>
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<td>10</td>
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<tr>
<td><strong>SHUTDOWN</strong></td>
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<td>THRUST, lbf</td>
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<td>150,000 - 380</td>
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<td>26 - 40</td>
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<td>RUN TANK PRESS., psia</td>
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<tr>
<td>GIMBAL DISPLACEMENT, degrees</td>
<td>10</td>
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<tr>
<td>GIMBAL RATE, degrees/s</td>
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<tr>
<td>GIMBAL ACCELERATION, degrees/s</td>
<td>20</td>
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</tbody>
</table>

Propulsion system operating characteristics were established for the run tank designs for start-up, steady state, and shutdown conditions. For both the 2-engine and 3-engine designs, the total thrust ramps up from 0 to 150,000 pounds in about 1 minute. Assuming a propellant condition of saturation at 16 psia, about 10 psi pressurization is required to provide NPSH to the engine turbopumps and to account for line entrance losses, line losses, and nuclear radiation heating of the propellant during line transit. The gimbal angular displacements, slew rates, and accelerations were estimated by adding 2 degrees displacement to the gimbal requirements determined for engine-out events, assuming conditions at the end of start-up.

For steady state, it was assumed that the total thrust could vary from full thrust with all engines operating to an engine-out condition with the active engine(s) throttled to 75 percent thrust. The specific impulse and maximum burn times were assumed to be unchanged from current specifications. For the planned mission, it was estimated that the propellant vapor pressure would rise approximately 14 psi due to nuclear radiation heating of the propellant. The gimbal requirements are the same as at the end of start-up.

The shutdown thrust reduces to a minimum of 190 pounds for the 75,000 lbf NERVA engines. It was estimated that this minimum requirement would scale linearly for the 50,000 lbf engine. The cooldown pulse rate will vary from steady flow to the frequency required at that condition at the point cooling can be terminated (0.0001). The tank pressures and gimballing requirements at the start of shutdown would be the same as for steady state.
The NTR engine interacts with the vehicle and its support systems through the: engine controllers, engine sensors, control feedback loops, vehicle health management systems, thrust structure etc. Each of these interface elements is affected in both design and operation by the propulsion system configuration. The thrust structure for example, is sensitive to propulsion system configuration (Run Tanks vs Boost Pumps, etc.) which affect its design on the ground (for access during integration assembly and checkout) and on orbit (depending on assembly philosophy, assembled vs docked vs modular propulsion system design). The other major consideration in the system interface impacts unique to NTR engine based propulsion systems is the radiation field. The propellant feedlines for example are affected by engine in the traditional manner, but with NTR one must also account for operation in an intense radiation environment (propellant heating in lines). Each of the primary interface elements are subject to optimization to minimize mass while maximizing safety and reliability. These systems together have a significant impact on the vehicles performance and design approach and should be integrated into any propulsion system design effort.
This configuration utilizes two 75 kib thrust nuclear thermal rocket engines hard-coupled to the core tank. Because of the large ullage volume in the core tank upon restart on some missions, an inordinate amount of pressurization gas would be required to supply the turbopump NPSH for engine restart. Accordingly, the propellant in the core tank is allowed to remain at saturated conditions and boost pumps are used to supply the pressure differential required to provide the NPSH and accommodate the entrance and line losses, as well as the nuclear radiation heating of the propellant as it flows through the line. The boost pumps are powered by turbine drives which run on pressurized gas. Once engine start is achieved, hydrogen gas is bled from the engines and used to run the boost pumps. At the end of each burn the pressurization bottles will be refilled to repeat the procedure for the next engine start.

The core tank, engines, and thrust structure form one unit and are launched together. Due to the fact that this is one unit, the core tank will be shortened by approximately 11.7 meters to accommodate the engines. Extendable nozzles would minimize the launch vehicle shroud volume losses for this configuration. The engine spacing used for this configuration was the same as determined for the run tank version. This provides a distance of 6 meters between the engine centers which is more than the 5 meter minimum required to minimize neutronic coupling impacts. This spacing allows one engine to gimbal inboard a maximum of 9 degrees with the other engine in the neutral position. The overall length of this configuration from start of thrust structure to engine exit is 17.2 meters.
### Mass Properties

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<th>ITEM</th>
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<tr>
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NASA LeRC NPO/ASAO reference weights were used for the engine and shield assemblies. Other weight estimates were developed by General Dynamics Space Systems (GDSS) and are estimated from existing Centaur system weights and/or NLS presubmission weights. All structural weights were calculated using Aluminum Lithium (Al 2090, p = 0.092 lb/in^3). Inertank adapters were assumed to be semi-monocoque construction. Machined isogrid adapters could be significantly lighter if no frequency/stiffness problems exist. The intertank adapter will likely have many cutouts for fuel lines and/or access. An additional 25% was added to the basic structural weight in order to account for additional localized structure needed around cutouts. An additional 10% contingency factor was added to the GDSS developed weights. The NASA LeRC NPO/ASAO weights were supplied with contingency included.
This configuration utilizes three 50 klb thrust nuclear thermal rocket engines hard-coupled to the core tank. Because of the large ullage volume in the core tank upon restart on some missions, an inordinate amount of pressurization gas would be required to supply the turbopump NPSH for engine restart. Accordingly, the propellant in the core tank is allowed to remain at saturated conditions and boost pumps are used to supply the pressure differential required to provide the NPSH and accomodate the entrance and line losses, as well as the nuclear radiation heating of the propellant as it flows through the line. The boost pumps are powered by turbine drives which run on pressurized gas. Once engine start is achieved, hydrogen gas is bled from the engines and used to run the boost pumps. At the end of each burn the pressurization bottles will be refilled to repeat the procedure for the next engine start.

The core tank, engines, and thrust structure form one unit and are launched together. Due to the fact that this is one unit, the core tank will be shortened by approximately 9.9 meters to accomodate the engines. The engine spacing used for this configuration was the same as determined for the run tank version. This provides a distance of 5.2 meters between the engine centers which is more than the 5 meters required to minimize neutronic coupling impacts. This spacing allows one engine to gimbal inboard a maximum of 8 degrees with the other engine in the neutral position. The overall length of this configuration from start of thrust structure to engine exit is 15.4 meters.
PROPULSION SYSTEM B-3
Mass Properties

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<td>- External Shields</td>
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NASA LeRC NPO/ASAO reference weights were used for the engine and shield assemblies. Other weight estimates were developed by General Dynamics Space Systems (GDSS) and are estimated from existing Centaur system weights and/or NLS predesign weights. All structural weights were calculated using Aluminum Lithium (Al 2090, ρ = 0.092 lb/in^3). Intertank adapters were assumed to be semi-monocoque construction. Machined isogrid adapters could be significantly lighter if no frequency/stiffness problems exist. The intertank adapter will likely have many cutouts for fuel lines and/or access. An additional 25% was added to the basic structural weight in order to account for additional localized structure needed around cutouts. An additional 10% contingency factor was added to the GDSS developed weights. The NASA LeRC NPO/ASAO weights were supplied with contingency included.
All the structural components were based on aluminum-lithium construction. To be conservative semi-monocoque structures were used for the core tank aft skirt and conical adapter until more stress analysis can be performed.

The thrust structure consists of the core tank aft skirt and the conical adapter. The sizing of these structures was based on launch loads. For the hard-coupled propulsion system design, the core tank aft skirt, conical adapter, and engines would be launched fully assembled to the core tank. Launch loads from the fully loaded core tank would be transferred through the aft skirt and into a payload adapter bypassing the conical adapter. The conical adapter would have to transfer launch loads from the engine mass into the payload adapter. Launch accelerations were assumed to be 3.0g axial and 1.5g lateral for a 300 kib type heavy lift launch vehicle.
A propellant feed system with boost pumps guarantees a sufficiently high net positive suction head at the propellant inlet to the main engine pumps without tank pressurization. The boost pumps are started by gaseous hydrogen from a storage bottle to initiate rotation of the boost pump turbine drive. Once the engine turbopump head is established, gaseous hydrogen is fed back from the engine cooling jacket outlet to bootstrap the propulsion system propellant head. Two independent main turbopumps were chosen for each engine to guarantee safe engine operation in case of failures in one pump system. The pumps are powered by preheated gaseous hydrogen in an expander cycle arrangement for simplicity and high reliability.

The propellant valves are generally electromechanical. Due to the large size propellant lines and requirements for fast shut-off the main tank shut-off valve is pneumatically controlled.

Gaseous hydrogen is used for the pneumatic control because it can operate with a single 3.5 ft diameter low pressure (300 psia) gas storage bottle that can be continually recharged with hydrogen by feeding liquid hydrogen from the tank through an electric heater or gaseous hydrogen from the engine during engine operation. This bottle can also be used for restart of the boost pump.

Each engine in a two or three engine configuration as shown has its own independent propellant feed system; however, other options are possible.
### PROPULSION SYSTEM OPERATING CHARACTERISTICS - B-2,3

#### START-UP

<table>
<thead>
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#### STEADY STATE

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#### SHUTDOWN

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<td>GIMBAL ACCELERATION, degrees/s(^2)</td>
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<tr>
<td>BOOST PUMP DELTA P, psid</td>
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The propulsion system operating characteristics were established for start-up, steady state, and shutdown conditions. For both the 2-engine and 3-engine designs, the total thrust ramps from 0 to 150,000 pounds in about 1 minute. Assuming a propellant condition of saturation at 16 psia, the boost pump provides an additional 10 psid to provide NPSH to the engine turbopumps and to account for line entrance losses, line losses, and nuclear radiation heating of the propellant during line transit. The gimbal angular displacements, slew rates, and accelerations were estimated by adding 2 degrees displacement to the gimbal requirements determined for the run tank designs with engine-out events and an additional 2 degrees to account for the reduced displacement of the engines from the c.g. for the boost pump vehicle designs.

For steady state, it was assumed that the total thrust could vary from full thrust with all engines operating to an engine-out condition with the active engine(s) throttled to 75 percent thrust. The specific impulse and maximum burn times were assumed to be unchanged from current specifications. For the planned mission, it was estimated that the shield could be designed to allow the propellant vapor pressure to rise approximately 24 psi due to nuclear radiation heating of the propellant. The gimbal requirements are the same as at the end of start-up.

The shutdown thrust reduces to a minimum of 190 pounds for the 75,000 lbf NERVA engines. It was estimated that this minimum requirement would scale linearly for the 50,000 lbf engine. The cooldown pulse rate will vary from steady flow to the frequency required at that condition at the point cooling can be terminated (0.0001). The tank pressures and gimballing requirements at the start of shutdown would be the same as for steady state.
The NTR engine interacts with the vehicle and its support systems through the: boost pump turbine drive, boost pump turbine return, engine controllers, engine sensors, control feedback loops, vehicle health management systems, thrust structure etc. Each of these interface elements is affected in both design and operation by the propulsion system configuration. The thrust structure for example, is sensitive to propulsion system configuration (Run Tanks vs Boost Pumps, etc.) which affect its design on the ground (for access during integration assembly and checkout) and on orbit (depending on assembly philosophy, assembled vs docked vs modular propulsion system design). The other major consideration in the system interface impacts unique to NTR engine based propulsion systems is the radiation field. The propellant feedlines for example are affected by engine in the traditional manner, but with NTR one must also account for operation in an intense radiation environment (propellant heating in lines). Each of the primary interface elements are subject to optimization to minimize mass while maximizing safety and reliability. These systems together have a significant impact on the vehicle's performance and design approach and should be integrated into any propulsion system design effort.
A qualitative assessment was made to compare the 4 different engine cluster configurations studied. Some performance advantage can be attributed to the 2-engine cluster designs because the higher thrust engines (75,000 lb) have a somewhat better thrust-to-weight ratio. Also, the boost pump design should be somewhat less weight than the run tank design because less structure is required.

The 3-engine installations provide significantly higher mission reliability, since it would be possible to continue a mission even after the failure of one engine. The mission would have to be aborted if only 1 engine survives, as would be the case for a 2-engine installation.

The mission operations are simplified with a boost pump, since the engine can be started at any time, whereas the run tanks have to be topped-off before restarting the vehicle with a run tank. Also, the complication of changing over to a core tank supply after start is eliminated with the boost pump design.

The run tank design has somewhat more complicated on-orbit coupling operations due to the necessity of coupling the individual propulsion modules to the aft core tank. With the boost tank design, the engines are hard connected to the aft core tank.

Since the 2-engine cluster designs require higher thrust than the 3-engine designs, the development costs will be higher and the development schedule somewhat longer. A major consideration is the cost of the ground test facilities, which is a function of engine size.

While a weighted scoring was not attempted, it appears that the boost pump design is a somewhat better choice than the run tank design. A 3-engine cluster design appears to be a much better design choice than a 2-engine.

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<th>MISSION OPERATIONS</th>
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<td></td>
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<tr>
<td>2 ENGINE CLUSTER</td>
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<tr>
<td>3 ENGINE CLUSTER</td>
<td>+</td>
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<td>+</td>
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<td>+</td>
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Identify Engine-Out Impacts on Propulsion Module and Vehicle

- Define/Modify Propulsion System Requirements to Accommodate Engine-Out (RID's)

- Identify Propulsion Module Technology Requirements

This section examines the impacts of engine out, documents requirements on the engine and vehicle to survive the event, and identifies new propulsion module technology needs.

Our analyses defines mission phases where failures could occur. Thrust vector control requirements to correct for an engine-out condition in both two and three engine configurations are given. Mission performance penalties, in terms of ΔV and additional propellant required to abort, is assessed. Propulsion system failure causes, symptoms, and remedies are examined. The requirements on the surviving propulsion module are defined, and vehicle impacts are discussed. Propulsion module technology requirements are defined, and suggested additions or modifications to the propulsion system baseline are summarized.
A Mars Transfer System (MTS) with multiple nuclear thermal rocket engines departs from Earth parking orbit on a 150-day trip to Mars. The MTS captures at Mars for a 90-day stay, then leaves Mars on a 310-day return, performing a Venus swingby enroute. The MTS then captures into Earth parking orbit.

Main engine failure can occur at any of the three mission phases: TMI, MOC, or TEI. During TMI the main engines are utilized for a single perigee burn from Earth parking orbit. Main engines are used again for MOC. The TEI burn is initiated after the 90-day stay on the surface. The failed nuclear thermal rocket engine(s) remains with the vehicle during the entire mission duration.

Our analysis assesses performance impacts due to engine out by calculating relative additional propellant mass and reactor burn time delta's to complete the reference mission on time. For the reference mission, total propellant used was approximately 1,036,000 lbs, while nominal reactor burn time is approximately 1.76 hrs.

The amount of additional propellant required to compensate for engine-out depends on which phase of the mission it occurs. For the TMI phase, full thrust is maintained until escape velocity is reached, then the engine-out condition occurs. For the other two mission phases, engine failure occurs prior the injection or capture maneuvers.

