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SPACE STATION COMMONALITY ANALYSIS

MODIFICATION No. 11 to NASA CONTRACT NAS8-36413 DPD-658, DR-03

Boeing Aerospace
Huntsville, Alabama

Final Report

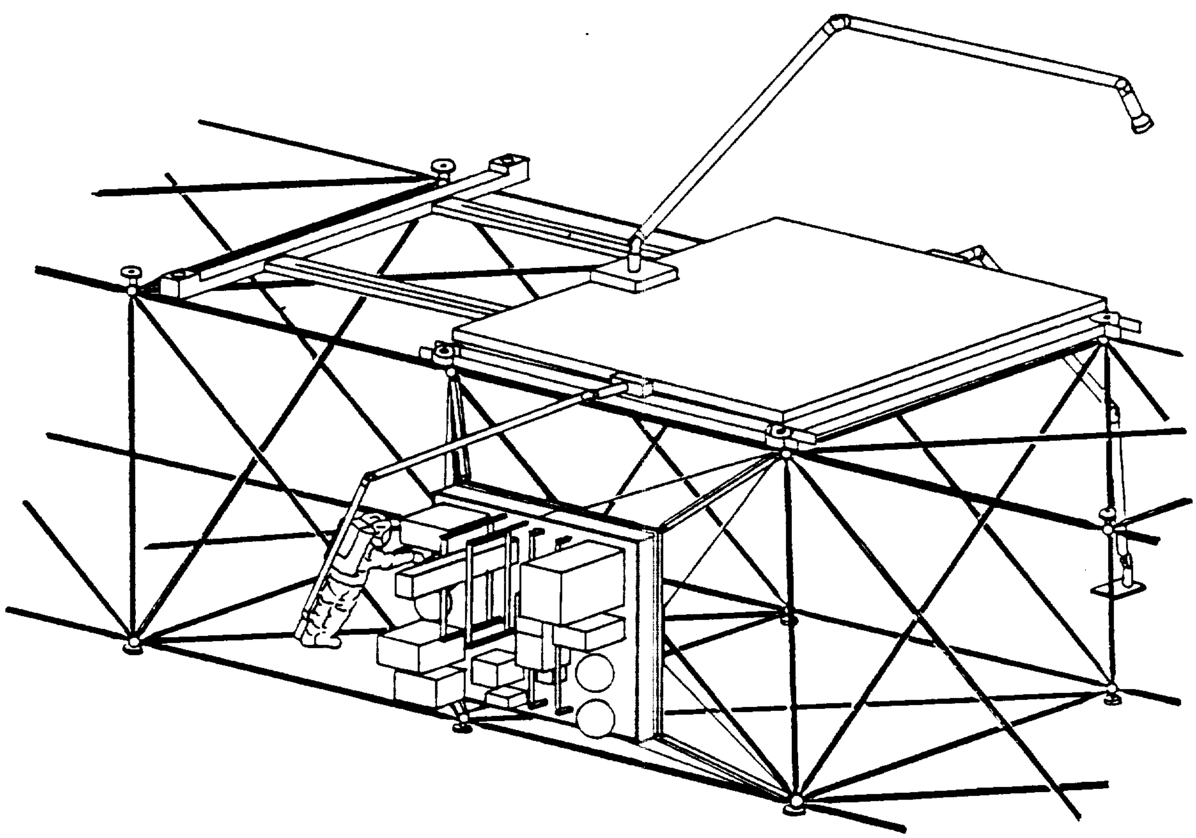

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TABLE OF CONTENTS

FIGURES

COMMONALITY ANALYSIS CONTRACT NAS8-34613
MODIFICATION NO.11 TECHNOLOGY EXPERIMENTS COMMONALITY

1.0 STUDY BASIS AND OBJECTIVES

This study was conducted on the basis of a modification to Contract NAS8-36413, Space Station Commonality Analysis. This effort was initiated in December, 1987 and completed in July, 1988.

The objective of the study was to investigate the commonality aspects of subsystems and mission support hardware while technology experiments are accommodated on board the Space Station in the mid-to-late 1990s, considering two types of missions:

(1) Advanced solar arrays and every storage, and

(2) Satellite Servicing.

The point of departure for definition of the technology development missions was a set of missions described in the Space Station Mission Requirements Data Base. (MRDB):

TDMX 2151 Solar Array/Energy Storage Technology

TDMX 2561 Satellite Servicing and Refurbishment

TDMX 2562 Satellite Maintenance and Repair

TDMX 2563 Materials Resupply (to a free-flyer materials processing platform)

TDMX 2564 Coatings Maintenance Technology

TDMX 2565 Thermal Interface Technology

Issues to be addressed according to the Statement of Work included modularity of programs, data base analysis interactions, user interfaces, and commonality. The study was to consider state-of-the-art advances through the 1990s and to select an appropriate scale for the technology experiments, considering hardware commonality, user interfaces, and mission support requirements. The study was to develop evolutionary plans for the technology advancement missions.

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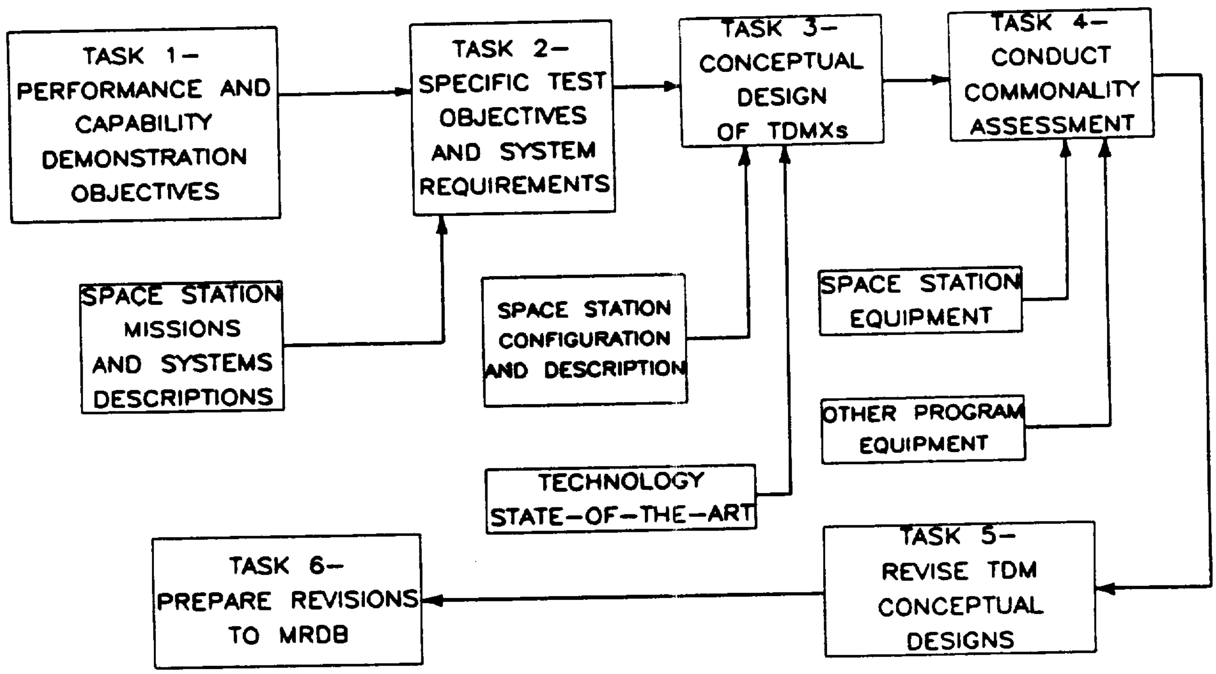
2.0 SUMMARY OF TECHNICAL APPROACH

The technical approach is diagrammed in figure 2-1. We began the study in each mission area by establishing performance and capability demonstration objectives. These were then extended to specific test objectives and system requirements by systems engineering functional analysis and requirements allocation, taking into account Space Station systems and their capabilities. For these early tasks, we used the information available in the MRDB. We began our commonality investigation by evaluating commonality of requirements and concluded that there was enough commonality among TDMXs 2561, 2562, and 2563 to merit combining them into a single mission.



Based on requirements, we conducted conceptual designs of the technology development missions, taking into account the existing and projected technology state-of-the-art in each area, and using Space Station configuration and capability information to guide the definition. Also, we applied commonality ground rules to use Space Station or other off-the-shelf hardware wherever practical. We then conducted commonality assessments to look for other areas where commonality could be applied. Since we began with a ground rule of commonality, this was a minimal task. Finally, we revised the TDM conceptual designs and prepared revisions to the MRDB. In the case of satellite servicing, the revisions to the conceptual designs responded mainly to a desire to develop a more evolutionary approach to the technology development.

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3.0 ROADMAP TO THE FINAL REPORT

This report is organized in three principal sections: Sections 4 presents results of the analysis of solar array/energy storage mission; Section 5 presents results for satellite servicing, including TDMXs 2561, 2562 and 2563; and Section 6 presents results for coatings maintenance, TDMX 2564. We concluded as summarized in Section 7 that thermal interface technology development, TDMX 2565, could be accomplished through ground testing in vacuum chambers, with results of the development fed into the satellite servicing technology program when ground-based development is complete. Section 8 presents overall conclusions of the study.

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4.0 SOLAR ARRAY/ENERGY STORAGE TECHNOLOGY (TDMX 2151)

4.1 Objectives and Benefits of Technology Advance

The large area of a solar electric power system for a Space Station in low Earth orbit is by far the greatest contributor to drag forces on the Space Station. Drag causes gradual decay of the orbit, requires resupply of propellant for reboost to cancel the effects of orbit decay, and effectively sets the minimum operating altitude for a Space Station based on minimum acceptable orbit lifetime without reboost. The possibility of significantly reducing solar collector area through increase in solar-to-electric conversion efficiency has been the principal motivator for future development of solar thermal-cycle electric power systems. However, if the operating efficiency of a solar photovoltaic array could be increased to about 21% (compared to 12% - 14% for the Space Station baseline), the potential for drag reduction would rival that likely attainable by thermal-cycle systems. Solar array technologies now in the laboratory have the potential for such efficiency increases.

While the solar collector creates the principal size and area problems for Space Station power, the energy storage system (batteries) contributes most of the weight. Advances in battery technology could significantly reduce the weight of the electrical power system. If improvements in battery efficiency can also be obtained, these will help to reduce array area.

Advances in battery technology have implications far beyond those applicable to Space Station. Batteries now in the laboratory promise improvements in energy storage specific energy (kilowatt-hours per kilogram) as much as a factor of six better than present technology. Significant weight savings in future satellites, platforms, and manned exploration vehicles will occur. Since performance of these systems is often weight-limited, energy storage weight savings can greatly increase achievable performance.

4.2 Study Objectives and Task Summary

The objective of this part of the study was to define a solar array/energy storage technology development mission (TDM) to be accommodated on the baseline Space Station. This included the development of the mission objectives, description and operations to a level sufficient to identify reference Space Station support and commonality requirements.

The study provided a preliminary definition of a solar array/energy storage TDM to be performed on the revised baseline configuration Space Station. It identified the technologies which will be ready for development and demonstration in the mid-1990s time frame, identified mission commonality requirements, presented a program plan to develop a Space Station mission, and provided data for incorporation into the Space Station Mission Requirements Data Base (MRDB) as TDMX2151.

The objectives of the study were accomplished by performing the following six tasks:

Task 1 - Set Performance Goals: The objective of this task was to develop a set of specific demonstration goals for a solar array and energy storage technology development mission. A state of the art assessment was conducted to select technology options to be considered and to assess technology readiness by the mid-1990s. Technology options were selected based on reasonable expectation of achieving TDM goals and objectives.

Task 2 - Develop TDM Requirements: The objective of this task was to develop a preliminary TDM requirements baseline to serve as a basis for conceptual design. The selected technologies were reviewed to determine the requirements necessary for their operation. TDM requirements development was conducted in parallel with initial steps of conceptual design in order to ensure that requirements were compatible with reasonable mission/system designs.

Task 3 - Conceptual Design of TDM: Configuration general arrangements for the TDM flight experiment were developed based on TDM requirements allocated to flight experiments, and on the Space Station Phase 1 baseline configuration. General arrangements were reviewed for compatibility with Space Station operations, and Space Station configuration constraints. TDM operations concepts were defined that include delivery, installation and checkout, experiment operations and data acquisition.

Task 4 - Conduct Commonality Analysis: The objective of this task was to develop a cost-effective level of commonality between the TDM and Space Station for equipment, procedures, support equipment, and mission/payload control requirements.

Task 5 - Revise TDM Conceptual Design: The objective of this task was to produce the final configuration, description, equipment list, supporting data, and program descriptions and schedules for the TDM, incorporating commonality concepts defined in Task 4.

Task 6 - Prepare MRDB Inputs: The objective of this task was to prepare inputs to the Mission Requirements Data Base (MRDB) to document the mission description information and requirements.

The five tasks were accomplished in approximately three months as shown in the accompanying schedule Figure 4.2-1 An interim review was held April 21, 1988, approximately one month after work began on this portion of the study. The final review was held June, 1988.

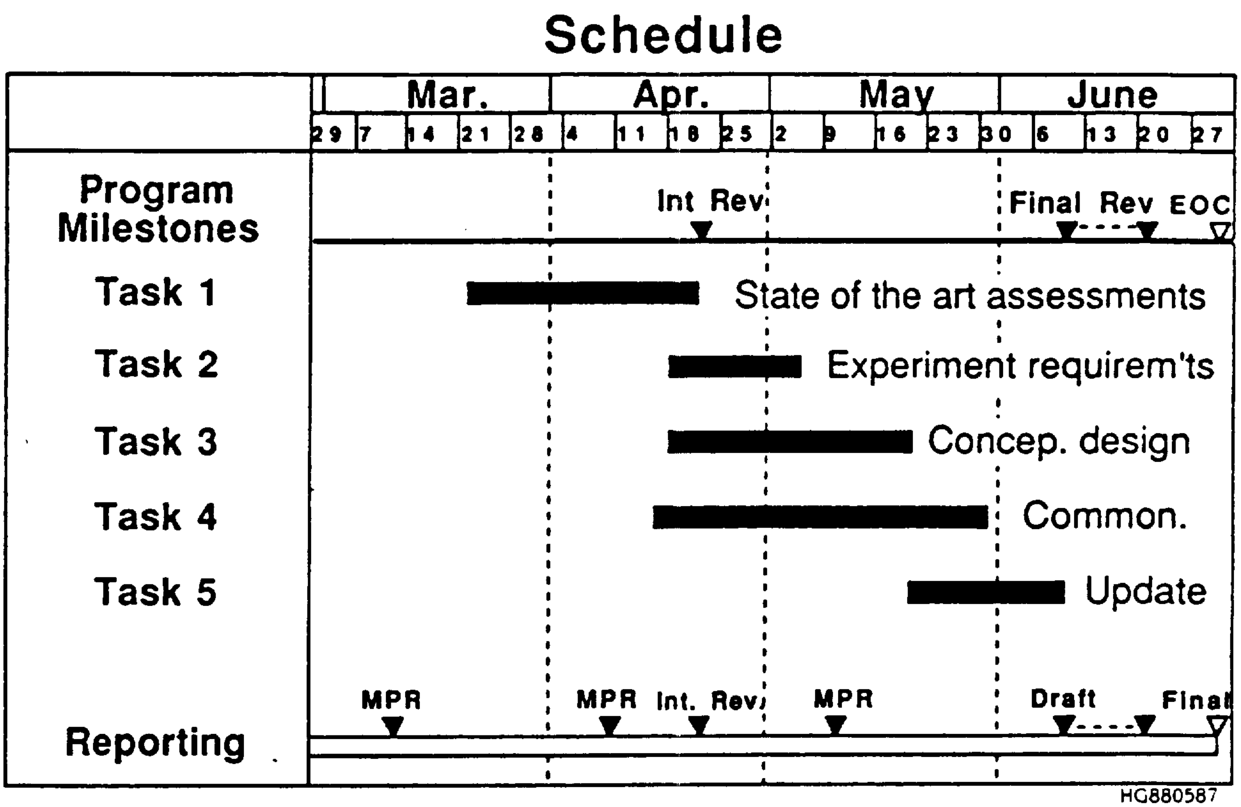


Figure 4.2-1. Space Station Commonality Analysis Solar Array/Energy Storage Technology

At the beginning of the study, a number of ground rules were established regarding the location of the experiment on Space Station, the capabilities of the experiment and the technologies that would be considered.

First, because of the limited amount of space available on the Phase I Space Station and the necessity for continuous pointing at the Sun, the baseline location for the experiment is on the main Space Station power boom, outboard of the alpha-joint.

It was further presumed that the Space Station will operate in a drag cancellation mode (continuous or nearly continuous thrusting to overcome aerodynamic drag). If the Space Station operates in a periodic reboost mode, a second panel will be required to balance the aerodynamic moment about the center of mass. Drag calculations are summarized in Figure 4.2-2. (The second panel need not be electrically functional.)

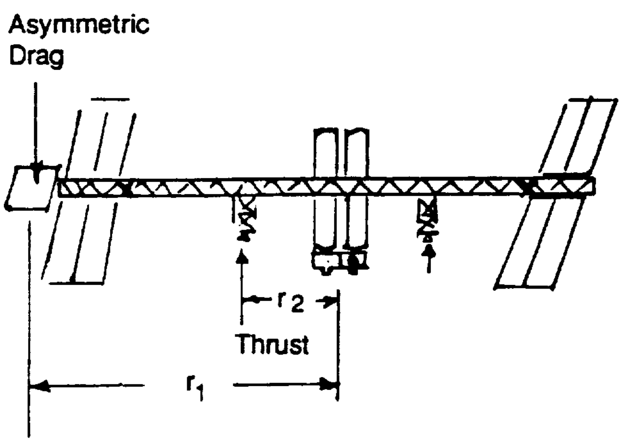
LOG_{10} ATMOSPHERE DENSITY AT 400 KM = -10.7 (SHORT TERM MAX)
= -11.7 (NOMINAL)

$$\text{AERO Q(STM)} = \tfrac{1}{2} \times 10^{-10.7} \left(\frac{398601}{6378+400}\right) \times 10^{6} = 5.8 \times 10^{-4}\ \text{PA} = 1.2 \times 10^{-5}\ \text{PSF}$$

AREA = 50 M^2 ARM $\simeq$ 50 M CD $\simeq$ 2.2

WORST-CASE DRAG = $5.8 \times 10^{-4} \times 50 \times 2.2 = 0.06$ N (0.014 LB_F)

TORQUE $\cong$ 3 N-M $\simeq$ 10,000 N-M-SEC/REV



DRAG MAKE-UP THRUST IS ADEQUATE AS LONG AS

$\Delta/A > R_1/R_2 \geq 4$

WHERE Δ IS NOMINAL SPACE STATION DRAG AREA, A IS TEST PANEL AREA.

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Figure 4.2-2. Drag Analysis

The solar array was sized to provide enough power to adequately demonstrate solar array and energy storage technologies. A precursor STS flight experiment for the selected energy storage and solar array technologies is recommended.

Standard Space Station cabling will be used whenever possible to provide commonality and to be consistent with the Space Station power management and distribution system (PMDS). The experiment will have the capability of contributing power to the PMDS.

No thermal energy storage techniques were considered in this study. The energy storage technologies considered were: batteries, regenerative fuel cells, super conductor magnetic storage and flywheels.

4.3 Performance Goals

4.3.1 State-of-the-art assessment

A state-of-the-art survey was performed on solar array technology and energy storage technology. Solar array technology considered both the solar cell and the solar panel/wing.

Leading candidates for solar cell evaluation are tabulated in Figure 4.3.1. These include indium phosphide and tandem cells because of their potential for high performance. Indium Phosphide cells can also be annealed at temperatures down to 100° C.

Solar Cells

- InP solar cells
- Tandem solar cells
  - GaAs/Germanium
  - CLEFT cells
  - $CuInS_2$
- Amorphous silicon cells
- V-groove GaAs cells

Solar Array

- High voltage
- On-array annealing
- Solar/Thermal/Photovoltaic (STPV)
- Solar concentrators
  - Mini cassegrainian concentrator (MC^2)
  - Light funnel
  - Newton
  - Fresnel lens
  - Slats

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Figure 4.3-1. Candidate Concepts For Solar Cells And Arrays

Leading candidates for solar array concepts included concentrators because of their high unit area performance, and the solar/thermal/photo-voltaic (STPV) which may have the highest performance capability of all.

Good efficiency and low cost are the primary discriminators. Structural and electrical impact are secondary in nature because no significant impact is anticipated for the concepts considered. Weight is a secondary discriminator because the potential concepts would not impose large weight penalties.

Indium phosphide solar cells are expected to have an efficiency of 21%. They have a low sensitivity to charged particle degradation, and they can be annealed at 100° C. At this time, they are early in their development, and much work is to be done before they are ready for space application.

Tandem (multiple band gap) cells are under development by a number of companies and are expected to have performance efficiencies in the low 20s. Existing cells are mechanically bonded together, but monolithic cells are under development. Current projected performance levels are presented in Figure 4.3-2. They are expected to be relatively expensive.

Year	Comp-onent	BOL 28° Efficiency	Operating Temp.	Temp. Factor	Radiation Deg. Factor 1E15 e/cm^2	Σ of UV, Micro-meteorite, Test, Mismatch Mech.Damage	EOL Operating Efficiency
1987	GaAs CIS Module	17 3 20	55°C	.93 .84	.85 1.0	.90 .95	12.09 2.39 14.5
1988	GaAS CIS Module	18 4 22	53°C	.93 .85	.85 1.0	.90 .95	12.81 3.23 16.0
1992	GaAS CIS Module	20 4 24	51°C	.94 .86	.85 1.0	.90 .95	14.38 3.27 17.7
2000	GaAS CIS Module	23 4 27	48°C	.95 .88	.85 1.0	.90 .95	16.72 3.34 20.1

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Figure 4.3-2. Tandem Cell/Module Performance Analysis

Amorphous silicon cells are low in cost and have reduced radiation degradation. However, their low efficiency results in increased area and drag. For this reason they are not recommended for the Space Station application where drag reduction is important.

The surface of V-Grooved GaAs Solar Cells is grooved to obtain improved light absorption. (Light is refracted around the front-surface contact fingers.) They also have improved radiation tolerance. The major disadvantage of these cells is increased manufacturing cost.

High-voltage solar arrays are feasible using encapsulation techniques, zero-potential ground planes, or a combination of these features. The major advantage of a high voltage design is the reduced I^2R losses. The zero-potential ground plane is probably the preferred method of achieving plasma protection because it is not vulnerable to micrometeoroid damage, as is an encapsulated array. In either case, manufacturing complexity is increased, and a high voltage design would require a redesign of the Space Station solar array power conditioning equipment.

Solar cells can be annealed during a mission to reduce charged particle radiation degradation by building local heaters into the solar array design. Alternatively, an external annealing heater could be used. The annealing would be done sequentially on groups of cells with one to two minutes required per group. The built-in heaters increase manufacturing complexity, and space station housekeeping would be provided with an additional task during the annealing process. Charged particle degradation is not a primary issue for Space Station solar arrays. While demonstration of annealing on a Space Station experiment would be useful, this process is probably not a candidate for actual Space Station applications.

The Solar/thermal/photovoltaic (STPV) concept provides a spectral shift that transforms the solar spectrum to one more aligned with the spectral response of the solar cell. This concept has the potential of achieving efficiencies greater than 35 percent, and this would be significant in terms of reduced area and drag. Development work is slow, and a sample test needs to be performed in orbit to demonstrate the capability of this concept. Improved solar array pointing accuracy would be required, and testing has proved difficult. Presently, this concept is not a prime candidate for technology demonstration during early Space Station operations.

Solar array concentrators offer improved performance per unit area, but also increased unit weight. Most concentrators require improved pointing accuracy, but some do not. Some concepts, but not necessarily all, could result in increased cost.

Weapon-hardened concentrator concepts were judged to be too heavy for space station application.

Integral solar cell covers have been under development for years with little success. One major problem has been the difference in thermal expansion coefficients between the cover and the cell. This is significant when covers are electrostatically bonded to the cells. A second problem has been the inability of the cover material to stand up to the space environment. This has been the primary problem of spray coatings or coatings applied by other means.

Two integral cover concepts being investigated were considered. A cerium-doped glass (CMZ glass) is being developed by Pilkington (of England) for use with electrostatic bonding. This material has a thermal expansion coefficient very close to that of the cell. The second material is a silica/alumina mixture being developed for the Air Force. The thermal expansion coefficient of this material can be varied by changing the mixture ratio.

Our conclusions for solar array technology are:

- Improved solar cells and covers can be incorporated into the space station solar array with minimal impact, and therefore should be considered for future application.

- High-voltage solar arrays are feasible, and could reduce I^2R losses, but the reduction in solar array area would not be worth the penalties of increased manufacturing complexity and a redesign of Space Station power conditioning equipment.

- On-array annealing of solar cells is feasible but may not have a significant payoff for Space Station because of the low charged particle environment.

- Concentrator concepts can provide improved unit area performance with reduced drag. However, a stiffer structure is required to achieve proper pointing. This could result in an overall weight increase.

A state-of-the-art survey was performed on energy storage devices, and concepts were identified as potential options for advanced Space Station power systems.

Three main categories exist as summarized in Figure 4.3-3: chemical batteries, regenerative fuel cells, and flywheels. These devices were evaluated on the basis of technical performance, weight, volume, and cost.

Energy Storage

- Chemical batteries
  - Sodium/Sulfur (NaS)
  - Rechargeable lithium
    - Lithium/Titanium Disulfide (Li/TiS_2)
    - Lithium/Molybdenum Disulfide (Li/MoS_2)
  - Nickel/Hydrogen (NiH_2)
- Regenerative fuel cells
  - H_2 - O_2
  - H_2 - Br_2
- Flywheels
  - Metals
  - Composite materials

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Figure 4.3-3. Solar Array/Energy Storage Candidate Concepts

Primary considerations for energy storage are high energy density (to reduce weight and volume) and a large number of charge/discharge cycles. For a 5-year LEO mission, nearly 30,000 charge/discharge cycles are required. Another important parameter, low impedance, reduces heat generation and also reduces size. Durability, simplicity, and low cost are always concerns for any energy storage device.

Sodium sulfur batteries have excellent energy densities, at least five times greater than the Space Station baseline nickel hydrogen batteries (120 WH/kg@ 60% depth of discharge, predicted). The Air Force has issued contracts for the development of cells with a capability of high cycle life, and for fully functional space-qualified batteries. Eagle-Picher Industries and Hughes Aircraft Company are in the process of developing sodium sulfur 50AH cells/batteries. More than 5000 cycles have been demonstrated. Development is on schedule, and space-qualified batteries are expected to be available in the mid-1990's.

The main disadvantage of the sodium-sulfur battery is the high operating temperature (350°C) of the cell itself, although the external surface of the cell will not be hot.

A typical sodium sulfur cell is shown in figure 4.3-4. The cell is long and cylindrical, and operates at approximately 350°C. The open-circuit voltage is approximately 2.08 volts, and a nominal voltage under load is approximately 1.85 volts.

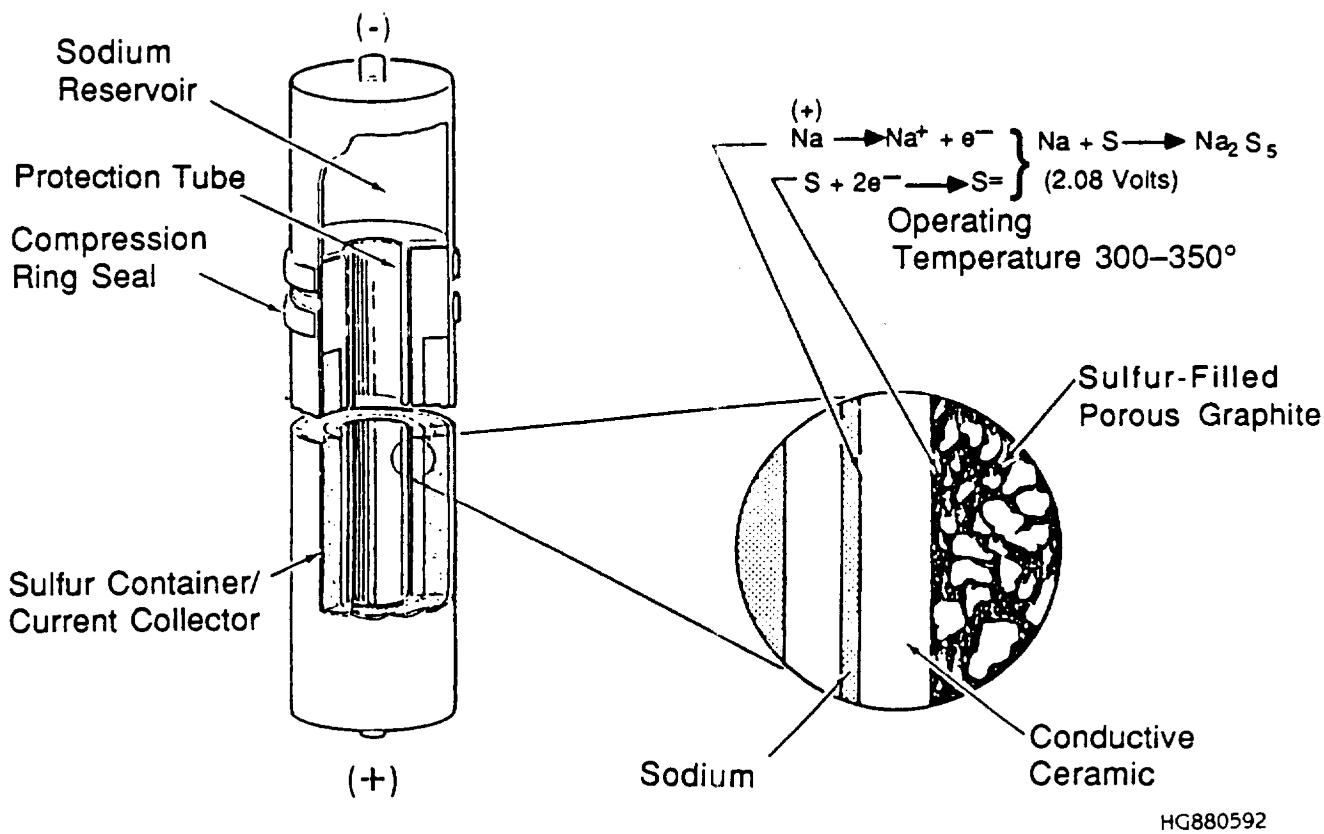

Figure 4.3-4. Sodium-Sulfur Cell

Rechargeable lithium batteries also have good energy densities and excellent charge retention capabilities. However, they are not being developed at a rate comparable to sodium sulfur development.

Their advantages are:

- Low self discharge; excellent charge retention
- 70-85 WH/Kg energy density

Disadvantages are:

- Limited cycle life, 100-300 cycles

  (1000-3000 cycles possible at 50-30% DOD)
- Lithium is a flammable metal - safety concerns
  - Thermal management crucial issue
- Only 5AH exprimental cells available

  (50 AH for Li/MoS^2)
- Wide voltage range 2.6 to 1.6V
- Not space qualified

Therefore, the number of charge/discharge cycles is significantly lower than that required for a LEO mission. Much work and significant funding would be required to space-qualify a rechargeable lithium battery by the mid-1990's. This possibility does not appear feasible at this time.

A 50 AH rechargeable lithium cell using a molybdenum disulfide cathode is featured in Figure 4.3-5. Cell components are spirally wound around a central mandrel. Thermal management is a critical issue in lithium cell design, as lithium is a very reactive metal and potentially dangerous if not handled properly. The operating voltage range is wide, 2.7 to 1.6 volts, so voltage regulation is an important factor to consider.

50 AH Aerospace Cell Design Uses Central Mandrel

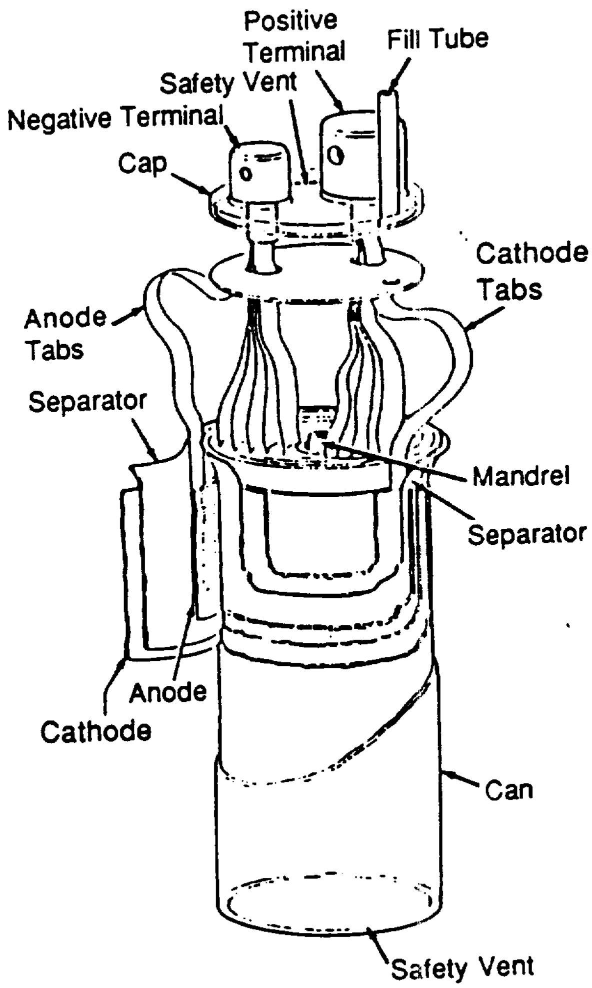

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Figure 4.3-5. Rechargable Lithium Molybdenum Disulfide Cell

Bipolar nickel-hydrogen batteries are an advanced concept of a proven electrochemical couple. The new design provides greater energy densities at twice the depth-of-discharge available with standard nickel-hydrogen batteries. These batteries feature reduced packaging weight and improved cooling.

Figure 4.3-6 depicts a simplified bipolar cell stack. The entire cell stack is placed inside a single pressurized vessel. A nominal discharge voltage is approximately 1.25 volts per cell.

Exploded View of a Typical Bipolar Stack Arrangement

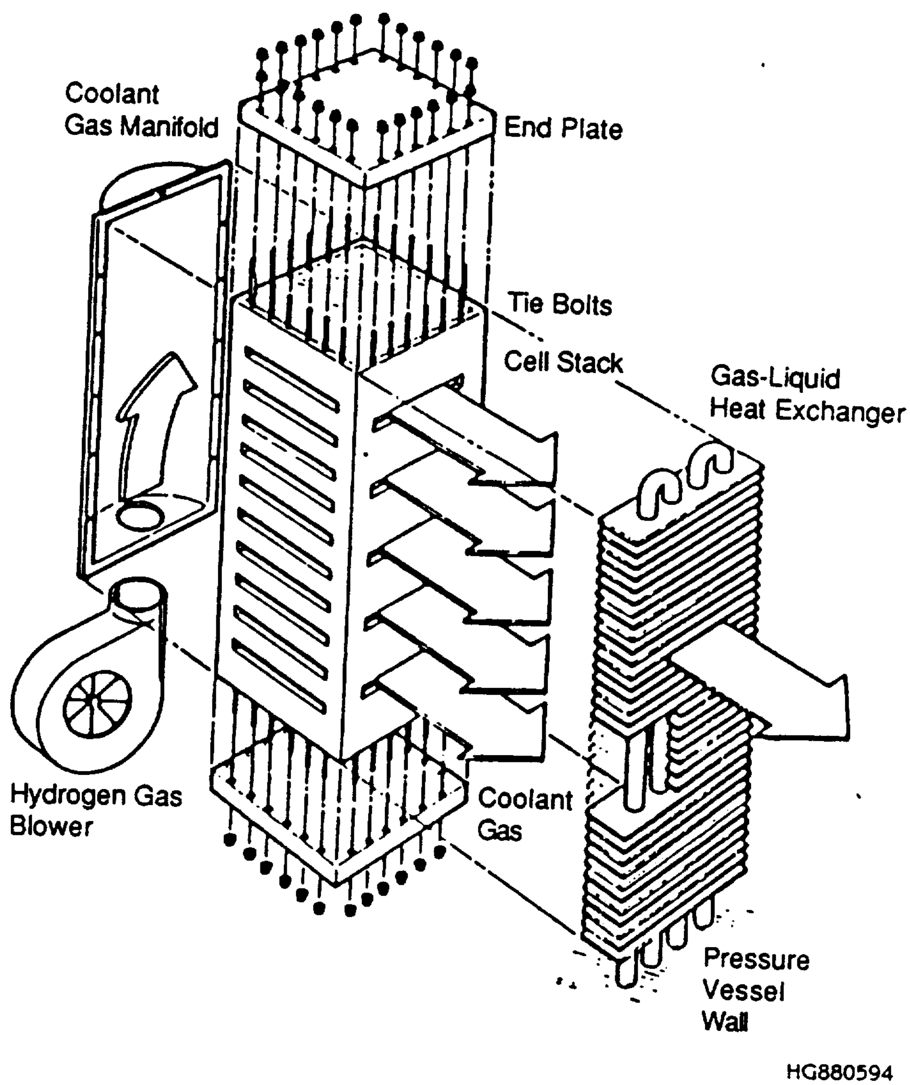

Figure 4.3-6. Advanced Nickel Hydrogen

Regenerative fuel cells have received considerable attention in the past few years as potential energy storage devices for space applications. They have a high power capability and can withstand large numbers of charge/discharge cycles. Difficulties have been encountered, however, as a result of the complexity of the devices.

The very high energy density potential of regenerative fuel cells is only realized for long energy delivery times such as might be the case for a lunar base. For the Space Station light/dark cycle they offer little advantage.

A schematic diagram of a hydrogen-oxygen regenerative fuel cell is shown in figure 4.3-7. The hydrogen and oxygen tanks could be cooled cryogenically to reduce weight and volume, but this adds significantly to the complexity.

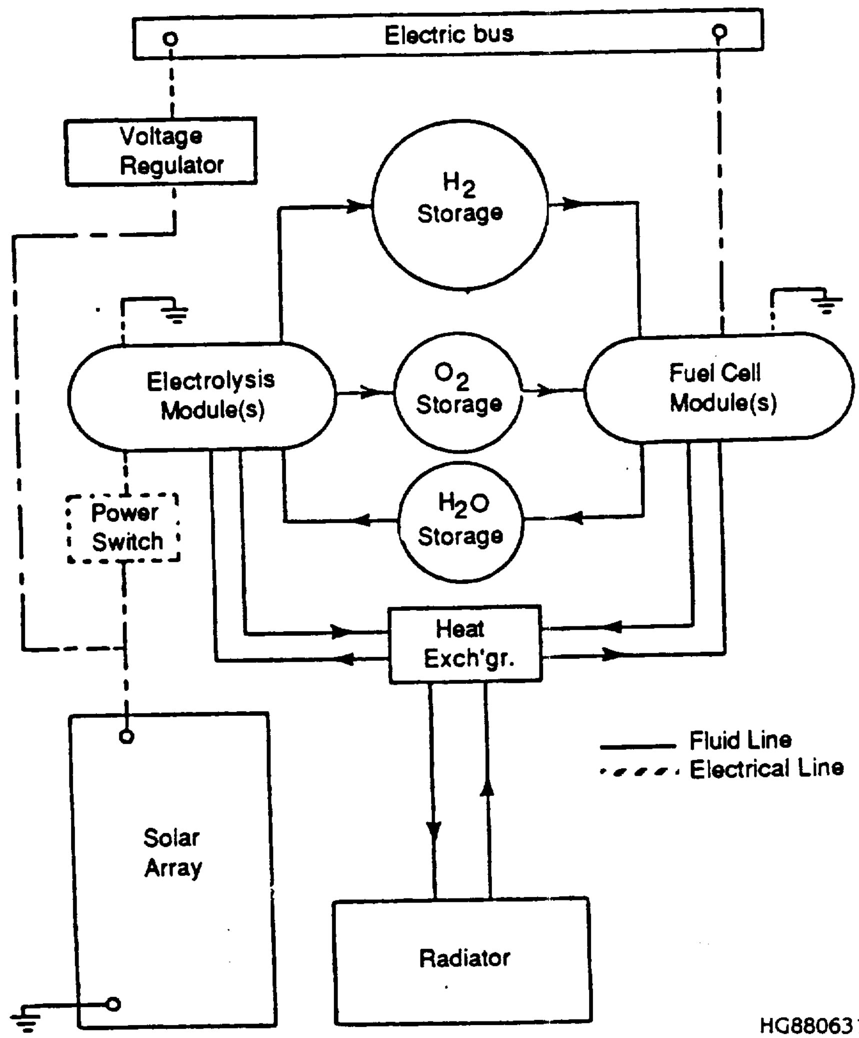

Figure 4.3-7. Regenerable Fuel Cell

Flywheels have good efficiency, and the development of new, stronger, lightweight materials provide the potential of high energy densities and long shelf lives. The devices are relatively complex, however, and do require some maintenance. Additional complications exist in counteracting the torquing and angular momentum effects of a single operating device.

New advanced energy storage devices have higher energy densities and can therefore reduce the weight and volume of existing space station nickel-hydrogen batteries. Based on present state-of-the-art technology and development work planned for the near future, it was decided that sodium sulfur batteries and bipolar nickel-hydrogen batteries are the candidates most likely to be available and space-qualified in the mid-1990's timeframe.

4.3.2 Technology Selection

Our selections for TDMX2151 experiment included a primary and secondary concept for both the solar array and energy storage:

- Concentrators
- Advanced Solar Cells
- Sodium Sulfur Batteries
- Bi-Polar Nickel Hydrogen Batteries

Selections were based primarily on their performance capability and their present development level.

4.3.3 Technology Readiness Assessment

A fresnel lens concentrator was selected as the primary solar array concept because of its relatively high power per unit area and light weight. This concept is being developed by Entech, Inc, with funding by NASA/Lewis.

The Dome Lens Photovoltaic Module conceptual design is illustrated in Figure 4.3-8. The dome is made of glass, with the fresnel lens bonded to the inside. The fresnel lens is presently made of DC92-500 silicone adhesive, but future plans call for the development of a glass lens. In either case, the lens is not subject to degradation from atomic oxygen because it is located on the inside of the dome. The dome is made of CMX or CMZ glass and will stand up well to the space environment. A GaAs cell is used and will see a radiation intensity of approximately 109 suns. The purpose of the cell cover is to improve the cells absorptivity by directing sunlight (with prisms) around the grid lines of the cell.

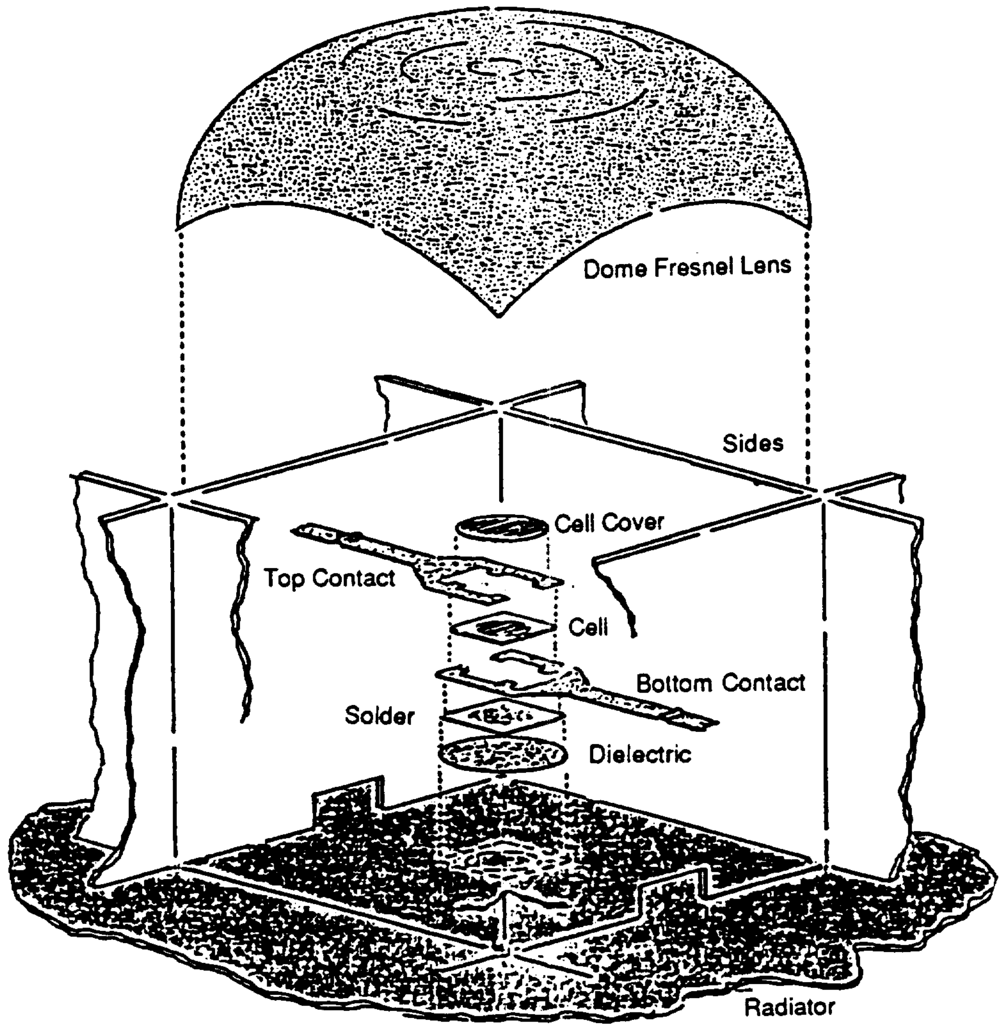

Figure 4.3-8. Dome Lens PV Module Conceptual Design

Electrical contacts are made to the top and bottom of the cell by solder bonds. Only a dielectric and appropriate adhesives separate the cell from the backside radiator. The sides of each unit concentrator are designed to properly position the dome/lens and to provide structural stiffness to the panel section.

A panel section has 14 concentrator units on a side for a total of 196 units per section as shown in Figure 4.3-9. Considering the unit size, this results in a section size of 0.518 M on edge with an area of 0.268 square meters.

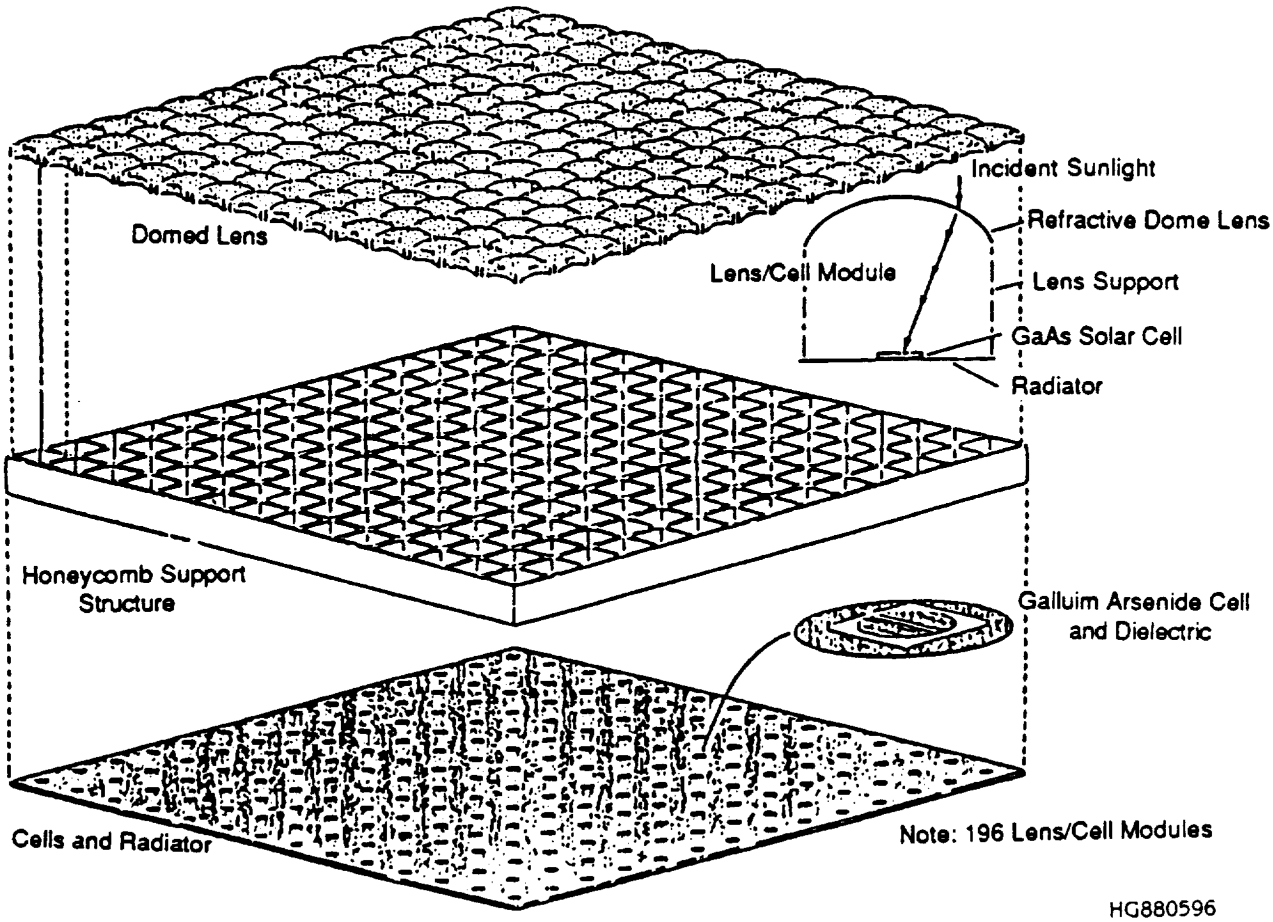

Figure 4.3-9. Panel Section Design

Each concentrator unit measures 3.7 cm on a side, for an area of 13.7 sq cm. Considering concentrator losses and cell performance, each unit provides 360 mw of power.

A weight breakdown is given in Figure 4.3-10 for each component for a basic weight of 2.5 kg per square meter. Studies of structural requirements (for other applications) have indicated a structural weight of 0.7 kg per square meter, thus producing a net weight of 3.2 kg per square meter.

Unit	
Area	= 13.7 cm^2
Conc Ratio	= 109
Power Out	= 360mw

Section

Number of Units = 14 × 14 = 196

Size = 3.7 cm × 14 = 0.518M

Area = $(0.518M)^2$ = $268M^2$

Power = 196 units × 0. 36w each = 70.56 w

Unit Power = 70.56w/0.268M^2 = 262 W/M^2

$$\text{Unit Weight} = \frac{(262.96\ W/M^2)}{(3.2\ kg/M^2)} = 82.2\ W/kg$$

Weight	(Kg/M^2)
Lens	0.68
Radiator	0.54
Cell/Mount/Intercom	0.05
Honeycomb Matrix	0.62
Attachments	0.14
Adhesions	0.34
Misc.	0.13
Subtotal	2.5
Structure	0.7
Total	3.2 Kg/M^2

CP GX/GW/D127/DISC 41

Figure 4.3-10. Basic Fresnel Lens Concentrator Parameters

Considering the unit size, the 196-unit panel section has a section size of 0.518 M on edge with an area of 0.268 square meters. The panel section provides a power of 70.56 watts which is equivalent to 262 watts per square meter. Considering the unit weight given above, this results in 82.2 watts per kilogram.

The relative solar irradiance profile, as received at the surface of the cell, is plotted in Figure 4.3-11. The cell is oversized to allow for a reasonable orientation error. An allowable orientation error of 5 degrees would enable the experiment to use the presently prescribed alpha and beta joints for solar array orientation.

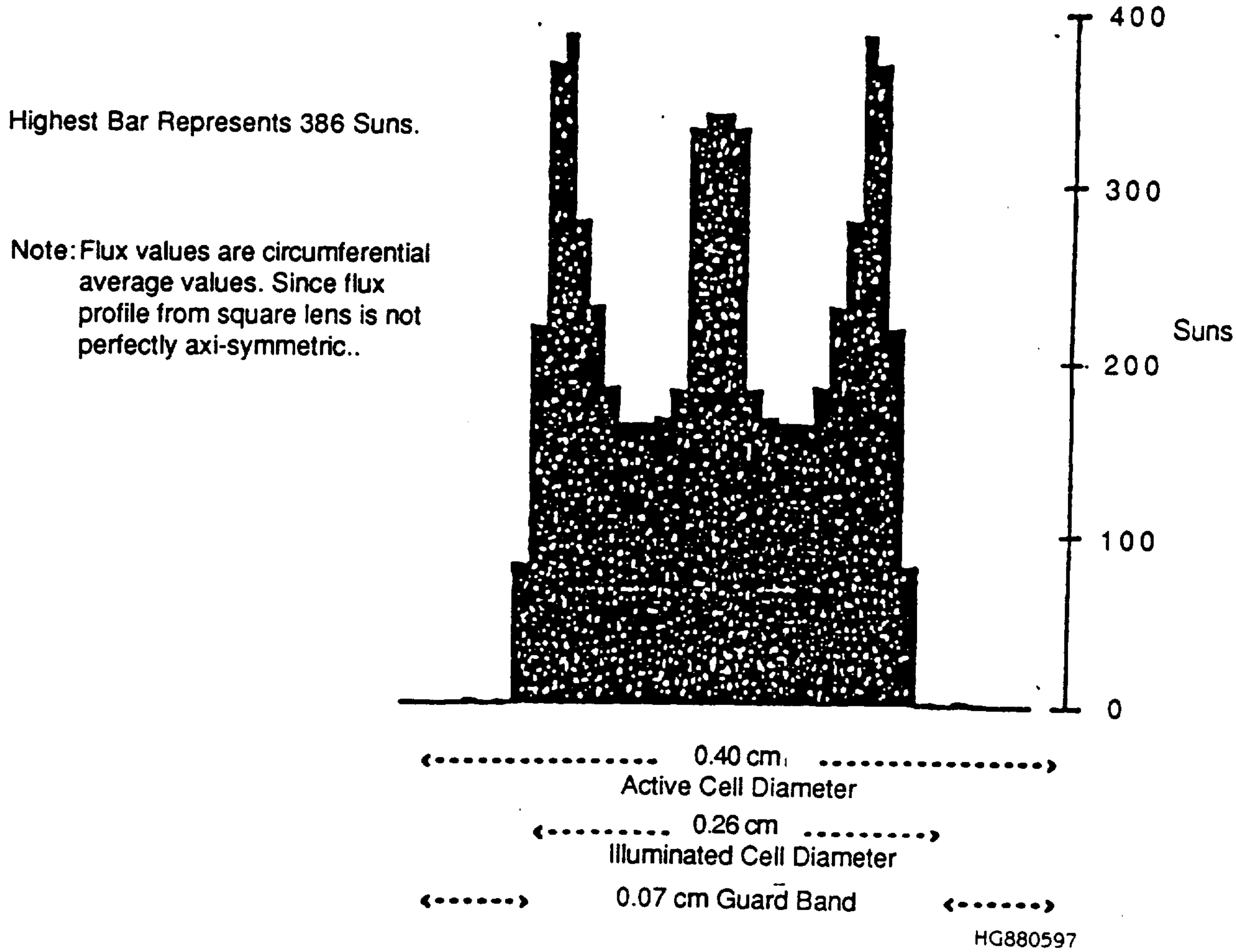

Figure 4.3-11. Calculated Irradiance Profile For Dome Lens

Figure 4.3-12 shows the selected design of square-aperture dome lens optical concentrater. The solar cell is located at the focus.

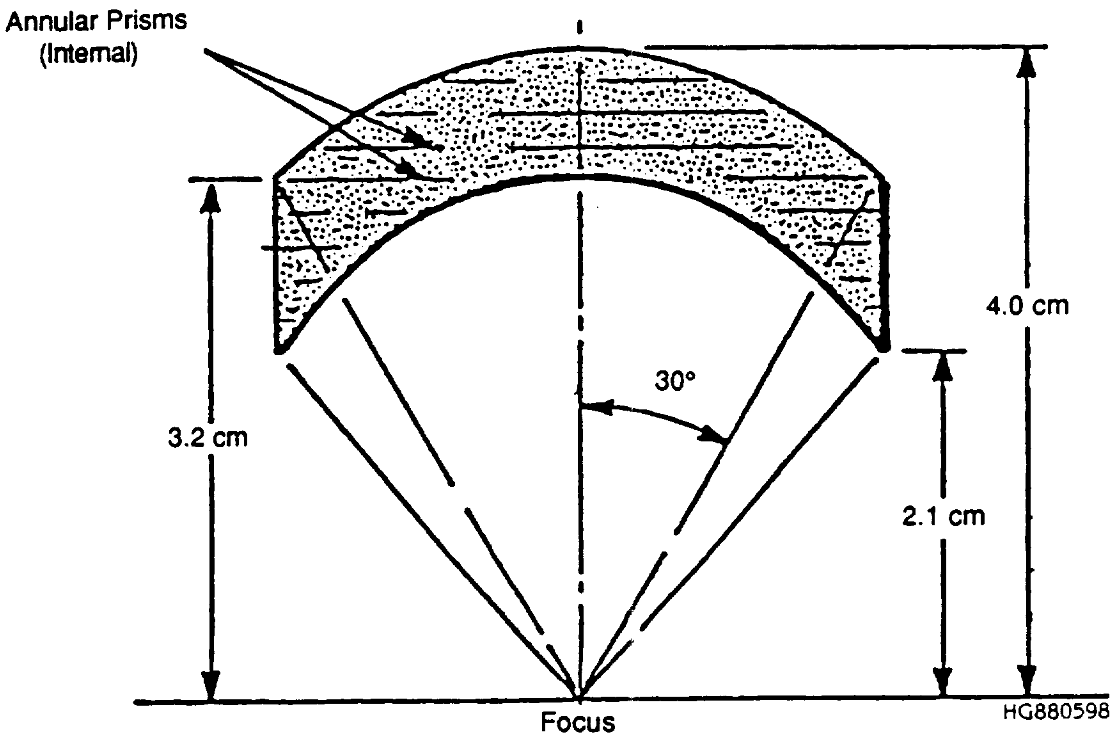

Figure 4.3-12. Selected Design of Square-Aperture Dome Lens Optical Concentrator

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Previously developed structure concepts for space station application are shown in figure 4.3-13. Based on this design, a structural weight of 0.7 kilograms per square meter was derived.

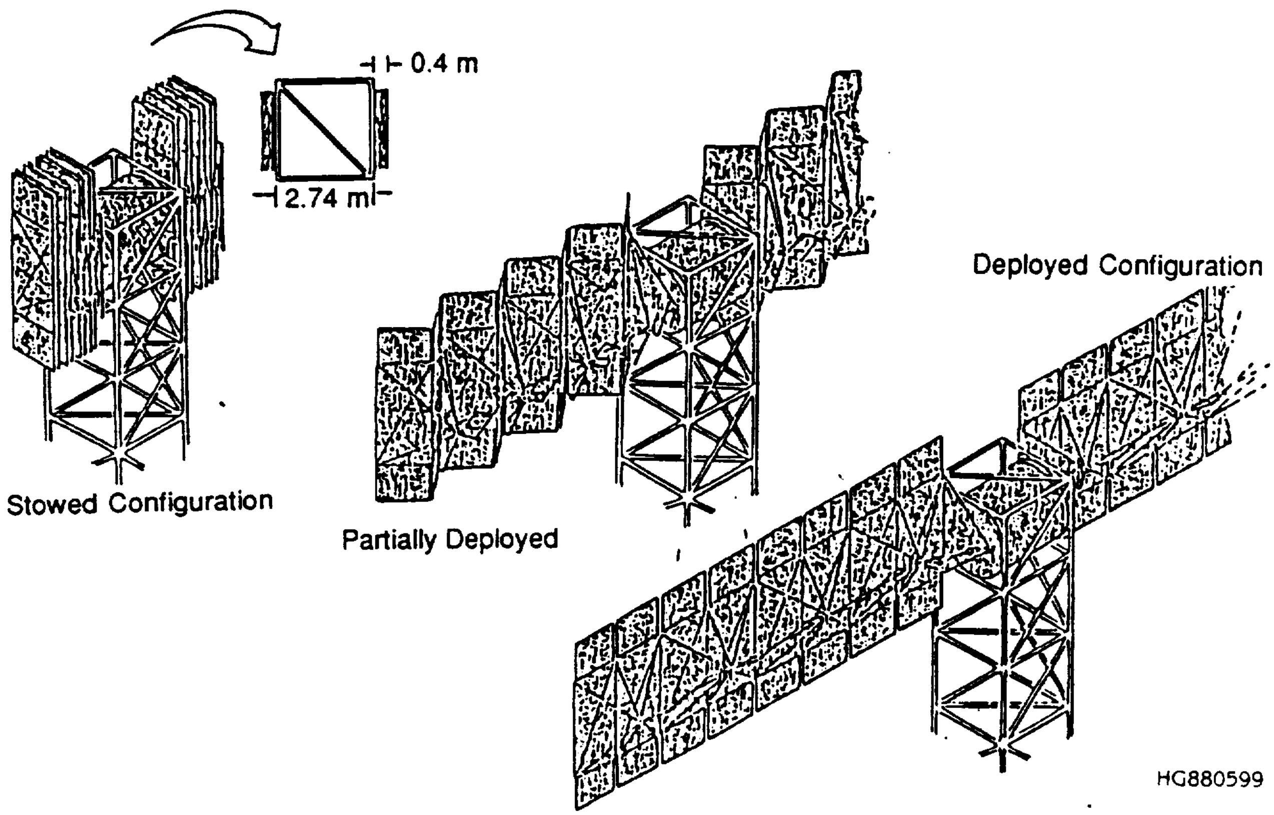


Figure 4.3-13. Structural Concepts For Space Station Application

Projected improvements in planar and concentrator solar arrays versus calendar time are shown in terms of watts per square meter in figure 4.3-14. The concentrator has the advantage in terms of unit area performance, and this would result in lower drag.

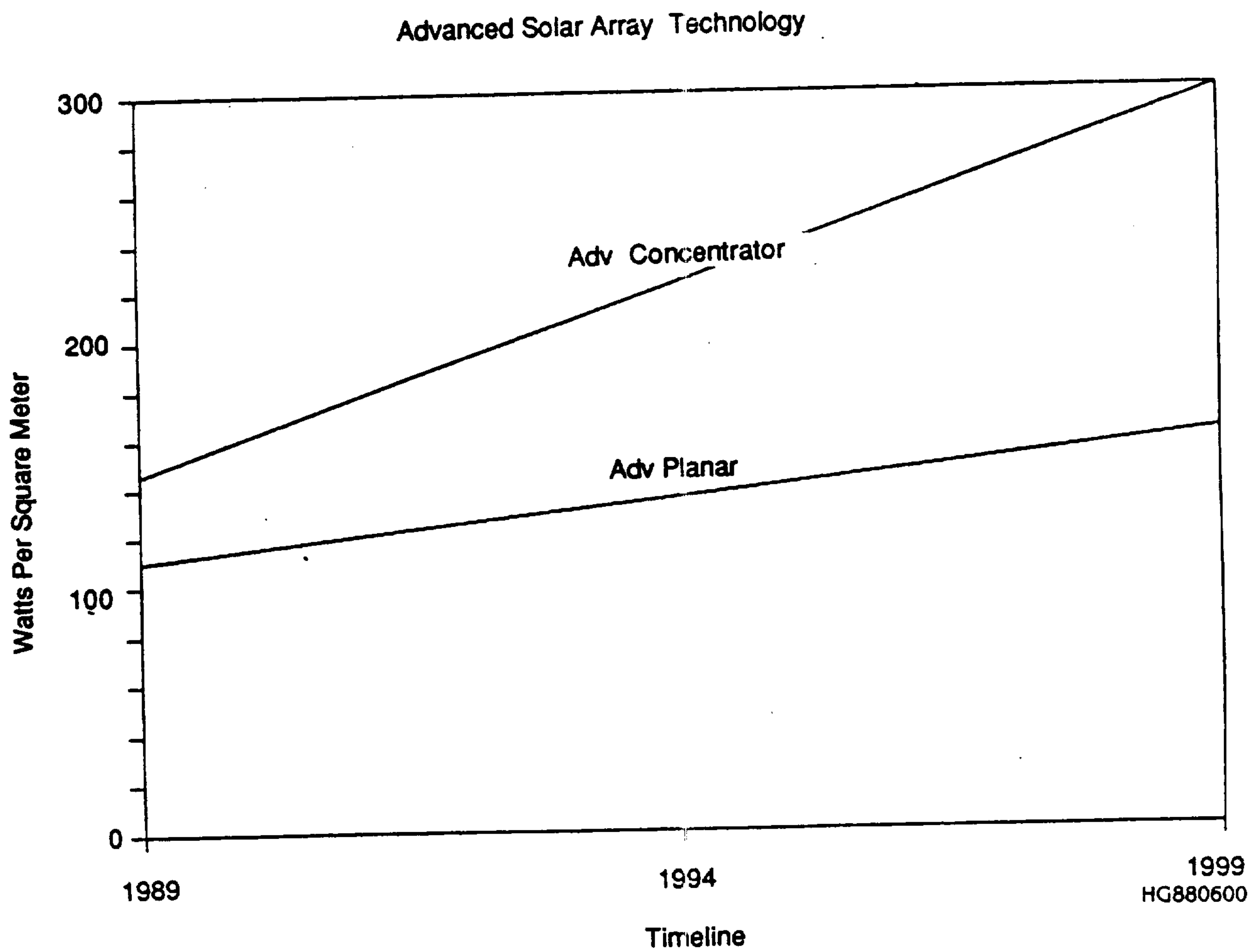


Figure 4.3-14. Projected WISQM Improvements

Projected improvements in planar and concentrator solar arrays are shown in terms of watts per kilogram versus calendar time in figure 4.3-15. This criterion favors the planar solar array.

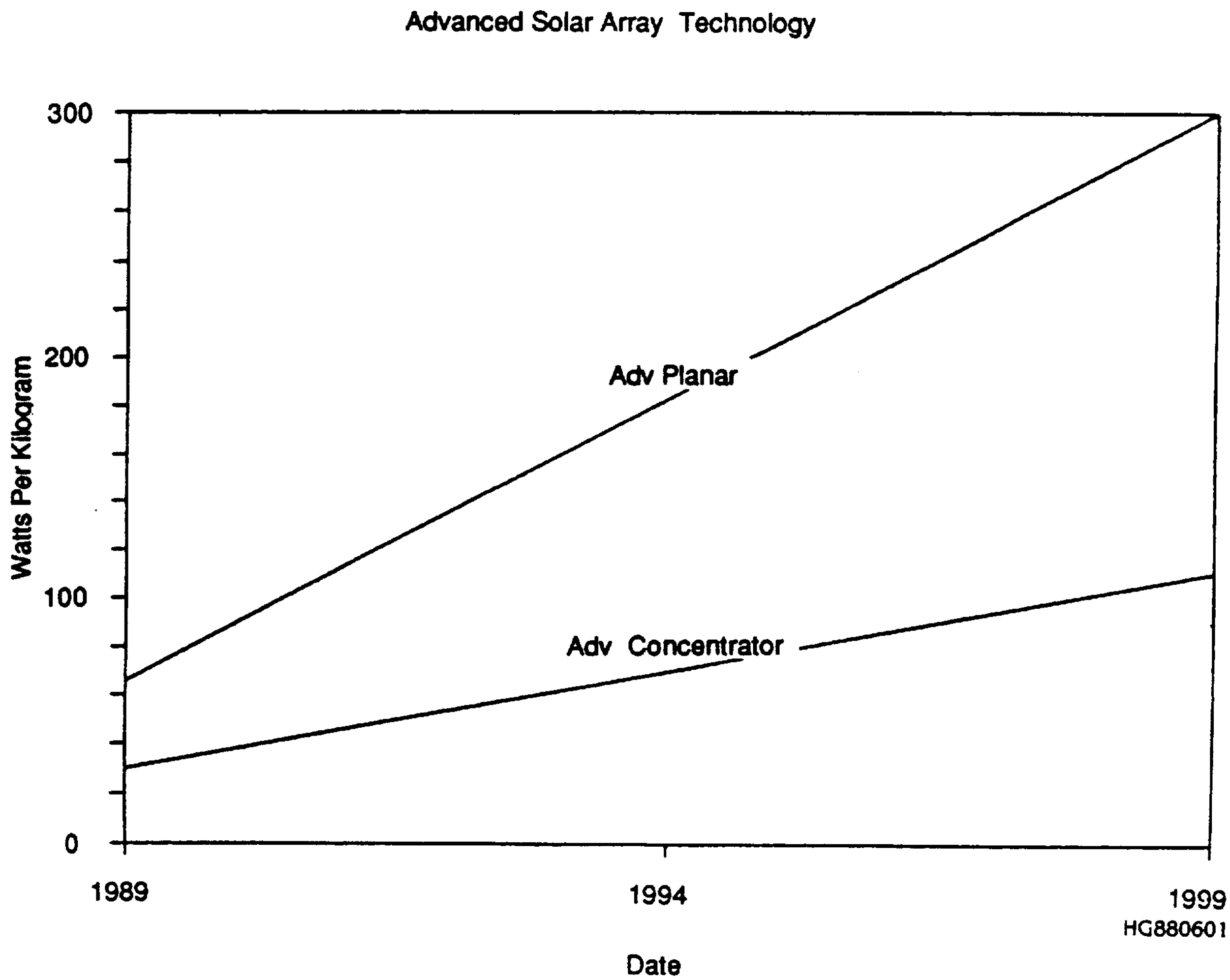


Figure 4.3-15. Projected W/KG Improvements

Advanced solar cells (and covers) were selected as the secondary concept because of the significant work in progress on cell development, and because this concept is easiest to implement for an improved solar array.

There are a few solar cell (and cover) concepts that have the potential of being leading contenders in the mid 1990s. Indium phosphide is presented in this section only as an example of what an advanced solar cell may deliver in terms of performance capability.

Figure 4.3-16 shows historical and projected data versus calendar time. Included are solar cell efficiency and specific solar array power. Solar cell efficiency is at standard --- air mass zero (AMO), 25 C --- beginning of life (BOL) conditions. Specific power is under identical conditions except for an array temperature of 55 C. A major increase in solar cell performance is expected if tandem and multiple bandgap (multi-junction) cells are developed to their expected potential.

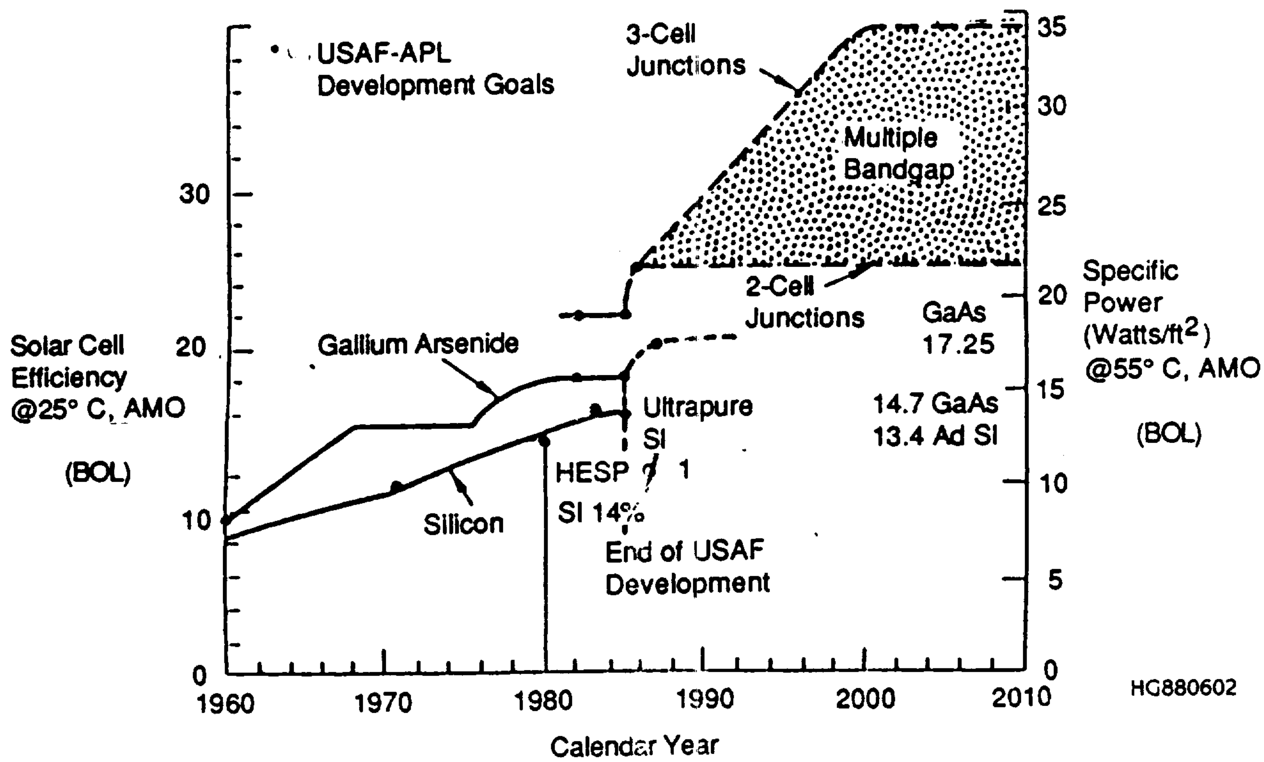


Figure 4.3-16. Solar Cell/Array Trends

Solar cell efficiency at standard conditions (AMO, 25 C) is plotted in Figure 4.3-17 versus bandgap energy level for a few contending design types. As evident from the plot, indium phosphide (InP) has the second highest potential in efficiency. Following figures show other characteristic that justify the InP cell as a leading contender. However, this does not necessarily imply that an InP cell would be the selected candidate in the 1990s time frame.

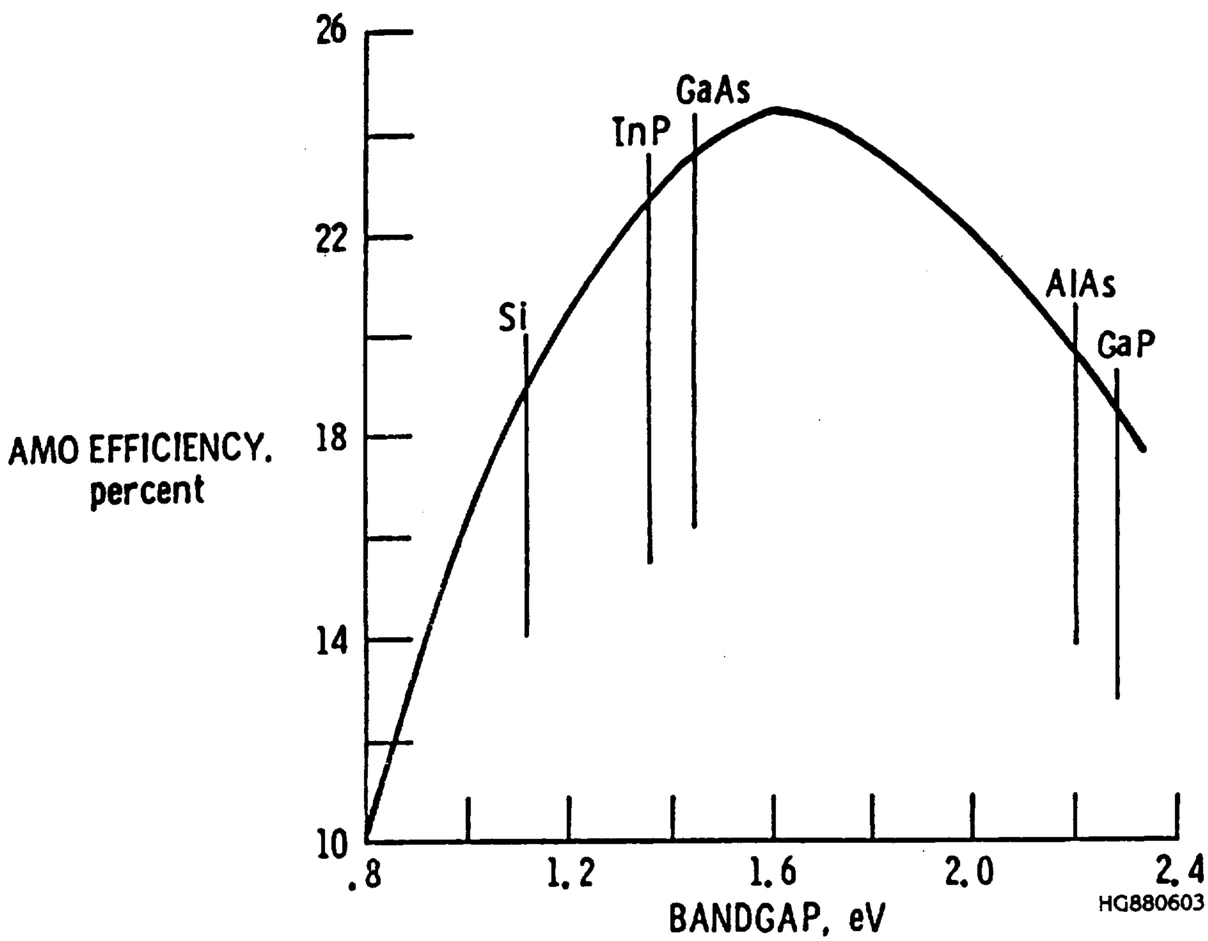


Figure 4.3-17. Efficiency Versus Bandgap In Amo

Figure 4.3-18 shows solar cell efficiency at standard conditions as a function of bandgap energy level for silicon (Si), indium phosphide (InP), and gallium arsenide (GaAs) cells. The solid line and symbols represents the predicted efficiency capability, and the open symbols represents the efficiency achieved to date. (The two symbols for silicon represent different cell configurations.)