The three engine case affords the least impact for a single engine-out condition. A factor of 2.0 - 2.2 less additional propellant and reactor burn time is required relative to the two-engine case. For two and three engine cases, propellant and burn time impacts are greatly reduced for MOC and TEI failures relative to TMI. This is a result of the reduced gravity well at Mars (.38 of Earth's).

The vehicle with a boost pump based propulsion system was also analyzed. Its Δpropellant and Δburn times were 8% and 3% lower respectively than the run tank case.
The above table summarizes the most serious problems that can occur for a nuclear propulsion system with a run tank; however, although they are also the least likely to occur. Many small failures can occur in the support systems undetected and without having any impact on the operation of the propulsion system because of the redundancy and safety features built into the systems. In most cases reduced thrust or safe abort is possible.

The propulsion system includes not only the main engine hardware, but also integrated support systems containing numerous valves, electrical switches, regulators, high pressure gas storage systems, etc. all of which are carefully chosen for specific functions and arranged in multiple combinations to guarantee safe, reliable and accurately controlled operation of the overall propulsion system. Problems associated with the propulsion system is therefore not only related to the main hardware components but also to the many components of the integrated support systems. Problems and failures in the overall system are most often related to the support systems and are detected by instrumentation and behavior of the support systems.

Problems and failures in NTR systems are related mostly to the systems and components which are similar to conventional chemical rockets, which makes it easier to analyze the NTR systems based on past experience. With failures in the reactor, it may be possible to continue safe operation at reduced power for even extended periods of time, since reactor life is greatly increased at reduced power.

Where practical, electromagnetic valves and actuators were chosen for high reliability and fast response. Electromagnetic hardware has been demonstrated to be better performing than pneumatic or hydraulic systems in many applications. Problem areas are mostly related to the controllers in the system which therefore require a large degree of redundancy built into the control systems.
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Various vehicle factors have to be considered in determining the thrust vector control requirements for a space vehicle.

A basic consideration is the flight path steering requirement. For an orbit launched vehicle, this requirement is minimal and is not critical.

A number of alignment factors, including thrust differential, vehicle c.g. offset, non-uniform depletion of propellants, engine-out, tank jettisoning, and engine jettisoning, require adjustment of the thrust vector.

Propellant sloshing and vehicle elastic motion may be coupled and must be considered in that context.

While a comprehensive survey has not been accomplished for this study, it was recognized that the engine-out event could have a major impact on the requirements. Accordingly, an assessment of this particular factor was made to obtain an indication of the magnitude of the requirement.
SURVIVING PROPULSION MODULE REQUIREMENTS

- Reactor burn time requirements could increase by 7% to 35% for the three engine case and 16% to 70% for the two engine case.

- If no engine jettison capability incorporated in design, surviving propulsion modules must be able to function in intense radiation and thermal environment caused by disabled engine.

- Gimbal requirements for run tank designs are worst for 1 of 2 engines out at maximum displacement, rate and acceleration of 7.5°, 3.5°/sec and 20°/sec² respectively.

- Gimbal mechanism must be robust and capable of vehicle control for extended duration at or near the maximum engine out null position of 5°.
Various vehicle factors have to be considered in determining the thrust vector control requirements for a space vehicle.

A basic consideration is the flight path steering requirement. For an orbit launched vehicle, this requirement is minimal and is not critical (unless a meteor/debris avoidance system is included).

A number of alignment factors, including thrust differential, vehicle c.g. offset, non-uniform depletion of propellant, engine-out, tank jettisoning, and engine jettisoning, require adjustment of the thrust vector.

Propellant sloshing and vehicle elastic motion may be coupled and must be considered in that context.

While a comprehensive survey has not been accomplished for this study, it was recognized that the engine-out event could have a major impact on the requirements. Accordingly, an assessment of this particular factor was made to obtain an indication of the magnitude of the requirement.
A dynamic guidance and control simulation was developed to examine the vehicle control response and engine gimballing requirements for the case where one engine of 2 fails. The vehicle mass distribution was examined and it was concluded that the worst case could be approximated with the Mars transfer injection maneuver tanks jettisoned and the aft core tank full of propellant. Instantaneous shutdown of the faulted engine also was assumed for the worst case.

A control loop was formulated and typical PID (Proportional Integral Differential) control gains were applied to obtain what appeared to be favorable results. As indicated in the plots of the results, the maximum vehicle alignment excursion is less than 1 degree (occurring about 4.5 seconds into the transient) and the excursion rate is about 0.3 degrees/second (at 2 seconds after engine thrust termination). The maximum engine gimbal response is approximately 7.5 degrees (at 3.5 seconds) requiring a maximum gimbal rate of about 3.5 degrees/second (at about 1 second). The engine-out null position is about 5 degrees parallel to the radial position vector of the faulted engine.

The control simulation maximum gimbal acceleration is about 20 degrees/second$^2$. This compares with a total deflection rate of 114 degrees/second$^2$ for the Centaur engines.
A dynamic guidance and control simulation was developed to examine the vehicle control response and engine gimballing requirements for the case where one engine of 3 fails. The vehicle mass distribution was examined and it was concluded that the worst case could be approximated with the Mars transfer injection maneuver tanks jettisoned and the aft core tank full of propellant. Instantaneous shutdown of the faulted engine also was assumed for the worst case.

A control loop was formulated and typical PID (Proportional Integral Differential) control gains were applied to obtain what appeared to be favorable results. As indicated in the plots of the results, the maximum vehicle alignment excursion is less than 0.5 degree (occurring about 4.5 seconds into the transient) and the excursion rate is less than 0.2 degrees/second (at 2 seconds after engine thrust termination). The maximum engine gimbal response is approximately 4 degrees (at 3.5 seconds) requiring a maximum gimbal rate of 2 degrees/second (at about 1 second). The engine-out null position is about 3 degrees parallel to the radial position vector of the faulted engine.

The control simulation maximum gimbal acceleration is about 10 degrees/second². This compares with a total deflection rate of 114 degrees/second² for the Centaur engines.
CONCLUSIONS - THRUST VECTOR CONTROL REQUIREMENTS

- ENGINE-OUT REQUIREMENTS:

<table>
<thead>
<tr>
<th>Requirement</th>
<th>Value 1</th>
<th>Value 2</th>
</tr>
</thead>
<tbody>
<tr>
<td>Displacement, degrees</td>
<td>7.5</td>
<td>4</td>
</tr>
<tr>
<td>Rate, degrees/second</td>
<td>3.5</td>
<td>2</td>
</tr>
<tr>
<td>Acceleration, degrees/s²</td>
<td>20</td>
<td>10</td>
</tr>
<tr>
<td>Null, degrees</td>
<td>5</td>
<td>3</td>
</tr>
</tbody>
</table>

- THE ENGINE-OUT REQUIREMENTS FOR THE BOOST PUMP DESIGNS WILL BE GREATER AND SHOULD BE ANALYZED.

- TANK AND ENGINE JETTISON CONDITIONS COULD BE SIGNIFICANT AND SHOULD BE ANALYZED.

- ALLOWANCE OF ABOUT 2 DEGREES APPEARS TO BE ADEQUATE FOR OTHER REQUIREMENTS.

The thrust vector control requirements have been determined for the engine-out event for the vehicle designed with a run tank. The displacement, gimbal rate, gimbal acceleration, and null position are about twice as great for the 3 engine installation with the run tank design as they are for the 2 engine installation of the same basic design. For the designs with the run tank, the requirements do not appear to be excessive. The requirements for the boost pump propulsion system designs could be significantly greater, however, and the should be analyzed.

The requirements for tank and engine jettison events could also be significant and should be analyzed. In any case, however, these events would not be concurrent with the engine-out event and therefore should not increase the overall gimbaling requirements.

Some other conditions and events, such as propellant sloshing, could be concurrent with engine-out. Such additional requirements should not be major, however, and probably can be covered with a nominal allowance of, say, 2 degrees.
TECHNOLOGY REQUIREMENTS

1) Robotic Coupling Tools/Techniques - On-orbit Assembly Of Core Tanks & Propulsion Modules

2) On-orbit Propellant Transfer - Top-off Propellant Tanks For Maximum Capability Missions

3) Boost Pumps - Six Times Mass Flow Of Centaur, GH2 Turbine Drive

4) Radiation Hardened Thrust Vector Controllers And Engine Controllers - Gamma Heating And Charged Particle Upsets

5) Run Tank Vent/Fill Systems - Vent GHe From Run Tank For In-space Restarts

6) Mixing Conditions With Bulk Heated Propellant - Predict LH2 Temperature For Turbopump Restart Conditions


8) Integrated Health Monitoring & Built-In Test - Reduce In-space Checkout Time/ Cost, Automatically Compensate For Failed/Off-nominal Conditions

Robotic coupling tools/techniques refers to the need for an OMV-type tug with mechanical manipulation capabilities. This tool would be used to mate core tanks, auxiliary tanks, propulsion modules and other components after delivery by launch vehicle to an assembly orbit. On-orbit propellant transfer capability would be needed to compensate for boiloff during assembly periods, and undertaking done to keep within launch vehicle delivery constraints. Boost pumps were developed for Centaur. However, those needed for an NTR stage would need to maintain six times their mass flow. Centaur turbopumps were driven by Hydrogen Peroxide. Those for an NTR stage might require special materials since driven by GH2 from the engine cooling jacket. Radiation hardened controllers for the engine and fluid system valves will be required Normal Centaur electronics would be subject to gamma heating damage and charged particle upset. Run tank vent/fill systems must be developed. At mission start, the run tank is pressurized with GHe. After MECO 1, propellant from core tanks is to fill and flow thru the run tanks to the engines. For minimum ullage, pressurization gases need to be vented from the run tank before it receives with the core propellant. Mixing conditions with bulk heated propellant need to be examined with CFD codes and experimental simulators able to model and predict existence of either stratified or circulation mixed propellant conditions. Unlike circulation mixed fluid, stratified propellant could allow non-uniform temperature of LH2 delivered to turbopump inlets over the duration of tank drain. Additional tank pressurization, which could impact design, would be needed to ensure net positive suction head conditions were met at all times. An engine jettison system to discard a dysfunctional propulsion module would reduce the mass of the vehicle, and thus the performance impact of engine-out. It would eliminate the possibility of a dead module staying critical and physically distorting to obstruct gimballing the remaining module. IHM and Smart Bit allow automated checkout to reduce the cost of in-space operations. It also allows off-nominal and inipient failure conditions to be compensated for at electronic speed.
CONCLUSIONS

- A cluster of multiple propulsion modules coupled to a core tank is feasible
- Hard coupling multiple nuclear thermal rocket engines to a core tank is an attractive alternative
  - Boost pumps utilized for engine start/restart
  - Upper and lower core tanks
  - Propulsion system integrated and checked out on the ground for single launch on an HLLV
- Three engine cluster appears to be more desirable than two engine cluster
  - Higher reliability
  - Less performance penalty in engine out scenario
  - Reduced reactor burn time requirements in engine out scenario
  - Lower thrust engines may cost less to develop
- May not be desirable to abort with 1 of 3 engines out, after TMI
  - Minimal reactor and propellant penalties, great mission success benefits
A number of major issues have to be resolved in order to adequately specify the propulsion system for a nuclear space transfer vehicle. The number and thrust of the engines must be determined based on an analysis of the initial and ultimate mission requirements, the cost of ground test facilities, and launch manifest considerations, as well as the reliability and engine-out capability of the engine cluster. The vehicle abort strategy and resolution of the requirement for single vs. dual turbopumps, in turn, are dependent upon the decision on the number of engines in the cluster.

The necessity for reactor cooling of a faulted engine can be avoided and abort mission performance can be improved if the faulted engine can be jettisoned. The experience available from the launch of 500 Atlases with jettisoned booster engines should be applied to determine what jettison features can be used with the clustered nuclear rocket propulsion system.

A major weight savings can be achieved if the engine thrust chamber can be developed with a variable internal shield. This is particularly critical with a clustered engine installation where side shielding is required to protect the adjacent run tank (or far side of the core tank bottom, in the case of the design with a boost pump). Accordingly, the feasibility of designing the thrust chamber with a variable internal shield should be explored.

On-orbit issues include coupling of the propulsion module to the vehicle and propellant transfer. The requirements for these operations should be analyzed in the context of design and development implications.

In order to establish the number and type of control and sensor interface connectors that must be provided, a definition of the engine checkout and health monitoring requirements must be derived. These requirements could have a major impact on the concept and location of the connector panels used for coupling the propulsion module to the vehicle.
SUMMARY AND CONCLUSIONS

- Engine jettison capability is a must
  - Jettisoning failed engine will improve reactor and propellant abort margins as well as reduce radiation and thermal protection requirements on the surviving propulsion module(s)

- Radiation hardening of engine/TVC controllers should result in substantial weight savings
  - The alternative is local electronics shielding or side shields on reactor
  - Ability to harden/locally shield will drive selection of TVC actuator

- Side shielding of reactor may offer substantial design/operations benefits
  - Reduce disk shield mass
  - Simpler installation on ground or in orbit

- TVC actuator displacement and gimbal rate and power requirements are within current state of the art
  - Displacement and rate calculated, actuator power by analogy

SUMMARY AND CONCLUSIONS

- Propulsion system design is dependant on man-rating requirements
- On orbit propellant transfer for tanking/topping propellant tanks
- Run tanks should be launched empty
  - Large surface area/volume ratio and on orbit assembly time
  - Reduce structural mass requirements

- Integrated health management is a must for any NTR
  - IHM/Smart Btt architecture implications will have significant impact on propulsion system mass and reliability

- The Earth to Orbit lift and volume constraints, coupled with on orbit operation significantly affect the propulsion module/system design
NTP Comparison Process

Nuclear Propulsion Technical Interchange Meeting

Sandusky, OH
October 2, 1992

Robert Corban
Nuclear Propulsion Office
NASA Lewis Research Center
The systems engineering process is shown above is for the concept definition phase of the program. The process involves three major elements: requirements definition, system definition, and consistent concept comparison. The requirements definition process involves obtaining a complete understanding of the system requirements based on customer needs, mission scenarios, and NTP operating characteristics. A system functional analysis is performed to provide a comprehensive traceability and verification of top-level requirements down to detailed system specifications and provides significant insight into the measures of system effectiveness to be utilized in system evaluation. The second key element in the process is the definition of system concepts to meet the requirements. This part of the process involves engine system and reactor contractor teams to develop alternative NTP system concepts that can be evaluated against specific attributes, as well as a reference configuration against which to compare system benefits and merits. Establishing the evaluation criteria will be extremely challenging and critical to the entire evaluation and selection process. Due to the various disciplines required and many goals the system will be required to achieve, an iterative and participative team approach must be utilized. Various methodologies exist for evaluating a comprehensive set of evaluation criteria: analytic hierarchy process (AHP), multiple-attribute-utility method (MAUM), and weighted-outranking method (WOM), but these provide little structure in identifying the key criteria. Quality function deployment (QFD), as an excellent tool within Total Quality Management (TQM) techniques, can provide the required structure and provide a link to the "voice" of the customer in establishing critical system qualities and their relationships. The third element of the process is the consistent performance comparison. The comparison process involves validating developed concept data and quantifying system merits through analysis, computer modeling, simulation, and, if required, rapid prototyping of the proposed high risk NTP subsystems. The maximum amount possible of quantitative data will be developed and/or validated to be utilized in the QFD evaluation matrix. If upon evaluation of a new concept or its associated subsystems determine to have substantial merit, those features will be incorporated into the reference configuration for subsequent system definition and comparison efforts.
Requirements Definition

- **Customers**
  - Understand who our customers
  - Need to understand customer needs & priorities

- **Functional Analysis**
  - AEC selected to provide requirements management
  - RDD-100 Systems Engineering Software
  - Functional hierarchy being developed
  - Requirements traceability & verification
  - "Living" Requirements Document established

**Customer**
A critical element of the process is the identification of the “customer(s)” and their particular desires for the NTP system. Those customers will consist of the President, Congress, the Nation's taxpayers, NASA management, and other government agencies concerned with the systems development and usage. These customers will most likely have different goals and objectives that must be understood and satisfied. The “voice” of the customers will be required to be part of the requirements definition process to guarantee their requirements are factored into the system.