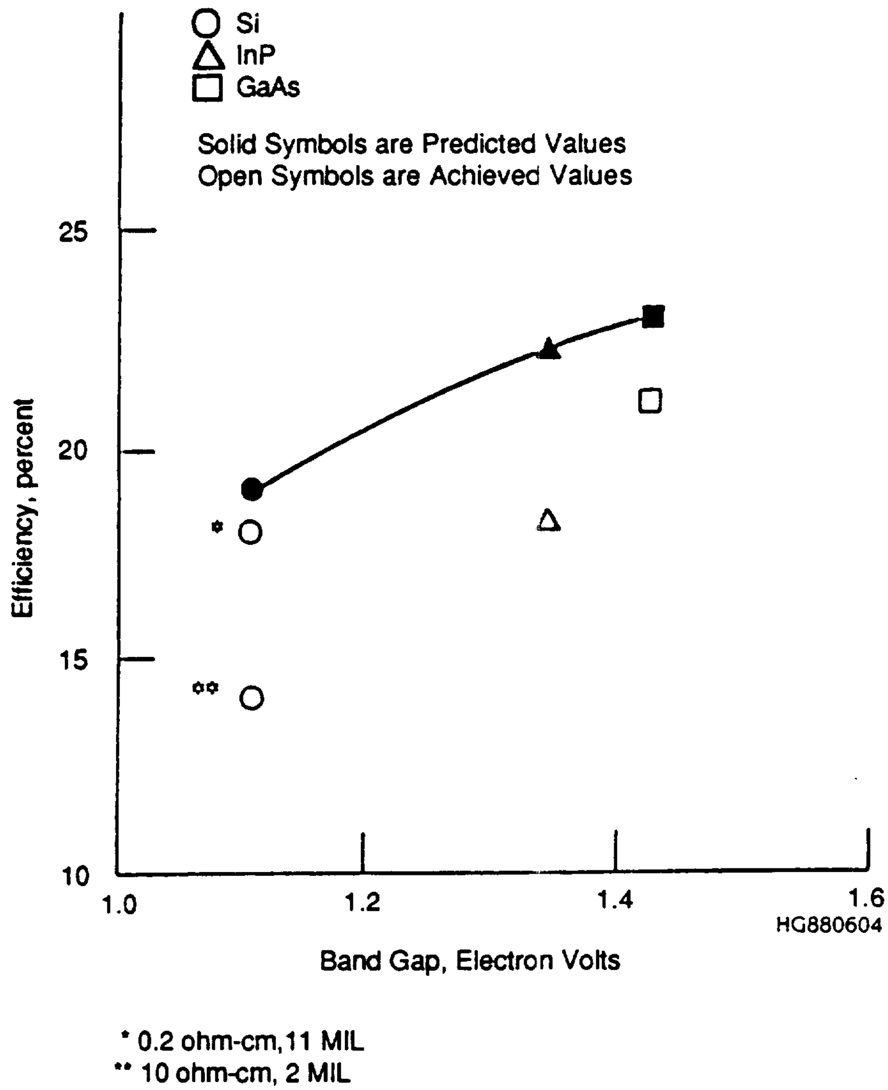


Figure 4.3-18. Predicted And Achieved AMO Efficiencies

The InP cell is shown to be the furthest from its predicted efficiency, but this cell is in its infancy compared to the others. Silicon is by far the most mature cell, with GaAs being second.

The overall message of this figure is that we can most likely expect InP cells to improve significantly relative to the other cells as their manufacturing/design techniques mature.

In Figure 4.3-19 the normalized power output of three leading cell types is plotted versus a 1-Mev electron fluence. The indium phosphide (InP) cell is much less sensitive to charged particle irradiation, compared to the other cells shown. The equivalent fluence (Si) for a 10-year GEO orbit is much more severe than the proposed space station orbit.

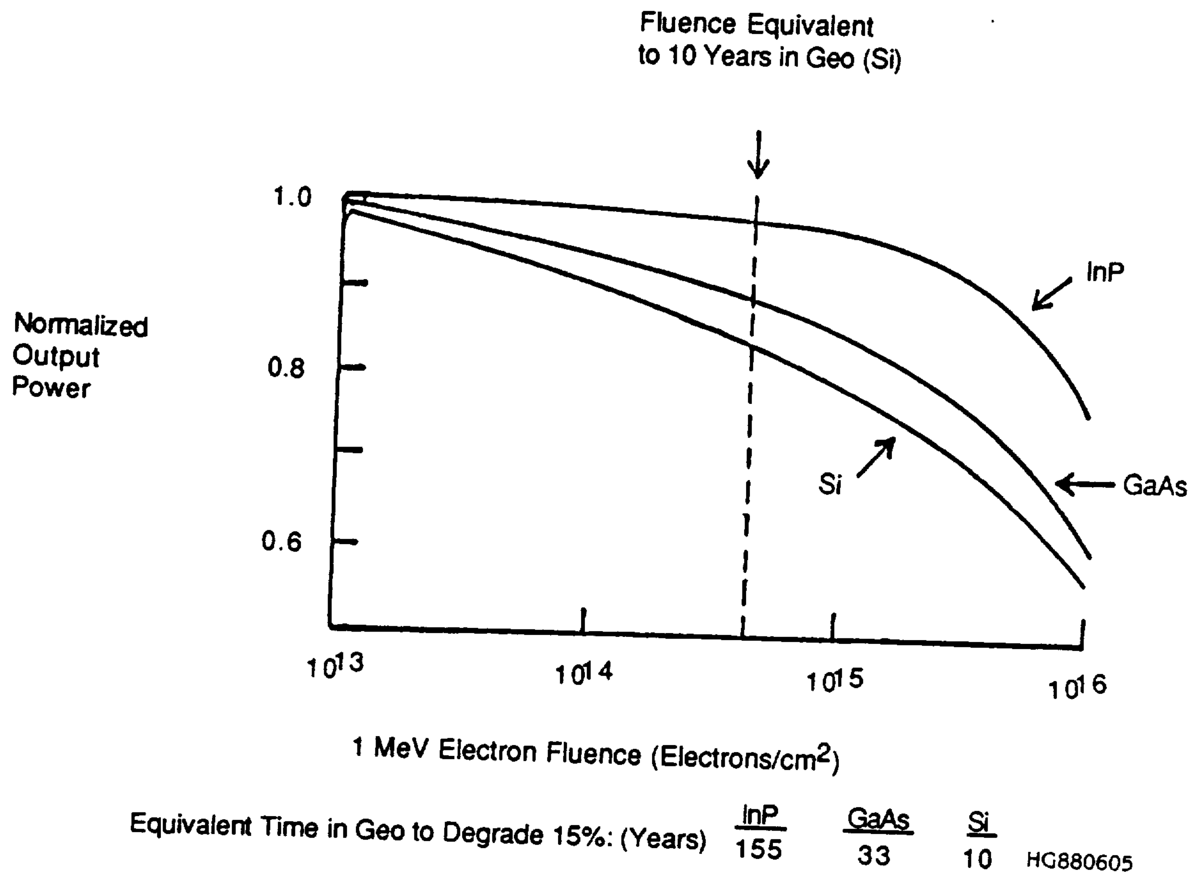
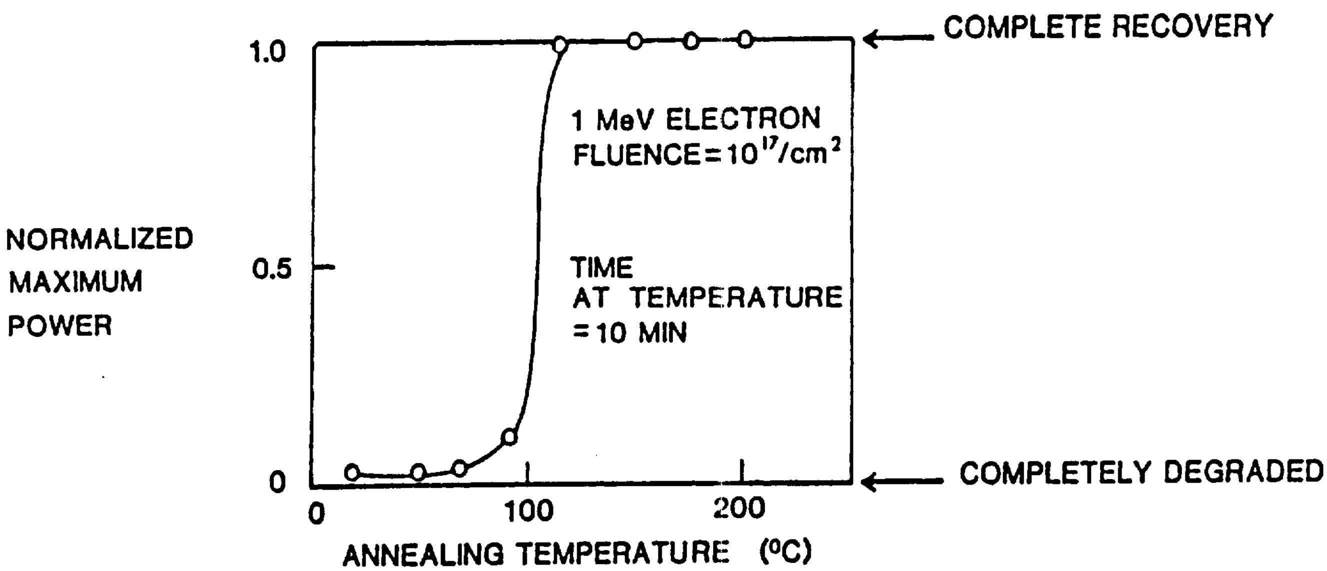


Figure 4.3-19. Radiation Resistance Comparison After Electron Irradiation

The annealing ability of indium phosphide (InP) solar cells at relatively low temperatures is illustrated in Figure 4.3-20. A 10-minute annealing cycle was performed on InP cells at different temperatures following exposure to a 1-Mev electron fluence that essentially reduced cell output to zero. The data shows that annealing cycles in the low 100s (deg cent) will essentially restore the cell to the pre-radiation performance level.



- LOW TEMPERATURE ANNEAL WILL NOT DEGRADE ARRAY COMPONENTS
- GaAs ANNEALS AT 200°C, SI AT 400°C CAUSING ARRAY DEGRADATION
- IN ORBIT ANNEAL POSSIBLE WITH InP

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Figure 4.3-20. Radiation Damage Removal In InP By Low Temperature Heating

The Sodium Sulfur LEO Cell program (F33615-86-C-2619 Air Force System Command, Wright-Patterson AFB) by Eagle-Picher Ind., Joplin, Mo. is scheduled to be completed by January, 1991. The Proof Cell of this activity provides a baseline concept, and the Laboratory Cell reflects the state-of-the-art concept for energy storage technology through the mid-1990s.

The goals and requirements for the sodium sulfur cell development are as follows:

- Low-earth orbit cycle capabilities
- Operation in weightless environment
- Structural design compatible with launch environment
- Lower weight and volume
- Improved seal reliability
- Reduced parts count
- Simpler fabricability of cells
- Improved performance
    - Greater cycle life
    - Decreased cell resistance
    - Tolerance to higher temperatures

The technology development is essentially on schedule and is expected to produce a suitable battery in time for this experiment.

Sodium Sulfur Cell Design - The sodium sulfur cell design is revolutionary to the extent that it provides radial sodium flow as opposed to axial flow, as in older cell designs, thus allowing high-rate operation. The improved design provides controlled/uniform radial flow in an opposing acceleration environment ranging from 0 G to 4 G. Improvements in the overall cell design are expected to provide a life expectancy of at least 30,000 cycles in LEO operation. The energy density is 139 WH/kg. A summary of design parameters is given in Figure 4.3-21

- **Anode: Nickel fibrex wick annulus with grafelt plug and sodium**
    - **Eliminates need for stainless steel safety tube.**
    - **Features include stiffness,malleability,weldability, reduction of parts and reduced weight**
    - **Capillary principles are used to advantage**
    - **Safety function is the limited sodium flow rate upon electrolyte fracture.**
    - **High-rate operation results from radial sodium flow vs axial.**
    - **LEO mission required to investigate zero gravity environment.**

- **Cathode: Graphite matrix with pore size #1 and #2 with grafelt spacer and sulphur**
    - **Capillary force $P = (2\sigma \cos \Theta)/R$,** **$\sigma$ = surface tension**
      Θ = contact angle
      R = Capillary radius
    - **Liquid with greater capillary force will occupy wick.**
    - **Relative capillary force related to wicking height.**

- **Electrolyte: Zirconia-Toughened Beta-alumina and alumina cloth.**
    - **Life limited by capacity throughout capability, 15,000 AH/CM^2 required**

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Figure 4.3-21. Basic Sodium Sulfur Cells Design

The detailed cell design is shown in Figure 4.3-22. All sodium sulfur cell components have been redesigned and integrated with improved manufacturing assembly methods. New design features provide improved ruggedness, reliability, and safety. Materials have been selected to meet the long-life objectives.

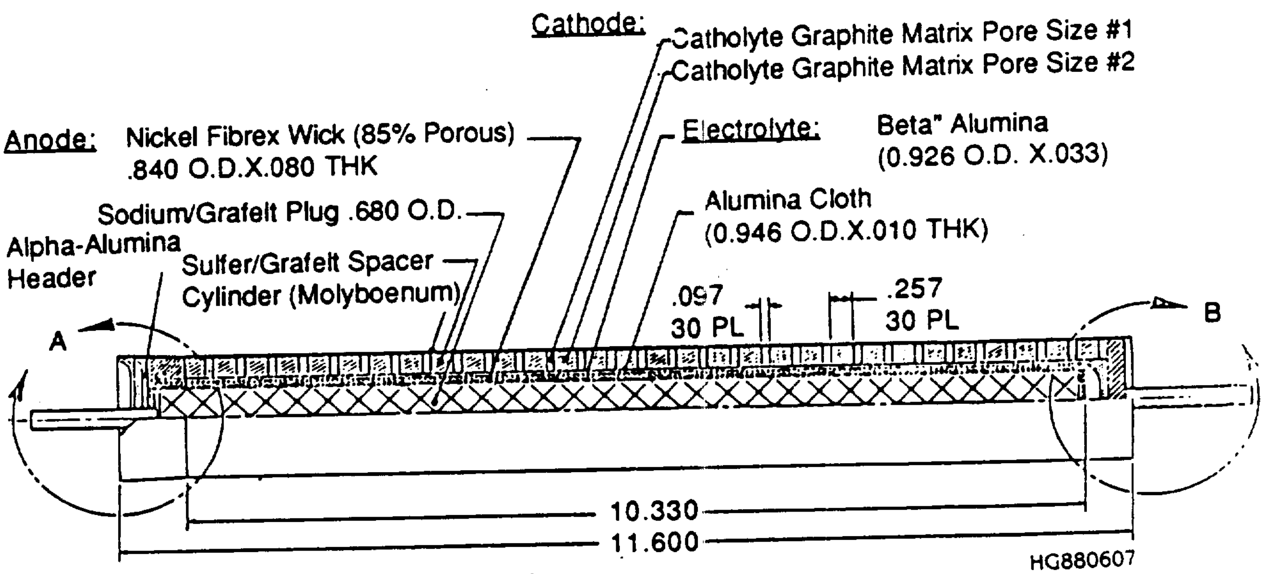


Figure 4.3-22. Detailed Cell Design

Eagle-Picher of Joplin, Mo. has fabricated and demonstrated Proof Cells having energy densities of 165 WH/kg, and these cells are undergoing a variety of tests. The development of Laboratory Cells is continuing, and they have been undergoing tests since Nov. 1987. The overall program is on the following schedule.

ACTIVITY	COMPLETION
* Proof cell testing	Sept 1988
* Technology development	Sept 1990
* Laboratory cell fabrication	Aug 1989
* Laboratory cell testing	Sept 1990
* Deliverable cell fabrication	Feb 1990
* Deliver 10 cells	June 1990
* Ship cell test stations	Feb 1990
* Final report	Jan 1991

In operation, the temperature is increased to 350° C over a ten-hours period or greater, before this class of battery is placed into operation. During normal operation the battery will require 50 watts of heat rejection by active heat removal through a thermal control radiator.

The chemical reaction at the terminals of each cell are:

CHARGE REACTIONS

At the positive:	$2\,Na$ -------->	$2\,Na^+ + 2e^-$
At the negative:	$3S + 2e^-$ -------->	S_3^{-2}
For the overall cell:	$2\,Na + 3S$ -------->	$Na_2\,S_3$

DISCHARGE REACTIONS

At the positive:	$2\,Na^+ + 2e^-$	--------> $2\,Na$
At the negative:	S_3^{-2}	--------> $3S + 2e^-$
For the overall cell:	$Na_2\,S_3$	--------> $2\,Na + 3S$

Testing in the zero-gravity environment is required to verify proper operation of the sodium sulfur battery wicking characteristics, flow rate and proper cathode mixing. It has been recommended that this verification be carried out on an early Shuttle flight.

BIPOLAR NiH_2 BATTERIES - The bipolar nickel hydrogen LEO multikilowatt advanced configuration system design by Lewis Research Center, Electrochemistry Technology Section, Cleveland, Ohio is a complete battery system that incorporates active cooling for thermal control. Reference to the design activity includes: NASA Technical Memorandum 82844, "Design of a 35-Kilowatt Bipolar Nickel Hydrogen Battery for LEO Applications" and NASA Technical Memorandum 83647, "Design of 1-kWh Bipolar Nickel Hydrogen Battery."

Goals set for the bipolar NiH_2 battery are:

- Active cooling
- Advanced electolyte control
- Advance O_2 recombination techniques
- High Voltage: > 100 VDC
- Capacity: 130 AH
- Weight savings: 20-30%
- Volume savings: > 50%
- Improved energy density: 42WH/kg
- Life: LEO 5 years (70% DOD for 30,000 cycles)

DEVELOPMENT ACTIVITY:

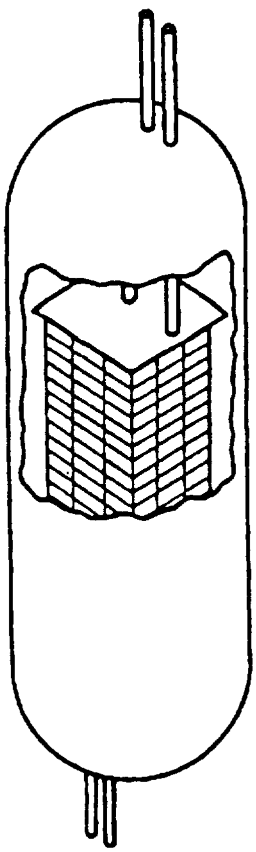
- Test to 28,000 @ 80% DOD (IPV)

These criteria are considered in the design of the battery cell. The major goal is to achieve 30,000 cycles at a depth-of-discharge (DOD) of 70 percent in a LEO operating mode.

The 160-volt design for this experiment is different from the NASA battery which has a nominal discharge voltage of 286 volts. The NASA design has 229 series cells versus 128 cells for the experiment battery.

Individual cells are assembled in a bipolar configuration and are contained within a single-pressure vessel. This optimized design has a 130-AH capacity and 42-WH/kg capability at discharge voltages greater than 100 volts.

The concept of bipolar power sources in fuel cells and redox systems has been used with success in other applications. The cell contains a bipolar plate, a hydrogen electrode with an associated gas flow screen, a separator, a "floating" nickel electrode, an electrolyte reservoir plate with catalyzed oxygen recombination wires, and a bipolar cooling plate. The medium for cooling can be hydrogen gas or a liquid coolant. The design is shown in Figure 4.3-23.



BATTERY SPECIFICATIONS FOR 160V SYSTEM

• POWER	21kWH
• LOAD VOLTAGE	160V
• THEORETICAL CAPACITY	130AH
• DISCHARGE CAPACITY/ AT 0.6 hr ECLIPSE	78AH
• DEPTH OF DISCHARGE	70%
• SERIES CELLS	128
• PLATE AREA	$390 In.^2$
• CELL THICKNESS	0.095In.
• STACK DIMENSIONS	27×27×19 In.
• BATTERY WEIGHT	885lb
• ENERGY DENSITY	42WH/kg AT 100% DOD
• VOLUMETRIC ENERGY DENSITY	$780WH/ft^3$
• VESSEL CONFIGURATION	TBD

Conceptual drawing of a 21KW bipolar constructed nickel hydrogen battery including specifications.

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Figure 4.3-23. Basic Bipolar NiIH$_2$ Battery Design

Small vents link the hydrogen electrode gas screen to the hydrogen gas reservoir shown on the left of Figure 4.3-24. Gas is created at this electrode/screen during charge to pressurize the vessel, and is subsequently consumed during discharge. The recombination of oxygen with hydrogen takes place on the opposite side of the nickel electrode. The high bubble-pressure separator forces the oxygen to contact the recombination sites located within the electrolyte reservoir plate. Proximity to the cooling plate facilitates the vapor's return to water which in turn is wicked back to the separator.

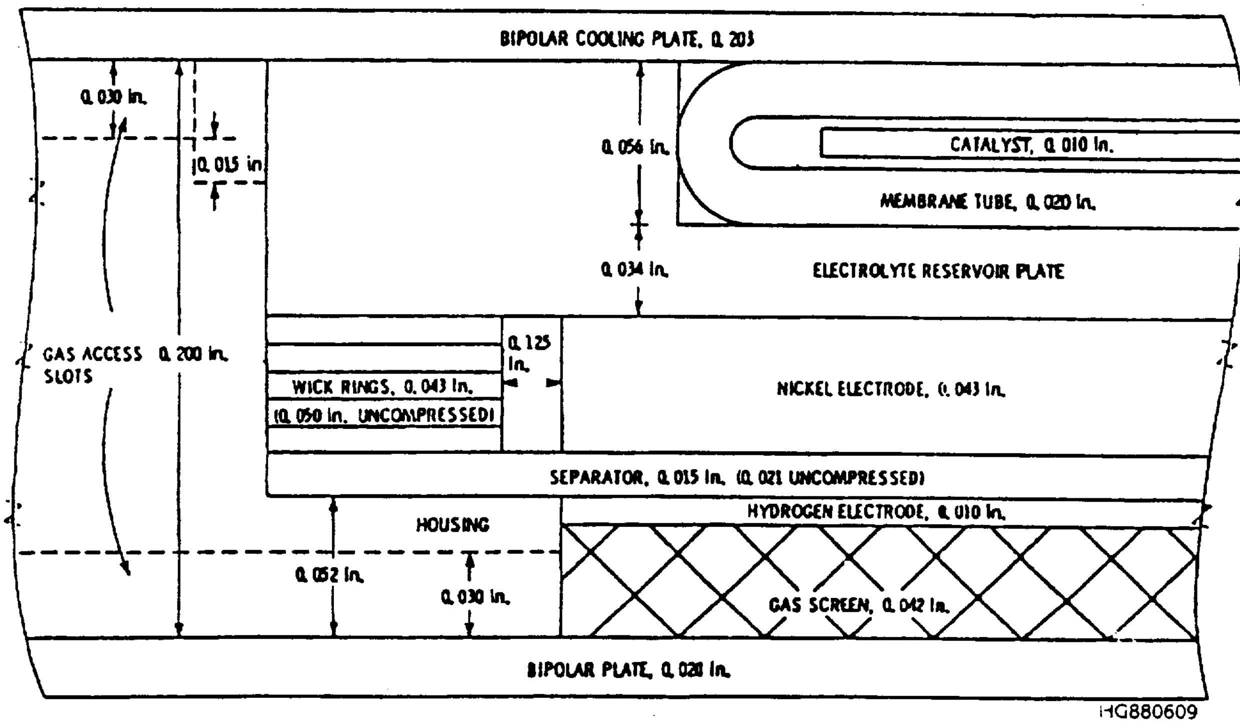

Figure 4.3-24. Unit Cell Cross Section

Each cell is housed in one frame and sealed to maintain material balance.

The bipolar NiH_2 battery features high AH capacity, high voltage, good energy density, and a 5-year life in LEO.

The initial study was to design, evaluate and construct a bipolar battery stack, and was initiated in 1981 at NASA Lewis. The Concept Verification Program included design and assembly during the first phase, and characterization and cycle testing during the second phase. The preliminary design of a 35-kW bipolar nickel hydrogen battery was completed in 1982 by NASA Lewis. The pre-prototype 1-kWH bipolar nickel hydrogen component/battery design and evaluation was completed in 1984, and fabrication and testing was completed in 1987. A 120-volt design is scheduled for completion in late 1989.

The bipolar nickel hydrogen battery program at Ford Aerospace Corporation and Whittaker-Yardney Power Systems (NASA Lewis funding) has included the design, fabrication and testing of an actively cooled, 10-cell, 75-AH battery. This program is to be completed in early 1991.

The discharge reactions at the hydrogen (negative) electrode gas screen consists of hydrogen and OH ions combining to form water and free electrons. At the nickel (positive) electrode, a nickel-oxyhydroxide molecule combines with water to form nickel hydroxide and the OH ion.

The charge reactions at the hydrogen electrode consists of the electrolysis of water into hydrogen gas and OH ions. The nickel electrode combines nickel hydroxide and OH ions to form the nickel-oxyhydroxide molecule, water and free electrons.

CHARGE REACTIONS

At the positive:	$Ni(OH)_2 + OH^-$	------->	$NiOOH + H_2O + e^-$
At the negative:	$H_2O + e^-$	------->	$1/2\ H_2 + OH^-$
For the overall cell:	$Ni(OH)_2$	------->	$NiOOH + 1/2\ H_2$

DISCHARGE REACTIONS

At the postive:	$NIOOH + H_2O + e^-$	----->	$Ni(OH)_2 + OH^-$
At the negative:	$1/2\ H_2 + OH^-$	----->	$H_2O + e^-$
For the overall cell:	$NIOOH = 1/2\ H_2$	----->	$Ni(OH)_2$

4.3.4 Performance goals

One objective of this experiment is to demonstrate the basic weight and packaging characteristics of a photovoltaic/energy storage system that approaches or exceeds a solar dynamic system. The key areas of investigation are improved solar array and energy storage capability. Secondary issues are improved packaging, reduced losses, and efficient power processing.

The second objective is to demonstrate reduced life cycle cost (LCC), including improved life and reduced degradation, improved redundancy and redundancy management, and solar array producibility.

A comparative analysis was performed to show the approximate differences in collection area between the solar dynamic and solar photo-voltaic concepts for an experiment power demand of 25 kW. A low- and high-performance design is shown for each technology in Figure 4.3-25. The current state-of-the-art (SOA) design has the largest area. The high performance photovoltaic design is shown to be in between the low- and high-performance solar dynamic designs.

	Dynamic-Hi	Dynamic-Low	Photovoltaic-Low (Current S.O.A.)	Photovoltaic-Hi (TDM Targets)
Solar Flux	1.353kW/m2	(Same)	(Same)	(Same)
Collection Efficiency	0.64	0.6	0.9	0.9
Conversion Efficiency	0.4	0.25	0.14	0.23
Distribution Efficiency	0.9	0.9	0.9	0.9
Storage Efficiency	0.90	0.80	0.6	0.7
Degradation Factor(BOL/EOL)	1.1	1.15	1.1	1.05
Orbit Period	95 Min.	(Same)	(Same)	(Same)
Dark Period	36 Min.	(Same)	(Same)	(Same)
Load	25KW	(Same)	(Same)	(Same)
Collector Area (M2)	148.0	277.4	361.5	194.9

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Figure 4.3-25. Solar Power System Area Efficiency Comparison

Other factors beneficial to photovoltaic designs include their long history of successful operation, and the fact that high-powered rotating equipment has yet to operate in space.

4.4 Experiment Requirements

Requirements for the TDMX2151 experiment are provided in this section. General requirements are provided in addition to specific requirements for each of the four selected concepts. The resultant experiment can use any combination or all of the selected concepts.

The overall general requirements include the basic functional and operational interfaces of the experiment with the space station. General requirements for the solar array, energy storage devices, and the radiators are:

- General
  - Interfaces
  - Performance measurement systems
  - Experiment power management & distribution system
  - Thermal management system
  - Drag and/or mass balance?
- Solar Arrays
  - Clearance envelope
  - Two-axis solar tracking
  - Clear line-of-sight to Sun

- Energy Storage
  - Clearance envelope
- Radiators
  - Clearance envelope
  - Single axis gimbal (with proper orientation)

A simplified functional block diagram of the experiment is shown in Figure 4.4-1 including the solar array, drive and slip ring assembly; the battery and charge control unit; the power control module; the load regulation unit; the dynamic load; and the processor control unit.

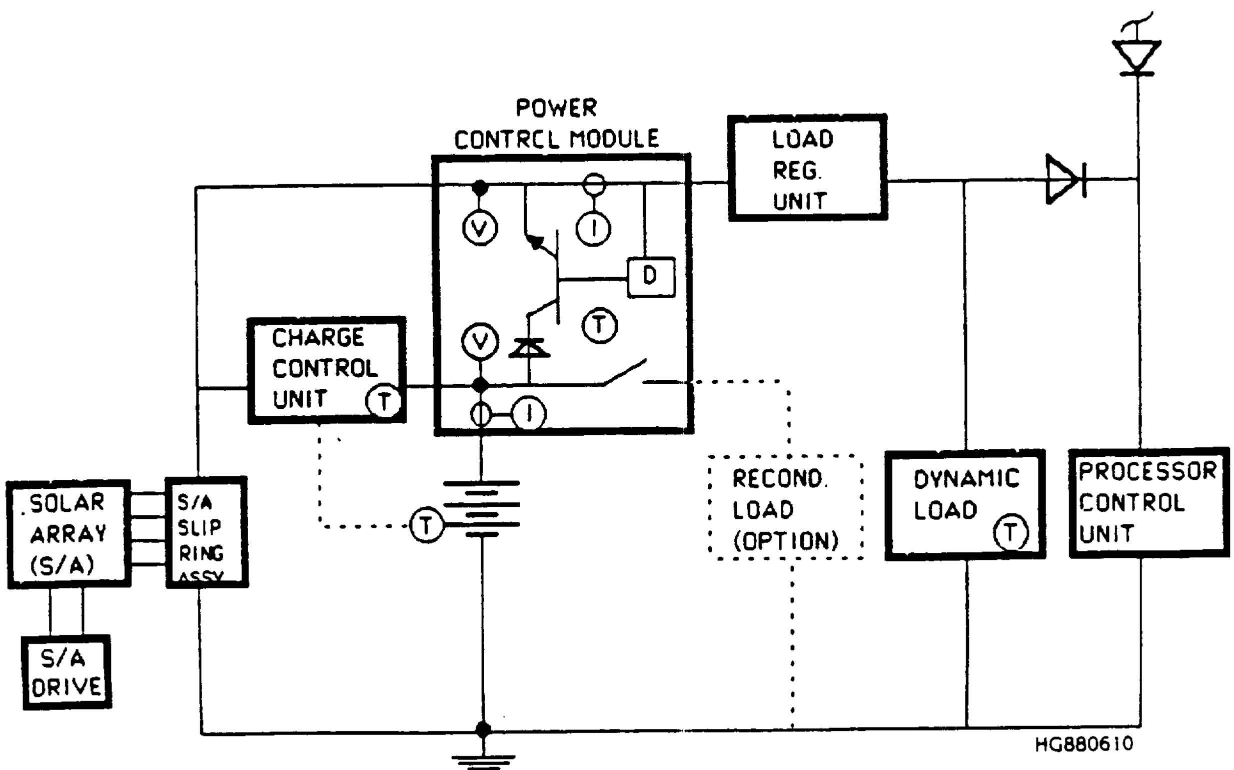

Figure 4.4-1. Solar Array/Energy Storage Experiment (Functional Block Diagram)

The experiment is located outside of the alpha joint, and requires a beta joint for complete solar tracking. The slipring assembly is required because rotation is continuous. The charge control unit regulates all energy into the battery, and can compensate for battery temperature, voltage, and state of health.

The power control module controls all power between the energy sources and storage devices, and monitors key performance parameters. The dynamic load dissipates battery energy during eclipse, thereby cycling the battery to a prescribed depth of discharge. (The basic load value would be selected as a function of which energy storage concept(s) was selected for the experiment.) Periodically (e.g. once per week) the dynamic load will cycle through a programmed sequence of power ranges to support measurement of solar array and battery current-voltage characteristics. This test, as with all experiment operations, would be controlled by the processor control unit.

Space Station power will be used to initiate experiment preparation and operation. Experiment operation will then become autonomous.

The solar array requirements are dependent on the battery selection because one experiment function is to cycle each battery to its maximum recommended depth of discharge (DOD). The selected batteries have different DOD requirements and different output voltages, so their power outputs are also different. As a result, the solar array size required to replenish the battery charge is a function of the battery type.

The required solar array power demand is calculated for the sodium sulfur battery and the bipolar nickel hydrogen. For each case we calculated the battery output, the battery charging power, and the solar array power demand.

There are many other solar array requirements having to do with electrical and mechanical interfaces, structural dynamic coupling, degradation, and other design parameters. Most other requirements are similar, if not identical, for all space solar arrays, and are not significant to this study.

The maximum recommended DOD for sodium sulfur batteries is assumed to be 80 percent. Based on a cell capacity of 50 ampere hours, the output is 40 ampere hours. An average output voltage of 1.845 volts per cell provides a battery voltage of 169.7 volts, which equates to 6.8 kWH output from the battery. The discharge voltage allows for 6 percent cell failures before degrading to the minimum nominal discharge voltage of 160 volts.

Considering a discharge regulator efficiency of 95 percent, the maximum power demand of the dynamic load is 10.8 kW. The calculations are:

Assumption: Maximum DOD of 80% for Na/S batteries

Battery Output:

$$t_E = 36 \text{ min.} = 0.6 \text{ Hrs.}$$

$$AH_O = (AH_C) \quad (DOD) = (50AH) \quad (80\%) = 40AH$$

$$Vo = (N_S)\ (V/cell) = (92) \quad (1.845.\ Volts/cell) = 169.7V$$

$$Eo = (AH_O)\ (V_O) = (40AH) \quad (169.7V) = 6.8KWH$$

$$(P_L)DYN = \frac{E_O \cdot \eta_{DCR}}{t_E} = \frac{(6.8KWH)\ (0.95)}{0.6H} = 10.8kW$$

The power required for charging the battery is dependent on the charge time, the average charge voltage, and the ampere hour input. Based on a 100 percent ampere hour efficiency, the battery requires 40 ampere hours for recharge. The charge voltage of 2.23 volts per cell results in a battery voltage of 205.2 volts, and the entire charge energy is 8.2 kWH. The available charge time was assumed to be 5 minutes less than the daylight segment (0.9 hours), and this results in an average charging power of 9.1 kWH as follows:

$$t_{sun} = 59 \text{ min.}$$

$$t_{CHG} = 54 \text{ min.} \quad = 0.9 \text{ hrs.}$$

$$AH_I = AH_O = (40AH)$$

$$V_{CHG} = (N_S)(V/cell) = (92)\ (2.23V.\ volts/cell) = 205.2Volts$$

$$E_{CHG} = (V_{CHG})(AH_I) = (205.2V)(40.AH) = 8.2kWH$$

$$P_{CHG} = \frac{E_{CHG}}{t_{CHG}} = \frac{(8.2KWH)}{0.9H} = 9.1\ kW$$

The total solar array power demand is based on the battery charging power, the charger efficiency, and the instrumentation loads. The charger efficiency is 95 percent, and the instrumentation load is approximate 100 watts. These parameters result in a total solar array power output requirement of 9.7 kW as shown in Figure 4.4-2.

Solar Array Power Demand:

$$\eta_{CHG} = 0.95$$

$$P_{SA} = \left[\frac{P_{CHG}}{\eta_{CHG}} + P_{L\,INST} \right]$$

$$P_{SA} = \left[\frac{9.1KWH}{0.95} + 0.1KWH \right] = 9.7KW$$

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Figure 4.4-2. Power/Energy Sizing Analysis

The maximum recommended DOD for bipolar nickel hydrogen batteries is assumed to be 70 percent. Based on a cell capacity of 130 ampere hours, the output is 91 ampere hours. An average output voltage of 1.25 volts per cell provides a battery voltage of 160 volts, which equates to a battery output of 14.6 kWH.

Considering a power condition efficiency of 95 percent, the maximum power demand of the dynamic load is 21.1 kW.

Assumption: Maximum DOD of 80% for Na/S batteries

Battery Output:

$$t_E = 36 \text{ min.} = 0.6 \text{ Hrs.}$$

$$AH_O = (AH_C) \quad (DOD) = (130AH) \quad (70\%) = 91AH$$

$$Vo = (N_S) \ (V/cell) = (128) \ (1.25V/cell) = 160V$$

$$Eo = (AH_O) \ (V_O) = (91AH) \quad (160V) = 14.6kWH$$

$$(P_L)DYN = \frac{(E_O) \cdot (\eta_{DCR})}{t_E} = \frac{(14.6kWH) \ (0.95)}{0.6H} = 21.1kW$$

The charging power of the battery is dependent on the charge time, the average charge voltage, and the ampere hour input. Based on a 95 percent ampere hour efficiency, the battery requires 95.5 ampere hours for recharge. The charge voltage of 1.4 volts per cell results in a battery voltage of 179 volts, and the entire charge energy is 17.0 kWH. The available charge time of 0.9 hours results in an average charging power of 18.8 kWH.

$$t_{sun} = 59 \text{ min.}$$

$$t_{CHG} = 54 \text{ min.} = 0.9 \text{ hrs.}$$

$$AH_I = (AH_O) \cdot (1.05) \ (91AH) \quad (1.05) = 95.5$$

$$V_{CHG} = (N_S) \ (V/cell) = (128) \ (1.4V/cell) = 179 \text{ Volts}$$

$$E_{CHG} = (V_{CHG}) \ (AH_I) = (179v) \quad (95.5AH) = 17.0kWH$$

$$P_{CHG} = \frac{E_{CHG}}{t_{CHG}} = \frac{(17.0kWH)}{0.9H} = 18.8 \text{ kW}$$

The total solar array power demand is based on the battery charging power, the charger efficiency, and the instrumentation loads. The charger efficiency is 95 percent, and the instrumentation load is approximately 100 watts. These parameters result in a total solar array power output requirement of 19.8 kW as shown in Figure 4.4-3.

$$\eta_{CHG} = 0.95$$

$$P_{SA} = \left[\frac{P_{CHG}}{\eta_{CHG}} + P_{L_{INST}} \right]$$

$$P_{SA} = \left[\frac{18.8KWH}{0.95} + 0.1KWH \right] = 19.8KW$$

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Figure 4.4-3. Power/Energy Sizing Analysis

The present Space Station beta gimbal as shown in Figure 4.4-4 is also suitable for the experiment solar array gimbal. The reported accuracy is ± 2 degrees, and when combined with the alpha gimbal (a reported accuracy of ± 3 degrees), a suitable overall pointing accuracy is provided for the experiment. The advanced solar cell concept is rather insensitive to angles under 10 degrees, and the domed concentrator concept can be designed to tolerate a misorientation of 5 degrees.

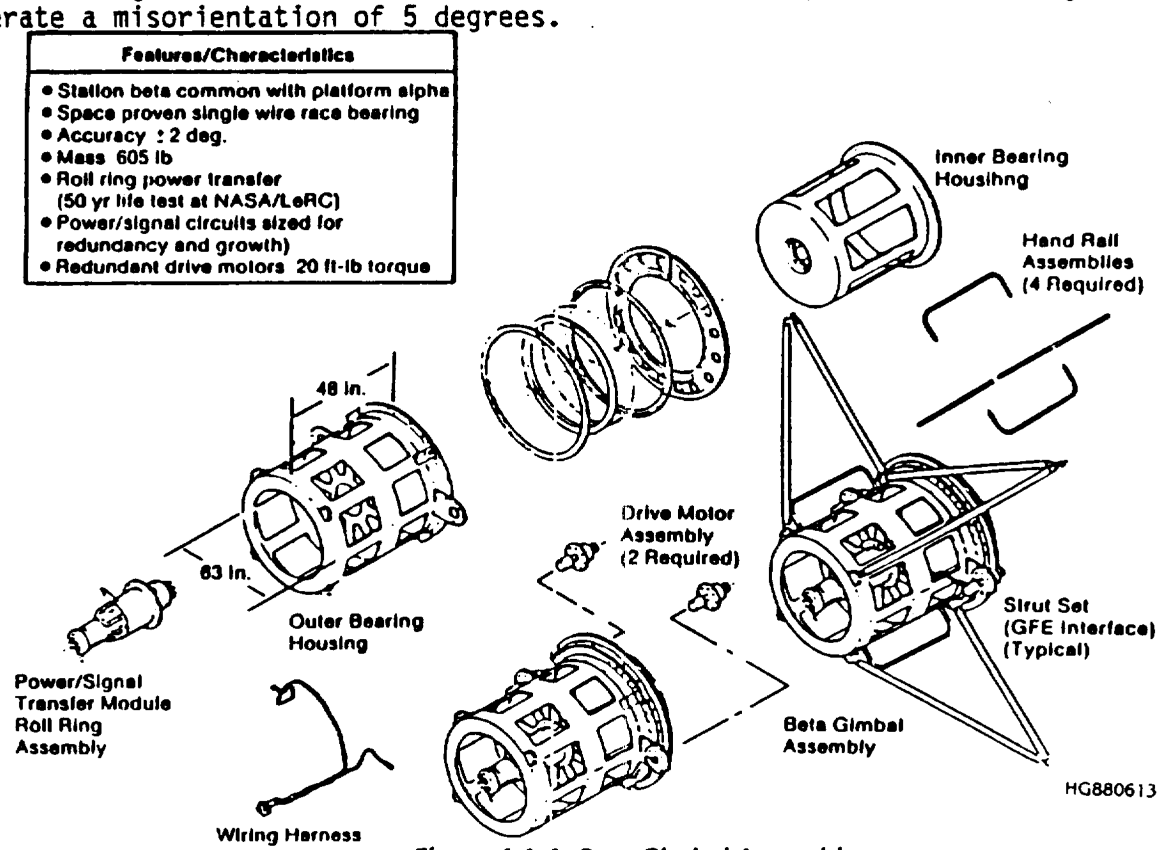

Figure 4.4-4. Beta Gimbal Assembly

The experiment requirements for the domed concentrator concept include the solar array sizing, the basic panel/wing design, and the wing weight. The wing is sized to provide the 9.7 kW required for the sodium sulfur battery and associated instrumentation. The wing area would have to be twice as great to power the 19.2 kW required for the bipolar nickel hydrogen battery.

The basic solar array size is a function of the required solar array power demand at end of life, the associated environmental factors, the basic solar cell efficiency, the packing factor, and the solar intensity. Environmental degradation is only 5 percent because the charged particle fluence is very low in the expected space station orbit. The 23 percent cell efficiency is based on an effective solar irradiance of 100 suns in the concentrator, and a solar cell temperature of 100 C. A 90 percent packing factor was used.

The resultant solar array area requirement is 36.4 square meters, or 391 square feet, as shown in Figure 4.4-5.

$$A_{SA} = \frac{P_{SA}\ (Fe)}{(\eta_c)(\eta_p)(S)}$$

$$= \frac{(9.7\text{kW})\ (1.05)}{(0.23)\ (0.9)\ (1.353\text{kW/M}^2)}$$

$$A_{SA} = 36.4\text{M}^2 = 391\ \text{ft}^2$$

FACTORS:

Fe: Environment

η_c: Charge eff

η_p: Packing

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Figure 4.4-5. Solar Array Sizing Analysis Summary (Concentrator)

The 196-unit panel section measures 0.518 meters on each side for a total area of 0.268 square meters. This is the basic manufacturing unit of which the panel and wing segments would be fabricated.

Panel and wing design is illustrated in Figure 4.4-6.

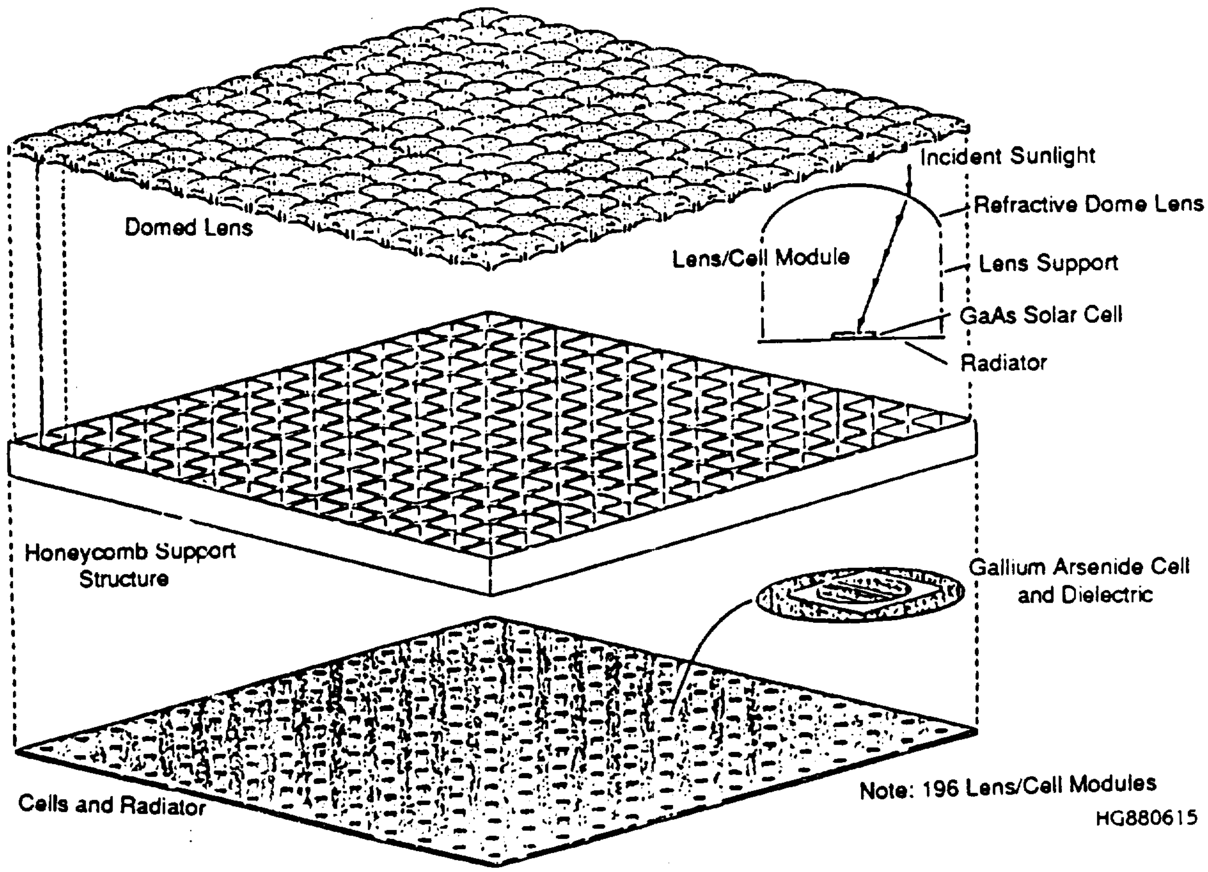
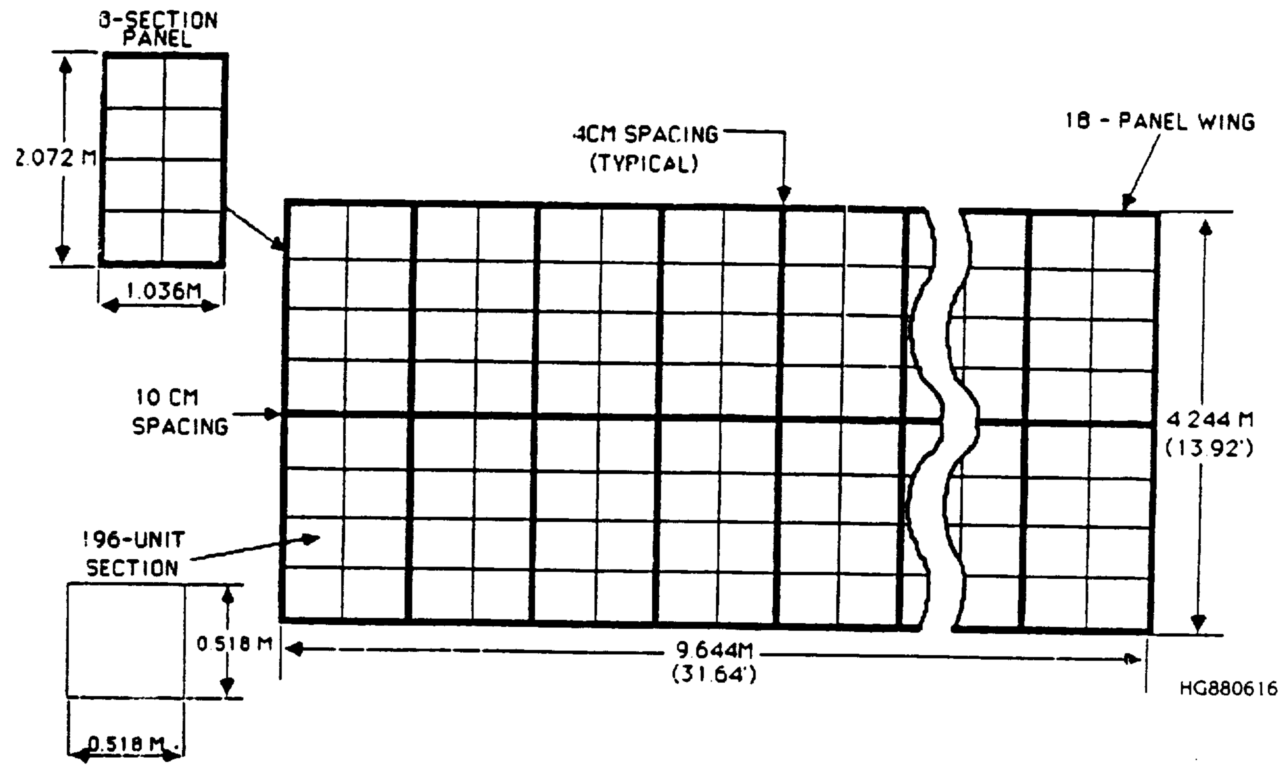


Figure 4.4-6. Panel Section Design

The basic design of the experiment wing is shown in figure 4.4-7 for the domed concentrator concept. The wing contains 18 panels, with each panel comprised of 8 panel sections. Each panel section contains 196 concentrator units (14X14). A spacing of 4 centimeters was considered between panels in the long direction, resulting in an overall wing length of 9.644 meters. A 10



JRB-82-12

Figure 4.4-7. Solar Concentrator Panel/Wing Design

centimeter spacing was considered between panels in the other direction for an overall wing width of 4.244 meters.

The dome concentrator wing has 18 panels of 8 sections each for a total of 144 sections. Each section has an area of 0.2683 square meters for a total area of 38.6 square meters. Considering a section weight of 2.5 kilograms per square meter, the 144 sections will weigh 96.6 kg. A structural unit weight of 0.7 kilograms per square meter results in a structural weight of 27.0 kg. The resulting wing weight is 123.6 kg for the experiment.

It is considered likely that a thorough analysis and design of a domed concentrator would disclose items that would add to the above weight. However, an extensive analysis and design effort is beyond the scope of this study.

The experiment requirements for the advanced solar cell concept include the solar array sizing, the basic panel/wing design, and the impact on wing weight. As with the concentrator concept, the sizing calculation is based on a 9.7 kW demand for charging the sodium sulfur battery. Should the bipolar nickel hydrogen battery be selected for the experiment, the active length of the solar array would have to be doubled to meet the 19.2 kW demand of that battery.

The basic solar array size is a function of the required solar array power demand at end of life, and the associated environmental factors, the basic solar cell efficiency, the packing factor, and the solar intensity. Environmental degradation is only 5 percent because the charged particle fluence is very low in the expected space station orbit. The 19 percent cell efficiency is based on a conservative extrapolation of current production cell capability. A 90 percent packing factor was used.

The resultant solar array area requirement is 44.0 square meters, or 474 square feet.

Assuming a blanket width comparable to the 14.2 feet of the baseline solar array, the resultant minimum blanket length is 33.4 feet as shown in Figure 4.4-8.

$$A_{SA} = \frac{P_{SA}\ (Fe)}{(\eta_c)(\eta_p)(S)}$$

$$= \frac{(9.7\text{kW})\ (1.05)}{(0.19)\ (0.9)\ (1.353\text{kW/M}^2)}$$

FACTORS:

Fe: Environment

η_c: Charge eff

η_p: Packing

$$A_{SA} = 44.0\text{M}^2 = 474\ \text{ft}^2$$

$$W = 14.2\ \text{ft} = \text{BASELINE}$$

$$L \approx \frac{A_{SA}}{W} \approx \frac{372\ \text{ft}^2}{14.2\ \text{ft}} = 33.4\ \text{ft}$$

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Figure 4.4-8. Solar Array Sizing Advanced Cells

The basic design of the experiment wing for the advanced solar cell concept is shown in figure 4.4-9. The wing contains two blankets of 14.20 feet in width by 20.00 feet in length, with active cells covering the inter 17.50 feet. Each blanket contains 16 panels in length by 3 panels in width for a total of 48 panels. The 3-panel row at each end contains no solar cells, resulting in 42 active solar panels. All wing dimensions common with are the baseline solar array with the exception of wing length (the 20.00 feet measurement in line with the boom).

Because baseline components and fittings would be used where ever practical for the advanced solar cell concept to achieve maximum commonality, a calculation of wing weight for the experiment would result in a value that appears excessively heavy because a full-wing complement is not used. Blanket weight essentially comprises 54 percent of the wing weight, and this concept considers nothing to improve the remaining 46 percent of the wing.

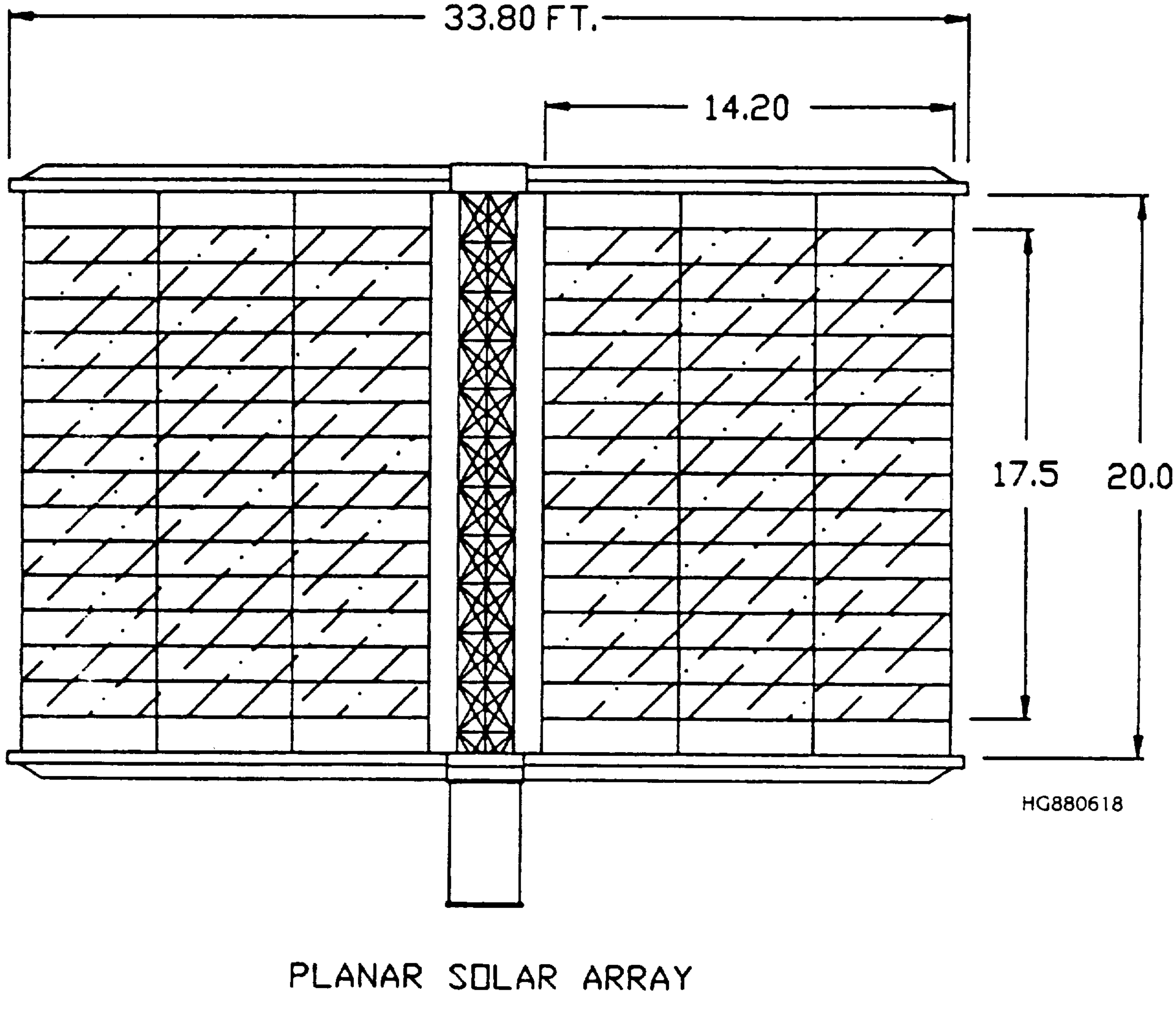

Figure 4.4-9. Wing Design (Advanced Solar Cells)

To show the benefits of this concept on the space station, the plot of Figure 4.4-10 shows only wing blanket weight and the potential weight savings versus solar cell efficiency --- based on a presumed baseline solar cell efficiency of 14 percent. Assuming a 19 percent solar cell efficiency as considered in this study, a savings of approximately 80 kilograms per wing could be achieved.

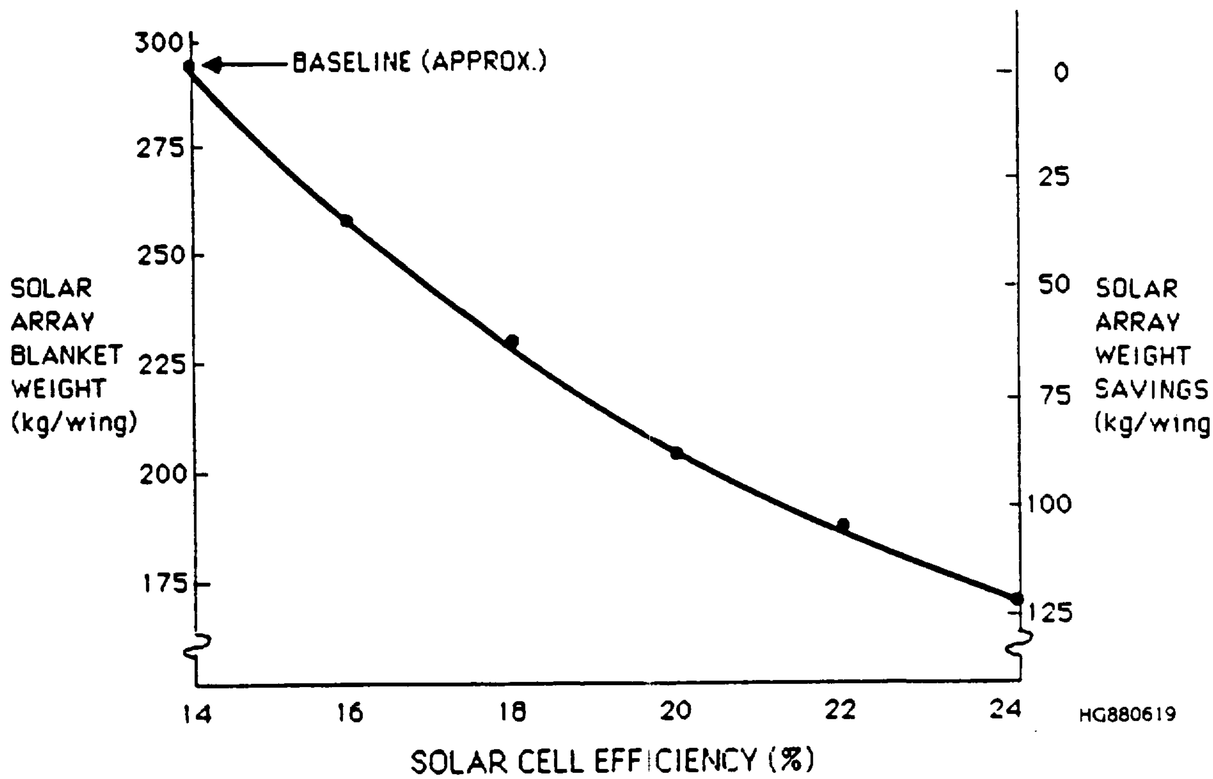

Figure 4.4-10. Potential Weight Savings For Advanced Solar Cells

The experimental requirements for the sodium sulfur battery are presented in this section.

The mid 1990s projected state-of-the-art advanced technology electrical energy storage system should be tested in low-Earth-orbit environment to aid in the development and demonstration of compliance with the design specification data to accompolish the optimum design goals as stipulated below:

1) high gravimetric and volumetric energy desity
2) high voltage-160 vdc nominal for LEO
3) reliability
4) 5 year life, LEO 30,000 cycles @ 80% DOD for Na/S battery
5) cost effective design
6) easy to manufacture, test and integrate
7) easily scaled, in voltage and A-hr capacity
8) thermal control

The design criteria of the sodium sulfur battery are listed in figure 4.4-11. Key criteria include 30,000 cycles over five years of life at a depth-of-discharge (DOD) of 80 percent. The battery is launched in the frozen state, and then heated to 350 C for operation in orbit.

DESIGN: Analysis, Fabricate, Testing and Demonstrate

350° C cell
5 year life
30,000 cycles (28,832/5 years @ 180 nautical miles, 54.8 min. charge, 36.4 min. discharge)
0.999 @ 0.98 confidence level
20 freeze/thaw cycles
30% packaging factor/battery configuration
50 AH cell
63 WH/lb. to 80% DOD (139 WH/kg)
50 WH/lb. minimum at 2 C discharge rate (110.23 WH/kg)
71 WH/lb. to 90% DOD (157 WH/kg)
6 m ohms/cell

Environment - (Frozen State)

-20 to 110° F ambient
3.45 psia to 15.53 psia
36 inch drop (transportation)
30 G's @ 11 ms duration (handling)
5 minutes 15 G's (launch)
0.25 G^2/Hz peak @ 300 to 1000Hz
1000 G's peak (pyrotechnic shock)

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Figure 4.4-11. Leo Sodium/Sulfur Battery (350° C) Design Criteria

The sodium sulfur battery is sized in Figure 4.4-12 for potential incorporation into a revised Space Station operating at 160V. A 60 percent DOD is assumed, and a 5 percent margin for cell failures. The resulting design has 93 series cells per battery. Each photovoltaic power module requires 2 batteries, for a total of 184 cells. This results in 736 cells for the 4 power modules. Total battery weight is 434 kg (957 lbs).

BATTERY SIZING FOR 160V

$$I_{total} = \frac{95000\ W}{160\ V} = 594\ A$$

Use 12 parallel strings @ 49.5 A/string

Cell Discharge Voltage ≈ 1.845V between 10% & 60% DOD

Assuming Battery Discharge Controller $\eta = 0.99$

Number of Cells (assuming 5% redundancy)

$$n_o = \frac{(160\ V)\ (105\%\ \text{for redundancy})}{(1.845\ V/\text{cell})\ (0.99\ \text{BDC efficiency})} = 92\ \text{cells/string}$$

Battery Voltage on Discharge = 170 V with no cells failed
= 161 V with 5 cells failed

Charge rate = C/2
Cell charge voltage = 2.230 V/cell

BATTERY - PACKAGING

Need 736 50 AH cells. (2 Batteries/Photovoltaic Power Module. = 100 AH x 80% DOD = 80AH or 22.4 kW) (2 x 92 cells = 184 cells x 4 power modules = 736 total cells)

Ball park weight for battery system:
(736 cells) (1.0 lb/cell) (1.30 packaging factor) = 957 lb

Lower weights are possible at increased risk (lower reliability and/or decreased life)

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Figure 4.4-12. Basic Leo Battery Design Sodium/Sulfur Battery (350° C)

Sizing the sodium sulfur battery for the experiment is based on an average discharge voltage of 160 VDC. The number of series cells is based on a voltage of 1.845 volts per cell. The 92 cells of the 50 AH battery have a net weight of 54.4 kg (120 lbs) as summarized in Figure 4.4-13.

Battery Sizing Requirement:

Cell charge voltage = 2.23V @ 30°C/cell

n_0 = (160V) (105% for redundancy) / (1.845V/cell) (0.99 efficiency) = 92 cells

30,000 cycles @ 80% DOD, LEO 5 year life

80% depth of discharge (DOD)

50 AH capacity

40 AH @ 80%

170 volts @ 1.845V/cell (10-60% DOD)

161 volts @ 1.845 V/cell (10-60% DOD), with 5 cells failed)

6.8k WH @ 80% capacity

C/2 charge rate

BATTERY - PACKAGING

Need 92 50 AHcells (1 Battery/Photovoltaic Power Module)

Ball park weight for battery system:
(92 cells) (1.0 lb/cell) (1.30 packaging factor) = 120 lb

Lower weights are possible at increased risk (lower reliability and/or decreased life)

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Figure 4.4-13. Experiment Battery Design Sodium/Sulfur Battery (350° C)

The 92 cells of the sodium sulfur battery are packaged into an ORU of standard base dimensions (23" X 25"), but the height is increased to 14 inches to accommodate the full cell length as shown in Figure 4.4-14. The cells are enclosed in a 36-layer thermal blanket. Heater elements are located in the base plate, which is also in physical contact with the thermal control radiator.

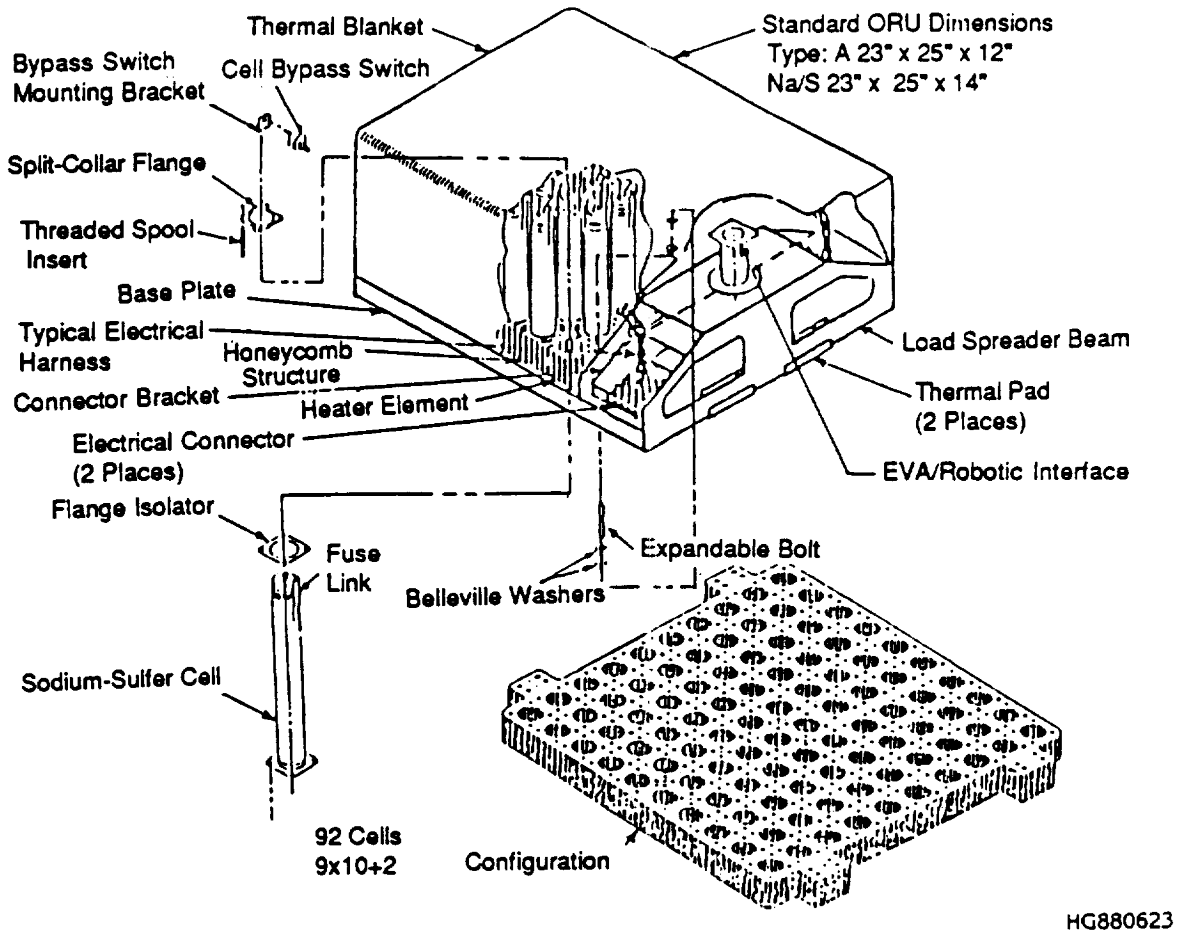

Figure 4.4-14. 92-Cell Battery Assembly

The experimental requirements for the nickel hydrogen battery are presented in this section.

The mid 1990s projected state-of-the-art advance technology electrical energy storage system should be tested in low-earth-orbit environment to aid in the development and demonstration of compliance with the design specification data to accompolish the optimum design goals as stipulated below:

1) high gravimetric and volumetric energy desity
2) high voltage-160 vdc nominal for LEO
3) reliability
4) 5 year life, LEO 30,000 cycles @ 70% DOD for Ni/H2 battery
5) cost effective design
6) easy to manufacture, test and integrate
7) easily scaled, in voltage and A-hr capacity
8) thermal control

Sizing the nickel hydrogen battery for the experiment is based on an average discharge voltage of 160 VDC. The number of series cells is based on a voltage of 1.25 volts per cell. The 128 cells of the 130 AH battery have a net weight of 402.3 kg (885 lbs) as summarized in Figure 4.4-15.

BATTERY SIZING REQUIREMENTS

Cell charge voltage = 1.40V @ 30°C/cell

n_0 = 160V/1.25V/cell = 128 cells

30,000 cycles @ 70% DOD, LEO 5 year life

70% depth of discharge (DOD)

130 AH capacity

91 AH @ 70%

160 volts @ 1.25V/cell (10-60% DOD)

15k WH @ 70% capacity

25 AH charge current

42 WH/kg @ 100% DOD energy density

BATTERY - PACKAGING

Need 128 130 AH cells. (1 Battery/Photovoltaic Power Module)

Ball park weight for battery system:
(128 cells) (5.41 lb/cell) (1.278 packaging factor) = 885 lb

Lower weights are possible at increased risk (lower reliability and/or decreased life)

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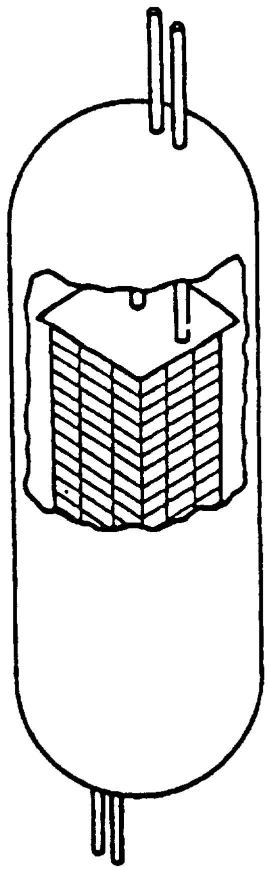
Figure 4.4-15. Experiment Battery Design Bipolar Nickel-Hydrogen Battery

The 128 cells of the nickel hydrogen battery are packaged into one common pressure vessel as shown in Figure 4.4-16. Stack dimensions are 27"x27"x19", and the plate area is 390 square inches. The battery will be cooled by active cooling techniques.



BATTERY SPECIFICATIONS FOR 160V SYSTEM

• POWER	21kWH
• LOAD VOLTAGE	160V
• THEORETICAL CAPACITY	130AH
• DISCHARGE CAPACITY/ AT 0.6 hr ECLIPSE	78AH
• DEPTH OF DISCHARGE	70%
• SERIES CELLS	128
• PLATE AREA	390In.^{2}
• CELL THICKNESS	0.095In.
• STACK DIMENSIONS	27x27x19 In.
• BATTERY WEIGHT	885lb
• ENERGY DENSITY	42WH/kg AT 100% DOD
• VOLUMETRIC ENERGY DENSITY	780WH/ft^3
• VESSEL CONFIGURATION	TBD

Conceptual drawing of a 21KW bipolar constructed nickel hydrogen battery including specifications.

Figure 4.4-16. Basic Bipolar Ni/H$_2$ Battery Design

The experiment assembly is illustrated in Figure 4.4-17 and is comprised of the solar array, energy storage, dynamic load, and experiment control. The Integrated Equipment Assembly (IEA) dictated the commonality of the experiment equipment, which is packaged in standard or non-standard Operational Replacement Units (ORUs). The sodium sulfur battery was packaged in an ORU of non-standard height (14" vs 12") to allow room for the full cell length.