**NTP Requirements**
The current top-level requirements for NTP for meeting currently envisioned SEI missions for cargo and piloted Mars missions have been in development over the past two years. A “living” requirements document has been developed with an on-going review process that incorporates current NTP team revisions and suggestions and begins to obtain a complete customer “voice” in the process. The current requirements have been incorporated by Analytical Engineering Corporation (AEC) into Ascent Logic’s powerful systems engineering software the Requirements Driven Development (RDD™) System Designer. This will allow for functional analysis, traceability, component-to-functions mapping, model behavior analysis, and failure propagation analysis.

**Functional Analysis**
AEC will be employing a methodology known as Enhanced Modern Structured Analysis (EMSA) in the analysis of the NTP systems. It will permit a logical structuring of all system functions in a top-down hierarchical decomposition to draw out all the requirements the system must meet while also providing insight for the system-level model developers and technologists. Various options will be provided to display the logical sequences and relationships of operational and support functions that lead to the fulfillment of each NTP function. Time dependent functions will be coupled with behavior models to allow for time-critical functional analysis. This analysis will also develop the basis for establishing functional interfaces and identify system relationships required in meeting SEI mission goals.
System Definition

NTP System Alternatives
Efforts were funded in 1992 by NASA to develop consistent state-of-the-art NTP concept data based on the same mission and engine requirements to permit an apples-to-apples comparison. Four alternative concepts were examined by various contractors to evaluate concept feasibility, thrust level implications in the range of 25,000 to 75,000 lbf, test facility requirements, manned mission impacts, key component technologies required, and an industrial approach to developing the system within the next decade. The four concepts examined were each defined based on a specific nuclear fuel element concept consisting of NERVA-derived, CERMET, Particle Bed, and a "twisted-ribbon" fuel element developed by the CIS.

Reference Concept
A reference concept will be utilized to help determine quantitative benefits of alternative engine concepts or subsystem. Significant past efforts on the NERVA concept combined with well understood improvements makes the current NERVA-derived concept the logical choice for the initial reference engine. The use of a reference concept will help in determining the benefits of alternative approaches to better quantify the risk, cost, performance, and schedule impacts.

System Attributes
The process required for evaluation and selection of a single NTP concept must be able to provide a structure that encourages the participation of many various disciplines and provides a focus on the customer needs. The attributes will not be honored if they are not obtained in a participative manner. Quality Functional Deployment, also referred to as the "house of quality," has demonstrated an advantage in providing a systematic and structured approach to achieving high quality systems. QFD identifies the most important system characteristics, relates characteristics directly to requirements, and identifies which characteristics need to be controlled. The current process will concentrate on only providing a system attributes matrix for NTP concept evaluation due to the extensive training, "cultural shock," and laborious nature in implementing QFD. But, with the goal within NASA to provide "faster, better, and cheaper" systems through Total Quality Management (TQM), the initial use of QFD can be expanded to provide the discipline required to achieve this ambitious goal.
Reference NTP Engine

The reference NTP concept shown above was defined by the Rocketdyne/Westinghouse team. The reference concept is based on a 50,000 pound engine utilizing dual turbopumps, 200:1 nozzle expansion and composite fuel within the NERVA fuel element configuration operating at 2700 K and a 785 psi chamber pressure. This NERVA reference engine shown is preliminary at this point. An initial reference engine and associated database will be determined in the next few months.
QFD Benefits

QFD was developed in Japan in the late 1960's in response to a recognized lack of "quality" in the definition/design process. The foundation for QFD is in the belief that systems should be designed to reflect customer needs and desires, thus requiring all disciplines to work closely together from the time a system is first conceived. Quality Functional Deployment, also referred to as the "house of quality," has demonstrated an advantage in providing a systematic and structured approach to achieving high quality systems. QFD identifies the most important system characteristics, relates characteristics directly to requirements, and identifies which characteristics need to be controlled. QFD provides a significant number of benefits in obtaining a quality product. Some of those benefits are shown above.
The QFD matrix, as shown in the example developed in the space transportation main engine (STME) program, begins with the customer needs, or wants, in phrases that describe the system and its characteristics in their own words. The wants are often grouped into areas of overall customer concerns that typically can include primary, secondary, and tertiary levels. Not all preferences are equal and the customer’s needs must be weighted based on discussions with the customers. The top of the QFD matrix lists those engineering characteristics that are likely to affect one or more of the customer needs. These characteristics should describe the system in measurable terms. The body of the matrix is filled with symbols indicating the strength of the customer needs in relationship with the engineering characteristics. On the right-hand side of the matrix, current reference concept’s level of meeting customer expectation and opportunities for improvement are determined. The rating of customer needs along with the number and strength of the matrix relationships provides the weighting for the engineering characteristics.
Consistent Comparison

- Integrated Government Team Formed
  - Develop & implement system performance modeling
  - State-of-the-art computational techniques
- Provide and/or Verify Quantitative Data
- Perform Risk and Failure Analysis
- Utilize Established Government Cost Models

The consistent comparison element of the process must provide and/or verify the quantitative data upon which the concepts will be evaluated. This data must be based on consistent assumptions, groundrules, and requirements. The data provided must also be independently verified to ensure proper analysis has been completed. The fundamental tools that assist the systems engineer in this process are the system performance and cost models, and quantitative risk assessments.

An integrated Government team has been formed to develop and implement a strategy for modeling NTP system performance. The modeling team was formed in order to integrate state-of-the-art computational resources and techniques, along with a diverse knowledge base, into simulations of NTP system performance. A parametric NTP model will be used to predict the system performance for all defined NTP concepts on a consistent basis. The model will also provide steady-state performance data for use in SEI mission analysis and evaluate system design perturbations. Transient evaluations, such as start-up and shut-down, will also be performed as the data and models become available. This will provide a means to evaluate the quantitative benefits to the system based on proposed subsystem and component improvements.

Risk, schedule, and cost analysis will be performed in addition to the performance assessments. The RDD™-100 systems engineering tool will be coupled with the Failure Environment Analysis Tool (FEAT) to assist in the identification of hardware and software failure effects on the entire system. This will ensure that the concept complies to redundancy, reliability, and safety requirements. Cost analysis will utilize established Government cost models to quantify cost benefits to the system upon the implementation of an alternative.
NUCLEAR THERMAL PROPULSION

TECHNOLOGY
Nuclear Thermal Propulsion Technology Overview

James R. Stone

NASA/LeRC Nuclear Propulsion Office

NUCLEAR THERMAL PROPULSION TECHNOLOGY ELEMENTS

Innovative Technology

High-Risk High-Payoff Technologies for 2nd & 3rd Generation Systems

NTP Enabling Technology

Technology Integration

Nuclear Sub-system Technologies

Non-nuclear Sub-system Technologies

Crit Tech Tst Pln
Fuel Fab/Prod
Nuc Fuel Tests
Non-nuc Fuel Tests
MIL Irradn Tests
Fuel Elem. Tests
Nuc. Furnace Tests
Neutronics, I&C

Propellant Mgmt

NTP Technology 1504501-1

Early Emphasis
NTP Focused Technology Status
Innovative Technology

- Model Development: Graduate Research
- Vapor-Core Modeling & Experiments: INSPI
- Gas-Core Simulation Facility: LeRC
- PBR Stability Modeling: MIT
- PBR Materials Modeling: Univ. New Mexico

NTP Focused Technology Status
Enabling Technology

- NOZZLES
  - CFD Model Development (3-D Navier-Stokes): LeRC
  - Nozzle Alternatives & Optimization Experiments: LeRC
  - Molecular CFD Plume Model Development: LeRC
- THURBOPUMPS
  - Low-NPSH Pumping Technology: MSFC
  - Materials Evaluation: MSFC
  - 3-D Navier-Stokes CFD Model Development: LeRC
- STRUCTURES
  - Probabilistic Model Development: LeRC
- INSTRUMENTATION & CONTROLS
  - High-Temperature Sensors: LeRC
- MATERIALS
  - Preliminary Sample Prep & Expts: LeRC & MSFC
Non-nuclear Material

- Goal is usable materials database
- Results needed early to support design and analysis work
- Advanced and commercially available materials to be studied
- Develop required processing and characterization facilities
- Tie-in with Base R & T work

Instrumentation, Controls, and Health Monitoring

- Large advances since NERVA, needed for autonomous ops
- Details of overall system architecture TBD
- Plan to build off on-going efforts in chemical engine area
- Current LeRC effort concentrating on sensors
- Good progress to date with SiC
Turbopumps

- LeRC working flow and performance modeling
  - Evaluation and modification of existing codes
  - Will use TPA testbed to validate model
- MSFC working hardware specifics
  - Evaluation of concept options, materials, technologies
  - Bearing options being studied

Nozzle and Extension

- CFD modeling of internal flow
  - Fluid, thermal, chemical behavior
  - 2-D work done for various temp and thrust ranges
  - Plan to expand results to 3-D, other nozzle forms
Nozzle and Extension

• Alternative nozzle design evaluation
  • Study of various alternative nozzle forms
  • Goal is performance and packaging improvements
  • Small scale tests to take place in mid 93
  • Promising results flowed back into CFD effort

Nozzle and Extension

• Probabilistic Structural Modeling
  • Large Expansion ratio nozzles (>200:1) cannot be ground tested
  • Develop analytical ability to be able to launch with assured reliability
  • Apply available prob struct modeling methods to NTR nozzle
  • Input CFD results, fabrication process uncertainties
  • Develop probabilistic QA criterion
  • Develop design spec for nozzle and extension
Exhaust Plume Characterization

- Content and behavior of exhaust plume critical to man-ratability
- LeRC developing validated numerical simulation capability
  - CFD not sufficient
  - DSMC with finite difference Boltzmann techniques
- Will accomodate various nozzle shapes, species, conditions
  - Experimental validation planned

Summary

- NTP is key to SEI
- Non-nuclear technologies vital to NTP
- Critical technologies identified
- Work begun, preliminary results available
- Efforts will continue in an evolutionary manner
Silicon Carbide Semiconductor Technology for High Temperature and Radiation Environments

Lawrence G. Maty

Figure 1: This talk will describe silicon carbide technology and its potential for enabling electronic devices to function in high temperature and high radiation environments. The talk will be given by Dr. Lawrence G. Maty of the Instrumentation and Control Technology Division, NASA Lewis Research Center.

SILICON CARBIDE

A Crystalline material with unique properties

- Abrasive
- Structural
- Refractory
- Semiconducting
  - Wide energy bandgap
  - High breakdown voltage
  - Low dielectric constant
  - High thermal conductivity
  - Able to dope both N and P type
  - High saturated electron drift velocity

Figure 2: Silicon carbide (SiC) has many unique properties. The LeRC research program is exploring the semiconductor properties of SiC. The wide energy bandgap of SiC allows it to function at high temperatures, the high breakdown voltage and high thermal conductivity of SiC suggests that power devices and radiation hard devices will be possible, and the low dielectric constant and high saturated electron drift velocity of SiC opens the possibility of high frequency devices. SiC can be doped both n- and p-type for electronic device fabrication.
The properties of the two most common polytypes of SiC (3C-SiC, also called beta or cubic SiC and 6H-SiC, also called alpha SiC) are compared with the properties of commercially available semiconductors and diamond. The commercially available semiconductors were judged unable to meet the 600°C temperature goal the LaRC program, while diamond technology was considered to be too far in the future.
Figure 5: Prior to 1989, SiC researchers had to use small irregular-shaped Lely SiC samples for device studies. Now, Cree Research, Inc., a small company in Durham, North Carolina, has made one inch SiC substrates commercially available.

Figure 6: The difficulty in producing SiC substrates is that SiC does not melt. Therefore, the SiC boule crystal growing technique involves the sublimation of SiC powder. The SiC boule growth is carried out in a high temperature furnace where after the SiC powder sublimes, the SiC vapor is transported along a temperature gradient, and then deposits onto a SiC seed crystal which is at a cooler temperature.
Figure 7: During the 1980s, the LoRC SiC program developed a chemical vapor deposition technique for the heteroepitaxial growth of 3C-SiC onto silicon substrates. This 3C-SiC material has many defects because of the mismatch in material properties between the SiC and silicon. However, the chemical vapor deposition process works well for the homoepitaxial growth of 6H-SiC onto 6H-SiC substrates. Our process uses silane as the silicon source, propane as the carbon source, and hydrogen as the carrier gas. For doping the SiC epilayers, nitrogen gas produces n-type while trimethylaluminum vapor produces p-type SiC. Hydrogen chloride gas is used during a pregrowth etch. The growth temperature is 1450°C. A radio-frequency generator heats the graphite susceptor. The growth process takes place at atmospheric pressure.

Figure 8: A photo of the LoRC chemical vapor deposition system. The time sequence and flow rates of all process gases are computer controlled.
Figure 9: Device structure and room temperature current-voltage characteristics for a 6H-SiC pn junction diode that was fabricated at LaRC. Etching of the SiC to produce the mesa-style pn junction configuration was accomplished by reactive ion etching; there are no known wet chemical etchants for SiC. The reverse diode characteristics demonstrated very low leakage out to the breakdown voltage of around 1000 volts. The forward diode characteristics displayed the rather large turn-on voltage associated with the wide bandgap of SiC.