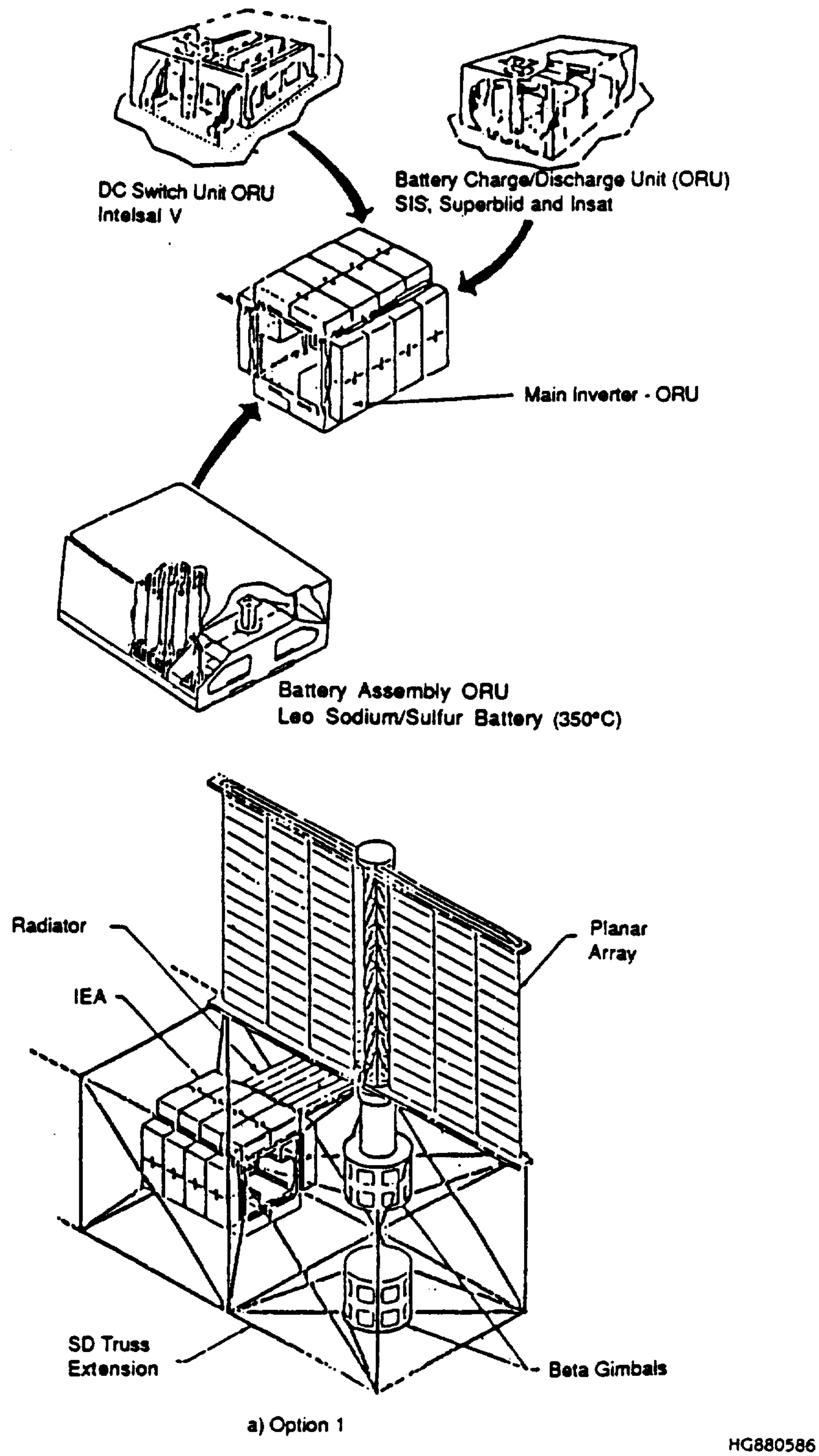


Figure 4.4-17. Experiment Assembly

4.5 Experiment Conceptual Design

The proposed location for the Solar Array/Energy Storage TDM is at one end of the Space Station power boom as shown in Figure 4.5-1. It would be mounted to the truss extension to be used for the solar dynamic power system on the Phase II station. The truss, utility trays and the beta gimbal would be delivered to the Space Station early for use by this TDM. A solar array technology demonstration experiment would be attached to this beta gimbal. A second beta gimbal could be located on the opposite side of the truss for a second solar array experiment. This location provides the required viewing and pointing without adding a dedicated two-axis pointing system. The power output of the experiment could also be conveniently added to the power obtained from the station arrays.

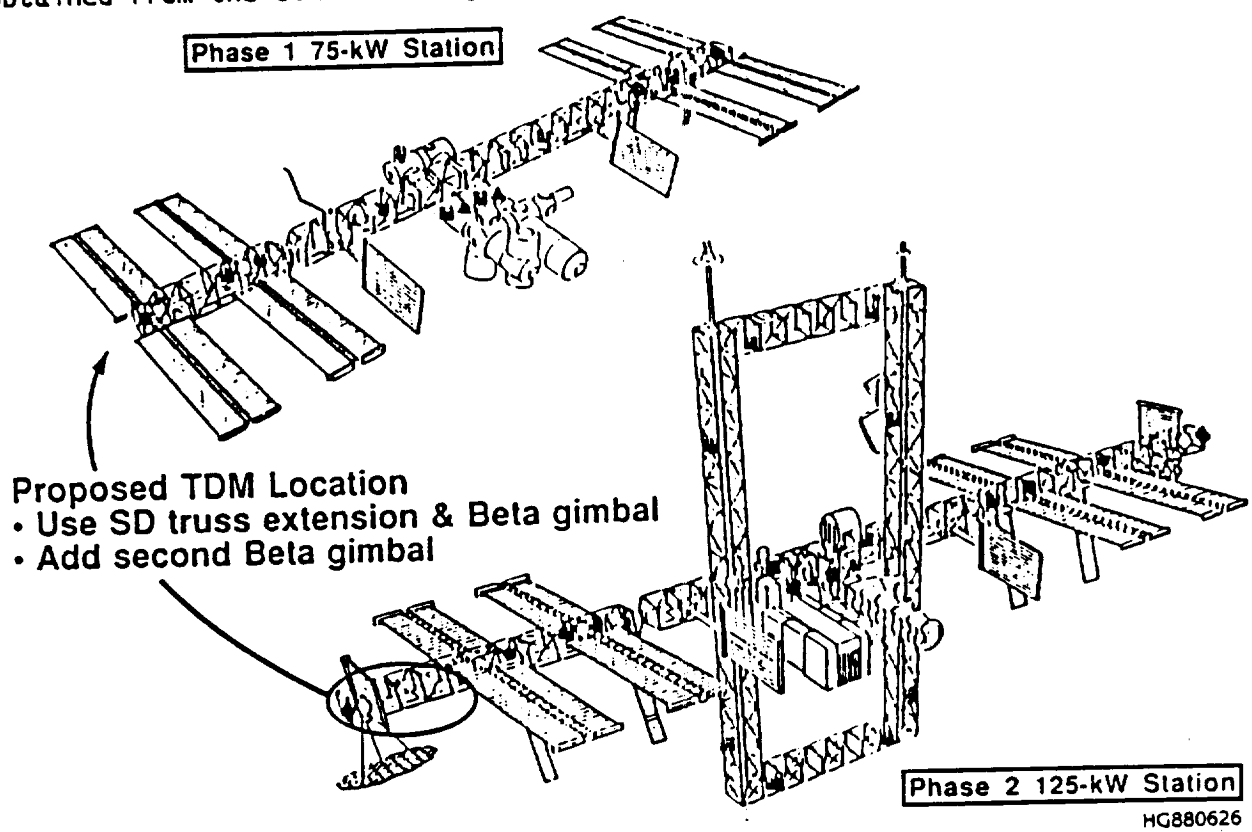


Figure 4.5-1. Solar Array/Energy Storage Experiment Location

Details of the experiment configuration are shown in the drawing (Figure 4.5-2) of the last two bays of the truss extension. Both solar array experiments identified as promising candidates are shown. The TDM would be less complex and less expensive if the two solar array technologies and battery concepts were demonstrated sequentially instead of simultaneously. However, for this study, both solar array concepts and both battery technologies will be included.

The planar array and the concentrator array are shown attached to their respective beta gimbals on opposite sides of the truss. The planar array storage and deployment hardware are the same as the baseline station arrays. Only the solar cells and possibly the blanket to which they are attached are unique to the experiment. The Integrated Equipment Assembly (IEA) concept proposed for Space Station is used to contain both the required electronics and the experimental energy storage technology experiments. The equipment radiator attached to the IEA is also common to Space Station (possibly with less surface area).

If the experiments do not require the number of ORUs provided by the IEA, a portion of an IEA can be attached to the truss using the Payload Interface Adaptor (PIA) and Station Interface Adaptor (SIA) concept planned for the attachment of other payloads to the Space Station truss. Not shown in the drawing are the connecting utility trays that contain power and instrumentation cabling.

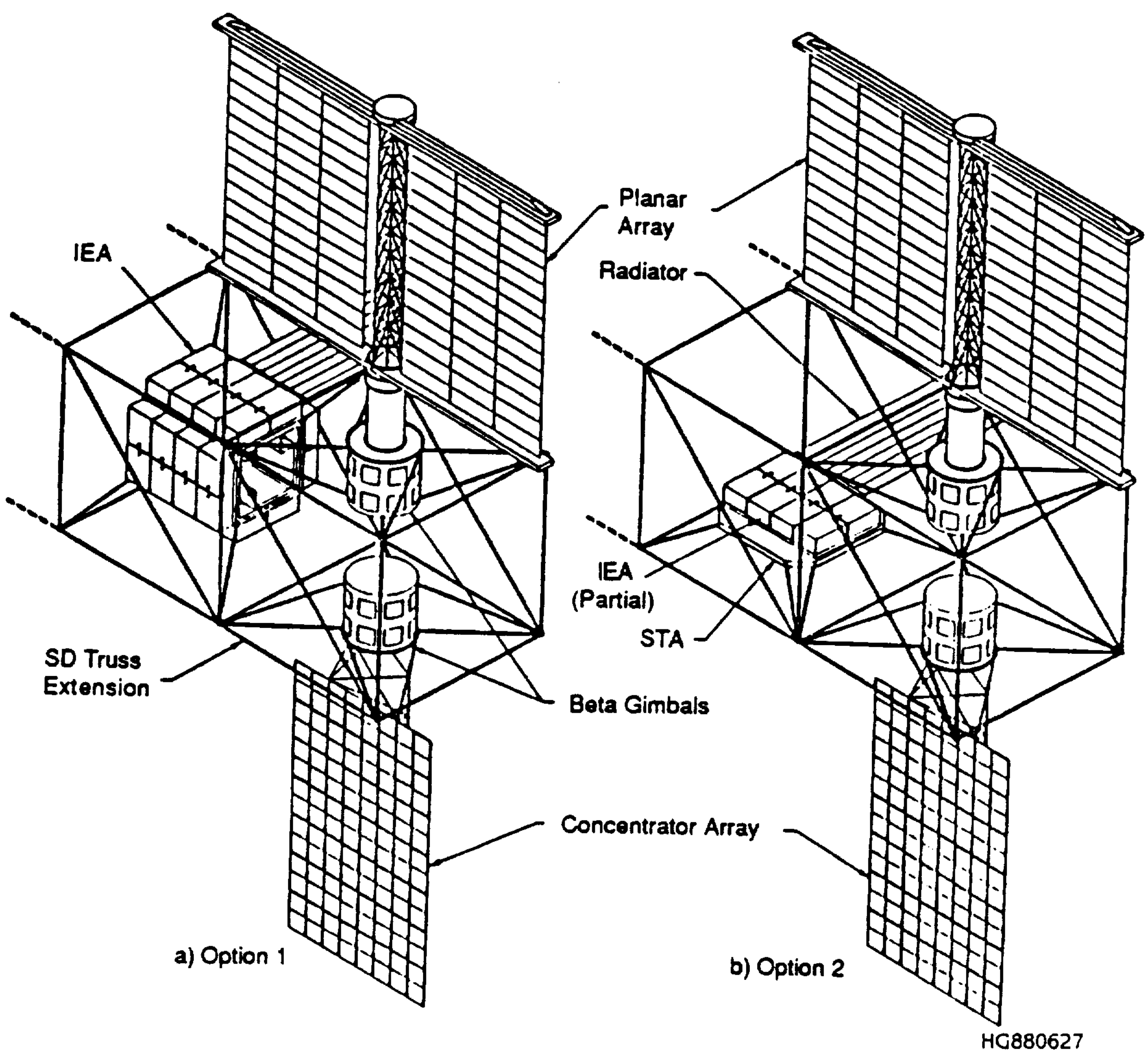


Figure 4.5-2. Solar Array/Energy Storage TDMX2151-Configuration

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Estimates of the weights of experiment components and flight support equipment are shown in Figure 4.5-3. The truss, utility tray and launch support equipment weights are based on the results of a previous study to define an STS flight experiment to demonstrate the assembly of the Space Station truss. Solar array and battery weights were derived during this study. The Beta gimbal and IEA weights were obtained or derived from Space Station Power System reports. The total weight of the packaged experiment is 10,255 lb (4638 kg).

Component	Wt. (lb)	Mass (kg)
Truss (5 bays)	948	430
Utility trays	1732	785
Beta gimbals (2)	1210	549
IEA (less batteries)	500	227
Planar array assembly	432	196
Concentrator array assembly	271	123
Sodium-Sulfar batteries	120	54
Bi-polar NiH2 batteries	885	401
Raditor panels	260	118
Truss component canister	368	167
Utility tray cradle	500	227
IEA/solar array cradle	3000	1361
Totals	10255	4638

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Figure 4.5-3. TDMX2151-Weight Summary

Figure 4.5-4 shows the proposed responsibilities for the operation of the TDM. Transportation of the packaged experiment is accomplished using the Orbiter RMS and the MRMS under the control of IVA crew members. The experiment is assembled via EVA with an IVA crew member controlling the MRMS and monitoring the work. Once assembled the experiment is deployed and checked-out from the ground with some assistance of the Space Station crew to verify on-board monitoring capabilities. Primary responsibility for the monitoring of experiment performance lies with the ground crew. Status of the experiments can be monitored on the Space Station, and, for safety, contingency alarms will also be available at the Station. The ground crew will also be responsible for conducting periodic tests of the experiments to obtain data for determining performance profiles. These will be used to determine the need for maintenance, the performance of which may require the assistance of the station crew. Primary responsibility for contingency planning and operations is shared with both the ground and the Station crews. Some contingency operations may require EVA.

Operation	Responsibility		
	Space Station		Ground
	EVA	IVA	
• Experiment Assembly	P	S	
• System Checkout		S	P
• Experiment Performance		S	P
• Periodic Performance Profiles			P
• Maintenance	S	S	P
• Contingencies	S	P	P

P = Primary S = Secondary HG880629

Figure 4.5-4. TDMX2151 Operations

The experiment and its supporting truss structure will be assembled by two EVA astronauts with assistance from the mobile remote manipulator system (MRMS). Experiment assembly is accomplished in three eight-hour days. The tasks accomplished in each of the three days and the corresponding man-hours needed for each is shown in this table.

The ground rules and basis for the operations and times are as follows:

- o EVA starts when battery power is turned on.
- o All EVAs include 2 EVA crew members and 1 IVA crew member.
- o Final EVA tasks end with connection of SCU
- o No contingency time is included, assumes everything goes smoothly.
- o Strut and node assembly times are based on EASE/ACCESS video tape analysis and "SAVE" study estimates.
- o "SAVE" times are multiplied by 3 to account for the lack of a construction fixture and crew foot restraints at assembly site.
- o Experiment packaging is based on RI concept for the PV modules.
- o Utility trays with utilities (electrical cabling, etc.) are packaged in a separate container.
- o The truss components (24 nodes, 30 longerons, 30 battens, 25 diagonals, 16 gimbal mounting struts and 8 IEA mounting struts) are in a separate container.

- o The MRMS will be capable of transporting the entire package from the docked Orbiter to the assembly site in one trip.
- o The RMS and MRMS will be controlled IVA from the Orbiter and SS cupola, respectively. All unloading and MRMS loading will be controlled by IVA RMS/MRMS crew members.
- o A safety wire used to assist crew translation is available along the length of the truss.

Day 1 - The experimental hardware is removed from the Orbiter bay and delivered to the end of the power boom using the MRMS. This task will be accomplished remotely by two IVA astronauts, one operating the Orbiter RMS and the other operating the MRMS. This task will be accomplished in one 8-hour shift.

Day 2 - After EVA preparations, which include donning suits, depressurizing the airlock, egress and translation to the construction site, the astronauts assemble the 5 truss bays. The MRMS is used to position one of the astronauts to aid in the assembly. Next, the two beta drives are installed on opposite sides of the last truss bay. The two solar array experiments are then attached to their respective beta gimbals (the solar arrays may be delivered already attached to their beta drives, which will save some assembly time). Installation of the integrated equipment assembly (IEA) within the fourth truss bay is accomplished next. The second day concludes with the installation of utility trays on the first truss bay after which the astronauts return to the Hab. The elapsed time for Day 2 is 6.4 hours.

Day 3 - EVA preparations are the same as those for Day 2. The assembly of the experiment continues by completing the installation of the utility trays in bays 2 through 5 and connecting the cabling to the beta drives. The equipment radiator is then attached to the IEA. With the experiment assembly complete, the astronauts secure all equipment containers and support hardware to the MRMS and return to the Hab. The elapsed time for day 3 is 7.2 hours.

Top level planing schedules for a precursor shuttle flight test and for TDMX2151 are shown in Figure 4.5-5. It also shows how the two programs are phased with respect to each other and also with the Space Station Program.

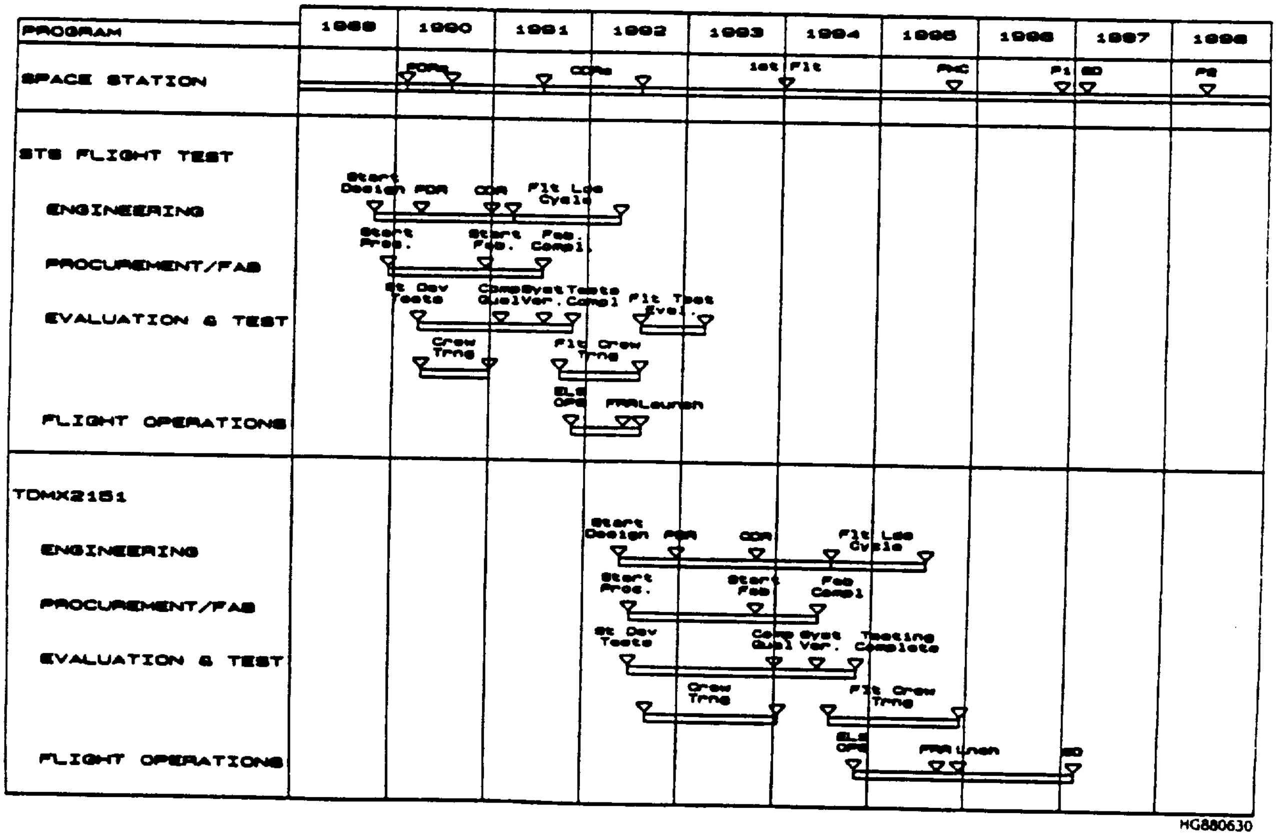

Figure 4.5-5. TDMX2151: Solar Array/Energy Storage Master Phasing Schedule

The STS flight test program covers approximately three years beginning in 1989, with PDR in early 1990, CDR at the beginning of 1991, and launch in mid-1992. Results from the flight test evaluation following the flight will be available to support the TDMX2151 program PDR and CDR.

The four and a half year TDMX2151 Program begins in mid-1992, during the time that the STS precursor test flight is being readied for launch at the Eastern Launch Site (ELS). The PDR occurs at the end of 1992. The CDR is near the end of 1993, just prior to the first Space Station delivery flight. The launch of the experiment occurs at the end of 1995, one or two months after Space Station permanently manned configuration (PMC). This will allow the experiment to operate for one year before the solar dynamic power system is delivered to the station in early 1997. A long operating time is needed for the experiment so that life cycle estimates, performance measurements and degradation characteristics can be determined.

4.6 Commonality with Space Station Equipment

As shown in Figure 4.6-1, the experiment makes maximum use of equipment that is common with Space Station. The truss structure, beta joints, planar array container & deployment mechanism, equipment radiator, power & instrumentation cabling, and Integrated Equipment Assembly (IEA) concept are envisioned to be the same as Space Station. The planar array will use new technology solar

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cells, and the blanket to which they are attached may, of necessity, be unique. The concentrator array, which requires more rigidity than the planar array concept, will be unique. Much of the equipment within the ORUs attached to the IEA are envisioned to be somewhat different from the corresponding equipment for Space Station due to the new technologies being demonstrated. However, many of the components that are used in the equipment may be common. The battery control module and charge control unit may be common with Space Station, but may require some modifications.

Equipment Item	Common	Unique
• Truss structure	X	
• Beta joints	X	
• Planar array container & deployment mechanism	X	
• Planar array		X
• Concentrator array and support structure		X
• Equipment radiator (attached to IEA)	X	
• Power & instrumentation cables	X	X
• Integrated Equipment Assembly (IEA) concept	X	
• Equipment contained in ORUs		
• Energy storage batteries		X
• Battery control module	?	
• Charge control unit	?	
• Load regulator unit		X
• Dynamic load		X
• Power distribution/switching		X
• Processor control unit		X

? - May be common or modified space station equipment

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Figure 4.6-1. Commonality With Space Station Equipment

4.7 Mission Requirements Data Base (MRDB)

The MRDB input form has been completed and is included in this report. The data shown assumes that both solar array technologies and both battery concepts are demonstrated in the TDM. Changes to the MRDB are summarized in Figure 4.7-1.

- **Solar array**
  - **Advanced solar arrays to be demonstrated by attaching the arrays at the end of the phase I boom**
  - **Solar concentrator technology arrays to be demonstrated on the opposite side of the truss**
  - **Use the power generated in the Space Station PMAD system**
- **Energy storage**
  - **Advanced energy storage techniques to be tested are:**
    - **Sodium - sulfur**
    - **Bi-polar Ni/H_2**
  - **Energy storage techniques to be tested at the same time as the solar array to be tested**

Figure 4.7-1. Changes to MRDB

The first page of the MRDB form, shows the proposed launch in late 1995 with the experiment operational throughout 1996. Experiment objectives are listed along with a brief description of the experiment. Estimates of TDMX2151 experiment characteristics and Space Station support requirements are contained in the subsequent pages.

4.8 Summary and Conclusions

TDMX2151, a solar array/energy storage technology development mission has been defined during this study. A state of the art assessment identified two solar array technologies and two energy storage technologies that show promise for significant performance improvements as shown in figure 4.8-1. Advancements in the technologies chosen are being made under on-going development programs in the aerospace industry, and will be ready for flight testing in the mid-1990s time frame. The four technology experiments have been incorporated into a TDM that has been designed to maximize commonality with Space Station equipment. As previously mentioned, the complexity and cost of the TDM could be reduced by demonstrating one solar array technology and one battery concept at a time.

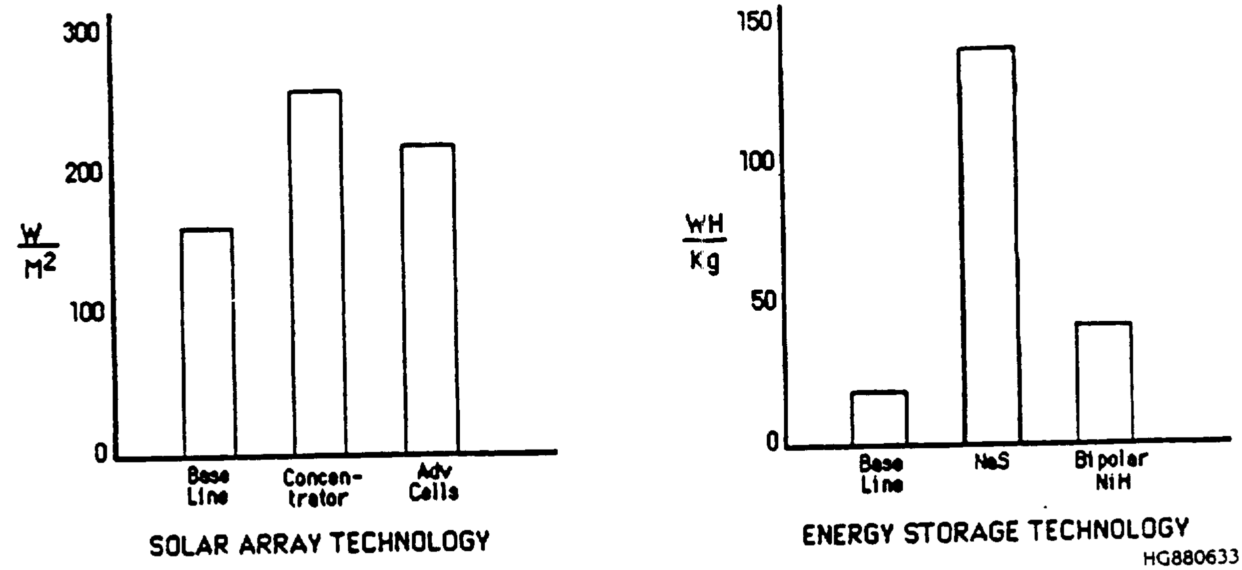

Figure 4.8-1. Performance Comparison

Once installed on the Space Station, the facility may be capable of accommodating other TDMX missions listed in the MRDB. These are identified on figure 4.8-2, which lists some of the other TDMXs listed in the MRDB or presented at the NASA RT&E Workshop that relate to power generation and its interaction with the space environment that may be able to use some of the TDMX2151 equipment or facilities.

- TDMX2152 - Large Space Power Systems Technology
- TDMX2511 - Space Power Systems Environmental Interference
- TDMX2512 - High Voltage in Space Plasma
- Environmental Interactions Experiment *
- Radiator Technology

* From NASA In-Space Research, Technology & Engineering (RT&E) Workshop, October 1985

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Figure 4.8-2. Experiments That May Share TDMX2151 Facilities

The result of this study is a preliminary definition of the Solar Array/Energy Storage Technology Development Mission, TDMX2151. The experiment must now be designed in more detail to define the details of its configuration and to identify its requirements more precisely.

We also recommended that a precursor STS flight test be conducted to reduce technological risks associated with the concentrator array experiment and the battery concepts. This Shuttle flight experiment needs to be defined in more detail.

Other TDMXs have been identified as being potential users of a facility similar to that required for TDMX2151. Therefore, an investigation into the ability of a single facility to accommodate several TDMXs is recommended.

5.0 SATELLITE SERVICING TECHNOLOGIES (TDMXs 2561, 2562, 2563, 2564, and 2565)

5.1 Objectives and Benefits of Technology Advancement

Most historical and current spacecraft systems are designed to last for a specified mission lifetime and then be abandoned. Some do not last for their intended duration; some are still returning valuable data or performing valuable services at the end of their design life. Often, a satellite's useful life ends because of something that should be easily repairable or replaceable: batteries wear out; propellants are depleted. Most actual failures should, in principle, be easily repairable; blown fuses, broken wires or connectors, failed power supply components. Even such complex things as chip failures in computers are routinely and quickly repaired in ground systems.

Extensive repairs were carried out on Skylab by the astronaut crews. Unscheduled maintenance and repair has been the rule on Shuttle and Spacelab flights. Shuttle missions have retrieved, repaired, and recovered satellites even when servicing provisions were not designed in. Some new spacecraft are being designed for servicing: the Hubble Space Telescope, Space Station. As systems grow in complexity and cost, our ability to assure long service life without servicing diminishes and the value of servicing increases.

For the foreseeable future, many if not most spacecraft assets will be in locations where human servicing (i.e. by EVA) is not possible or is very expensive. Only in instances where crew are on location because of a shuttle flight that can reach the asset (but was not paid for by the servicing mission), or where crew are on location by virtue of being onboard the Space Station, is human servicing affordable. Teleoperation and robotics offers a potential means of making servicing more available. The design impact, however, must be modest for satellite program managers to accept it. One can easily show by expected value analysis that the cost of putting in servicing provisions should be no more than a few percent of the satellite program cost. Unfortunately, most past studies of teleoperated or robotic servicing have approached the problem from the servicing point of view and put most of the design burden on the satellite side of the interface.

Investing in servicing technology is a strategic investment. The short-term payoff is small. The long-term payoff is significant extension of the useful life of space assets. Since the acquisition and deployment of space assets is a budget-limited process, in the long run the affordable inventory of working assets is proportional to the average service life. it is reasonable to expect advancement of servicing technology to do better than double the average service life, probably much better. <u>An effective servicing technology, including teleoperated and robotic servicing, could more than double the inventory of working space assets in the U.S. space program</u>.

5.2 Study Objectives and Tasks Summary

The objective of this part of the study was to define technolgy development missions accomplishing the objectives of TDMXS 2561, 2562, 2563, 2564, and 2565. Our approach to the effort began by reviewing the MRDB descriptions of these TDMXs to extract mission requirements. In addition, we noted that TDMX 2063 (on-orbit spacecraft assembly & test) has many requirements common with the servicing TDMXs 2561-5.

We prepared a comparative requirements definition and completed analysis of requirements commonality. This led us to group TDMXs 2561, 2562, 2563, and 2565 together. TDMX 2564, coatings maintenance, has unique requirements and was kept as a separate mission. We then developed initial concepts for each of the missions. In particular, the group of four servicing TDMXs led to a concept for a servicing technology demonstration satellite test article. This test article could also satisfy the needs of TDMX 2063.

At the midterm briefing, the COR asked us to look into low-cost alternatives even if they would only satisfy a portion of the identified requirements. We also, by this time, had obtained information on the GRO and AXAF spacecraft, which had been identified as possible test articles for these TDMXs. We re-examined the requirements in a task/goal matrix format and developed an evolutionary approach to the servicing technology missions. This also led to recognition that TDMX 2565, the thermal interface technology mission, could be satisfied almost entirely by ground testing. It was therefore defined as a separate mission.

The analysis was completed by defining logic networks and strawman schedules for the missions, determining space station accommodation requirements and impacts, and preparing MRDB draft inputs.

The task flow is shown in Figure 5.2-1

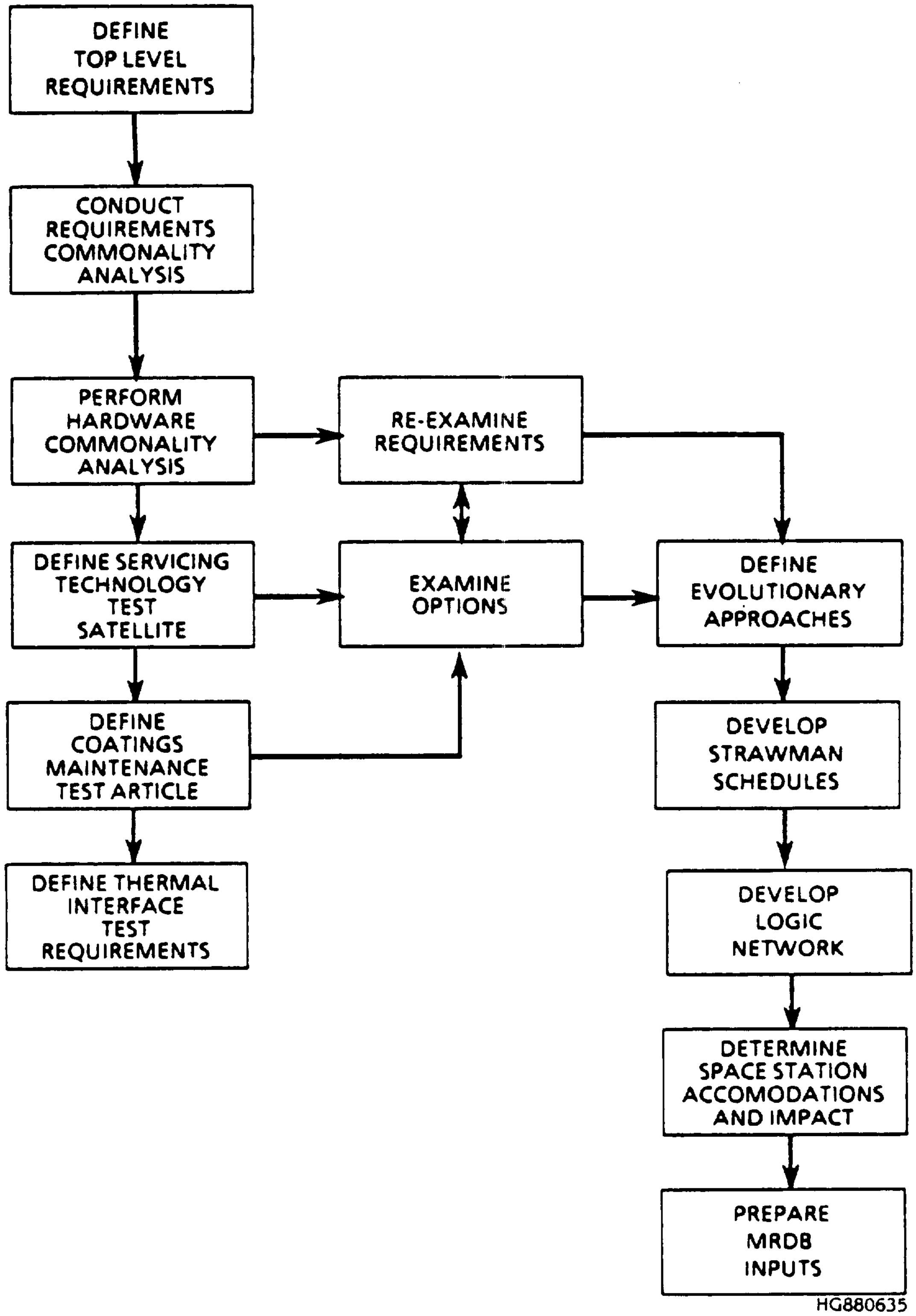

Figure 5.2-1. Satellite Servicing Overall Task Flow

5.3 Performance Goals

5.3.1 State-of-the-art assessment

Spacecraft servicing has a long history, a few highlights of which are listed in Figure 5.3-1. Ongoing programs, also listed, will accomplish further advances in the servicing state of the art. Historical and planned capabilities are summarized on the chart. Determination of servicing technology needs must take into account where the technology will be at the time the technology program begins.

Apollo/LRV Fender	EVA on Lunar Surface
Skylab	Extensive EVA and IVA repairs
Shuttle/Spacelab	IVA repairs
Solar Max	• Rendezvous and capture • EVA ORU changeout and component repairs
COMSAT Retrieval	Capture, Berthing and Safing
EASE/ACCESS	Structural Assembly
Hubble Space Telescope	Instrument and Subsystem Remove and Replace (EVA)
OMV	Remote Rendezvous, Docking, Placement and Retrieval
Space Station	• Assembly of Complex Vehicle • Indefinite Life Maintenance (IVA and EVA)

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Figure 5.3-1. Servicing History And Plans

5.3.2 Technology selections

Our estimate of servicing technology needs shown in Figure 5.3-2 was developed as a consequence of re-examining the requirements and options after the study midterm.

Need	Rationale
• Design requirements on Space Station servicing facility	• Minimize cost • Maximize safety and efficiency
• Minimize impact on satellite design	• Make servicing provisions a routine design practice
• Self-test and diagnostics techniques	• (Same)
• Remote teleoperation and robotics	• Enable servicing in GEO, polar, and other orbits where manned access is expensive or impractical
• Cryogenics fluids replenishment	• Extend life of systems with cooled sensors
• Assembly of complex, precision structures	• Large antennas and optical systems • Lunar/Mars exploration Vehicles

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Figure 5.3-2. Servicing Technology Needs

The servicing technology program should be conducted in time to provide design requirements for the Space Station Phase II servicing facility. This will avoid costs and lead to a safer, more efficient facility by enabling better tailoring of requirements.

Past efforts at design for satellite servicing have tended to impact satellite design in favor of simplifying servicing systems and operations. This makes servicing "non-user friendly" and servicing provisions have been designed in to only a few spacecraft. For servicing to become a routine design practice, it must be perceived by satellite designers as practical, available, and minimal impact.

This need also drives the next items, self-test and remote capability. Most satellites are not easily crew-accessible; remote operations require not only the remote servicing capability but also confidence, before the servicing mission is dispatched, that the necessary service is known and will restore the satellite to normal operation.

Transfer of cryogenics, principally liquid helium, is important to satellites with special sensors such as long-wave IR devices, e.g. SIRTF. This technology need can best be addressed by adding cyrogenic helium transfer to TDMX 2572.

Future exploration and science missions will require assembly of large, precision structures. Some will have special requirements such as assurance that aeroshell heat shields will not leak hot shock layer gas. Large structures construction is addressed by TDMX 2064 and 2461.

5.3.3 Technology readiness

Technology Readiness was not specifically addressed. The requirements analysis described in section 5.4 relates to technology readiness.

5.3.4 Performance

Performance goals were described in terms of test objectives, sine the performance goals are generally to demonstrate functional and opertional capabilities.

Top-Level objectives of this set of technology demonstrations are summarized in Figure 5.3-3 as taken from the MRDB. TDMX 2063 is included here as it has many requirements common with the satellite servicing TDMXs.

MRDB #	Title	Objective
TDMX 2063 (REF)	On-Orbit Spacecraft Assy/Test	Demonstrate and Verify the Feasibility of On-Orbit Assembly and Test of Spacecraft. • Construction • Contamination Removal/Control • EVA • IVA • Manipulator Operations • ORU Service/Maintenance • Refuel • Teleoperations • Test and Measurement • Tether Operations • Visual Operations
TDMX 2561	Satellite Servicing and Refurbishment	Demonstrate and Verify the Capability to Refurbish and Resupply LEO Operational Satellites. • Retrieve • Berth in Servicing Facility • ORU Changeout • Refueling • Other Repair/Refinishment • Replace in Operational Orbit

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Figure 5.3-3. Spacecraft Servicing Objectives From MRDB. (Sheet 1 of 3)

MRDB #	Title	Objective
TDMX 2562	Satellite Maintenance and Repair	Develop and Demonstrate Capability to Service Free-Flying Satellites at Space Station Inside Unpressurized Hangar. (Use AXAF) • Retrieve • Change Out Modules • Replenish Fluids • Spacecraft Checkout • Return to Operational Orbit
TDMX 2563	Materials Resupply	Develop and Demonstrate Capability to Retrieve and Resupply Material Samples or Modules from a Materials Processing Platform. Validate Remote Servicing Capability of OMV Using Smart Front End.

SPACECRAFT SERVICING OBJECTIVES
(SHEET 2 OF 3)

MRDB #	Title	Objective
TDMX 2564	Coatings Maintenance Technology	Demonstrate On-Orbit External Space Station System Refurbishment Using Active Beam Technology. • Assess Normal Contamination and Degradation • Cleaning • Resurfacing • Recoating
TDMX 2565	Thermal Interface Technology	Develop and Demonstrate Techniques to Remove Components From a Cold Plate Heat Sink in Orbit Via EVA and Replace Them, Restoring Adequate Thermal Conductivity by Proper Replacement of Fill Material.

SPACECRAFT SERVICING OBJECTIVES
(SHEET 3 OF 3)

Satellite Servicing
TDMX Objectives

Combined Spacecraft Servicing Objectives/ Requirements

- Parts Storage
- S/C Construction
- EVA
- IVA
- Manipulator Operations
- ORU Servicing Maintenance
- Refueling
- Teleoperating
- Test and Measurement
- Alignment Checks
- Visual Operations

- Berth in Servicing Facility
- ORU/Module Change-Out
- Component remove and Replace
- Thermal Fill Material Remove and Replace
- Replace Fluids
- Inspection, Repair and Refurbish
- On-Orbit OMVModule Replacement
- Return to Operational Orbit

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Satellite Servicing
TDMX Objectives

TDMX 2063 On-Orbit Spacecraft Assembly/Test	TDMX 2561 Satellite Servicing& Refurbishment	TDMX 2562 Satellite Maintenance & Repair	TDMX 2563 Materials Resupply	TDMX 2565 Thermal Interface Technology
• S/C Construction	• S/C Retrieval	• S/C Retrieval	• OMV Docking	• EVA
• Contamination Remove/Replace	•Berth in Servicing Facility	• Change of Modules	• On-Orbit OMV Module Replacement	• Remove & Replace Components
• EVA	• ORU Change-Out	• Replenish Fluids	• Platform (S/C) Co-Orbit Space Station	• Remove & Replace Fill Material
• IVA	• Refueling	• S/C Check-Out		• Verify Proper Thermal Conductivity
• Manipulator Operations	• Repair/Refurbish	• Return to Operational Orbit		
• ORU Servicing/ Maintenance	• Replace S/C in Operating Orbit	• OMV Docking		
• Refueling	• S/C Check-Out	• Use of Servicing Umbilical		
• Teleoperations				
• Test and Measurement				
• Tether Operations				
• Visual Operations				
• Release to Orbit/Co-Orbit				
• Alignment Checks				
• Storage on Space Station				

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Figure 5.3-4 presents a comparative listing of the objective of each TDMX considered for evaluation as part of the satellite servicing analysis. Also shown is a listing of the combined objectives as they would apply to a single supporting test article.

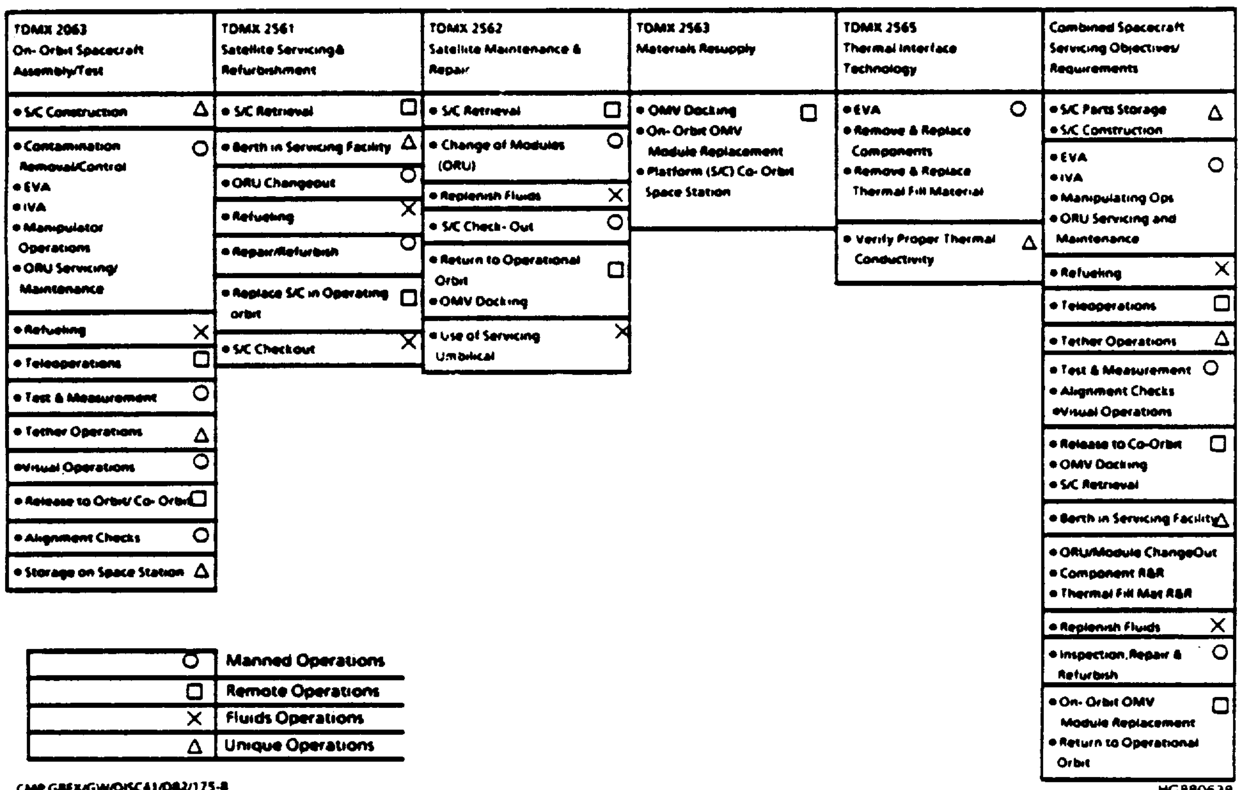

Figure 5.3-4. Spacecraft Servicing TDMX Objectives

5.4 Experiment Requirements

The study determined that the complete set of objectives of servicing technology development could only be met by developing a spacecraft test article to use in the demonstrations.

In response to questions raised at the mid-term briefing, alternatives to the use of a full system test article were examined even if they did not completely satisfy all the needs for technology demonstrations. This was to learn what could quickly and inexpensively be used to do precursor testing with the Shuttle before the Space Station and in the early phase of the Space Station development. Candidate systems include AXAF, GRO, sounding rocket payload packages and balloon payload packages. The AXAF and GRO will be active satellites for up to 15 years. It is not reasonable that these systems will be open for use as technology experiments while they are still active. Therefore, precursor missions are limited to sounding rocket packages and balloon packages. Data for figure 5.4-1 comes from the active German TEXUS sounding rocket program and the planned European Mikroba balloon packages (by the time of this briefing the experimental flight of the Microbia should have taken place). The figure gives an indication of the demonstrations that can be done with these facilities.

Combined Spacecraft	AXAF Preliminary	GRO	Sounding Rocket Package	Mikroba* Balloon Package
• S/C Parts storage	Yes	Yes	Yes	Yes
• S/C Construction	Support equipment assembly	No	No	No
• EVA	Yes	Yes	Possible	Possible
• IVA	Yes(Module Package)	Yes(Module Package)	Limited	Limited
• Manipulating OP'S	Yes	Yes	No	No
• ORU Servicing and Maintenance	Possible (not planned)	Possible (not planned)	N/A	N/A
• Refueling	Yes	Yes	N/A	N/A
• Teleoperations	Through robotic arm	Through robotic arm	N/A	N/A
• Tether operations	Possible (not planned)	Possible (not planned)	Possible	Possible
• Test and measurement	Yes	Yes	Yes	Yes
• Alignment checks	Yes EVA,IVA, Manipulator	Yes EVA,IVA, Manipulator	Possible IVA only	Possible IVA only
• Visual operations	Yes EVA,IVA, Manipulator	Yes	Yes	Yes
• Release to co-orbit	Yes	Yes	No	No
• OMV Docking	No	No Grapple Present	No	No
• S/C Retrieval	Yes	Yes	No	No
• Berth in servicing Facility	Yes	Yes	No	No
• ORU Module change-out	Yes (limited)	Yes (limited	No	No
• Component R&R	By IVA	By IVA	Yes, IVA	Yes, IVA
• Thermal fill material Remove & replace	No, must be designed in	No, must be designed in	No, must be designed in	No, must be designed in
• Replenish Fluids	Not Known	No	Yes	Yes
• Inspect, repair & refurbish	Limited	Limited	Limited	Limited
• On-orbit OMV Module Replacement	No	No	No	No
• Return to operational orbit	Yes	Yes	N/A	N/A
• MPS Payload	No	No	Yes	Yes
• Life Expectancy	15 Years	2 Years with 2-Year Extensions up to 15 years	------------	------------
• NASA 1700.7	Yes	Yes	Some	Some

* Mikroba carries sounding rocket and getaway special (GAS) sized packages

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Figure 5.4-1. Assessment Of Baseline Objectives

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Figure 5.4-2 summarizes a requirements analysis in which the functional requirements for demonstration of servicing technology are described in terms of (1) the issues and problems addressed; (2) the means of demonstration; and (3) the importance of accomplishing the demonstration or the risk accepted by not accomplishing it. Three questions were asked: (1) what type of issue is addressed by the demonstration task, (2) how can the demonstration best be accomplished, and (3) how important is it? Each question was asked for each demonstration requirement. The requirements are taken from the right hand side of Figure 5.3-4.

•	Objective/ Requirement	Issue/ Problem(s) (Demonstration Need)	Demonstration Status/Method(s)	Risk/Importance
1	S/C Parts Storage	Minimize crew member EVA translation from worksite to storage areas. Need a means of providing parts to work crews. Also must restrain/contain parts	• Carry-along pallets • Pallets moved and positioned by RMS • "Lazy Susan" device on large pallets for parts/equipment access • Solar Max had carry along pallet plus RMS handling of large ORU	• Tools, techniques and support equipment must be integrally designed and tested • Safety: collision, getting trapped, getting loose (e.g. from tether) suit damage • Work efficiency

2	EVA	• Safety and efficiency of EVA operations
		• Benefits/problems of enclosed work areas (hangars)
		• Ability to do intricate work and specialized tasks
			• Solar Max repair
			• Ease/access
			• Propellant transfer demo
			• COMSAT retrieval
			• HST (planned)
			• Space Station assembly (planned)
			  • Tests in simulated/ temporary hangar
			  • Specialized tasks such as electronics card changeout; welding
				• Important to further refine understanding of in-space maintainability design requirements
				• Must ensure/validate safety of planned operations

3	IVA	IVA repair of equipment	Will probably normally evolve with space station ops	No special requirements identified
4	Manipulation Operations	Manipulating satellites to install them in servicing facility, e.g. after OMV retrieval. Using manipulators to set up for servicing, minimizing EVA	Use OMV and RMS to set up servicing configurations. This could use a dummy spacecraft	Safety and efficiency thru minimizing EVA
5	ORU Servicing and Maintenance	(a) Selecting servicing approach and technology to minimize impact on satellite design for servicing (b) Selecting optimum replacement levels (c) Maximizing utility of teleoperation and robotics to minimize EVA	This calls for new design of test spacecraft to get greatest benefit. Test/validate jointly optimized satellite designs and servicing technologies.	Enabling technology for making serviceability a routine design practice for spacecraft.
6	Refueling	• Remote teleoperated or robotics refueling operations: Biprop, Hydrazine • Refueling non bladder systems	• Hydrazine transfer demonstration on shuttle flight. (Bladder system) • Demonstrate remote system by actual transfer • Demonstrate transfer into non-bladder systems	• Remote transfer enhances safety • Leak prevention • Non-bladder systems

7	Teleoperations	• Ability of teleoperations to accomplish servicing
		  • Remove/replace
		  • Delicate/intricate tasks
		  • Vision
		  • Time delay
		  • Testing/diagnostics
			• Most of this can be done in the lab (on the ground) with final verification in flight
			• Needs to be worked with FTS program
				Enabling technology for routine maintenance and servicing

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Figure 5.4-2. Requirements Analysis

	Objective/ Requirement	Issue/ Problem(s)	Demonstration Status/Method(s)	Risk/Importance
8	Tether Operations	Management of crew and equipment tethers to avoid tangling, fouling and other hazards	• EVA demonstrations with actual servicing scenarios • Can be done at space station with dummy satellite or no satellite	• Tethers are essential for safety • Coordination of mobility, safety, handling, and restraints • Safety and efficiency
9	Test and Measurement	• Ability to test and diagnose an operational spacecraft for FDIR • Accessing test points and applying test routines • Calibration, alignment and adjustment • Verification of successful repair and operational readiness	• Some of this can be done on an AXAF or GRO • Developing combined design/servicing approach for test points and measurements will require new design	Enabling technology for making servicing a routine design practice for spacecraft
10	Alignment Checks	• Structural alignment (jigs and adjustments) • Optical • Electrical/RF • Teleoperation/robotics vs human/EVA	• Devise appropriate tools • Demonstrate use • Some can be done in NBF and ground-based labs	• Some servicing jobs require this: • Structural assembly • Servicing optical systems • Servicing communications systems
11	Visual Operations	This is a subset of other tasks and capabilities	Inherent in other demonstrations	N/A
12	Release to Co-orbit	Not an issue - will be demonstrated on OMV program	N/A	N/A
13	OMV Docking	Not an issue -will be demonstrated on OMV program. A suitable test spacecraft could serve as a "practice" OMV target	OMV return to space station with spacecraft	Essential to OMV operations but expected to be demonstrated as part of OMV flight test
14	Spacecraft Retrieval	Not an issue -will be demonstrated on OMV program. A suitable test spacecraft could serve as a "practice" OMV target and payload	OMV return to space station with spacecraft	Essential to OMV operations but expected to be demonstrated as part of OMV flight test
15	Berth in servicing facility	A retrieved satellite must (a) if flying under its own power, be grappled by RMS and berthed (b) if retrieved by OMV, handed off to RMS and berthed. This is not an OMV program demonstration requirement	Demonstrate using OMV and test spacecraft. Requires a "flyable" test spacecraft	Enabling for those servicing missions that involve retrieval of a satellite and return to space station for servicing
16	ORU/Module Changeout	Need to develop and demonstrate changeout methods and design approaches that minimize impact on spacecraft and maximize utility of teleoperation and robotics	Was accomplished on Solar Max and planned for HST See 5 and 7 above	Enabling - See 5 and 7 above

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Figure 5.4-2. Requirements Analysis (cont'd.)

	Objective/ Requirement	Issue/ Problem(s)	Demonstration Status/Method(s)	Risk/Importance
17	Component Remove and Replace	Issue is between (a) Bringing ORUs into space station thru airlock for IVA service (b) EVA remove and replace (c) Teleoperated/robotic remove and replace	Was accomplished EVA on solar MAX and planned EVA for HST. Needs a test spacecraft designed to test alternative approaches	Important to advance state-of-the-art but not needed for basic servicing
18	Thermal Fill Material Remove and Replace	Same as 17	• No flight experience • This can be done with a simple test piece, or as a designed-in test on test spacecraft	• Enabling for component R&R where thermal fill material is involved
19	Replenish Fluids (a) Pressurized gases (b) Cryogenic helium. Task does not include cryogenic propellant transfer	• Devise and demonstrate an efficient method, e.g. thermal/supercritical, recharging of gas systems • Devise a feasible method of replenishing cryogenic helium systems in zero gravity	• Recharging high pressure gas with higher-pressure gas has been demonstrated but is inefficient • No experience with thermal/supercritical or cryo-helium replenishment • This can use either a special test article or a designed-in test on a test spacecraft	• Cryo-helium replenishment is essential to servicing systems such as SIRTF which require cryo-helium for instrument cooling
20	Inspect, repair and refurbish	This is a subset of prior tasks, especially 5, 7, 16, and 17	N/A	N/A
21	On-orbit OMV Module Replacement	Demonstrate remote, teleoperated servicing using OMV and FTS	This is the final flight demonstration referred to in item 7. Needs test spacecraft	Enabling for remote teleoperated servicing, e.g. at GEO
22	Return to Operational Orbit	Not an issue - will be demonstrated on OMV program	N/A	N/A
23	MPS Payload	This is a variation on 21, specifically the ability to change out an MPS payload: (a) Change out an MPS experiment on a free-flyer; (b) Change out an MPS product module on a prototype production unit carried by a free-flyer	Requires a test spacecraft with simulated or actual MPS payloads, and OMV/FTS	Enabling for MPS free-flyer platform operations

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Figure 5.4-2. Requirements Analysis (cont'd.)

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This analysis provided the basis for an evolutionary approach to servicing technology mission definition.

Several options were considered for meeting the servicing technology demonstration requirements, including existing spacecraft (used as test articles), existing or specially created test articles, and a test spacecraft designed exclusively for servicing technology flight testing. In order to ascertain the relative merits of the options, we continued the requirements analysis as summarized in Figure 5.4-3. Responses on Figure 5.4-2 were categorized according to the keys below the columns.

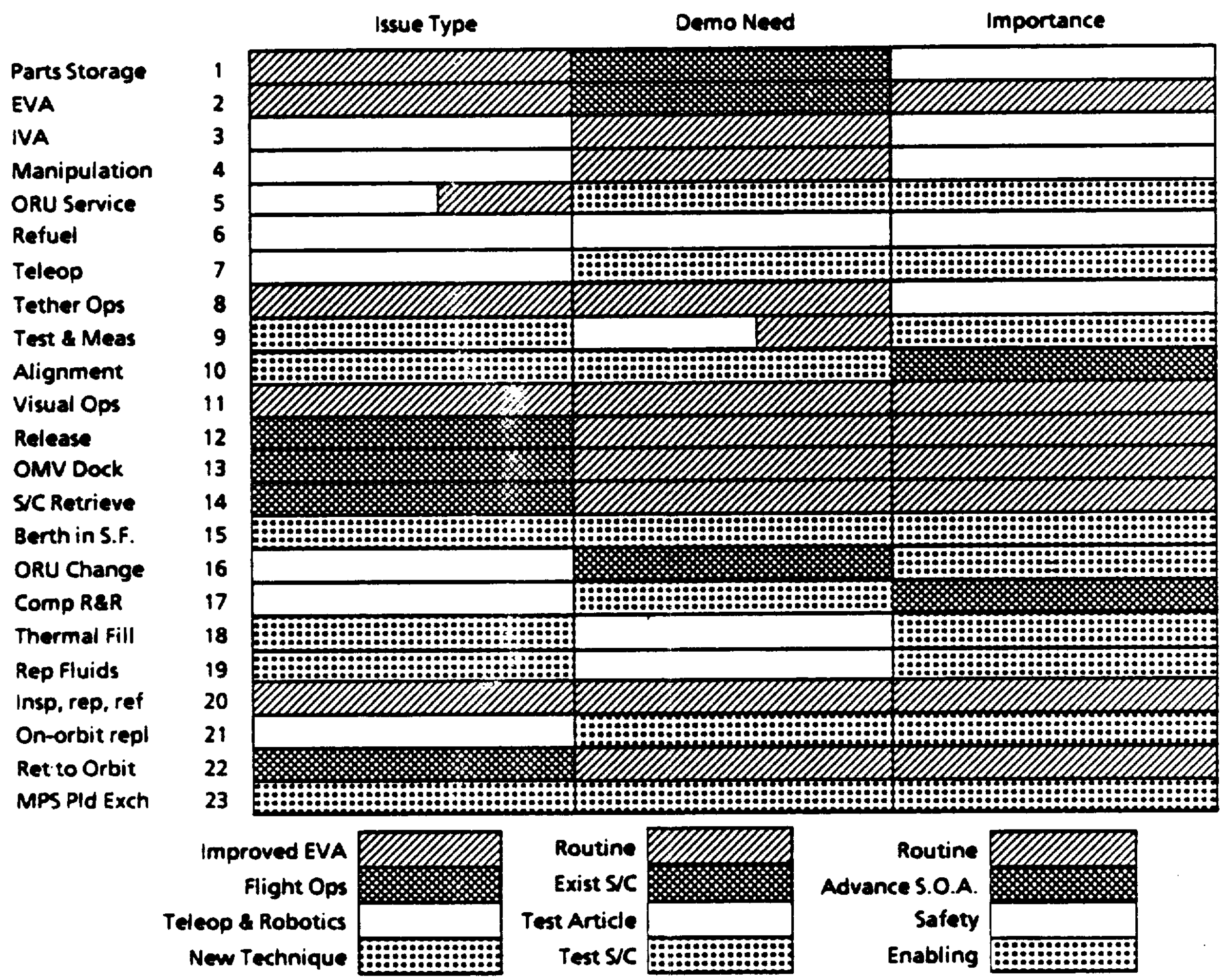

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Figure 5.4-3. Summary of Servicing Technology Requirements Analysis

"Improved EVA" and "flight operations" items are expected to mature with or without a special technology program, in the course of maturing space flight operations. "Teleoperation and robotics" and "new techniques" need a servicing technology program (this applies, for teleoperation and robotics, to those techniques unique to satellite servicing). Under the second column, "routine" means a demonstration that will occur as a routine part of servicing technology development. "Existing spacecraft" means that an existing spacecraft such as GRO could be used (it does not mean we believe

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the owners of the existing spacecraft would permit it to be so used). "Test article" means that a complete spacecraft is not needed for the demonstration; "test S/C" means a test spacecraft designed specially for servicing technology development. Under the third column, "routine" means the same as under column 2; "advance S. O. A." means the item would advance the state of the art of servicing; "safety" means that improved safety, typically of EVA, or reduced needs for EVA, will be obtained; and "enabling" means that a new servicing capability will result.

From this we concluded that some important demonstration needs require a special-design test spacecraft but that much important technology development can be done before such a spacecraft is built. This analysis pointed towards an evolutionary servicing technology program.

Figure 5.4-4 correlates servicing objectives described earlier to the functional requirements derived from the TDMX descriptions, and derives the main thrust of a servicing technology program.

	Servicing Facil Design Reqts	Min Impact on Sat Des	Self Test and Diagn Tech	Remote Teleop and Robotics	Cryo Fluid Replen	Assy Complex Precision Str
Parts stor/spares mgmt	•			•		
EVA	•	•				•
IVA		•				
Manipulation	•	•		•		•
ORU Serv	•	•		•		
Refuel	•	•				
Teleoperation	•	•		•		•
Tether Ops	•					•
Tests amd Meas	•	•	•	•		
Alignment	•	•		•		•
Visual Ops	•					•
Release to Co-Orbit	•					
OMV Dock		•				
S/C Retrieve		•				
Berth in S.F	•			•		
ORV Change	•	•		•		
Comp R&R	•	•	•	•		
Thermal Fill		•		•		
Repl Fluids	•	•		•	•	
Insp, rep, refurb	•	•		•	•	
On-orbit Repl.		•		•	•	
Ret to Orbit				•		
MPS Pld Exch		•		•		

Add Cryo Helium to TDMX2572 TDMX2064, 2461

Servicing Technology Program

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Figure 5.4-4. Correlation of Servicing Objectives to Functional Requirements

Other TDMX missions in the data base address cryogenics fluids replenishment and assembly of complex structures. These are noted on the chart. The servicing technology need not address these objectives, assuming that the noted TDMXs become approved programs. TDMX 2572 does not mention cryogenic helium; this should be added to that mission to develop the technology for helium replenishment in helium-cooled systems.

The servicing technology program should concentrate on the three objectives noted. In so doing, it will also determine and validate design requirements on the Space Station Phase II servicing systems.

5.5 Satellite Servicing Test Article Concept Design

5.5.1 Design concept

Figure 5.5-1 shows a combined satellite servicing test article in the front and back projections. The test article was derived from the unpressurized logistics carrier (ULC), used for Space Station rack, propellant and servicing gas transport. The ULC will be covered in 1/8th inch thick light weight aluminum panels for micrometeoroid and debris protection. The propellant ring will be fitted around the outer circumference and be fed by the propellant rack tank. An inner ring separating the two rack spaces will be the isolated electrical buss and standard electrical lines and connections for each section, including power lines, avionic controls, and payload support. All connections, electrical and mechanical will be plug-in quick disconnects. Communications and data will be transmitted through S-Band and Ku-Band antennae and be controlled by the avionics section. The avionics section will also control guidance, navigation and satellite control including the attitude control activation and deactivation, flyback and co-orbit maneuvers. The power will be supplied by two solar panels that are sections of the space station array panels. These will be deployed and stowed by extendable booms which can fully retract to the side of the test article. The arrays will flex and gimbal slightly to account for the beta angle drift.

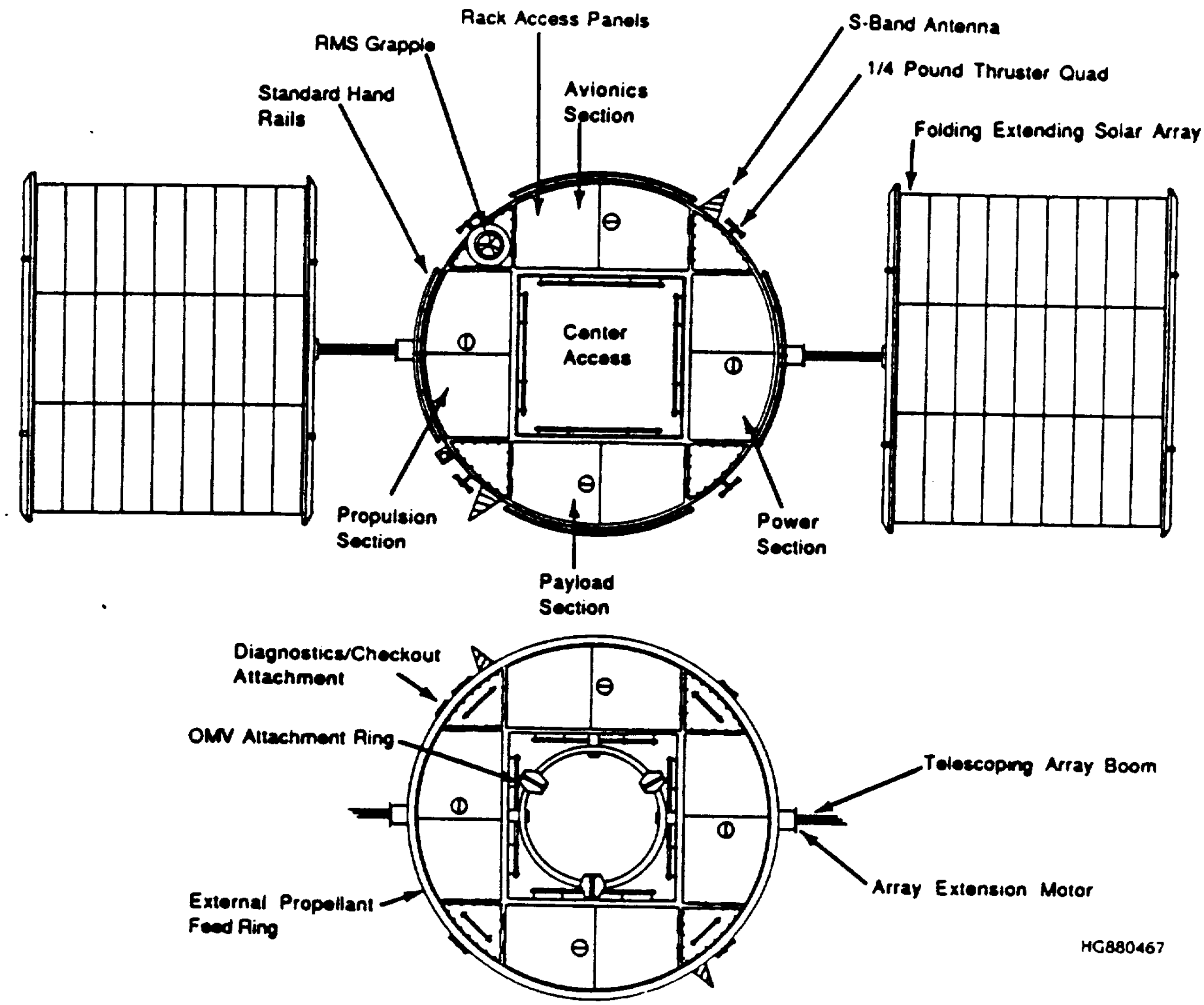

Figure 5.5-1. Satellite Servicing Test Article Plan Views

The power will be stored onboard by Space Station designed Ni/H_2 batteries, and distributed through the center buss ring. The baseline payload is considered to be protein crystal growth, a low power materials processing facility. It is considered to be self-contained except for power and, possibly cooling, which may be provided by heat pipes around the outside of the vehicle's electrical buss central support. The thermal load of the vehicle will be handled by spacecraft reflective/conductive coatings or the heat pipe arrangement, if required. The Orbit Maneuvering Vehicle (OMV) can be used for test article transport, and will attach to the test article in the attachment ring at one face of the article. The provision is made for the use of the Remote Manipulator System (RMS) with the placement of the grapple post at the corner of the articles' opposite face.

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The views of the Test Article in Figure 5.5-2 shows the Ku-Band antennae, the same type as used with the Space Shuttle, and the rack access into the test article. Rack access is from both sides of the test article, however with the OMV docked to the test article access to that side is restricted. Rack change out on the OMV docking side must be done with the test article restrained by the RMS. Human or robotic access can be done through the open center of the vehicle, although a part of this area could be adapted for additional payload space.

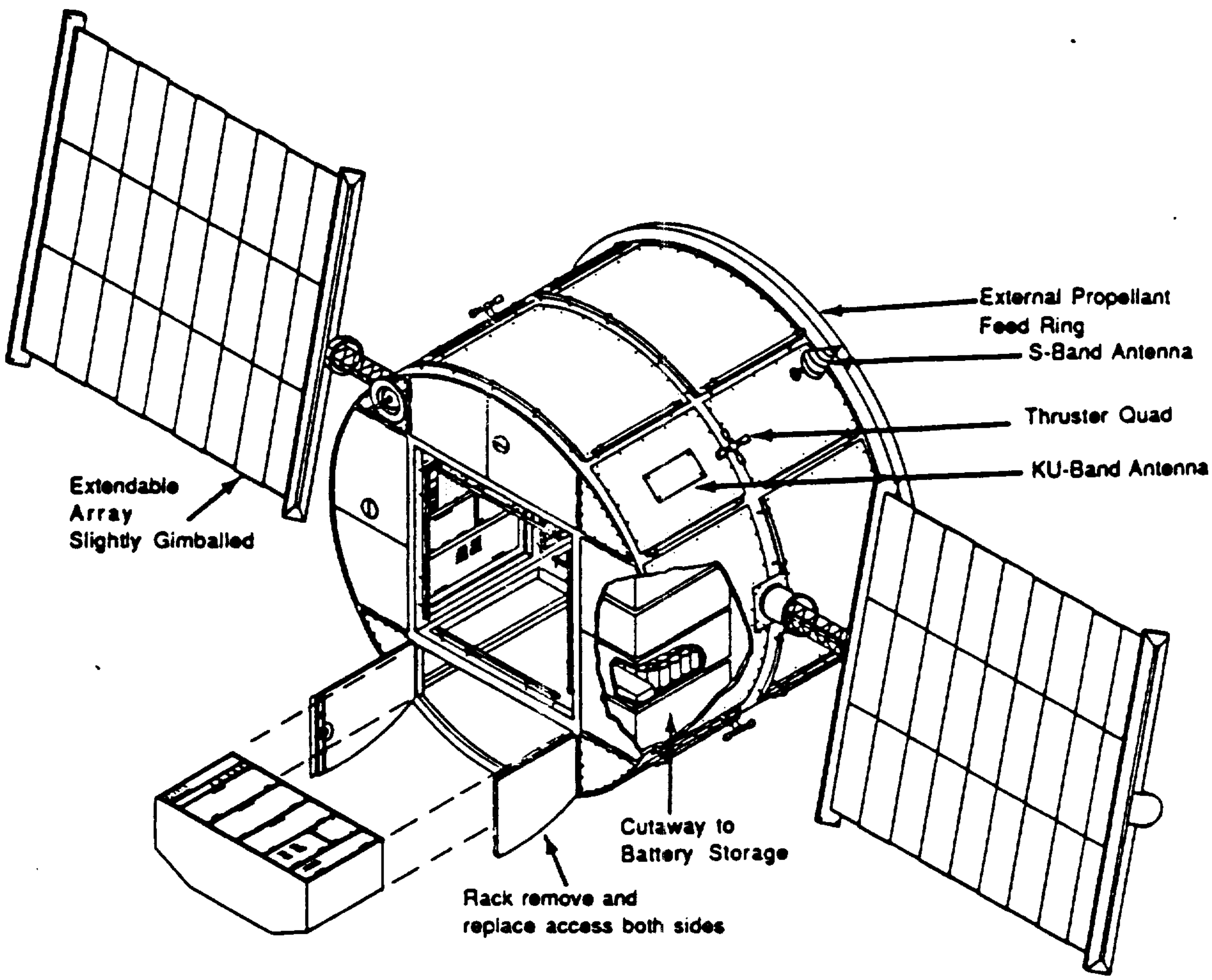

Figure 5.5-2. Satellite Servicing Test Article Angled View

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5.5.2 Satellite Servicing Test Article Equipment List

The Test Article consists of four main subsections-propulsion, avionics, payload support and the Power System. The Propulsion system uses a Gaseous nitrogen "cold gas" propellant in a pressure feed system. Nitrogen was chosen as a safe propellant to use with EVA operations. The control of the system will be through either an independent controller, or more likely, as a portion of the avionics system. This controller will tell the propellant system when it is enabled and disengaged, to prevent the test article from operating the thrusters during OMV docking operations, during test article assembly, and EVA involved rack change out operations. There is further discussion of the propulsion system hardware later in the text. The Avionics section contains the guidance, navigation and control system, the test article data and communications systems, and the propulsion system thruster controls. The test article's relative position will be determined by the star scanner and position sensors (sun sensors); the internal position sensing will be done with the Transfer Orbit Stage (TOS) Honeywell H760 3-accelerometer, 3-laser gyro system (to be space qualified for TOS). The GN&C computer will evaluate the test article's position and command thruster operations. The communications link is through the S-Band and Ku-Band antennae and processed through the communications computer. This system will be linked to the GN&C computer to accept commands for test article thruster operations involving, flyback to the Space Station, thruster enable/disable, attitude correction, test article health relay to the Space Station/Ground and data transmittal. All computers systems will use the Space Station Embedded Data Processing Computer (EDP).

Payload Support will vary according to the payload carried. Data from the payload will be fed into the avionics computer to transmit data (payload operations completion) and monitor the health of the payload. Some data recording may be done but it will be the responsibility of the payload system to record data or state the data requirements. This will also be true of the power conditioning and cooling systems. The payload will be as autonomous as possible. The power system will consist of solar arrays made of standard Space Station array sections. They will be mounted on telescoping booms that will extend the box containing the arrays away from the plume discharge of the thrusters (to present minimum blockage of the thruster plumes), and deploy the array. The array may be therefore be stowed close to the test article during boost phase, work on the vehicle or when flight assisted by the OMV. The arrays may be slightly gimbaled or twisted to follow the 57 degree range of the beta angle of the Sun, which will be tracked by a set of sun sensors in a control loop. The power distribution system will feed the other sections along the central electrical bus ring, to which all systems will connect.

5.5.3 Potential Users

We developed our estimate of the potential users for the combined satellite servicing test article in each of three possible use modes: as an MPS carrier, as an astronomy/Earth observation platform and as a space environments platform.

As an MPS carrier - the commercial companies listed in Figure 5.5-3 for all three possible facilities are ones that have expressed interest in such investigation. In fact some of the companies have applied for JEAs or TEAs with NASA in these fields. Most notable of these are McDonnell-Douglas, Battelle, Rockwell International, 3M and Hercules. Government health agencies may require large protein crystals to study that may require extreme isolation (HIV, Alzhiemers, legionnare's disease, etc.), for which this system could be used.

- Materials Processing
  - Protein Crystal Growth
    - Commercial
      McDonnell-Douglas
      Battelle
      Rockwell International
      Texas Medical Center
      Scripps Institute
      Summa Medical
      Lovelace Medical Foundation
      Upjohn
    - Government Agencies
      Atlanta Center for Disease Control
      National Institute of Health
      U.S. Department of Agriculture
      McDonnell-Aircraft Company
      Burroughs-Wellcome
      Dow Chemical
      Dupont
      Merck
      Schering
      Smith, Kline and French
  - Organic and Polymer Crystal Growth
    - Commercial
      3M
      General Motors
      Goodyear
      Vulcan Rubber
    - Government Agencies
      Department of Energy
  - Bio-reactor/incubator
    Phillips Petroleum
    Hercules
    GTE
    Celanese

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Figure 5.5-3. Combined Spacecraft Servicing Test Article Potential Users

- **Astronomy/Earth Observation**
  - **Astronomy**
    - **Universities with Astronomy Curriculum**
    - **Smithsonian Insitute (For Data Augmentation)**
  - **Earth Observations**
    - **Universities**
    - **Forestry Services**
    - **Mining Companies**
    - **Large Scale Agriculture Concerns**
- **Space Environmental Data**
    - **NASA Space Station**
    - **Observation Class Spacecraft Contractors**
    - **Communications Class Spacecraft Contractors**

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Figure 5.5-3. Combined Spacecraft Servicing Test Article Potential Users (cont'd.)