Silicon Carbide Junction Diode

Accomplishment: Highest reported operational temperature (600 °C) for any p-n junction diode device. Significantly improved characteristics above 400 °C. Demonstrates high quality 6H-SiC epitaxial film growth processes.

Benefits: Silicon carbide diodes (p-n junctions) are basic building blocks from which all future silicon carbide electronic devices will be developed.

Figure 10: The diode array photo shows SiC diodes of different sizes. The diode sizes range from 50 to 400 microns in diameter. The 6H-SiC diode demonstrated excellent current-voltage characteristics when tested to 600°C. As a semiconductor material, SiC can clearly perform at elevated temperatures. The SiC pn junction is a basic building block from which all future SiC electronic devices will be developed.
Figure 11: A photo documenting the operation of a 6H-SiC pn junction diode at 600°C. One diode, of the many on the chip, is being examined on a probing station equipped with a heating stage. The forward biased diode is emitting blue light while the heating stage is glowing cherry-red at 600°C.

Figure 12: As seen in figure 11, SiC is an LED material. This photo documents for the first time, that both 6H-SiC blue LEDs and 3C-SiC green-yellow LEDs can be produced on a single chip. The ability to fabricate a 3C-SiC LED is an indication of the high quality of the 3C-SiC material.
SILICON CARBIDE MOSFET

Milestone: Develop and demonstrate a high temperature, (400 °C), 6H-SiC metal-oxide-semiconductor field effect transistor (MOSFET)

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.

Figure 13: A depletion-mode 6H-SiC Metal-Oxide-Semiconductor Field-Effect Transistor (MOSFET) was demonstrated to an operating temperature of 500°C. SiC has silicon dioxide as its native oxide, so many of the silicon oxidation techniques are directly importable to the SiC technology. The current-voltage characteristics for this MOSFET are not yet ideal because the device structure and oxide growth processes have not yet been optimized.

Silicon Carbide JFET Radiation Response

Figure 14: SiC is expected to be a radiation-hard semiconductor. Work performed at the Harry Diamond Laboratories demonstrates that, yes indeed, SiC is radiation-hard. 6H-SiC Junction Field-Effect Transistors (JFETs) were exposed to both gamma and neutron radiation. The JFET experienced little effect from the gamma radiation and was still functioning after an exposure of $10^{16}$ neutrons per cm$^2$. The JFET also performed better at the elevated temperature of 300°C than at room temperature after the neutron exposure.
AREAS REQUIRING TECHNOLOGY ADVANCEMENT
(FOR HIGH TEMPERATURE APPLICATIONS)

- Metallization (electrical) contacts
- Passivation and dielectric layers
- Wire attachment
- Packaging
- Circuit board technology

Figure 15: Several areas still require technology advancement before SiC is ready for high temperature and/or high radiation environments. The LeRC program is supporting research in the areas of metallization, passivation and dielectric layers, wire attachment, and component packaging. Ultimately circuit board technology must be developed.

CONCLUDING REMARKS

- Need for 600 °C electronic devices
- SiC is the semiconductor of choice
- Significant SiC crystal growth progress
- Discrete devices (diodes and MOSFETs) demonstrated
- Several challenging areas await
- SiC is on its way

Figure 16: Concluding Remarks: The LeRC program believes that a need for high temperature (600°C) and/or high radiation-hard electronic devices exists. The semiconductor of choice is SiC because of its many unique properties and the fact that diamond is still far in the future. During the past ten years, significant progress has been made in the advancement of SiC technology. Key progress has been made in the SiC crystal growth process. This progress has allowed device scientists to fabricate prototype SiC electronic devices with exciting characteristics and thus, LeRC researchers feel that SiC, as an electronic material, is definitely on its way. However, as is probably evident, there are still a number of challenging areas of research to be pursued.
NTR PLUME MODELING

Presented to the
Nuclear Propulsion Technical Interchange Meeting

October 21, 1992

D. BYERS, Chief, Low Thrust Propulsion Branch
C.-H. CHUNG, Principal Investigator
R. STUBBS, Chief, Computational Methods for Space Branch

COMPUTATIONAL FLUID DYNAMICS (CFD)
FOR PLUME ANALYSIS

MOLECULAR FLUID MECHANICS

- THE VAST MAJORITY OF CFD DEALS WITH GASES WHICH ARE ADEQUATELY DESCRIBED BY THE CONTINUUM THEORY, I.E., THE NAVIER-STOKES EQUATIONS.

- IN RAREFIED GAS FLOWS, A MOLECULAR MODEL IS APPROPRIATE, REQUIRING DIFFERENT TECHNIQUES.
  - DIRECT SIMULATION MONTE-CARLO (DSMC)
  - FINITE DIFFERENCING OF THE BOLTZMANN EQUATION

- MOLECULAR CFD IS REQUIRED FOR:
  - NOZZLE LIP AND CRITICAL BACKFLOW REGIONS
  - PLUME / SPACECRAFT INTERACTIONS
  - GROUND TESTING
MOLECULAR CFD CHARACTERISTICS

- DSMC TECHNIQUES TRACK A LARGE NUMBER OF MOLECULES (OF ORDER $10^5$ TO $10^7$) AND MODEL THEIR INTERACTIONS STATISTICALLY.

- COMPUTATIONALLY INTENSIVE

- DR. CHAN-HONG CHUNG HAS DEVELOPED AN ENHANCED DSMC CODE WITH MULTI-SPECIES CAPABILITY, ALLOWING MORE ACCURATE CALCULATIONS OF SPECIE SEPARATION.

DIRECT-SIMULATION MONTE-CARLO (DSMC) METHOD

A computer technique to model low density gas flows by concurrently following the motion and intermolecular collisions of representative molecules.
INTEGRATION OF
DSMC AND NAVIER-STOKES COMPUTATIONS
Fig. 1  Density profile along the line parallel to exit plane
Fig. 4 Molecular hydrogen temperature profile along the line parallel to exit plane.
## LOG10(Number Density)

<table>
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<th>Value</th>
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<td>6.00</td>
<td>5.50</td>
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Area Ratio = 100  
Throat Radius = 28 cm  
Lip Thickness = 4 cm

## Velocity (m/sec)

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</table>

Area Ratio = 100  
Throat Radius = 28 cm  
Lip Thickness = 4 cm
Fig. 2 Density profile along the line parallel to exit plane

Fig. 3 Degree of separation along the line parallel to exit plane
COMPUTATIONAL FLUID DYNAMICS
FOR
NUCLEAR THERMAL PROPULSION

Presented to the
Nuclear Propulsion Technical Interchange Meeting
October 21, 1992

Robert M. Stubbs
Suk C. Kim
SPECIFIC ENTHALPY OF HYDROGEN AND MOLE FRACTION OF H AS A FUNCTION OF CHAMBER PRESSURE AT A CHAMBER TEMPERATURE OF 3,600 K

SPECIFIC IMPULSE AS A FUNCTION OF CHAMBER PRESSURE

$T_C = 3,600 \text{ K}$
**RPLUS**

- **DEVELOPED AT NASA-LEWIS**
- **A NAVIER-STOKES CODE WITH FINITE RATE CHEMICAL KINETICS CAPABILITY**
  - LU-SSOR
  - 9 SPECIES, 18 REACTIONS, \((H_2, O_2)\) COMBUSTION SYSTEM
  - 3-D, (ONLY 2-D AXISymmETRIC REQUIRED HERE)
SPECIFIC IMPULSE AS A FUNCTION OF CHAMBER PRESSURE

\[ T_C = 3,600 \text{ K} \]

![Graph showing specific impulse as a function of chamber pressure.

WALL CONFIGURATIONS OF NOZZLES "A", "B", AND "C"

ALL HAVE:
- \( R_{\text{TROAT}} = 0.28 \text{ m} \)
- \( A_e/A_T = 100 \)
- THROAT TO EXIT LENGTH = 7.6 m
Axial Distributions on the Centerlines

Radial Distributions at the Exit
### TABLE 4. Specific Impulse of NTP

Nozzles which have been scaled to produce, at each Temperature, approximately equal Thrust.

<table>
<thead>
<tr>
<th>Temp (K)</th>
<th>PC=10 atm</th>
<th>P_C=1.0 atm</th>
<th>P_C=0.1 atm</th>
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<tbody>
<tr>
<td></td>
<td>Tc, (K)</td>
<td>r_t=0.28 m</td>
<td>r_t=0.8854 m</td>
</tr>
<tr>
<td>2700</td>
<td>901.61</td>
<td>899.48</td>
<td>903.14</td>
</tr>
<tr>
<td>3200</td>
<td>1024.33</td>
<td>1037.21</td>
<td>1072.47</td>
</tr>
<tr>
<td>3600</td>
<td>1144.22</td>
<td>1183.39</td>
<td>1223.17</td>
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</table>
TABLE 5. Specific Impulse for variously sized NTP Nozzles with $T_C=3600$ K, $P_C=1.0$ atm.

<table>
<thead>
<tr>
<th>$r_t$ (m)</th>
<th>$\text{lsp, (lb}_T\text{s/lb}_m$)</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.28</td>
<td>1151.57</td>
</tr>
<tr>
<td>0.8854</td>
<td>1183.39</td>
</tr>
<tr>
<td>2.8</td>
<td>1220.41</td>
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**SUMMARY**

- **CFD Simulations Predict Lower Specific Impulse Values for the Low Pressure Nuclear Thermal Rocket Than One-Dimensional, Inviscid Analyses.**

- **The Low Pressure Concept Shows More Promise at Higher Temperatures Than at Lower Temperatures, Because of the Greater Amount of Dissociation.**

- **Smaller Nozzles Show Larger Viscous Losses, Especially at Low Pressures; Therefore, Performance Gains Are Associated With Larger Nozzles.**

- **Advanced CFD Codes Such as RPLUS (3D, Navier-Stokes, Chemical Kinetics), With Their Ability to Simulate Real Gas Effects, Provide the Designer With Powerful Tools to Analyze the Entire Flow Field and Calculate Global Performance Values.**
ON-GOING WORK

NOZZLE WALL FILM-COOLING STUDIES

- EFFECTIVENESS OF HYDROGEN FILM COOLING
  ● AMOUNT OF HYDROGEN
  ● OPTIMIZATION OF FILM PLACEMENT

- PERFORMANCE IMPACT
  ● INTERACTION WITH PRIMARY FLOW
  ● LOSSES IN SPECIFIC IMPULSE

Tw : wall temperature (K)

Fig. 4: Temperature variations along the nozzle wall with and without film-cooling.
Probabilistic Structural Analysis for Nuclear Thermal Propulsion

Dr. Ashwin Shah
Sverdrup Technology

(presented by J.R. Stone, LeRC/NPO)

CERTIFICATION OF SPACE NUCLEAR PROPULSION SYSTEM NOZZLE WITH ASSURED RELIABILITY

OBJECTIVE: To develop a methodology to certify Space Nuclear Propulsion System Nozzle with assured reliability
Certification of Space Nuclear Propulsion System Nozzle with Assured Reliability

Uncertain boundary condition

Uncertain structural response

Uncertain pressure load

Uncertain thermal gradients

Uncertain material behavior

Uncertain shape

Height

Diameter

471
**ADVANTAGE OF PROBABILISTIC STRUCTURAL ANALYSIS**

**Case I: Wider scatter in stress and strength**

Safety factor = 60/40 = 1.5  
Reliability = 0.99  
Reliability is small because shaded overlapping area is large due to wider scatter.

**Case II: Narrow scatter in stress and strength**

Safety factor = 60/40 = 1.5  
Reliability = 0.999  
Reliability is large because shaded overlapping area is small due to narrow scatter.

Reliability of structure depends on scatter in stress and strength.  
Probabilistic approach accounts for scatter in stress and/or strength rationally.

---

**Space Nuclear Propulsion System Nozzle**

**Uncertainties in the Random Variables**

<table>
<thead>
<tr>
<th>Random Variable</th>
<th>Coefficient of Variation / Standard Deviation</th>
<th>Distribution</th>
</tr>
</thead>
<tbody>
<tr>
<td>Pressure</td>
<td>5%</td>
<td>Normal</td>
</tr>
<tr>
<td>Geometry: X-Coordinate</td>
<td>0.25 ln</td>
<td>Normal</td>
</tr>
<tr>
<td>Geometry: Y-Coordinate</td>
<td>0.25 ln</td>
<td>Normal</td>
</tr>
<tr>
<td>Geometry: Z-Coord. (Height)</td>
<td>0.25 ln</td>
<td>Lognormal</td>
</tr>
<tr>
<td>Thickness</td>
<td>2.5%</td>
<td>Normal</td>
</tr>
<tr>
<td>Temperature</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Inside Surface</td>
<td>5%</td>
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<tr>
<td>Layer 2</td>
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<td>Layer 3</td>
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<tr>
<td>Layer 4</td>
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<tr>
<td>Outside Surface</td>
<td>5%</td>
<td>Normal</td>
</tr>
<tr>
<td>Modulus of Elasticity</td>
<td>5%</td>
<td>Weibull</td>
</tr>
<tr>
<td>Coefficient of thermal Expansion</td>
<td>2.5%</td>
<td>Normal</td>
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<tr>
<td>Strength</td>
<td>4%</td>
<td>Weibull</td>
</tr>
</tbody>
</table>
Probability of the natural frequency being less than 66.7 Hz = 0.999

Therefore, to achieve a reliability of 0.999, the frequency of exciting force should be larger than 66.7 Hz to avoid resonance.

Variables controlling the scatter of natural frequency within 66.7 Hz are thickness, modulus of elasticity and mass density. Therefore a tighter tolerance for the thickness and material properties are essential.
SNPS Nozzle
CDF of Effective Stress in the shell

Sensitivity of primitive variable uncertainties
SNPS nozzle - shell stress

To control the stresses in the shell and achieve higher reliability, the uncertainties in the inside surface temperature should be reduced.
Work in progress:

- Modelling of NERVA base model with coolant tubes
- Development of pseudo-super element to reduce the size of the global model to achieve computational speed and accuracy
NON-NUCLEAR MATERIALS ASSESSMENT

"IDENTIFY AND EVALUATE CANDIDATE MATERIALS FOR USE IN NTP TURBOMACHINERY AND PROPELLANT FEED SYSTEM APPLICATIONS."

THE APPROACH WAS TO DEVELOP AND IMPLEMENT DETAILED TEST PLANS AND EVALUATION CRITERIA FOR EFFECTS OF A HOT HYDROGEN ENVIRONMENT ON NTP CANDIDATE MATERIALS.

THE FOLLOWING MATERIALS WERE SELECTED FOR SURFACE EROSION TESTING:
- INCONEL 718 (BASELINE MATERIAL)
- IMAR 246 (IMPROVED MARAGING STEEL)
- NASA 23 (NICKEL BASED STEEL)

THE FOLLOWING TEST PLANS WERE DEVELOPED:
- STATIC HOT HYDROGEN TESTING UP TO 1000 C AND 5000 psi. TEST TO INCLUDE:
  - TENSILE PROPERTIES
  - LOW CYCLE FATIGUE
  - CREEP
  - FRACTURE TOUGHNESS
- FLOWING HOT HYDROGEN TESTING AT TEMPERATURES UP TO 1000 C FOR MICROSTRUCTURE CHARACTERIZATION OF EXPOSED MATERIALS

MATERIAL SAMPLES WERE TESTED AT AUBURN UNIVERSITY IN A 700 C HYDROGEN ENVIRONMENT WITH ADDITIONAL TESTS TO BE PERFORMED AT MSFC, ALSO AT 700 C.

MSFC FACILITY IS CAPABLE OF MATERIAL EXPOSURE TESTING UP TO 980 C AND 5000psi IN A HYDROGEN OR HELIUM ENVIRONMENT.

NON-NUCLEAR MATERIALS ASSESSMENT

The objective of the MSFC materials effort is to identify and evaluate candidate materials for use in NTP turbomachinery and propellant feed system applications. The initial task was to develop a set of test plans and evaluation criteria that could be applied to screen candidate materials for application in NTR components. In order to be a viable candidate, the material must be resistant to degradation due to the effects of exposure to hydrogen. A set of baseline materials were selected which included Inconel 718, NASA 23, and IMAR 246. These material samples were provided to Auburn University for exposure in a 700 C hydrogen environment and characterization of the induced surface erosion. Similar hydrogen environment testing was performed at MSFC for obtaining the mechanical properties of the samples through in situ testing in 10 MPa (1500 psi) hydrogen at 700 C. In situ test capabilities include tensile strain properties, fatigue, crack growth, fracture toughness, creep, and four point bend.

NTP: Technology 476 MSFC

NP-TIM-92
NON-NUCLEAR MATERIALS ASSESSMENT

HIGH TEMPERATURE TESTING OF VARIOUS CARBIDE BASED COATINGS FOR APPLICATION TO TURBOPUMPS, TURBINE BLADES, FLOW SYSTEMS, AND NOZZLES HAVE ARE BEING PERFORMED. TaC, WC, & NbC HAVE BEEN EXPOSED TO HYDROGEN AT 1 ATMOSPHERE AND TEMPERATURES OF 830, 1350, & 1460 C TO DETERMINE % WEIGHT LOSS

<table>
<thead>
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<th>Material</th>
<th>% weight loss at temperature</th>
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<td></td>
<td>830 C</td>
</tr>
<tr>
<td>TaC</td>
<td>0.03</td>
</tr>
<tr>
<td>WC</td>
<td>0.07</td>
</tr>
<tr>
<td>NbC</td>
<td>0.10</td>
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</table>

SILICON NITRIDE AND ALUMINA CERAMICS HAVE BEEN TESTED FOR HIGH TEMPERATURE COATING APPLICATIONS. COMPARATIVE TESTING WAS PERFORMED IN AIR AND HYDROGEN AT AMBIENT TEMPERATURE AND AT 700 C FOR PERIODS OF 1 HOUR. 

A FOUR-POINT BEND FIXTURE WAS USED TO TEST BASIC MECHANICAL PROPERTIES OF MEAN STRENGTH AND WEIBULL MODULUS.

NON-NUCLEAR MATERIALS ASSESSMENT

High temperature testing of carbide based coatings for application to turbopumps, turbine blades, flow systems, and nozzles are being performed. These coatings include the carbides of Tantalum, Niobium, Tungsten, and Silicon. These materials are being exposed to elevated temperatures over the range of 830 C to 1460 C in a vacuum and hydrogen environment for periods of one hour. Analysis consists of in situ weight loss determinations and residual gas analysis with subsequent examination of microstructure via Scanning Electron Microscopy and Transmission Electron Microscopy. The objective of this investigation is to characterize microstructural changes in these materials as a result of exposure to hydrogen.

Further material evaluations involve the preparation of Silicon Nitride and Alumina, candidate ceramics for high temperature coatings, for comparative tests in air and hydrogen at room temperature and 700 C for periods of one hour. At MSFC a four point bend fixture was configured to perform in situ testing of these materials for determination of their basic mechanical properties of mean strength and Weibull modulus.
NTP TURBOMACHINERY TECHNOLOGIES

"DEVELOP AND VALIDATE ADVANCED TURBOMACHINERY TECHNOLOGIES AT THE COMPONENT AND TURBOPUMP ASSEMBLY LEVELS."

THE NASA REFERENCE SIZE WAS BOUNDED BY 50K AND 75K LB THRUST ENGINE.

THE APPROACH USED WAS TO ASSESS AND DEFINE TURBOMACHINERY TECHNOLOGY REQUIREMENTS FOR A SPACE BASED/START NUCLEAR THERMAL ROCKET ENGINE. MSFC AND LeRC SPECIALISTS COLLABORATED TO DEFINE AN INITIAL TECHNOLOGY PLAN FOR TPA TECHNOLOGY.

THE PLAN ADDRESSES:
- BEARINGS (FLUID FILM & FOIL)
  - SLOW START TRANSIENT FOR FLUID FILM BEARINGS
  - RUB TOLERANT MATERIALS FOR FLUID BEARINGS
  - ROLLING ELEMENT CAGE MATERIAL FOR THRUST BEARING (IF REQUIRED)
  - MAGNETIC BEARINGS
- SEALS
  - DEFINITION OF SEAL REQUIREMENTS
  - RUB TOLERANT MATERIALS
- EARLY NEED FOR A PROPELLANT FEED SYSTEM TEST BED
- EARLY DEFINITION OF TRANSIENT LOADS
- EVALUATION OF FEED SYSTEM IMPACTS ON TURBOMACHINERY

NTP TURBOMACHINERY TECHNOLOGIES

The objective of the MSFC turbomachinery technology task is to develop and validate technologies at the component and turbopump assembly level for application in a Nuclear Thermal Rocket engine. Marshall Propulsion Laboratory personnel collaborated with turbomachinery specialists at LeRC on the assessment of the technology requirements and priorities as well as an initial technology development plan. The ground rules provided that the engine size be in the range of 50K to 75K lb thrust and space based. Space base/start imposes a need to assess the requirement for low NPSH technologies.

The technology assessment and development plan addresses both fluid film and foil bearings. Current thinking is that rolling element bearings would not be used unless in a thrust bearing application. There exist, to date, little experience with either foil or hydrostatic bearings. Most experience addresses only fast start transient systems and, therefore, indicate a need for research in the application of fluid film/foil bearings in a prolonged start transient such as the NTR application. This also illustrates the need for further materials research for materials that would be wear resistant in a hydrogen and radiation environment. The main concern for rolling element bearings is for application as a thrust bearing where research is needed to identify a cage material that will survive the radiation environment. The application of a magnetic bearing could eliminate wear at startup and is also being considered.

Additional technology is also required in the area of seals. Questions must be addressed as to the need for a positive seal between the pump and turbine for pre and post operation.

A propellant feed system test bed is needed early in any TPA technology/advanced development program. A preliminary study has begun to assess the possibility of using existing turbomachinery and test stand hardware to facilitate the development of a test stand. The testbed is needed to evaluate transient operation and provide early definition of transient loads. This facility could also be used to assess feed system impacts on the turbomachinery. The system could also be used to evaluate TPA control and health monitoring technologies.
HIGH TEMPERATURE SUPERCONDUCTING MAGNETIC BEARING TECHNOLOGY

"DEVELOP AND VALIDATE ADVANCED TECHNOLOGY FOR HIGH TEMPERATURE SUPERCONDUCTOR (HTS) PASSIVE MAGNETIC/HYDROSTATIC BEARING"

GREATLY REDUCE, OR ELIMINATE COMPLETELY, THE EXPECTED WEAR TO A CONVENTIONAL HYDROSTATIC BEARING AS A RESULT OF NTR SLOW STARTUP TRANSIENT

SDIO CONTRACT WITH MTI JOINTLY FUNDED BY DARPA AND NASA

HTS TECHNOLOGY WILL ENABLE THE DEVELOPMENT OF A NEAR ZERO-WEAR BEARING WHEN COMBINED WITH FLUID FILM BEARING CONCEPTS.

CURRENTLY DESIGNING PROOF-OF-CONCEPT TEST RIG BASED ON MSFC SUPPLIED REFERENCE TPA PARAMETERS BASED ON J-2S TURBOMACHINERY.

TESTS AND MATERIAL RESEARCH ONGOING TO INCREASE HTS BEARING LOAD CARRYING CAPABILITY AT LH2 TEMPERATURES. HTS LOAD CAPABILITY HAS BEEN IMPROVED BY 450%.

MSFC INHOUSE EFFORTS ARE FOCUSED ON MEASUREMENT OF HTS MAGNETIZATION OVER TEMPERATURE RANGE FROM 25K TO 77K. NTP-TPA OPERATIONAL TEMPERATURE IS PREDICTED TO BE AROUND 30K.

HIGH TEMPERATURE SUPERCONDUCTING MAGNETIC BEARING TECHNOLOGY

The objective of the MSFC HTS technology task is to develop and validate advanced technology for High Temperature Superconductor (HTS) passive magnetic/hydrostatic bearings for application in a nuclear rocket engine. This bearing concept will greatly reduce, or eliminate completely, the expected wear to a conventional hydrostatic bearing as a result of the extremely slow startup transient of the NTR. This work was performed by Mechanical Technology Inc. under a SDIO contract funded jointly between NASA/MSFC and DARPA.

By combining HTS technology with that of fluid film bearings, it will be possible to suspend the pump shaft during the start-up and shut-down of the pump when the hydrostatic bearing is not fully functional. HTS stiffness has been improved by 450% during the course of this effort and further improvement to capabilities of >2000 lb/in2 is believed very possible. MTI was supplied reference parameters based on the J-2S turbopump in order to design a proof-of-concept test apparatus.

Additional inhouse efforts have focused on measurement of HTS magnetization over temperature ranges from 25K to 77K. The operational temperature of the turbomachinery for the NTR is predicted to be 30K.
NUCLEAR GAS CORE PROPULSION RESEARCH PROGRAM

a presentation to the
Nuclear Propulsion Technical Interchange Meeting '92
NP-TIM-92
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by
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NUCLEAR GAS CORE PROPULSION RESEARCH PROGRAM

Advanced Nuclear Propulsion Studies
- To develop a hydrogen properties package at temperatures 10 - 10,000 K and pressures 0.1 - 200 atm.
- To develop a transient simulation program for parametric studies and design analysis of high temperature nuclear rockets

Nuclear Vapor Thermal Rocket (NVTR) Studies
- To conduct nuclear and thermal design optimization of the NVTR fuel, fuel elements and core geometry
- To develop a system and parametric analysis code for the NVTR

Ultrahigh Temperature Nuclear Fuels and Materials Studies
- Determine properties of UF₄ and UF₄ mixtures nuclear fuels at temperature-pressure ranges of interest to advanced nuclear propulsion systems
- Measure/model high temperature compatibility of UF₄ with refractory carbides.

The objectives of these studies are to develop models and experiments, systems and fuel elements for advanced nuclear thermal propulsion rockets. The fuel elements under investigation are suitable for gas/vapor and multiphase fuel reactors.
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.
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 $C_p$. 
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.
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.
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 cernet fueled engine and INSPI's nuclear thermal vapor rocket.
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 $S_\Omega$ code. The axial power shape factor is an input for the simulation code.
Specific Impulse vs Chamber Pressure
INSPI-NTVR @ 75000lbf Thrust

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.
The Nuclear Vapor Thermal Rocket (NVTR) is an advanced thermal propulsion engine, using vapor or multiphase nuclear fuel, with predicted performance at the upper limits of solid core reactors. The NVTR also serves as base technology development toward high performance Gas Core Reactors.
Design Values

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Pump Flowrate (Total)</td>
<td>75.50 lb/s</td>
</tr>
<tr>
<td>Pump Discharge Pressure</td>
<td>3,369 psia</td>
</tr>
<tr>
<td>Number Of Pump Stages</td>
<td>2</td>
</tr>
<tr>
<td>Pump Efficiency (%)</td>
<td>78.26 %</td>
</tr>
<tr>
<td>Turbopump Rpm</td>
<td>70,000 RPM</td>
</tr>
<tr>
<td>Turbopump Power (Each)</td>
<td>8,082 HP</td>
</tr>
<tr>
<td>Turbine Inlet Temp</td>
<td>361 deg-R</td>
</tr>
<tr>
<td>Number Of Turbine Stages</td>
<td>2</td>
</tr>
<tr>
<td>Turbine Efficiency</td>
<td>81.51 %</td>
</tr>
<tr>
<td>Turbine Pressure Ratio</td>
<td>1.85 ---</td>
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<tr>
<td>Turbine Flow Rate (Each)</td>
<td>33.87 lb/s</td>
</tr>
<tr>
<td>Reactor Thermal Power</td>
<td>1,759 MW</td>
</tr>
<tr>
<td>Fuel Element Transferred Power</td>
<td>1,724 MW</td>
</tr>
<tr>
<td>Nozzle Chamber Temperature</td>
<td>5,580 deg-R</td>
</tr>
<tr>
<td>Chamber Pressure (Nozzle Stagnation)</td>
<td>1,500 psia</td>
</tr>
<tr>
<td>Nozzle Expansion Area Ratio</td>
<td>560.1</td>
</tr>
<tr>
<td>Nozzle Percent Length</td>
<td>123 %</td>
</tr>
<tr>
<td>Vacuum Specific Impulse (Delivered)</td>
<td>993.3 sec</td>
</tr>
</tbody>
</table>

Heat loads are as follows:
- Nozzle-con (total): 30.05 MW
- Nozzle-div (total): 22.97 MW
- Reflector (total): 35.00 MW

Note: Flows indicated are for one-half of system

Formulas:
- $P = \text{psia}$
- $T = \text{deg-R}$
- $W = \text{lb}_m/\text{sec}$
- $H = \text{BTU}/\text{lb}_m$
- $S = \text{BTU}/\text{lb}_m \cdot \text{R}$
NUCLEAR VAPOR THERMAL ROCKET PARAMETERS