As an astronomy/Earth observation platform - this could support small instrumentation packages that would augment known data or provide detailed study of one area or over a short spectrum range. It would be useful to universities that as a class or school project need an inexpensive detailed study of a small area of the sky or a small path across the ground. Within the trace of the test article's path condition of crops and forests may be monitored and soil identification could be made within a narrow range of spectral data.

As a space environments monitor - in circling the Space Station with a mass spectrometer the test article can identify the local field of contamination around the Station. This field may interfere with platform experiments involving observations or wavelength attenuations as a principal part of investigations or operations.

5.5.4 Performance and weights for test article

Three types of missions were evaluated for feasibility on the basis of their required delta velocity budgets. The first was the standard materials processing in space mission in which the test article co-orbits in the same plane with the Space Station, as much as 10 kilometers outward of the Space Station ahead or behind it. The second mission is a fly around mission that would circle the Space Station, possibly to evaluate the contamination levels around the Station. The third would involve an elevated orbit, placing the test article in a higher orbit for the length of the mission. It was found that after seven days the nodal regression of the elevated orbit caused such a plane change that a considerable delta velocity was required to bring the orbit of the test article back into the same plane as the Space Station. This would increase the required propellant load to an unrealistic level for this vehicle. The calculated changes in velocities are summarized in Figure 5.5-4, rounded off to the next highest integer for each of the listed orbit effects.

MPS Mission

		Km/sec
Δ	Altitude (inital) (10 Km)	6
	Perturbations ..	5
Δ	Plane change . (0.007 degrees/day)	10
Δ	Altitude	6
Δ	Docking	5
Δ	Contingency ...	5
	Total	37

Status: Acceptable

Fly Around Mission

		Km/sec
Δ	Apogee (Initial) (20 Km)	6
	Perturbations .. (10 days)	1
	Drag makeup .. (10 days)	2
Δ	Perigee (Initial)	6
Δ	Apogee	6
Δ	Perigee	6
	Docking	5
	Contingency ...	5
	Total	37

Status: Acceptable

Elevated Orbit

	Km/sec
Orbit insertion	5
Perturbations	5
Nodal regression (for 7 days)	34.5
Orbit Return Rendezvoues	5
Docking	5
Contingency	5
Total	59.5

Status: Not Acceptable

Figure 5.5-4. ΔV Budget

Figure 5.5-5 lists N2 propulsion component weights and quantities. One 289 lbm, 17.32 cubic foot composite tank holds 432 lbms of high pressure (maximum 4863 psia) GN2. 75 feet of one quarter inch tubing weighing 0.1 lbm per foot carries pressurized N2 to 16 individual thrusters. The thrusters themselves are actually a series of two solenoid 1/4 inch valves connected to a simple expansion nozzle. Additional components necessary for control, pressure regulation, safety,quick disconnect,mounting hardware and N2 refill capability are also listed with weight estimations for each. High pressure tubing and fittings are used upstream of the pressure regulator. An approximate weight total for the entire propulsion system including a fully charged tank is listed as 807 lbms.

• **Cold Gas (Nitrogen) System Consisting of:**

Item	Quantity	Weight (Lbs.)
N_2 Tank	1	289
N_2 Gas	1	432
Fill Port (1/4")	1	0.5
Burst Disk(1/4")	1	0.5
Vent Valve (1/4")	2	0.5
1/4" Valve	36(1.4 Lbs. Ea.)	50.4
Pressure Regulator	1	5.0
Thruster Expansion Nozzle	16(1.0 Lb.)	16
N_2 Filter	1	0.4
Quick Disconnects	1	1.0
1/4" Tubing	75 /Ft.@0.1 Lbs. Per FT. 7.5	
Mounting Hardware		5.0
Total		807.8

Figure 5.5-5. Satellite Servicing Propulsion Equipment and Weights

The schematic of Figure 5.5-6 shows the relative positions of the N2 cold gas propulsion system components. A Brunswick Corporation space qualified composite tank (Kevlar overwrap with a titanium liner) used on the space shuttle Orbiter holds N2 at 4863 psia and 80 degrees F. The tank is mounted in a removable rack accessible from the docking face of the vehicle. The tank is fitted with a vent valve and a burst disk. The tank is filled and refilled through a .25 inch fill valve teed of from the main .25 inch high pressure line leading from the tank to the closeoff valves just in front of the pressure regulator. Quarter inch diameter, 0.083 inch wall thickness stainless steel tubing is used for all 74 feet of pressurized lines carrying N2 from the tank to the thrusters. The pressure regulator reduces the tank pressure to a 120 psia line pressure. The 16 one pound-force thrusters (physically just a valve/expansion nozzle combination) are grouped in sets of four to provide full maneuverability for the vehicle. These thrusters serve as both "main propulsion" and "RCS" propulsion. Pressure and temperature sensors are positioned to indicate both tank and line conditions. Thruster firing indicators are also placed at each thruster. Mounting hardware is necessary for all components and for the quarter inch feed line ring which is mounted directly to the circular frame of the logistics module. Each valve (solenoid) is controlled by the guidance, navigation and control computer.

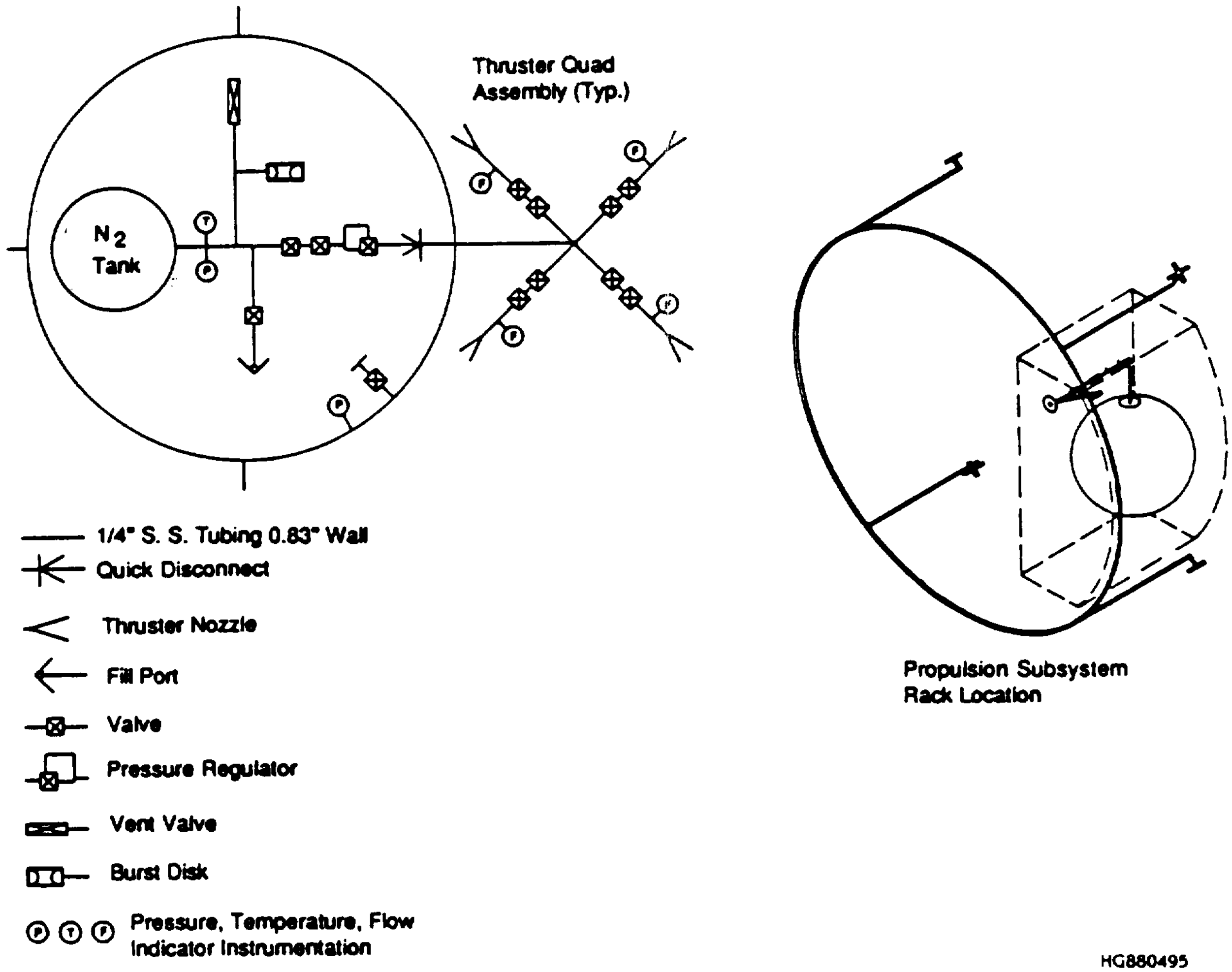

Figure 5.5-6. Propulsion System Schematic

The tanks fill valve mentioned above has an external access for refill during flight which necessitates a docking/mating connector for the system and also requires a "switch off" command from the refurbishment vehicle to prevent the propulsion system from firing in its attempt to regain positional stability during docking and experiment changeout.

Figure 5.5-7 is a listing of the estimated weights of the test article major components, based on either the expected weight of the component from Space Station sources or other spacecraft subsystems. The weight of the main support structure is that of the unpressurized logistics module (ULM). This is the current estimate from the Space Station logistics module group on the standard 14.7 feet by 8 feet ULM which carries 8 racks (two standard Space Station racks abreast). The external area of the ULM needs to be covered with 1/8 inch thick lightweight aluminum for protection against the O_2 atmosphere impingement and debris impacts. The racks and mountings will have to be secured in place and this is the proposed fitting weight. The propellant system includes the estimated weight of the propellant system and the support structure for the thrusters, the attachments for the propellant ring, and protection for the ring. The estimate for the power system weights are from the Space Station power supply system estimates. The MPS rack experiment is taken from the protein crystal growth revised weight given by the customer accommodations group of the Space Station. The avionics GN&C and communications weight estimate is based on the weight of the Inertial Upper Stage systems.

Item	Pounds	Kilograms
Unpressurized Logistics Module	2178	987.8
External Walls (1/8" Thick Aluminum)	300	136.1
Racks and Mountings	300	136.1
Propellant System	900	408.2
Power System:		
Batteries and Power Conditioner	500	226.8
Solar Array	87	39.5
Array Gimble	16	7.3
MPS Rack (Protein Crystal Growth)	441	200.0
Avionics-Communications and Control	700	317.5
Total	5422	2459.3

Figure 5.5-7. Satellite Servicing Weight Budget

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5.5.5 Test

Figure 5.5-8 is a table of the testing that must be accomplished to develop the test article for service.

Component Tests	Subassembly Tests	Subassembly Tests	Full System Tests	Ground Checkout Tests	Flight Tests
Array assembly actuation	Array actuation	Full array test	Array actuation	Subsystem c/o	Array actuation
Software development	Software tests: Tracking, GN&C startup-shutdown	Control tests-tracking simulation	Solar vacuum Vacuum	Computer checks	Communication c/o
Self-test system components	Self-test: Hardware/software	Self-test: RF/EMI tests	Self-test: mechanisms RF/EMI	Self-test:c/o	Self-test:c/o
Robotics: a) Prevention of "bent pin" condition for electrical connections	Robotics: flexibility with variations due to hardware differences	Robotics: access MPS interface Propulsion tests	Robotics: tests Assembly-neutral buoyancy with dummy system		GNC/propulsion suppression
b) Pattern recognition		Functional tests			
	Battery storage-power distribution	Communication tests	Docking tests (arm and OMV)		
		Accoustics/ vibration testing			

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Note: Components are basically off-the-shelf items and do not require component level tests.

Figure 5.5-8. Combined Spacecraft Servicing Testing Levels

Component tests- since most of the assemblies and sub assemblies are off-the-shelf items, very few components must be developed for this article. Those that must be developed are principally the ones related to demonstrating new technologies (robotics, self-test, software, and articulation of the solar array).

Subassembly tests- conducted principally to integrate the component development into functional subassemlies.

Subsystem tests- functional testing for subassemblies to work as subsystems independently and under the conditions that will be encountered in operations.

Full system tests- functional tests of the full test article for operations under conditions similar to those that it will encounter on-orbit or in transit flight.

Ground checkout tests- final on-ground checkout of mission critical operations and hardware.

Flight tests- functional on-orbit checkout of the critical systems prior to the test article service operations.

5.5.6 Evolutionary Program

While a rigorous analysis of the servicing technology requirements from the MRDB pointed to the test spacecraft just described, a step-by-step analysis of requirements showed the potential for an evolutionary program.

The logic network shown in Figure 5.5.9 presents an evolutionary servicing technology program.

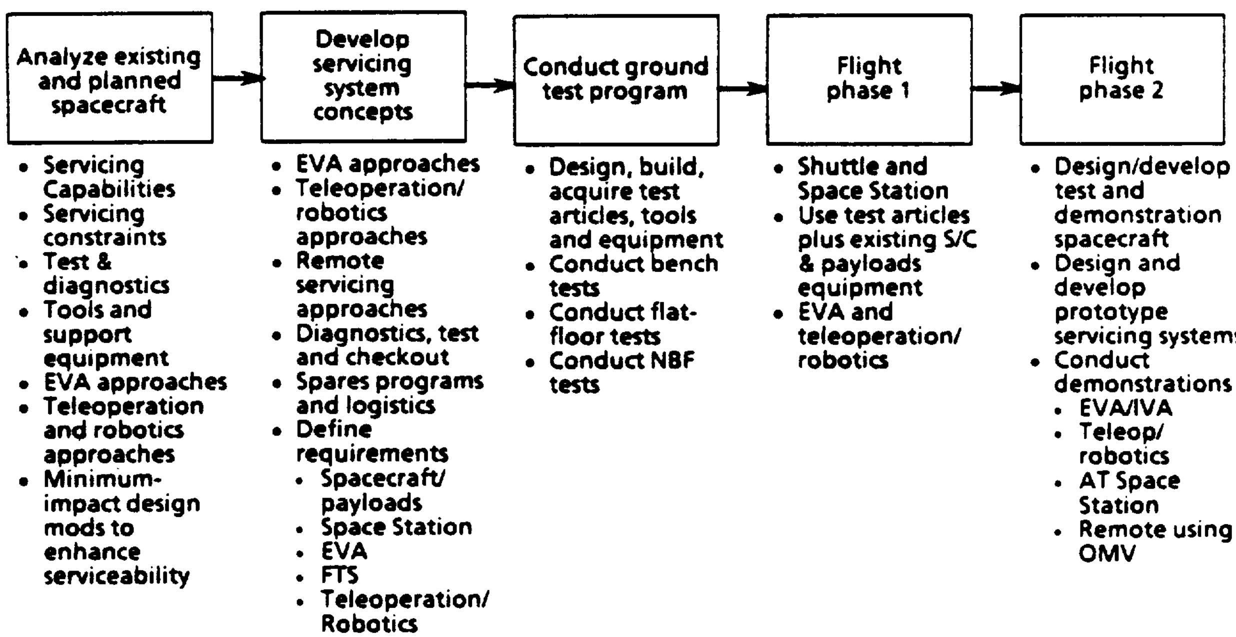

Figure 5.5-9. Servicing Technology Logic Network

The first step is a comprehensive, comparative analysis of existing and planned spacecraft to assess their suitability for servicing as designed, any servicing features included in the design, and any constraints to servicing created by the design. The second step is development of servicing concepts. These steps are aimed at minimizing the impact of servicing on spacecraft design (design the servicing technology to deal with, insofar as possible, the real world of contemporary spacecraft design rather than asking spacecraft designers to design in such a way as to make the servicing technology job easy), and to define what has to be done to make remote servicing practical.

Once the overall approach to servicing systems is defined, the program evolves through three typical experimental phases. It is our present perception that most of the requisite technology advancements can be accomplished by ground-based testing, and that the flight programs will accomplish and demonstrate successful integration of the elements of the technology in actual flight operations.

Phase 1 of the flight program is defined as everything that can be done without a specially-designed servicing technology spacecraft; phase 2 completes those demonstrations that require the specially-designed spacecraft. Phases 1 and 2 of this program are not the same as Phases I and II of the Space Station program. The servicing technology program should be completed before Phase II servicing capabilities of the Space Station are implemented, so that the derived design requirements and criteria can be folded into that program.

The schedule of Figure 5.5-10 follows the logic chart of the previous page and illustrates representative schedule phasing with development of operational servicing systems. It indicates that the study phases of this effort should begin next year in order to have Space Station tests ready to go when the Space Station begins operations according to current planning schedules.

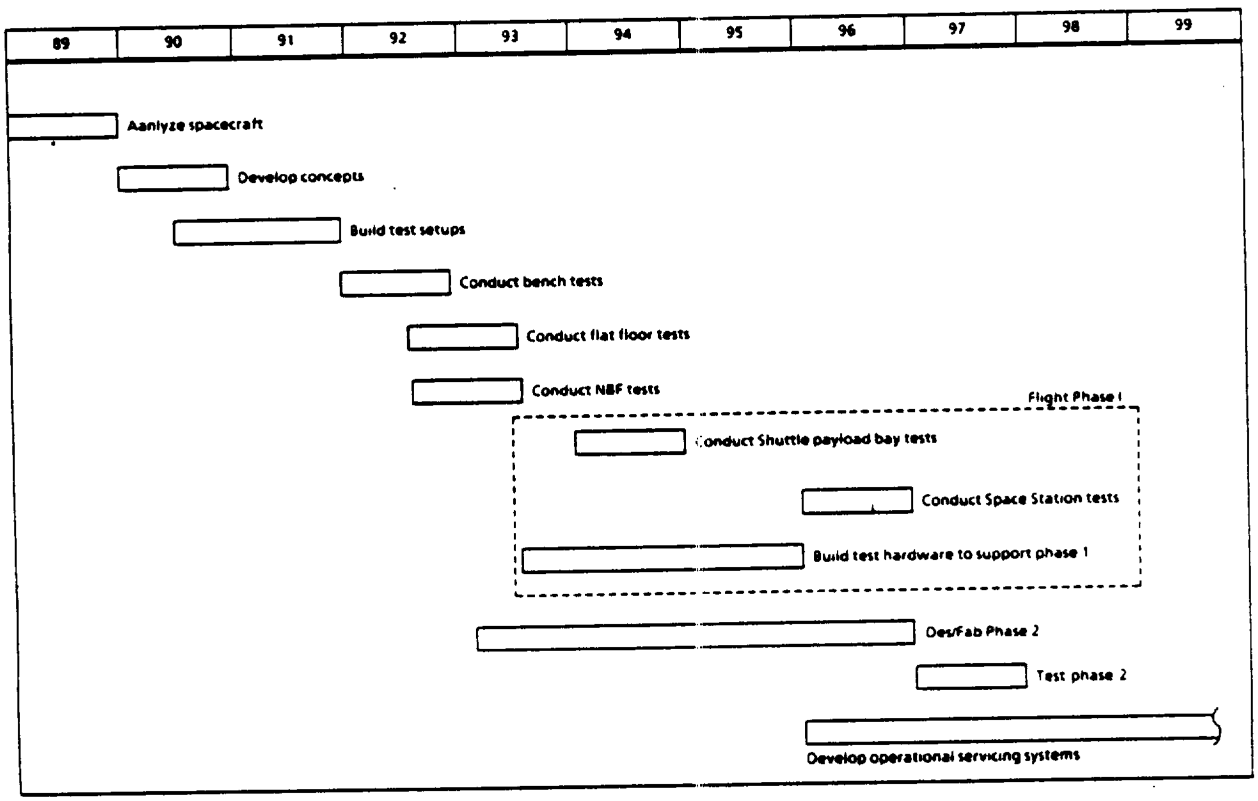

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Figure 5.5-10. Servicing Technology Strawman Schedule

5.5.7 Satellite Servicing Space Station Impacts

While no new design constraints would be imposed on the Space Station a range of standard support will be required.

The Mobile Remote Manipulator System (MRMS) will be required to support assembly on orbit, retrieval and rack changeout. EVA will be needed in the same operations. IVA will be needed to support data analysis both of the product and the test article self-test evaluations, handling of stowed equipment (prior to and after flight), as a communications link with the test article, product handling before and after flight, and in OMV teleoperations. With or without the satellite servicing facility emplaced on the Space Station, the test article needs to be berthed, fluids for the product (and possibly coolant) need to be transferred as does the gaseous nitrogen propellant. To the degree to which the test article and support systems are made robotic, these operations will be monitored and controlled. Communications has been mentioned previously, but one final function is the programming of the test article on-board computers with the mission programs needed for the mission operation which the test article is expected to conduct.

5.6 Commonality

The commonality potential for the satellite servicing test article was assessed during the conceptual design. Our approach was aimed at maximizing commonality, using new design only where existing or in-development hardware was not available. We reviewed available Space Station hardware as first choice then looked to other programs. As Figure 5.6-1 shows, the only new items are integration hardware such as brackets and secondary structure, and cold gas thrusters which are simple machined parts (nozzles).

- Propulsion System
  - Tank
  - GN_2 Propellant
  - Valves
  - Thrusters
  - Tubing
  - Supports (Valve and Plumbing)
  - Regulator (Pressure)
  - Controller
- Avionics
  - Position Sensors
  - Star Scanner
  - Avionics Computer
  - Communications Link
  - Communications Computer
  - Antennae
    - Ground Communications
    - Space Station Communications
    - OMV Communications
- Payload Support
  - Data Recorder
  - Power Conditioner
  - Mission Specific Support
  - Cooling System
  - Support Computer
- Power System
  - Solar Array Feeds
  - Solar Arrays
  - Solar Array Positioners (Sun Sensors & Control Loop)
  - Power Distribution System

Figure 5.6-1. Satellite Servicing Required Equipment List

We did not specifically review software commonality, but by using Space Station EDPs we should be able to use their software operating systems. Also, the applications generator software approach being explored by Space Station should be applicable to this test article. Considerable simplification of the software job would result.

5.7 Satellite Servicing Revisions to The MRDB

Recommended changes to the Mission Requirements Data Base are: Combine the objectives of TDMXs 2561, 2562, and 2563 with TDMX 2063, all of which have similar goals of developing Satellite Servicing techniques. This will allow the development of one coordinated Satellite Servicing TDMX effort.

Conduct precursor tests with smaller test articles to show the test and operations development directions. This will be of use in evaluating techniques before the completion of Space Station Phase I where they will need to be incorporated in the next generation design.

A full test article needs to be built to test the full requirements of a Servicing Bay before the structure is in place.

GRO and AXAF should not be considered for use as they are expected to function for a longer time then the test demonstrations can wait for them to become non-operational. Their reference should be removed from the TDMXs.

Include in the new MRDB TDMX on Satellite Servicing the capability to use and demonstrate Robotic operations on spacecraft and the need for spacecraft to demonstrate self-test and self-diagnose on-board problems and relay that to the support station.

Conduct the TDMX 2565 (Thermal Interface Technology) principally through in ground operations testing where selection of materials can be more rapidly done. Manipulation techniques can be practiced remotely in vacuum and in the Neutral Buoyancy tank. The TDM can then be supported by Shuttle operations at an earlier date than the Space Station operations will allow.

5.8 Cost

A preliminary cost estimate was made for the satellite servicing test article previoulsy illustrated in Figure 5.5-1 and 2. The Boeing PCM model was used, with an across-the-board estimate of 95% off-the-shelf hardware.

Item	Design & Dev	Manufacturing
Primary Structure	476	1383
Propellant System	186	946
Avionics	2850	5414
Electrical Power	1790	2612
Other Electronics	1461	707
Integration & Test	210	230
Subtotals	6973	11292
Spares		23
SE & I & Software	3768	
Systems Ground Test	5303	
Support & Test Equipment		446
Logistics	1544	
Liaison Engineering	240	
Subtotals	10855	469
Totals	17828	11761
TOTAL PROGRAM	29589	

Figure 5.8-1. Satellite Servicing Test Article Parametric Cost Estimate Dollars in Thousands

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6.0 COATINGS MAINTENANCE

Coatings maintenance was covered as a distinct subject in this study because the technology and techniques are diffferent from those associated with maintenance and repair of subsystems and components. Maintenance and repair usually involves removal and replacement of equipment, while coatings maintenance is done in place with specialized beam technologies.

6.1 Objectives and Benefits of Technology Advancement

Coatings are used on space systems for optical system, thermal control systems, and wherever else a coating can enhance the properties of a material or system. Space coatings are subject to degradation due tonatural and induced environments. Decontamination, replacement or restoration of coatings is a key part of extending the lifetime of space systems. The benefits are the same as those discussed in Seciton 5.1 In addition, there are benefits associated with related technology such as annealing of radiation damage in solar arrays.

6.2 Coatings Maintenance Study Objectives and Task summary

The recoating technique is a process that at present is a laboratory phenomenon. It is still open to investigation on how the technique occurs and on the range of substances that can be used in the recoating and the surface materials that these substances will work with. All the equipment for both the laser system and the plasma/ion system remain to be space qualified. For the beam systems themselves it is not known what surfaces the treatment, recoating and epitaxy will be most effective on. The surfaces must be down selected to those surfaces that can best be refurbished and under what conditions these operations may be done.

Contamination, both particulate and electromagnetic noise, will be produced by this experiment. The type of particulate will depend on the substances and materials used. We do not know how big the particles will be, how much will be produced and therefore what range of contamination will exist (how big an area will be affected) or how this will affect operations (will the contamination coat surfaces and must be operated away from optical instruments). Radio frequency noise will be generated by the plasma/ion system; we do not know what frequencies will be produced, how it should be shielded and how much of a potential problem it is.

Finally, if the system is successful and the problems can be solved can it be adapted to a robotic or automated system for large surface use?

This task was begun by reviewing the MRDB TDMX 2564 task description to derive the mission requirements. This became a much more focused effort than the satellite servicing missions, as there was demonstration hardware that had been built under the active cleaning technique program contract for Marshall Space Flight Center by Boeing Aerospace in Seattle. Research into this previous program yielded the information to evaluate the hardware capabilities of the plasma and ion systems. The laser system would be an adaptation of commercially available hardware that would satisfy the

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reannealing requirements of silicon and gallium arsenide solar cells (reference the article in Nature volume 303, 9 June 1883 " Laser Processing of Silicon", by Ian W. Boyd and John I.B. Wilson, Department of Physics, Heriot-Watt University, Riccarton, Currie, Edinburgh EH14 4AS, United Kingdom, and other articles). With this information a preliminary configuration could be defined and remaining development areas pinpointed, of which there are several. From this a development schedule was derived and an assessment of the impact to the Space Station made. The top level requirements were reevaluated in light of the current hardware development status and remaining work to be done, and the TDMX changes developed.

The overall study flow is shown in Figure 6.2-1.

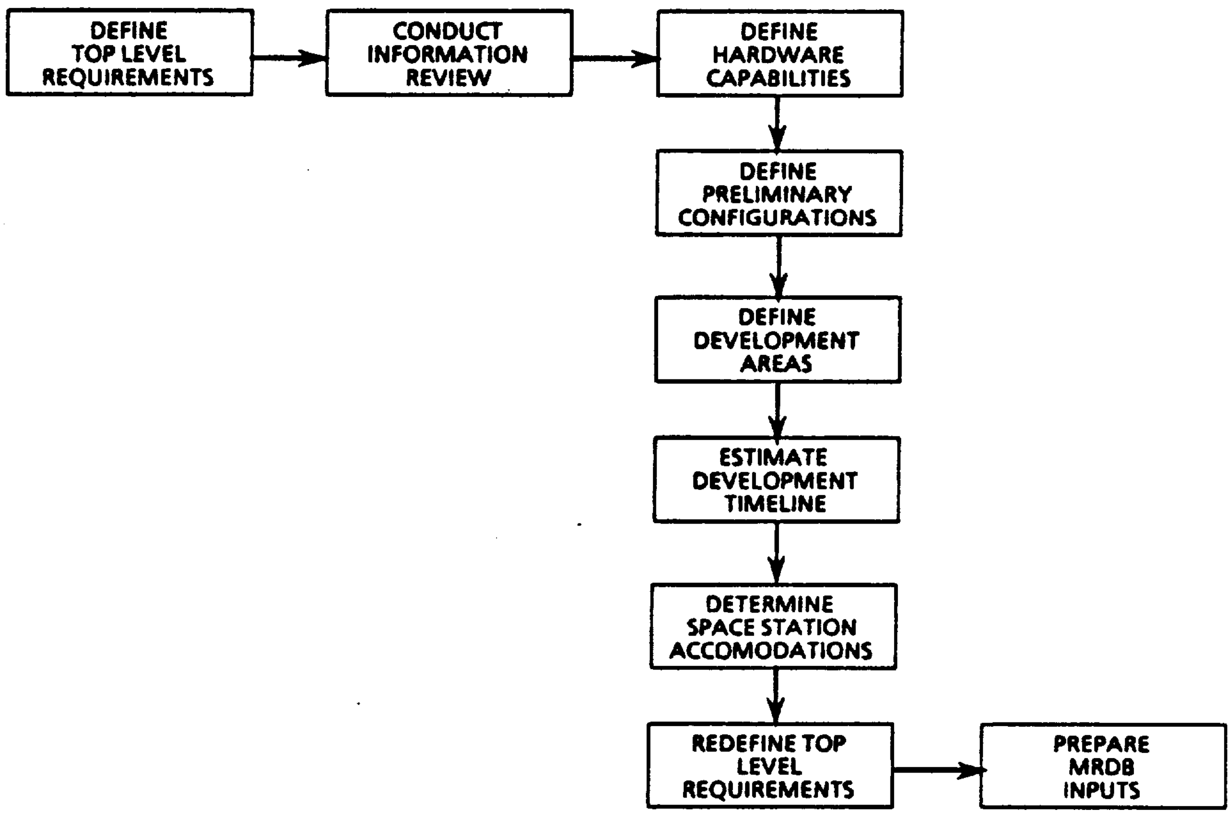

Figure 6.2-1 Coatings Maintenance Overall Task Flow

6.3 Coatings Maintenance Design

Figures 6.3-1, 2, and 3 present three views of the coatings maintenance technology demonstration test platform, showing the contents of the platform (which rests on a Space Station provided standard support platform), the elevation above the support structure and the relative size of the platform to the station and the EVA astronaut. The size of the equipment is taken from current ground test equipment without the vacuum support equipment not needed in space (10^{-4} to 10^{-5} torr is the nominal operating pressure for the plasma and ion beams, the Laser does not need this support). The plasma and ion beams heads share the same support system they will not operate at the same time. The elevation diagram shows the spacing that exists on the

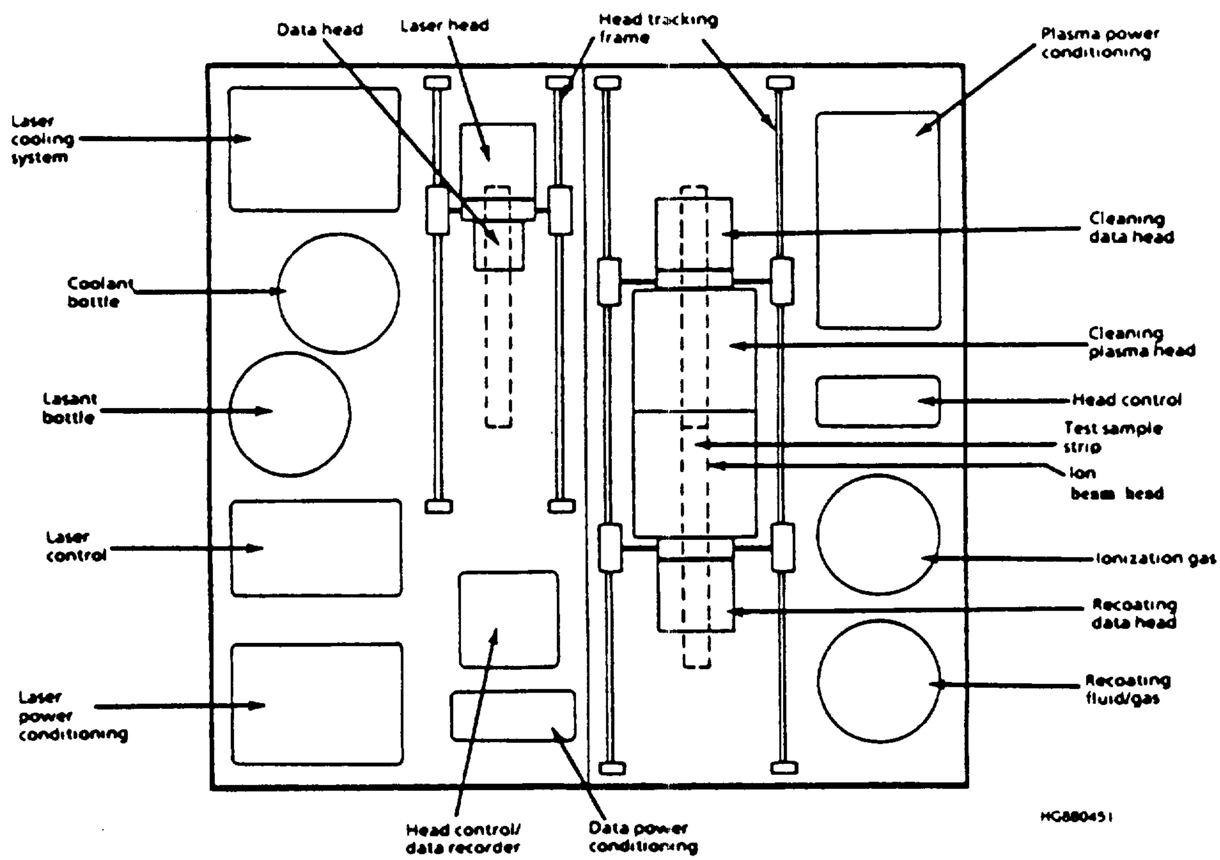

Figure 6.3-1. Coating Maintenance Technology TDMX Schematic Plan

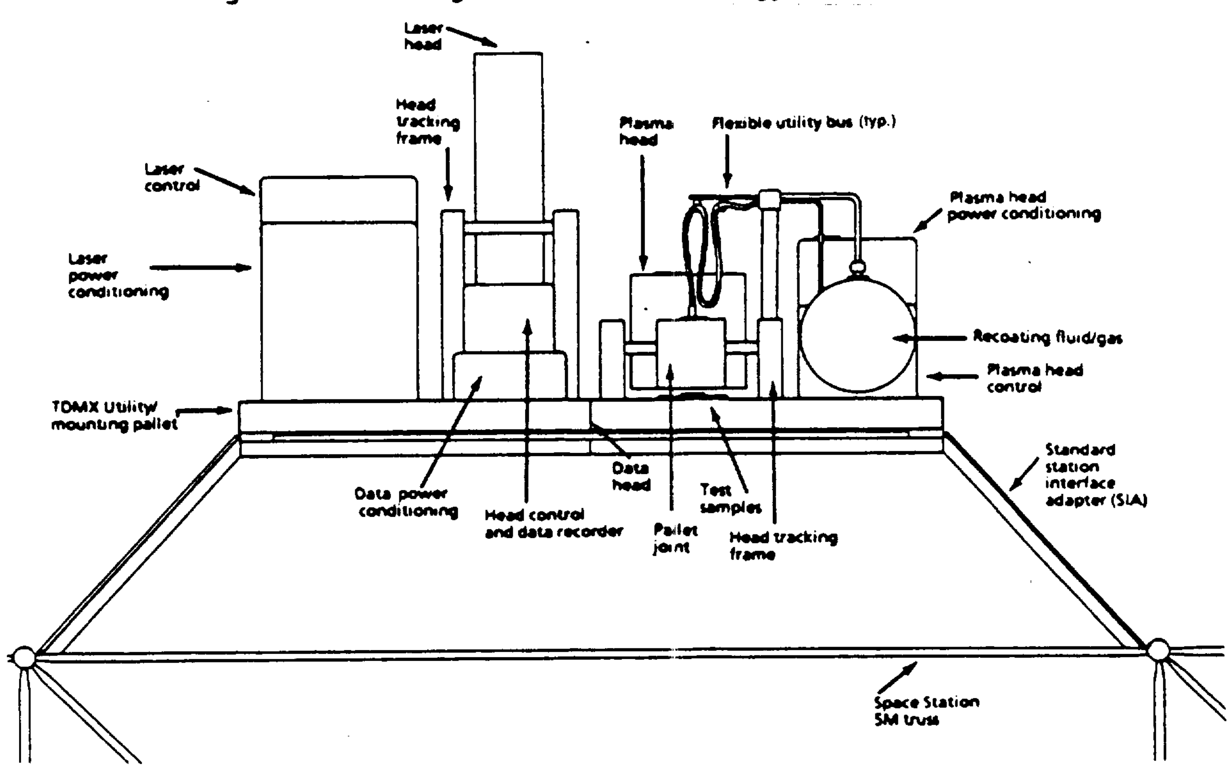

Figure 6.3-2. Coatings Maintenance Technology TDMX Schematic Elevation

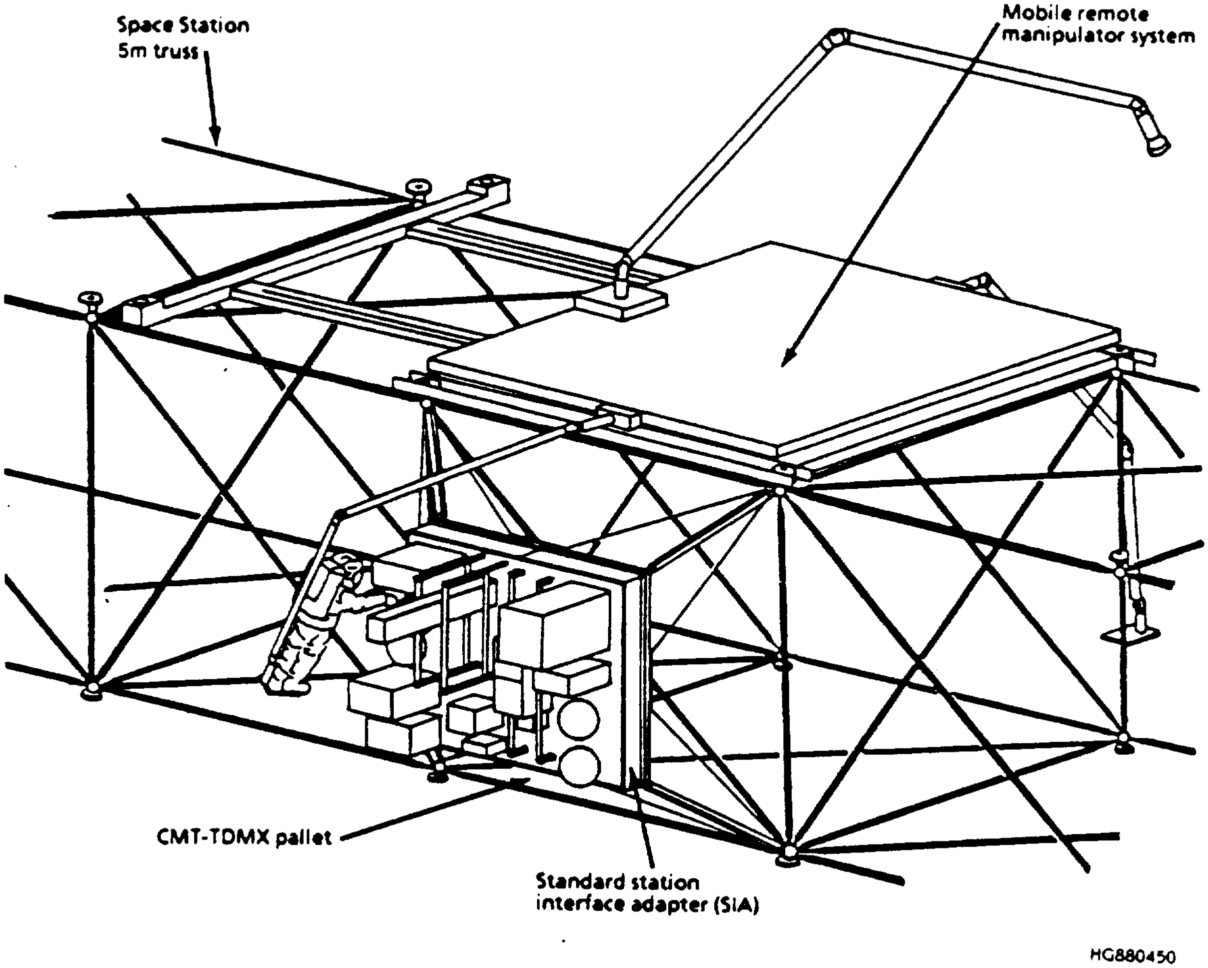

Figure 6.3-3. Coatings Maintenance Technology TDMX Space Station IN-Situ View

platform for access to the individual units. The final view shows the relative size position and EVA/MRMS access to the platform itself.