Fuel Pressure: 100 atm
Average Fuel Temperature: 4000 K
Maximum Element Heat Flux: 330 W/cm²
Nominal Element Length: 150 cm
Fuel Volume Fraction: 0.15
Coolant Volume Fraction: 0.15
Moderator Volume Fraction: 0.70
Fuel Element Power: 0.7 MWt
Element Heat Transfer Area: 2170 cm²
Reactor Core L/D: 1.5
Fuel Channel Diameter: 0.142 cm
Fuel Channel Sectional Area: 0.0158 cm²
Total Fuel Channel Area Per Element: 0.505 cm²
Fuel Element Sectional Area: 3.464 cm²
Element Diameter (across flats): 2.00 cm
Coolant Channel Diameter: 0.142 cm
Coolant Channel Sectional Area: 0.0158 cm²
Total Coolant Channel Area Per Ele.: 0.505 cm²
Core Volume (elements only): 1.053 m³
Core Power Density: 1330 MW/m³
Fuel Element Mass, Total: 1.35 MT
Forward Reflector Mass: 0.60 MT
Aft Reflector Mass: 0.51 MT
Radial Reflector Mass: 2.47 MT
Radiation Shield Mass: 0.90 MT
Total Reactor Mass: 5.83 MT
Misc Engine Components Mass: 0.9 MT
Total Engine Mass: 6.83 MT
Engine F/W: 5.0 MT

NVTR Cross Section

10/29/92
HIGH TEMPERATURE NUCLEAR FUELS AND MATERIALS STUDIES

- Experimental Studies Related to a Parallel Program Confirmed UF₄ Compatibility with:
  - W at temperatures up to 2200 K
    (Experiment and post-test analysis at T up to 3000 K in progress)
  - Mo at temperatures up to 2200 K
    (Experiment and post-test analysis at T up to 2600 K in progress)
  - C at temperatures up to 1800 K

- Detailed Thermodynamics Analysis Demonstrated Outstanding Chemical Compatibility Between UF₄ and WC, W₂C, Mo₂C at Temperatures up to 2600 K

- Thermodynamic Studies Revealed Outstanding Properties of UF₄ - UC₂ Mixture for NTVR Fuel

Compatibility of UF₄ at elevated temperatures with wall materials is a key to successful development of fuel element for NTVR. Experimental studies of UF₄ compatibility with a wide range of materials has shown promising results for Mo, W, and C. Thermodynamic analysis suggested outstanding chemical compatibility of WC, W₂C and Mo₂C at temperatures up to 2600 K. High temperature thermodynamics analysis has also revealed the outstanding stability of UF₄ - UC₂ system. Due to presence of carbon in UF₄ - UC₂ fuel mixture, better compatibility with the fuel element wall materials and gaseous fuel is expected.
Flow Instability in Particle-Bed Nuclear Reactors
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PRESENTED AT NUCLEAR PROPULSION INTERCHANGE MEETING
NASA LEWIS RESEARCH CENTER, OCTOBER 22, 1992

Abstract
The particle-bed core offers mitigation of some of the problems of solid-core nuclear rocket reactors. Dividing the fuel elements into small spherical particles contained in a cylindrical bed through which the propellant flows radially, may reduce the thermal stress in the fuel elements, allowing higher propellant temperatures to be reached. The high temperature regions of the reactor are confined to the interior of cylindrical fuel assemblies, so most of the reactor can be relatively cool. This enables the use of structural and moderating materials which reduce the minimum critical size and mass of the reactor. One of the unresolved questions about this concept is whether the flow through the particle-bed will be well behaved, or will be subject to destructive flow instabilities. Most of the recent analyses of the stability of the particle-bed reactor have been extensions of the approach of Bussard and Delauer, where the bed is essentially treated as an array of parallel passages, so that the mass flow is continuous from inlet to outlet through any one passage. A more general three dimensional model of the bed is adopted here, in which the fluid has mobility in three dimensions. Comparison of results of the earlier approach to the present one shows that the former does not accurately represent the stability at low Re. The more complete model presented here should be capable of meeting this deficiency while accurately representing the effects of the cold and hot frits, and of heat conduction and radiation in the particle-bed. It can be extended to apply to the cylindrical geometry of particle-bed reactors without difficulty. From the exemplary calculations which have been carried out, it can be concluded that a particle bed without a cold frit would be subject to instability if operated at the high temperatures desired for nuclear rockets, and at power densities below about 4 megawatts per liter. Since the desired power density is about 40 megawatts per liter, it can be concluded that operation at design exit temperature but at reduced power could be hazardous for such a reactor. But the calculations also show that an appropriate cold frit could very likely cure the instability. More definite conclusions must await calculations for specific designs.

* R.C. Maclaurin Professor of Aeronautics and Astronautics
Conclusions
Comparison of three quite different approaches to modeling the stability of the particle-bed reactor, all with consistent physical assumptions, shows that a complete linear stability such as that presented here is in fact necessary for reliable prediction of the stability of the particle-bed reactor. The approach, termed here the Parallel-Stream model, where the reactor is assumed to be composed of a series of channels coupled only at their inlets and outlets, does not accurately represent the stability at low Re, nor does it represent the effect of heat conduction in the bed.

The model termed here (perhaps somewhat naively) the Complete Model should be capable of accurately representing the effects of the cold and hot frits, and of heat conduction and radiation in the particle bed. It can be extended to apply to the cylindrical geometry of particle-bed reactors without difficulty.

From the exemplary calculations which have been carried out, it can be concluded that a particle bed without a cold frit would be subject to instability if operated at the high temperatures desired for nuclear rockets, and at power densities below about 4 megawatts per liter. Since the desired power density is about 40 megawatts per liter, it can be concluded that operation at design exit temperature but at reduced power could be hazardous for such a reactor. But the calculations also show that an appropriate cold frit could very likely cure the instability. More definite conclusions must await calculations for specific designs.

Acknowledgments
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References


SCHEMATIC OF PARTICLE-BED

GOVERNING EQUATIONS

\[ \nu_p = -\frac{b \epsilon^2}{\kappa} \frac{\partial T_p}{\partial x} \]  
\[ k = 150 \left( 1 - e^2 \right) \frac{\mu(T_p)}{e^3} \]  
\[ b = 1.75 \left( 1 - e^2 \right) \frac{1}{e^3} D_p^2 \]  
\[ \rho_p \frac{\partial T_p}{\partial t} = \frac{1}{\alpha D_p} \frac{\partial}{\partial x} \left( \alpha D_p \frac{\partial T_p}{\partial x} \right) + k_{eff} \nabla^2 T_p \]  
\[ k_{eff} = k_{cond} + k_{rad} + k_{conv} \left( \frac{1}{3} \right) \alpha D_p T_p^3 \]  
\[ h = \frac{25}{10} \left( 1 - e^2 \right) \frac{1}{e^2} + \frac{25}{3} \frac{1 - e^2}{e^2} \]  
\[ \rho_p \frac{\partial h_p}{\partial t} + \nabla \cdot \rho_p \rho \mathbf{u} = 0 \]  
\[ \rho \mathbf{u} \cdot \nabla \rho = 0 \]  
\[ \rho \mathbf{u} \cdot \nabla T = \rho \mathbf{u} \cdot \nabla T \]
NON-DIMENSIONAL GOVERNING EQUATIONS

\[\begin{align*}
\nu_0 &= b_1 \left( T_p \right)^2 \nu \quad b_2 T_p \rho_0 \nabla \phi \\
&= \frac{b_1 \rho_0(0) \nu(0) T_p}{\phi(0) \kappa} \left[ \frac{\rho(0) T_p}{\phi(0) \kappa} \right]^{1/2} \left( \frac{T_p}{T_T(0)} \right)^{1/2}
\end{align*}\]  

(10)

\[\begin{align*}
c &= \frac{\rho_0(0) \nu(0) c_p(0)}{\nu(0) T_T(0)} \\
H &= \frac{h}{h_T} = \frac{\nu}{\nu_T} \frac{T_p}{T_T} W = \frac{\nu}{\nu_T} T_p
\end{align*}\]  

(11)

\[\begin{align*}
\rho \frac{\partial T_p}{\partial t} &= \nabla \cdot \left( \rho c_p T_p \right) - \rho \left( T_p - T_T \right) \mathbf{J}
\end{align*}\]  

(12)

\[\begin{align*}
\frac{\partial T_T}{\partial t} &= \nabla \cdot \left( \rho c_p T_T \right) - \rho \left( T_T - T_T \right) \mathbf{J}
\end{align*}\]  

(13)

ZEROTH-ORDER OR STEADY SOLUTION

\[\begin{align*}
\rho g \nu_0 \frac{dT_T}{dx} &= \frac{q}{K} \frac{d^2 T_p}{dx^2} \\
T_T(0) &= 1 + q x
\end{align*}\]  

(9)

\[\begin{align*}
T_T(0) &= 1 + q x + \frac{q}{h} \\
p_0 &= \frac{P_0}{T_T(0)}
\end{align*}\]  

(10)

\[\begin{align*}
p_0^2 &= \frac{1}{(\nu + q) q} \left[ \left( 1 + q x \right)^2 + \frac{b_2}{q} \left( 1 + q x \right)^2 \right]
\end{align*}\]  

(11)

\[\begin{align*}
\rho_0 - P_0 &= \frac{L}{T_T(0)}
\end{align*}\]  

(12)
FIRST ORDER

\[ V_\rho = b \left[ T_\rho^2 + \sum_i u_\rho \cdot V_\rho \cdot T_\rho \right] \cdot b \left[ \sum_i \left( u_\rho + u_i \right) \right] \]
(14)

\[ \epsilon \frac{\partial T_\rho}{\partial t} = \lambda \left( T_\rho - T_0 \right) \cdot \left( T_\rho - T_0 \right) \cdot \left( T_\rho - T_0 \right) \cdot \left( T_\rho - T_0 \right) \]
(15)

\[ \frac{\partial \rho}{\partial t} + \rho \frac{\partial u_\rho}{\partial x} \quad \frac{\partial \rho}{\partial x} + u_\rho \frac{\partial \rho}{\partial x} = 0 \]
(17)

Variables are: \( \rho, \rho_0, T_\rho, u, u_\rho \) and \( T_0 \)

PARAMETERS.

Dimensionless Parameters :
- from 1n. \( \nu, b_1, b_2 \)
- from 3n. \( q, c, H, K_c, K_r \)

Operating Parameters :
- \( T_{\rho0} \) (exit) = 3000 K
- \( \rho_0 \) (exit) = 100 atm
- \( Q \) = 4 \times 10^{10} \text{ watt/m}^3

Design Parameters :
- \( I = 0.01 \text{ m} \)
- \( D_p = 0.005 \text{ m} \)

Stability Parameters :
\[ R_e = \frac{\rho_0(u_\rho(0))D_p}{\mu(0)} = \frac{D_pQ_I}{\rho_0u_\rho(0)\mu(0)} \]
q
APPROACHES TO INSTABILITY ANALYSIS

1) Parallel Flow Instability
2) Local Instability Analysis
3) Full Stability Analysis

PARALLEL-STREAM INSTABILITY

Instability is possible if $p_0(1)$ increases with mass flow density for fixed $Q$ and $p_0(0)$. Hence:

$$\frac{d\log(p_0(1))}{d\log(Re)} \bigg|_{Q,p_0(0)} > 0 \Rightarrow \text{INSTABILITY}$$
COMPLETE INSTABILITY MODEL

\[ p(x,y,z,t) = p(x) e^{i\omega t} \]

\[
\frac{d\rho}{dx} = \left( a_1 T_{G0} + b_2 \right) \rho_x \cdot \left[ a_1 \left( \frac{\partial u_x}{\partial x} \right) \right] T_g \cdot \left[ a_2 \frac{\partial u_x}{\partial x} \right] \rho
\]

\[
\frac{dT_g}{dx} = \left( T_{G0} - T_g \right) (\rho u_x + \rho u_y) T_g
\]

\[
\frac{d\rho}{dx} = \left( \frac{1}{T_{G0}} \right) \left( \frac{\partial \rho}{\partial T_{G0}} \right) \rho \cdot \left( \frac{1}{T_{G0}} \right) \left( \frac{\partial T_g}{\partial x} \right) T_g
\]

\[
\frac{dT_g}{dx} = \left( \rho \left( \frac{\partial T_g}{\partial x} \right) \right) (\rho u_x + \rho u_y) \rho
\]

\[
\frac{dP}{dx} = \frac{1}{K} \cdot \frac{dT_g}{dx} + \left( \rho u_x \right) \left( \frac{\partial T_g}{\partial x} \right) + \left( \rho u_y \right) \left( \frac{\partial T_g}{\partial y} \right)
\]

Approximate Case Neglecting Conduction in x:

\[
T_g = \left( \frac{1}{K} \cdot \frac{dT_g}{dx} \right) + \left( \rho u_x \right) \left( \frac{\partial T_g}{\partial x} \right)
\]
LOCAL INSTABILITY

\[ p(x, y, z, \lambda) = p(x, y, z, \lambda) \cdot \delta[k, \lambda] + \text{int} \]

Nominal Particle-Peel Operating Point

COMPLETE INSTABILITY MODEL

Boundary Value Solutions
- Cold Frit
- \( k = 1 \)
- 1 cm cold frit
- No cold frit

Nominal Pebble Bed Operating Point

Parallel-Stream
## NTP Reactor & Fuel Requirements

### Reactor Requirements

<table>
<thead>
<tr>
<th>Performance:</th>
<th>Fuel Requirements:</th>
</tr>
</thead>
<tbody>
<tr>
<td>Specific Impulse:</td>
<td>Fuel Temperature $&gt; 3000K$</td>
</tr>
<tr>
<td>Thrust-to-Weight:</td>
<td>Uranium Loading $&gt; 0.8$ g/cc</td>
</tr>
<tr>
<td>Single Burn Time:</td>
<td>Thermal &amp; Chemical Stability</td>
</tr>
<tr>
<td>Operating Life Time:</td>
<td>Low Diffusion Rates</td>
</tr>
<tr>
<td>Restart:</td>
<td>Thermal Shock Resistance</td>
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</table>

### Safety:

<table>
<thead>
<tr>
<th>ALARA radiation</th>
<th>FP retention</th>
</tr>
</thead>
<tbody>
<tr>
<td>Large margin to failure</td>
<td>High Melting Point</td>
</tr>
<tr>
<td>Redundancy</td>
<td>Robust Fuel Elements</td>
</tr>
<tr>
<td>Fast restart</td>
<td>Thermal Shock Resistance</td>
</tr>
</tbody>
</table>
TEMPERATURE vs. PERFORMANCE

- Burnup
- Swelling, Fission
- Gas Release, Chemistry
- Diffusion, Vaporization

Los Alamos

PROPULSION EFFICIENCY AND TEMPERATURE

SPECIFIC IMPULSE SQUARED vs. HYDROGEN EXIT TEMPERATURE (K)

- (U,Zr)C
- (U,Zr)C/Graphite
- Coated UC₂/Graphite

Los Alamos
ROVER FUEL TYPES

- UC Particles/Graphite Matrix PyC Coated UC Particles/Graphite Matrix
- ZrC Coating

UC, Particles
Graphite Matrix
Hydrogen

Carbide/Graphite Composite

URANIUM FUEL COMPOUNDS

<table>
<thead>
<tr>
<th>Property</th>
<th>UO₂</th>
<th>UC</th>
<th>UC₂</th>
<th>UN</th>
<th>U₂Zr₈C₃₂⁹</th>
</tr>
</thead>
<tbody>
<tr>
<td>Density, g/cc</td>
<td>10.96</td>
<td>13.63</td>
<td>11.68</td>
<td>14.32</td>
<td>8.01</td>
</tr>
<tr>
<td>U Density, g/cc</td>
<td>9.66</td>
<td>12.97</td>
<td>10.60</td>
<td>13.52</td>
<td>2.88</td>
</tr>
<tr>
<td>Melting Point, K</td>
<td>3100</td>
<td>2775</td>
<td>2710</td>
<td>3035*</td>
<td>3350</td>
</tr>
<tr>
<td>Thermal Expansion, 10⁻⁶ / K</td>
<td>10.1</td>
<td>11.2</td>
<td>12.0</td>
<td>8.9</td>
<td>7.6</td>
</tr>
<tr>
<td>(at 1273 K)</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Thermal Conductivity, W/cm K</td>
<td>0.035</td>
<td>0.23</td>
<td>0.07</td>
<td>0.25</td>
<td>0.3</td>
</tr>
<tr>
<td>(at 1273K)</td>
<td></td>
<td></td>
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</tbody>
</table>
UC-ZrC PSEUDO-BINARY PHASE DIAGRAM

MAJOR SOURCES OF DATA
U-Zr-C PHASE DIAGRAM

Uranium - Carbon Binary
Uranium Carbide - Zirconium Carbide Pseudo-Binary
Uranium Dicarbide - Zirconium Carbide Pseudo-Binary
Zirconium - Carbon Binary
Calculations - Chang Formulation
- Butt and Wallace
PHASE DIAGRAM "OPTIMIZATION"

SOLIDUS - LIQUIDUS CALCULATION

2473 K
ISO-ACTIVITY TIE-LINES
UC-NbC-ZrC PSEUDO TERNARY SYSTEM

MELTING POINT EXPERIMENTS

SAMPLE FABRICATION
COMPOSITION
FABRICATION
MEASUREMENT
ANALYSIS
PRELIMINARY MELT POINT COMPOSITIONS

U-Zr-C Ternary Diagram
3273 K ISOTHERMAL SECTION

SAMPLE FABRICATION TECHNIQUES

COLD PRESS, REDUCE, AND SINTER

ARC MELT

COMBUSTION SYNTHESIS
### Measured Melt Point Comparison

<table>
<thead>
<tr>
<th>Composition</th>
<th>Observed Melt Pt., K</th>
<th>Literature Value, K</th>
<th>Variance, K</th>
</tr>
</thead>
<tbody>
<tr>
<td>UC$_{1.0}$</td>
<td>2806</td>
<td>2793</td>
<td>+13</td>
</tr>
<tr>
<td>UC$_{1.4}$</td>
<td>2633</td>
<td>2673</td>
<td>-40</td>
</tr>
<tr>
<td>U$<em>{d}$Zr$</em>{6}$C$_{1.2}$</td>
<td>2683</td>
<td>2673</td>
<td>+10</td>
</tr>
<tr>
<td>U$<em>{d}$Zr$</em>{6}$C$_{1.2}$</td>
<td>2655</td>
<td>2673</td>
<td>-18</td>
</tr>
</tbody>
</table>

### Cryochemical Fuel Processing

```
UO$_2$ ➔ Graphite ➔ Binder ➔ Surfactant ➔ Water

 ballet mill slurry ➔ freeze droplets ➔ recycle reject

freeze dry microspheres ➔ sieve ➔ synthesize to UC ➔ sinter ➔ characterize

CH$_4$ ➔ CVD coatings ➔ characterize & release ➔ particle bed, extruded, or cermet fuel
```
CRYOCHEMICAL SPHERE FORMING ADVANTAGES

- Process is composed of a few simple steps
- Applicable to a variety of nuclear fuel concepts
- Porosity is likely a controllable variable
- Spheres >1000 μm diameter appear possible
- Rejected spheres are easily reused
- Re-using process fluids minimizes wastes
Laser Diagnostics for NTP Fuel Corrosion Studies

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Nuclear Propulsion Technical Interchange Meeting
NASA-Lewis Research Center, Plum Brook Station

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In order to generate the necessary propulsive performance, the nuclear reactor will operate at temperatures approaching 3000 K. Such temperatures, in combination with the high-pressure hydrogen propellant flowing through the core, provide a very hostile operating environment for the fuel material, particularly with respect to hydrogen-induced corrosion. Identifying the corrosion products as well as experimentally quantifying the corrosion rates of these fuels under a variety of conditions is critical to assess the expected lifetime of the reactor. The severe environment surrounding a fuel material under test is not conducive to probing by standard instrumentation. Thus alternative diagnostic techniques are required to provide real-time monitoring of corrosion products. Laser diagnostics offer a means of non-intrusively probing the high-temperature environment above the surface of the fuel to identify and establish spatial distributions and local concentrations of many of the anticipated corrosion species.
Solid solution refractory carbide fuel materials, such as U_xZr_{1-x}C_y, are being seriously considered as candidate NTP reactor fuels because of their high melting point and resistance to corrosion by hydrogen. Butt has calculated the equilibrium partial pressures above the surface of U_{0.05}Zr_{0.95}C_{1.07} during exposure to 1 atm of hydrogen over a temperature range of 2000 to 3200 K. The results show that in high temperature hydrogen, U(g), Zr(g), and various hydrocarbon species dominate the gas-phase products. Above approximately 2800K, acetylene and gas phase uranium and zirconium are predicted to be the primary vapor constituents. Confirmation of such predictions through experimental measurements is critical to the development of accurate corrosion kinetics models.

In the current study, Zr atoms were selected as the species to probe using laser techniques. This selection was influenced by several factors. First, the zirconium atom is predicted to be a major corrosion product. Second, the corrosion of U_xZr_{1-x}C_y in hydrogen may be rate limited by the transport of gas phase Zr away from the surface. Third, testing a laser diagnostic for zirconium obviates the need to perform experiments in a hydrogen environment or with uranium-containing fuel samples.


Laser-induced fluorescence (LIF) offers a highly sensitive technique for monitoring many of the NTP fuel corrosion products (including Zr) as well as for determining properties of the NTP exhaust. Quite simply, LIF can be viewed as an absorption of laser light, at a specific frequency, by an atomic or molecular species followed after some finite time by an emission from the excited state. This emission or fluorescence is, in general, at a different (typically longer) wavelength than the exciting laser light's wavelength. By viewing this off-resonance fluorescence, it is possible to avoid interference from scattered laser light.

Planar LIF or PLIF represents an extension of the LIF technique from a point or line diagnostic to a 2D or field measurement method. In general, the species-exciting laser beam is transformed into a thin sheet by the placement of cylindrical lenses into the optical train. A camera is typically used to collect the resulting fluorescence emission, perpendicular to the species-exciting laser sheet. The local intensity of the collected light can then be related to the concentration, temperature, or velocity of the target species. The non-intrusive nature of this technique, as well as its good spatial and temporal resolution, make it particularly well suited for application as a diagnostic in the high temperature NTP operating environment.
A PLIF scheme for measurement of corrosion species evolving during the test of an NTP fuel sample/element could take the above configuration. The laser sheet is passed through the high temperature hydrogen stream, which contains the various corrosion products, and contacts the surface of the fuel. Fluorescence emission from the chosen target species is then imaged producing a two-dimensional distribution map.
Production of zirconium vapor for subsequent illumination by a PLIF diagnostic, is accomplished by focusing a pulsed laser onto a ZrC target. This technique, known as laser ablation, represents a relatively simple method for producing gas phase samples of refractory materials.

The apparatus utilized for these experiments is displayed above. The cubical (~30 cm) ablation chamber contains five, window ports which allows optical access to its interior. The chamber is evacuated by a standard mechanical pump through both a liquid nitrogen and alumina-filled trap. The chamber can be backfilled with a variety of gases through a separate, flow regulated feed line. During each experiment, a slow flow of argon is maintained through this line and the chamber to minimize the buildup of particulate. A series of capacitance type manometers and thermocouple gauges are available to monitor chamber pressure. The base pressure for the chamber is 20 mtorr. Typical operating pressures are between 7 to 10 torr. The ZrC target specimen is positioned on a rotating table which is externally driven by a variable speed DC motor. Rotation of the target prevents the formation of a pit in the ZrC disk by action of the ablation laser.

A Lumonics model TE-860-4 excimer laser operating at 248 nm is used to produce the ablation pulse. Beam energies are on the order of 100 mJ/pulse and the laser is operated with a 5-10 Hz repetition frequency. The beam is brought to the ablation chamber by several high reflectance mirrors and focused at normal incidence onto the target using a 18.3-cm focal length quartz lens. The ablation spot size is approximately 1 mm² and the corresponding laser fluence at the target is ~670 MW/cm². The PLIF sheet is passed, at right angles, through the plume.
Plume emission images were recorded at different delay times. One representative ZrC plume emission image is displayed above. The ability to acquire such images represents a necessary step in the application of the PLIF technique. An intensified, gated uv camera (Xybion model ISG-250-U) with a 105 mm, f/4.5 quartz focusing lens was interfaced with an EPix Silicon MUX RGB frame grabber board with a programmable trigger option. The timing of the camera intensifier, frame grabber board, and excimer laser was controlled using a Stanford Research System model DG535 programmable delay/pulse generator. The above image was captured 50 ps after the excimer laser pulse. The ablation chamber’s argon background pressure was held constant at 7.5 torr and the camera gate width set at 20 μs. In these expanding plasmas, atom velocities can exceed $10^6$ cm/s at low background pressures (tens of mtorr) with neutral gas temperatures near the target surface approaching 15,000 K.
Temporally resolved ZrC plume emission spectra were recorded for regions near the target surface. The chamber pressure was maintained at 8 torr. A representative ZrC plume emission spectrum, recorded 10 µs after the excimer laser pulse, is shown above. This emission spectrum, which covers a wavelength range from 200.0 to 500.0 nm, is dominated by the presence of zirconium atom emission (several of the many Zr(I) lines are identified in the figure). No emission lines from other species, such as carbon atoms are identified in the emission traces.
By adjusting the delay time, it is possible to establish emission spectra at specific times during the plume expansion event. For example, ZrC emission spectra recorded for increasing delay times indicate that the plume emission intensity has reduced to essentially undetectable levels at approximately 1 ms (for a background pressure of 8 torr). Such a reduction is due to expansion cooling and quenching through radiative and collisional processes. Quantifying the temporal behavior of plume emission intensity is important for establishing the proper delay times to acquire PLIF images of the expanding zirconium vapor as it reduces interference from background emission.
PLIF Imaging of ZrC Plume

- Zr(I) ground state to excited state transition at 35515 cm⁻¹ pumped using Nd:YAG laser-pumped dye laser (R590 dye)
- Fluorescence emission monitored at 418.76 nm using filtered, gated, uv-intensified CCD camera coupled to frame-grabber board
- Image capture at different times after excimer pulse
- PLIF sheet 30.0 mm × <0.5 mm

PLIF images of the spatial distribution of the zirconium atom in the ZrC plume have been acquired. To capture these images, the Zr(I) ground state to the excited state (t³P₂, J=2) transition at 35,515.4 cm⁻¹ (281.5 nm) was pumped using the frequency doubled output of a Nd:YAG laser-pumped dye laser operating on Rhodamine 590 dye. Fluorescence emission at 418.76 nm, corresponding to the transition from the t³P₂ excited state to the b³F₂ (J=2) intermediate state at 11640.7 cm⁻¹, was captured with the uv-intensified CCD camera. A dye laser sheet approximately 30.0 mm by 0.5 mm was formed for passage through the plume by a lens combination consisting of a +100 mm (converging) lens and a -100 mm (diverging) focal length cylindrical lens and a 150 mm focal length spherical lens.
A PLIF image showing the spatial distribution of ground state zirconium atoms in the ZrC plume is shown above. The fluorescence emission was filtered using a GG395 filter. The image was recorded 625 μs after the ablation laser pulse with the camera gate width set at 20 μs. The spots observed in this image are pieces of hot (radiating) sputtered material from the target.
Summary

- Utilized a focused excimer laser to ablate material from ZrC targets
- Zr prevalent in plumes
- Temporally resolved CCD image of plume emission generated

\[ \tau_{\text{plume}} \sim 1 \text{ ms} \quad (P_{\text{Background}} = 8 \text{ torr}) ; \quad T_{\text{exc}} \equiv 12,000 \text{ K} \]

- PLIF utilized to successfully image Zr atom distribution in plume
- PLIF technique should be able to monitor Zr about fuel element under test

We have utilized a focused excimer laser to ablate material from ZrC targets for the purpose of developing appropriate laser-based diagnostics for gas-phase, corrosion products from hydrogen-exposed U\textsubscript{x}Zr\textsubscript{1-x}C\textsubscript{y} fuel elements proposed for NTP application. Temporally and spatially resolved emission spectra from the produced vapor plumes show the dominating presence of zirconium atoms. Temporally and spatially resolved images of the ZrC plume emission have also been recorded. The PLIF technique has been successfully used to image Zr atom distributions in the ablated ZrC vapor plume and thus could potentially be utilized to monitor Zr about fuel elements under test.
Future Activities

- Investigate other fluorescence excitation wavelengths for Zr
- Expose samples to rf-heated, hydrogen containing flow and probe flowfield around and downstream of sample with PLIF diagnostic
- Investigate other NTP fuel materials; (U-Nb-C) system
- Quantify Zr, Nb, etc. concentration in terms of fluorescence emission

Potential future activities include investigating other zirconium atom excitation wavelengths to ascertain the optimal transition for obtaining PLIF images in the ZrC ablation plumes. Following such determination, ZrC samples will be exposed to radio-frequency (rf) heated, hydrogen-containing flows and the PLIF diagnostic will be used to measure Zr atom distributions in the region surrounding the samples or in the nozzle exhaust flow.

For this study, a radio frequency discharge driven flow system will be used to produce a continuous, high temperature, chemically-clean gas stream. The unique feature of this system, which has been described by Wantuck, is the use of an inductively coupled plasma tube as a high enthalpy gas source. A 50 kW rf generator is used to supply power to the tube where the gas is heated to between 5000 to 10,000 K. The system configuration will allow two different modes of ZrC sample heating, namely, placement of the sample within the plasma tube for direct inductive/plasma heating or positioning downstream of the nozzle exit for heating by the gas stream. The first configuration approximates corrosion species distribution in an NTP exhaust flowfield. The second configuration best simulates propellant flow over a fuel element. In both cases, a dye laser beam, operating at the same wavelength employed for the plume PLIF illumination studies, will be used to probe the nozzle exit flowfield or the region surrounding the gas-stream heated ZrC sample.

SNTP Propellant Management System

Current SNTP Engine System Uses High Temperature Bleed Cycle

- Tank Shut-off Valve
- Boost Pump Ejector
- Turbopump Assembly
- Pump Discharge Shutoff Valve
- Turbine Control Valve
- Hydrogen Flowmeter

SNTP Cycle Selection

Full-Temperature Bleed Cycle is Lowest Engine System Mass with Minimal Isp Penalty

- No design interaction with reactor
- Allows light-weight radiation-cooled nozzle
- Lowest system complexity, potentially highest system reliability
- High-temperature, low-Z material minimize cooling in radiation environments
NTP System Components Have Unique Design Constraints

- High Ionizing Radiation Environment
- High Heat Load From Radiation Energy Absorption
- Restricts Use Of High-Z Materials
- Design Must Provide For Heat Removal

Bleed Cycle Presents Unique Design Requirements for Turbopump

- Moderate operating pressures (1350 psi)
  - Single-stage pump
  - Light pressure vessels
- High operating temperatures (2750 K)
  - Highly energetic working fluid
  - High-pressure ratio impulse turbine
  - High turbine temperatures
  - Large thermal gradients
- Environmental factors
  - Environmental heating — low-Z material
  - Limited elastomers selection
  - Hot-hydrogen embrittlement
- Use of bleed cycle and uncooled thrust nozzle results in substantial system weight savings.
Bleed-Cycle Turbopump
Uses Carbon-Carbon Components for Operation on 2750 K Gas

Carbon-Carbon Hot Section Housing
Carbon-Carbon Turbines
Titanium Shafting

Carbon-Carbon Nozzle/Plenum
Aluminum Pump and Inducer
Ceramic Rolling Element Bearings or Foil Bearings

SNTP Carbon-Carbon Turbine Wheel
Design is Based on Technology Developed on the ELITE Program

- Helical 2-D polar weave architecture
- Impulse blades
- 55,600 rpm
- 2750 K inlet temperature
- 45-percent design stress margin
- 26-percent design speed margin
Turbine Development Program

*High-Temperature, Carbon-Carbon Components Are Being Fabricated and Will be Tested at 2750 K*

- Turbopump Preliminary Design
- High-Temperature Component Design and Fabrication
- Hot-Turbine Spin Rig Test (HTSR)

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San Tan Hydrogen Test Facility

*Facility Constructed for Development of SNTP Hydrogen-Related Components*

- Turbopump, valves, internal reactor components
- Hot, two-phase, and cryogenic hydrogen capability
- Dedicated facility for non-nuclear NTP testing
- Company-funded construction
The objective of the SNTP program is to develop advanced nuclear thermal propulsion technology based on the particle bed reactor concept. A strong philosophical commitment exists in the industry/national laboratory team directed by the Air Force Phillips Laboratory to emphasize testing in development activities. This presentation focuses on nuclear testing currently underway to support development of SNTP technology.
This is the summary test logic for NTP Development that has been generally accepted in the propulsion community provided resources are available. It is very consistent with the SNTP approach. Because of the limited time for the presentation, I will concentrate on critical assembly tests, fuel nuclear tests, and fuel element tests in loops in an existing reactor. Given time, I would discuss each box.
Critical Experiment (CX)

Purpose:

To obtain neutronic information for engineering needs (Safety, Mechanical Design, Control, Power Distribution, Weight, Reactor Design)

Accomplishments:

* Movable test reactor built - critical on October 24, 1989
* Excess reactivity - good agreement with predictions
* Peek-A-Boo control scheme feasibility demonstrated
* Control, safety, and shim element worth determined
* Moderator temperature coefficient determined
* Near-Term Priority:
  - Hot RHQ experiment completion
  - Cold moderator follow-on experiment
  - Para/Ortho hydrogen worth

In the critical assembly test category, an operating low-power reactor called CX was developed and is operated in the Sandia Pulsed Reactor (SPR) facility. The goal of CX is to perform nuclear measurements that allow for neutronic codes and cross-sections to be benchmarked so that design margins, controllability, and limiting accident scenarios may be predicted and assessed for PBRs with reliability and certainty. The SNTP team successfully fulfilled all approval requirements imposed by DOE for operating a critical assembly. The CX achieved first critical in October 1989. Eight experiment campaigns have been performed to date with over 100 operations logged. Several of the major completed experiments have been listed in this vugraph.
Sandia and the SNTP Team Built and Operate a Low-Power PBR for Reactor Physics Experiments

This is a photograph of the CX assembly with the water moderated removed.
Moving on to fuel and fuel element tests, this figure shows the experiment data flow for tests on these key nuclear components. Some of the test acronyms are as follows:

- **PHT** = Particle Heating Tests
- **PNT** = Particle Nuclear Tests
- **NET** = Nuclear Element Tests
- **PIPET** = PBR Integral Performance Element Tester
- **GTA** = Ground Test Article (System-level test)

This sequence follows the test logic of going from components to subsystems to systems. The stepwise data achieved in these tests is key technology development and validation.
Particle Nuclear Test (PNT)

PURPOSE:
To conduct in-reactor testing of fuel particles to provide design verification, identify potential failure modes, and evaluate effects of manufacturing process variables.

ACCOMPLISHMENTS:
- 4 Capsule Tests
- 12 Fuel Holders
- 200,000 Fuel Particles
- 1800 - 3100K
- 150 - 600 Seconds
- 1 - 4 Cycles
- Fission Product Release Measurements

STATUS:
- Performance limits determined for baseline fuel. Integrated fission product release Model operational. Additional tests planned as fuel becomes available.

PNTs are performed on fuel particles to provide design verification, identify potential failure modes, and evaluate effects of manufacturing process variables. Since this vugraph was prepared, a fifth capsule test was performed on baseline particle fuels. This vugraph shows the range of test parameters. A computer code called HEISHI is now operational at Sandia to predict particle performance and to estimate fission product releases. The PNTs provide a considerable amount of data for code validation.
This is a fuel holder used in PNT experiments conducted to date. It holds one cubic centimeter of fuel particles.
This is a photo of a PNT fuel holder with the top cap removed after a test.
Several (typical 3) fuel holders are placed in an experiment capsule for irradiation. The capsule is placed in the central cavity of the Annular Core Research Reactor (ACRR). The ACRR is a pulse-type reactor that serves as the driver core for creating the desired experimental conditions. After an experiment is complete, the fuel is removed for post-irradiation examination.
PNT Particle Bed After Nuclear Irradiation in the ACRR

This photo shows a PNT particle bed after nuclear irradiation in the ACRR.
Nuclear Element Test (NET)
Closed Loop, In-Reactor Test Of A Complete Fuel Element In Flowing Cryogenic Hydrogen

Purpose:
- Demonstrate integration of fuel element technologies
- Test to full temperature capability
- Validate fuel element designs
- Support
  - PIPET/GTA development
  - Fuel development
  - Model verification
  - Safety analysis

Accomplishments:
- Test hardware designed and fabricated
- Unirradiated experiment assembly and flow loop performance characterized in helium
- Test reactor (ACRR) control capabilities demonstrated

Status:
- Net-0 testing in cryogenic hydrogen
  - Summer 1992
  - First fueled test (NET-1) early CY 1993

The next step is to assemble particles into a complete fuel element that can be tested incore in flowing hydrogen. The NET experiments provide this demonstration of integrated fuel element performance up to the limits of what environments can be achieved in existing reactor facilities. Since this vugraph was prepared, the unirradiated tests with cryogenic hydrogen in the NET-0 experiment capsule have been successfully completed. Six weeks after receipt of a fuel element, a nuclear test can be completed.

A computer code, F2D, has been developed and is used to predict the thermal-hydraulic performance of the NET element. Codes such as this are being used by the SNTP program to evaluate key thermal-hydraulic issues.

At the systems level, the SAFSIM Code has been developed. A separate paper on SAFSIM is being presented in a later session.
This photo shows the fuel element configuration to be evaluated in the first NET test.
This photo shows the inside portion of a NET capsule. It would be surrounded by the capsule containments for in-core tests.
In closing, I would like to note that extensive testing has already been performed and more will be conducted in the future using existing reactor facilities. However, eventually we will reach the limits of what we can do in existing facilities. A new ground test facility will be required for testing elements and reactor/engine systems. This vugraph shows the functions that must be performed by this facility. The environmental process to be reviewed by the next speaker outlines a key contribution to the decisions defining the scope and location of this ground test facility.
SNTP ENVIRONMENTAL, SAFETY, AND HEALTH

NASA-LEWIS NP-TIM-92

Presenter
Charles D. Harmon

21 October 1992
PROGRAM SAFETY POLICY

- POLICY DOCUMENT PUBLISHED IN MAY 1992

- OVERALL OBJECTIVES:
  - TO ENSURE THE MAXIMUM PROTECTION OF THE HEALTH AND SAFETY OF THE PUBLIC AND SNTP WORKERS
  - TO PROTECT PROPERTY FROM DAMAGE OR LOSS
  - TO PROTECT THE ENVIRONMENT FROM CONTAMINATION OR DAMAGE AS A CONSEQUENCE OF SNTP ACTIVITIES

PROGRAM SAFETY POLICIES

- SAFETY AND ENVIRONMENTAL PROTECTION WILL BE EXPLICITLY CONSIDERED AND INCORPORATED THROUGHOUT THE LIFETIME OF EVERY SNTP PROGRAM ACTIVITY.

- THE SNTP PROGRAM SHALL MEET ALL MANDATED, STATUTORY, AND LEGAL REQUIREMENTS FOR SAETY AND ENVIRONMENTAL PROTECTION.

- ADDITIONALLY, EVERY PRACTICAL EFFORT SHALL BE MADE TO MAINTAIN RISKS DUE TO RADIATION AND TOXIC MATERIAL EXPOSURES AS LOW AS REASONABLY ACHIEVABLE (ALARA)

- COMPLIANCE WITH THESE REQUIREMENTS WILL BE BASED ON THE PRINCIPLES OF DEFENSE-IN-DEPTH INVOLVING MULTIPLE PHYSICAL, PROCEDURAL, AND ADMINISTRATIVE BARRIERS.
Space Nuclear Thermal Propulsion
EIS Process

DEIS PUBLIC HEARING COMMENTS

- NEPA PROCESS:
  - INSUFFICIENT REVIEW PERIOD
  - SCOPE COMMENTS NOT INCLUDED
  - POSTPONE PENDING RELATED NEPA ACTIVITIES
  - REQUESTED ADDITIONAL PUBLIC HEARINGS
  - INAPPROPRIATE SCOPE
  - INSTALLATIONS NOT EQUALLY REPRESENTED
  - INSUFFICIENT STATEMENTS OF PURPOSE AND NEED

- PROVIDES OPTIONS FOR TESTING ALTERNATIVE FUELS
- REQUESTED CLARIFICATION RELATIVE TO INDEMNIFICATION
DEIS PUBLIC HEARING COMMENTS (cont'd)

- Questioned relationship to previous classified documents
- Impacts on Yucca Mountain not analyzed
- Potential impacts on employment not analyzed

Alternatives:
- Personnel requirements underestimated
- All suitable sites have not been included
- Transportation issues inadequately addressed
- Meteorological prerequisites not specific
- Non-nuclear alternatives not considered

DEIS PUBLIC HEARING COMMENTS (cont'd)

- Hazardous materials:
  - High level waste stream not identified
  - TRU waste certification misrepresented
  - Waste reduction techniques not addressed
  - WIPP capabilities over estimated
  - Some disposal methods are illegal
  - Liquid waste streams not analyzed
  - RCRA-listed waste not specified
  - Limits of INEL RCRA-B exceeded
  - LDR wastes not identified

- Potential greenhouse effect not analyzed
DEIS PUBLIC HEARING COMMENTS (cont'd)

- Proposal insensitive to desert ecology
- Native American issues not addressed
- Geology and soils:
  -- Seismic activities underestimated
  -- Volcanic activities underestimated
  -- Flooding potential underestimated
- Water resources:
  -- Potential drought effects not analyzed
  -- Groundwater uncertainties not identified

DEIS PUBLIC HEARING COMMENTS (cont'd)

- Health and safety
  -- ALARA principle not sufficiently explained
  -- Heat related safety issues not identified
  -- Accident analyses not bounding
  -- Insufficient design detail - containment & control
  -- Insufficient design detail - ETS
  -- Control room habitability not defined
  -- Inappropriate correlations to chest X-rays
  -- Fallout consequences to flora/fauna not discussed
DEIS PUBLIC HEARING
COMMENTS (cont'd)

- HEALTH AND SAFETY (cont'd):
  - HYDROGEN SAFETY ISSUES NOT ADEQUATELY ADDRESSED
  - HYDROGEN EMBRITTLEMENT ISSUES NOT CONSIDERED
  - CONSEQUENCES TO SOUTHERN UTAH NOT SPECIFIED
  - METHODS TO CALCULATE EXPOSURE NOT EXPLAINED
  - QUANTITY OF REACTOR CORES ON-SITE NOT SPECIFIED
  - REQUESTED A DISCUSSION OF MILK MONITORING PROGRAMS
  - CLARIFY HAZARDS OF LONG TERM LOW-LEVEL EXPOSURES

DEIS PUBLIC HEARING
COMMENTS (cont'd)

- HEALTH AND SAFETY (cont'd):
  - NO-THRESHOLD-LEVEL CONCEPT NOT APPLIED
  - INVERSION LAYER EFFECTS ON Be RELEASES NOT DISCUSSED
  - BACKGROUND DOSES ARE TOO HIGH
  - LACK OF RAIL SYSTEMS NOT IDENTIFIED
  - PROGRAM BENEFITS NOT COMPARED TO RISKS
  - EFFECTS FROM HIGH VOLTAGE LINES NOT DISCUSSED
  - DOE SAR PROCESS INADEQUATE
  - 20% NESHAP REPRESENTS SIGNIFICANT INCREASES
DEIS PUBLIC HEARING
COMMENTS (cont'd)

o HEALTH AND SAFETY (cont'd)

- 170 mRem LIMIT NOT EXPLAINED
- CHARTS DO NOT SUPPORT DECREASING DOSE CLAIMS
- 1980 CENSUS DATA USED FOR INEL

SPACE NUCLEAR THERMAL PROPULSION PROGRAM
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### SPACE NUCLEAR THERMAL PROPULSION PROGRAM
#### EIS SCHEDULE

**Update:** 10/31/92

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**Page 4**
# Nuclear Propulsion Technical Interchange Meeting

**Volume I**

**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.**

## Subject Terms
- Nuclear electric propulsion; Nuclear thermal propulsion; Nuclear propulsion; Nuclear rocket engines; Nuclear research and test reactors; Manned Mars mission; Test facilities; Models

## Security Classification
- Unclassified

## Number of Pages
- 568

## Security Classification of This Page
- Unclassified

## Limitation of Abstract
- Unclassified