The list of equipment in Figure 6.5-4 has been taken from the commercially available system components for the laser and the active cleaning technique functional test and development systems without the vacuum support equipment.

- Laser Beam
  - Laser, 1.06×10^{-6} meter/0.53×10^{-6} meter
  - Beam head
  - Laser Support System
  - Power Conditioner
  - Gas Supply
  - Control Electronics
  - Coolant System
  - Data Head
- Plasma Beam/Ion Beam:
  - Plasma Beam Head
  - Ion Beam Head
  - Data Heads (2)
  - Power Conditioner
  - Gas Supply
  - Recoating Material Supply
- Data Recorder
- Safety Interlocks
- Grounding System
- Equipment Platform
- Cable Supports for all beam and data recording means
- Space Ststion planned payload platform

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Figure 6.3-4. Coating Maintenance Required Equipment List

The laser beam equipment- The laser cnamber itself will form the basic component of the mobile beam head. It will be operable at 1.06×10^{-6} meter or 0.53×10^{-6} meter wavelengths both of which are used for solid state epitaxy with silicon and gallium arsenide. The candidate lasers for this system are argon, argon-ion, xenon, krypton, and Nd:YAG (solid state glass) laser systems. The required energy density will be 1.5 Joules/cm^2 maximum and pulsed at 20 to 30 x 10^{-9} second or at 0.1 x 10^3 Watts over a 2 millimeter beam radius in continuous wave operations moving at 2 centimeters per second (low power lasers). The laser support system will control the function of the laser including scan or pulse rate, coolant supply rate, laser power application, gas demand feed (all but the Nd:YAG glass laser) and laser beam operation and intensity, while the control electronics will control the beam head motion and rate, the data head scan rate, the data head operations, the data head motion and rate. The laser support system and control electronics may be the same system, possibly using an Embedded Data Processing unit design from the Space Station.

The plasma beam/ion beam equipment- the plasma and ion beam heads will share the same support equipment and not operate at the same time. The difference between the two systems is the addition of an ion generator to the plasma chamber on the active cleaning technique (ACT) system. Part of the conclusions of the ACT program are that the cleaning and refurbishment qualities was not completely dependent on the plasma gas. The same cleaning could be done with gases other than oxygen,like helium, hydrogen, argon, or nitrogen. The use of any of these gases would reduce handling risks associated with oxygen. The support systems would operate similar to the laser support/control electronics system, with the consideration for tandem head operations and the recoating gas/liquid that must be feed to the plasma head for recoating operations. This material can be silicones, hydrocarbon, florocarbon or one of several inorganic materials which will coat a surface if they are introduced into the plasma stream.

These systems will sit on a self contained platform with electrical connections running between it and the Space Station support platform that provides electrical connections between the platform and the Space Station. The support equipment will plug into the adapter platform.

Since this experiment can produce high voltages, laser coherent light and large radio frequency noise; grounding, shielding and safety interlocks must be carefully examined before this system is made operational.

While no new design constraints on the Space Station were identified, expected standard support will be required.

Assembly, test and initial operations will require EVA support and MRMS support. Data from the mobile data heads and the health of the operations will need to be monitored from inside the Station (IVA) through the standard Station platform connection, from which Station power will also be provided to the experiment. Sample retrieval and isolation will have to be done by EVA, as no on-orbit characterization by IVA should be done. The possible contamination fields (particulate and EMI) that the experiment will generate must be monitored either by IVA or EVA.

Impact areas are listed in Figure 6.5-5.

- **No new Space Station design requirements were identified**
- **Mission support required is within the Space Station capability**
  - **Data Transfer**
    - **-IVA**
  - **Sample Retrieval**
    - **-Robotics (FTS)**
    - **-MRMS**
    - **-EVA**
  - **Construction/Emplacement**
    - **-Robotics (FTS)**
    - **-MRMS**
    - **-EVA**
  - **Power from Space Station**
  - **EVA**
- **Contamination Field Present The Extent of Which is Not Known**

CMP GRFX/GW/DISC41/DSS/169-8

Figure 6.3-5. Coating Maintenance Space Station Impacts

The following testing must be accomplished to develop the platform for service.

- Component tests- since most of the assemblies and sub assemblies are items that must be space qualified, many components must be developed for this platform. This includes the technologies that are still to be developed such as the recoating system technique.

- Subassembly tests- conducted principally to integrate the component development into functional subassemlies.

- Subsystem tests- functional testing for subassemblies to work as subsystems independently and under the conditions that will be encountered in operations.

- Full system tests- functional tests of the full platform system for operations under conditions similar to those that it will encounter on-orbit or in transit flight.

- Ground checkout tests- final on-ground checkout of mission critical operations and hardware.

- Flight tests- functional on-orbit checkout of the critical systems prior to the platform operations.

Some of these tests that have been identified are shown in figure 6.3-6.

Development Tests	Component Tests	Sub Assembly Tests	Sub System Tests	Full System Tests	Ground Checkout	Flight Test
Recoating Technique	Recoating parts/test material selection	Recoating material introduction	Recoating system function	Full system actuation		
Mobile heads: plasma/ion laser data	Mobile heads: plasma/ion laser data	Mobile heads: separate subsystems	Mobile heads functional	Full mobile head actuation	Mobile head actuation	Mobile head actuation
Software Development	Software Subprogram checks	Software/hardware interface	Software full program checks	Command/ control tests	Control checks	Control checks
EMI/RF Noise	EMI/RF Noise component check	EMI/RF noise	EMI/RF noise	EMI/RF Noise		
Coronal field	Coronal field component check	Coronal fields	Coronal fields	Vacuum/Solar Vacuum		
		Cross-talk	Cross-talk			
Power conditioner	Power conditioner parts	Power conditioner	Power conditioner	Power distribution	Power up/down	Power up/down
		Communications/ data transmittal	Communications/ data transmittal	Communications/ data transmittal	Comm.	Comm.
			Acoustic Vibration	Acoustic Vibration	Remote control of operating	
				Assembly (Neutral Buoyancy)		

CMP GRFX/GW/DISC 41/074/169-8

Figure 6.3-6. Coating Maintenance Testing Levels

This experiment will principally support large space structures that are difficult to maintain without constant refurbishment. They are too large to return to the Earth for cleaning and recoating and, in space, they would consume valuable EVA time in inspection, hand cleaning and the development of hand recoating techniques or removing and replacing surface/ structure segments on a routine basis. If the system could be automated the operation could be carried out as a continuous operation. Such systems would include:

The following are the primary objectives of coatings maintenance technology demonsration:

- Evaluation of the surface deterioration- an evaluation of the damage done to the different surfaces at different periods of time will yield some measure of how often and how much the resurfacing and recoating can be or should be done.
- Cleaning different surfaces to evaluate the range to which this process or process series is applicable.
- Resurfacing of the different materials to evaluate the range of surfaces to which this technique is applicable.
- Recoating the different surfaces to evaluate the ability of the various surfaces to take the recoating process without flaws in the surface protection.
- The use of active beam technology to accomplish all of the above tasks.

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OF POOR QUALITY

Figure 6.5-7 is a brief top level development schedule with an estimation of progression of the testing levels. It can be seen that there is a considerable amount of time that must be devoted to the development of both the techniques of coatings maintenance and the hardware.

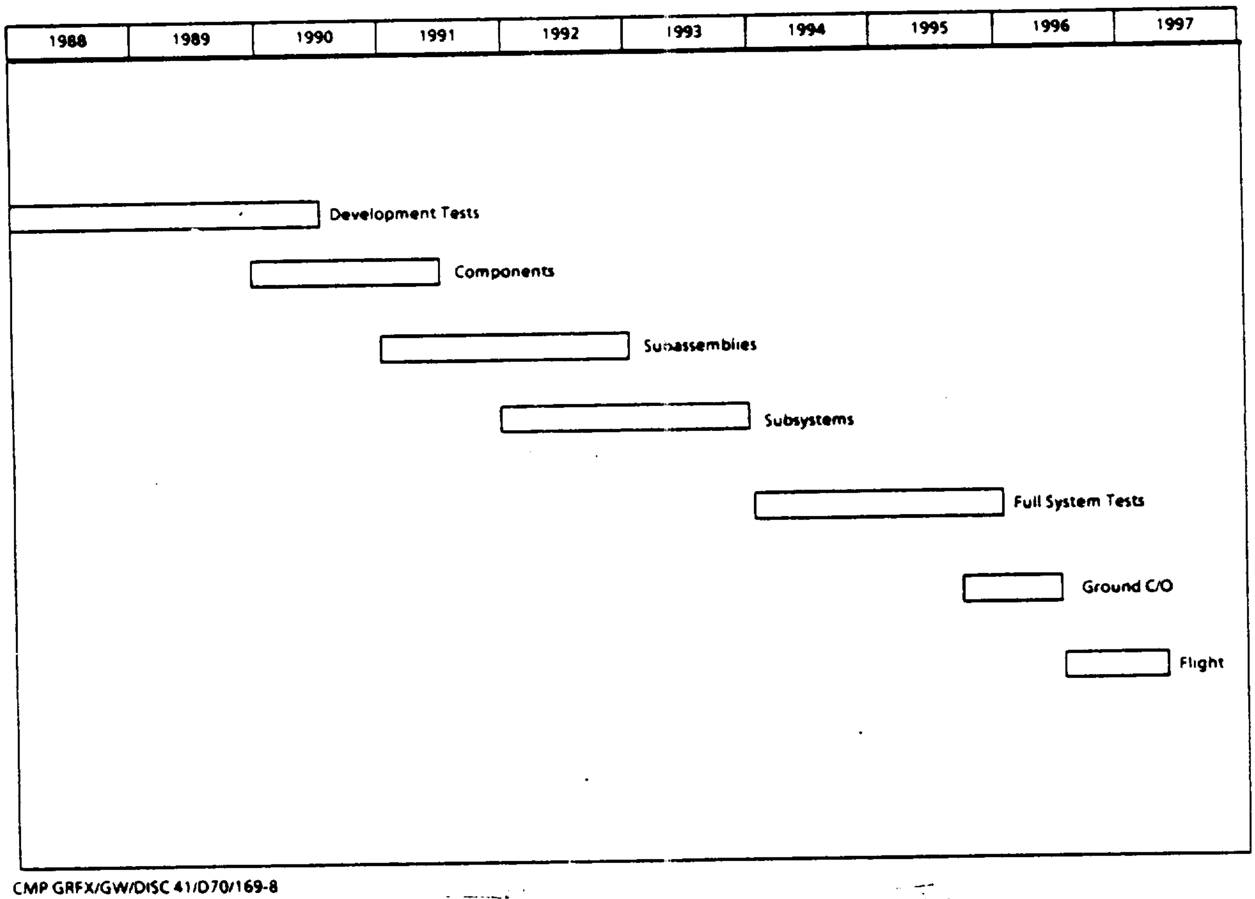

Figure 6.3-7. Coating Maintenance Development Schedule

- Development tests - 2.5 years
- (both of the techniques and hardware)
- Component testing - 1.5 years
- Subassembly testing - 1.5 years
- Subassemblies testing - 2 years
- Full system tests - 2.25 years
- Ground checkout - 9 months
- Flight tests to begin in 1996, estimated development time with testing overlap - 8.5 years

6.4 Commonality

The coatings maintenance TDMX can use planned space station external payload accommodations and other experiement support resources. The experiment itself involves new technology: standard components can be used in support and auxilary equipment.

6.5 Coatings Maintenance Revisions to the MRDB

The basic recommended revisions to the Mission Requirements Data Base TDMX 2564 are:

Since the Active Cleaning Technique program discovered that the species involved in the plasma field had little or no effect on the technique it is recommended that an inert gas or a gas such as Nitrogen be used as the Plasma/Ion gas. Nitrogen would be easier to handle and if used in the ion beam test provide information on strengthening materials on orbit by ion implantation.

Having a test operation of this nature, with its high potential for particulate contamination in the confined area of the Space Station, plus the size of the test equipment configuration, would prove an operational hazard to the interior equipment if not the crew. For the same reason (particulate contamination), on orbit characterization of the type of damage sustained to the coatings, the resultant surface cleaning/reannealing, and recoating should be limited to remote observation and data. The samples should not be brought in to the Space Station but stored outside for return to Earth.

Since many of the techniques of the operation are still unrefined and much of the hardware has yet to be space qualified, time must be allowed for their development. The ACT program noted that the ACT equipment produced considerable RF noise, the extent of which and its effects are yet to be fully determined. This RF noise must be shielded against for the protection of the astronauts and this too must be developed. While the schedule must account for these developments, the program must start soon if the techniques are to be fully developed in time to be of use to a Growth model Space Station.

The current MRDB TDMX 2564 gives a wide ranging list of materials and surfaces to be evaluated for the coatings maintenance techniques. This array of materials and surfaces should be down selected by ground test to select several starting test samples for evaluation, with samples representing the types of operations to be accomplished by these processes.

7.0 THERMAL INTERFACE TECHNOLOGY TDMX 2565

The materials that are currently used as cold-plate/component interface are thermal grease - a metal-filled grease, or silica gel (which has been know to get on and coat an astronauts gloves) and thin foil sheets, which can be the size of a desk top.

From John Pizzichemi (773-2767): Boeing-Seattle is now working on cold-plate technology. Four methods are currently under consideration for interface contract mechanisms:

1) Metal-filled gel

2) Copper or aluminum thin foil insert that deforms to provide component/cold-plate contact

3) "Comerics" - sheets of thermally conductive, electrically insulation material; "looks like an insulation gasket" and cuts the same way

4) high-polished surfaces - metal to metal contact or anodized surfaces (tend to become part and participle to each other - hard to remove after exposure to vacuum).

Use of the above, except for "Comerics", is labor intensive to apply, hard to remove (all) and may require the part being "scraped" off the surface (all), and the cold-plate cleaned and refurbished or the cold-plate removed and replaced.

What type of material to use is to be determined - should be determined in the ground testing, in vacuum chambers, with remore applications or manipulations from outside the chamber, in the neutral buoyancy tank, or man-rated vacuum chamber.

This might be a good candidate for robotic operation, particularly the remove and replace operation and the initial installation.

The final test could be and EVA/robotic action in the shuttle bay.

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8.0 SATELLITE SERVICING TECHNOLOGY CONCLUSIONS AND RECOMMENDATIONS

Our conclusions and recommendations place future servicing technology development in context with current and planned activities and programs, and were derived based on the servicing technology missions described in the data base. We recommend that the five servicing TDMXs we studied be reduced to 3, with TDMXs 2561, 2562, and 2563 combined into one mission.

We did not find any design impact on Space Station. We believe Space Station Phase I as presently planned can accommodate the flight phases of these missions. We were able to maintain near 100% commonality and use of off-the-shelf components except where new design is dictated because the technology is new. For example, the test articles and test spacecraft can use Station components and other off-the-shelf equipment. An experimental robotics servicer, however, would presumably be new design.

The proposed programs are evolutionary, with as much accomplished by ground test as possible.

There are many activities presently going on and planned that relate to servicing technology; most if not all of the NASA centers and JPL are involved. We recommend an agency-wide activity. This could begin with establishment of a servicing technology working group for exchange of data and plans, and development of a coordinated program plan.

Satellite servicing, if it became routine, and if it were extended to mission orbits not presently human-accessible through teleoperation and robotics, would provide significant benefits in mission capability extension and cost avoidance.

MISSION DESIGN

MISSION CODE: TDMX2151

PAYLOAD ELEMENT NAME: SOLAR ARRAY/ENERGY STORAGE TECH.

COUNTRY: USA NASA OAST (TDMX)___________

CONTACT: GEORGE MCKAY_________________
FM01_________________________
MARSHALL SPACE FLIGHT CENTER____
MSFC, AL 35812________________
PHONE: 205-544-1773__________________

STATUS: CANDIDATE

FLIGHTS:	92	93	94	95	96	97	98	99	00	01
EQUIPMENT UP (flights):	0	0	0	1	0	0	0	0	0	0
EQUIPMENT DOWN (no. of times)	0	0	0	0	0	0	0	0	0	0
OPERATIONAL DAYS (per flight)	0	0	0	10	365	0	0	0	0	0
OTV FLIGHTS	0	0	0	0	0	0	0	0	0	0

EARLY FLIGHT: __
LATE RETURN: __

OBJECTIVE:

TO DEMONSTRATE ADVANCED PHOTOVOLTAIC SOLAR ARRAY TECHNOLOGY WITH AREA EFFICIENCY AND SYSTEM WEIGHT/PACKAGING VOLUME CHARACTERISTICS APPROACH-ING OR EXCEEDING SOLAR DYNAMICS POTENTIAL AND TO DEMONSTRATE REDUCED LIFE CYCLE COSTS. TO DEMONSTRATE ENERGY STORAGE TECHNOLOGY IMPROVEMENTS FOR SPACE APPLICATIONS.

DESCRIPTION:

TWO ADVANCED PHOTOVOLTAIC POWER SYSTEM ARRAYS ARE ATTACHED TO THE END OF THE PHASE I STATION MAIN BOOM USING THE SOLAR DYNAMIC TRUSS EXTENSION IDENTIFIED FOR PHASE II. ONE EXPERIMENT DEMONSTRATES ADVANCED SOLAR CELL TECHNOLOGY USING A 5 KW PLANAR ARRAY. THE BETA GIMBAL, ARRAY DEPLOYMENT MECHANISM AND SUPPORT STRUCTURE ARE IDENTICAL TO THE PHASE I PV ARRAYS. THE SECOND EXPERIMENT DEMONSTRATES SOLAR CONCENTRATOR TECHNOLOGY AND IS ATTACHED TO A SECOND BETA GIMBAL LOCATED ON THE OPPOSITE SIDE OF THE TRUSS. THE INTEGRATED EQUIPMENT ASSEMBLY (IEA) CONCEPT ENVISIONED FOR THE PHASE I SPACE STATION WILL BE USED TO CONTAIN PERFORMANCE MEASURE-MENT, POWER CONDITIONING AND DISTRIBUTION ELECTRONICS, AN EQUIPMENT RADIATOR AND TWO ADVANCED ENERGY STORAGE TECHNOLOGY EXPERIMENTS. A DYNAMIC ELECTRICAL LOAD WILL ALLOW THE PERFORMANCE OF THE EXPERIMENTAL SYSTEMS TO BE DETERMINED. THE POWER GENERATED WILL ALSO BE AVAILABLE TO THE SPACE STATION PMAD SYSTEM.

TYPE NUMBER: 11

IMPORTANCE OF SPACE STATION: 10

NON-SERVICING OMV FLIGHTS (per year): __

ADD RESOURCES: 2

RESOURCE REFERENCE: __________

ORBIT

MISSION CODE: TDMX2151

ORBIT: 1 (If 1 is selected, skip remainder of Form 2)

APOGEE: __________ km + __________ km TOLERANCE
- __________ km

PERIGEE: __________ km + __________ km TOLERANCE
- __________ km

INCLINATION: __________ km , + __________ km TOLERANCE
- __________ km

LOCAL TIME OF EQUATOR CROSSING NODE: __________ hr __________ min

ASCENDING OR DESCENDING: __________

SPECIAL CONSIDERATIONS (ORBIT):

____
____
____
____

POINTING/ORIENTATION

MISSION CODE: TDMX2151

POINTING/ORIENTATION: 2 (If 1 is selected, skip remainder of Form 3)

VIEW DIRECTION: 2

If 4 selected, OTHER ______________

HOURS: 24

TRUTH SITES:

__SUN____

POINTING ACCURACY: ___18000__ arc sec

POINTING KNOWLEDGE: ____900__ arc sec

FIELD OF VIEW: ____360__ deg

POINTING STABILITY RATE: ____-_____ arc sec per sec

POINTING STABILITY: __________ arc sec

PLACEMENT: _________ arc sec

SPECIAL CONSIDERATIONS:

POINTING REQUIREMENTS SHOWN APPLY TO THE SOLAR CONCENTRATOR TECHNOLOGY EXPERIMENT. PLANAR ARRAY POINTING REQUIREMENTS ARE THE SAME AS THE S.S. PV POWER SYSTEM.

POWER

MISSION CODE: TDMX2151

POWER: 2

	AC	DC
OPERATING (KW):	______	_.03__
HOURS, PER DAY (OPERATING)	______	__1___
VOLTAGE:	______	______
FREQUENCY:	______	______
PEAK (KW):	______	_.05__
HOURS PER DAY (PEAK)	______	__.1__
STANDBY POWER (KW)	______	__0___

SPECIAL CONSIDERATIONS (POWER):

SPACE STATION POWER IS REQUIRED ONLY FOR INITIAL ORIENTATION OF ARRAYS. POWER GENERATED BY THIS TDM IS CAN BE SUPPLIED TO S.S. PMAD.

THERMAL

MISSION CODE: TDMX2151

THERMAL: 2

	ACTIVE		PASSIVE	
	OPER.	NON-OPER.	OPER.	NON-OPER.
MIN TEMP (C)			N/A	N/A
MAX TEMP (C)			N/A	N/A
MIN HEAT REJECTION (KW)			0	0
MAX HEAT REJECTION (KW)			0	0

SPECIAL CONSIDERATIONS:

THE EXPERIMENT USES ITS OWN THERMAL RADIATOR, HENCE NO SPACE STATION HEAT REJECTION IS REQUIRED.

DATA/COMMUNICATIONS

MISSION CODE: TDMX2151

ON-BOARD DATA PROCESSING REQUIRED: 1

If 1 (YES), this DESCRIPTION:

EXPERIMENT PERFORMANCE DATA, STATUS OF POWER DISTRIBUTION, HEALTH STATUS

ON-BOARD STORAGE (MBIT): __1___

STATION DATA REQUIRED:

TIME, POSITION, ALPHA AND BETA GIMBAL ANGLES, MAJOR EVENTS LOG__________

COMMUNICATION LINKS:

1. From: Station
   To: Ground

	Digital Data		Video Data		Voice
a. Generation rate (kbps)	__10___	:	_______	:	__N/A__
b. Duration (hours)	__1.5__	:	_______	:	_______
c. Frequency (per day)	__1____	:	_______	:	_______
d. Delivery time (hours)	__1.5__	:	_______	:	___0___
e. Security (yes/no)	__NO___	:	_______	:	_______
f. Reliability (%)	__95___	:	_______	:	_______
g. Interactive (yes/no)	_YES___	:	_______	:	__Yes__

2. From: Ground
   To: Station

	Digital Data		Video Data		Voice
a. Generation rate (kbps)	__10___	:	_______	:	__N/A__
b. Duration (hours)	__.1___	:	_______	:	_______
c. Frequency (per day)	___1___	:	_______	:	_______
d. Delivery time (hours)	__.1___	:	_______	:	___0___
e. Security (yes/no)	__NO___	:	_______	:	_______
f. Reliability (%)	__95___	:	_______	:	_______
g. Interactive (yes/no)	_YES___	:	_______	:	__Yes__

3. From: Station To: Free Flyer	Digital Data		Video Data		Voice
a. Generation rate (kbps)	_______	:	_______	:	__N/A__
b. Duration (hours)	_______	:	_______	:	_______
c. Frequency (per day)	_______	:	_______	:	_______
d. Delivery time (hours)	_______	:	_______	:	___0___
e. Security (yes/no)	_______	:	_______	:	_______
f. Reliability (%)	_______	:	_______	:	_______
g. Interactive (yes/no)	_______	:	_______	:	__Yes__

4. From: Free Flyer To: Station	Digital Data		Video Data		Voice
a. Generation rate (kbps)	_______	:	_______	:	__N/A__
b. Duration (hours)	_______	:	_______	:	_______
c. Frequency (per day)	_______	:	_______	:	_______
d. Delivery time (hours)	_______	:	_______	:	___0___
e. Security (yes/no)	_______	:	_______	:	_______
f. Reliability (%)	_______	:	_______	:	_______
g. Interactive (yes/no)	_______	:	_______	:	__Yes__

5. From: Station To: Platform	Digital Data		Video Data		Voice
a. Generation rate (kbps)	_______	:	_______	:	__N/A__
b. Duration (hours)	_______	:	_______	:	_______
c. Frequency (per day)	_______	:	_______	:	_______
d. Delivery time (hours)	_______	:	_______	:	___0___
e. Security (yes/no)	_______	:	_______	:	_______
f. Reliability (%)	_______	:	_______	:	_______
g. Interactive (yes/no)	_______	:	_______	:	__Yes__

6. From: Platform To: Station	Digital Data		Video Data		Voice
a. Generation rate (kbps)	_______	:	_______	:	__N/A__
b. Duration (hours)	_______	:	_______	:	_______
c. Frequency (per day)	_______	:	_______	:	_______
d. Delivery time (hours)	_______	:	_______	:	___0___
e. Security (yes/no)	_______	:	_______	:	_______
f. Reliability (%)	_______	:	_______	:	_______
g. Interactive (yes/no)	_______	:	_______	:	__Yes__

7. From: Platform To: Ground	Digital Data		Video Data		Voice
a. Generation rate (kbps)	_______	:	_______	:	__N/A__
b. Duration (hours)	_______	:	_______	:	_______
c. Frequency (per day)	_______	:	_______	:	_______
d. Delivery time (hours)	_______	:	_______	:	___0___
e. Security (yes/no)	_______	:	_______	:	_______
f. Reliability (%)	_______	:	_______	:	_______
g. Interactive (yes/no)	_______	:	_______	:	__Yes__

8. From: Ground To: Platform	Digital Data		Video Data		Voice
a. Generation rate (kbps)	_______	:	_______	:	__N/A__
b. Duration (hours)	_______	:	_______	:	_______
c. Frequency (per day)	_______	:	_______	:	_______
d. Delivery time (hours)	_______	:	_______	:	___0___
e. Security (yes/no)	_______	:	_______	:	_______
f. Reliability (%)	_______	:	_______	:	_______
g. Interactive (yes/no)	_______	:	_______	:	__Yes__

9. From: Station To: Shuttle	Digital Data		Video Data		Voice
a. Generation rate (kbps)	_______	:	_______	:	__N/A__
b. Duration (hours)	_______	:	_______	:	_______
c. Frequency (per day)	_______	:	_______	:	_______
d. Delivery time (hours)	_______	:	_______	:	___0___
e. Security (yes/no)	_______	:	_______	:	_______
f. Reliability (%)	_______	:	_______	:	_______
g. Interactive (yes/no)	_______	:	_______	:	__Yes__

10. From: Shuttle To: Station	Digital Data		Video Data		Voice
a. Generation rate (kbps)	_______	:	_______	:	__N/A__
b. Duration (hours)	_______	:	_______	:	_______
c. Frequency (per day)	_______	:	_______	:	_______
d. Delivery time (hours)	_______	:	_______	:	___0___
e. Security (yes/no)	_______	:	_______	:	_______
f. Reliability (%)	_______	:	_______	:	_______
g. Interactive (yes/no)	_______	:	_______	:	__Yes__

COMMENTS:

____
____
____
____

EQUIPMENT

MISSION CODE: TDMX2151

MODULE CODE: 1

SHARED FACILITY CODE: 0

MISSIONS: ________

____

____

____

EQUIPMENT LOCATION: 3 (END OF MAIN POWER BOOM__________)

DIMENSIONS (M)

Length ____3_____

Width or Dia. ___11_____

Height (or blank) ___24_____

VOLUME (M^3) __460_____

PKG DIMENSION (M)

Length ____8_____

Width or Dia. ___4.5____

Height (or blank) __________

PKG VOLUME (M^3) ___100____

LAUNCH MASS (KG) __5971____

ACCELERATION MAX. (g) __________

ATTACH POINTS: 1

SET UP CODE: 1 2 3

HARDWARE DESCRIPTION:

S.S. TRUSS EXTENSION, 2 BETA GIMBALS, 9 X 11 M PLANAR SOLAR ARRAY, 5 X 10 M CONCENTRATOR ARRAY, 3 X 3 X 3 M INTEGRATED EQUIPMENT ASSEMBLY (IEA), 3 X 8 M RADIATOR

CREW

MISSION CODE: TDMX2151

INITIAL CONSTRUCTION/SET UP: 1

TASK:

CONSTRUCT TRUSS, ASSEMBLE EQUIP., DEPLOY SOLAR ARRAYS, ASSEMBLE RADIATOR

PERIOD: __3___ days

IVA TOTAL CREW TIME: __21.6___ man-hrs

EVA PRODUCTIVE CREW TIME: ___27.1___ man-hrs

SKILLS:

SKILL TYPE

SKILL LEVEL	1	2	3	4	5	6	7
1							
2					2		
3							2

DAILY OPERATIONS: 0

TASK:

IVA CREW TIME PER DAY: __________ man-hrs

SKILLS:

SKILL TYPE

SKILL LEVEL	1	2	3	4	5	6	7
1							
2							
3							

MISSION CODE: TDMX2151

PERIODIC OPERATIONS: O

TASK:

IVA OCCURRENCE INTERVAL: ______ days

IVA CREW TIME/OCCURRENCE: _________ man-hrs

EVA OCCURRENCE INTERVAL: ________ days

EVA PRODUCTIVE CREW TIME/OCCURRENCE: __________ man-hrs

SKILLS:

SKILL LEVEL \ SKILL TYPE	1	2	3	4	5	6	7
1							
2							
3							

TEARDOWN AND STOW: __

TASK:

PERIOD: ______ days

IVA TOTAL CREW TIME: _________ man-hrs

EVA PRODUCTIVE CREW TIME: __________ man-hrs

SKILLS:

SKILL LEVEL \ SKILL TYPE	1	2	3	4	5	6	7
1							
2							
3							

MISSION CODE: TDMX2151

COMMENTS:

Primary experiment operation and monitoring will be from the ground. On-board monitoring may be available. Maintenance and contingency operation, if required, will be directed from the ground.

Typical example of skill type/level matrix input:

<u>Skill Types</u>

1. No Special Skill Required
2. Medical/Biological
3. Physical Sciences
4. Earth and Ocean Sciences
5. Engineering
6. Astronomy
7. Spacecraft Systems

<u>Skill Levels</u>

1. Task Trainable
2. Technician
3. Professional

If two medical/biological professionals are required, put 2 in second column, third row. No more than 6 skill types can be used for a given task.

SERVICING

MISSION CODE: TDMX2151

SERVICING: 1 (If 1 is selected, skip remainder of Form 9)

SERVICE INTERVAL (days): ________

CONSUMABLES:

TYPE:

WEIGHT: __________ kg

RETURN: __________ kg

VOLUME UP: __________ m^3

VOLUME DOWN: __________ m^3

POWER: __________ kw

HOURS FOR POWER: __________ hrs

EVA HOURS PER SERVICE: __________ hrs

TYPICAL TASKS (EVA):

IVA HOURS PER SERVICE: __________ hrs

LOCATION OF SERVICING: __________

TYPICAL TASKS (IVA):

SPECIAL CONSIDERATIONS:

CONFIGURATION CHANGES

MISSION CODE: TDMX2151

CONFIGURATION CHANGES: 1 (If 1 is selected, skip remainder of Form 10)

INTERVAL (days): ________

CHANGE-OUT EQUIPMENT:

TYPE:

WEIGHT: __________ kg

RETURN: __________ kg

VOLUME UP: __________ m^3

VOLUME DOWN: __________ m^3

POWER: __________ kw

HOURS FOR POWER: __________ hrs

EVA HOURS PER CHANGE: __________ hrs

TYPICAL TASKS (EVA):

IVA HOURS PER CHANGE: __________ hrs

LOCATION: __________

TYPICAL TASKS (IVA):

SPECIAL CONSIDERATIONS:

MISSION CODE: TDMX2151

SCIENTIFIC AIRLOCK:

TETHER:

VACUUM VENTING:

OTHER:

MISSION DESIGN

MISSION CODE: TDMX 2561, 2562, 2563 and 2063 Combined

PAYLOAD ELEMENT NAME: Satellite Servicing

COUNTRY: USA NASA OAST (TDMX)

CONTACT: George McKay
PM01
Marshall Space Flight Center
MSFC, AL 35812
PHONE: 205-544-1773

STATUS: Candidate

FLIGHTS:	92	93	94	95	96	97	98	99	00	01
Equipment Up (flights)	0	0	0	0	1	0	0	0	0	0
Equip Down (# of times)	0	0	0	0	0	0	0	0	0	0
Operational Days/Flight	0	0	0	0	10	60	60	60	60	60
OTV Flights	0	0	0	0	1	3	2	2	2	2

EARLY FLIGHTS ____
LATE RETURN ____

OBJECTIVE:

To demonstrate satellite construction, maintenance, repair, refurbishment, and resupply techniques; including the checkout, testing and self-diagnosis of on-orbit systems, the replacement of fluids (including fuel) and change-out of payload modules (by EVA and robotic operations) with retrieval from and return to an operational orbit.

DESCRIPTION:

The main Satellite Servicing Test Article will be an independent vehicle that will serve as a carrier for a functioning payload (primary facility a Materials Processing in Space Facility, but not limited to this type of payload). Several precurser missions will be performed with smaller payloads that may be carried in the Shuttle.

The main Test Article will use a gaseous nitrogen propellant with a system start-up and shutdown capability. It will use a section of the Space Station type solar array and battery storage (NiH batteries) systems for power. It will use an enclosed Space Station Unpressurized Logistics Module for structural framework. It will be supported by the Space Station OMV for orbit placement and retrieval when necessary, otherwise it will be capable of orbit maintenance.

TYPE NUMBER: 15

IMPORTANCE OF SPACE STATION: 8

ON-SERVICING OMV FLIGHTS (per year): 1 - 2

ADD RESOURCES: 1

RESOURCE REFERENCE: ______________________________

MRDB Form 1 (concluded)

ORBIT

MISSION CODE: TDMX2561, 2562, 2563 AND 2063 Combined

ORBIT: 2 (If 1 is selected, skip remainder of Form 2)

APOGEE:	400KM	+	10	KM	TOLERANCE
		-	10	KM	
PERIGEE:	400KM	+	10	KM	TOLERANCE
		-	10	KM	
INCLINATION:	28.5 DEGREE	+	____	DEG	TOLERANCE
		-	____	DEG	

LOCAL TIME OF EQUATOR CROSSING NODE: _____ HR _____ MIN
ASCENDING OR DESCENDING: _____

SPECIAL CONSIDERATIONS (ORBIT):

10km co-orbit with the Space Station

MRBD Form 2

POINTING/ORIENTATION

MISSION CODE: TDMX2561, 2562, 2563, and 2063 Combined

POINTING/ORIENTATION: TBD (If 1 is selected, skip remainder of Form 3)

VIEW DIRECTION: TBD
If 4 selected, OTHER_______________

HOURS: _____

TRUTH SITES:

POINTING ACCURACY: _______________arc sec

POINTING KNOWLEDGE: _______________arc sec

FIELD OF VIEW: _______________deg

POINTING STABILITY RATE: _____-____arc sec/sec

POINTING STABILITY: _______________arc sec

SPECIAL CONSIDERATIONS:

Pointing capabilities are dependent on the type of mission conducted.

MRBD Form 3

POWER

MISSION CODE: TDMX 2561, 2562, 2563 and 2063 Combined

POWER: 2

	AC	DC
OPERATING (KW):	____	1.5
HOURS, PER DAY (OPERATING)	____	24
VOLTAGE:	____	28
FREQUENCY:	____	
PEAK (KW):	____	1.75
HOURS PER DAY (PEAK)	____	0.1
STANDBY PWER (KW)	____	0

SPECIAL CONSIDERATIONS:

This is a self-contained test article acting in a Space Station co-orbit.

MRBD Form 4

THERMAL

MISSION CODE: TDMX2561, 2562, 2563, and 2063 Combined

THERMAL: 3

	ACTIVE OPER.	ACTIVE NON-OP	PASSIVE OPER.	PASSIVE NON-OP
MIN TEMP (C)	TBD	______	TBD	______
MAX TEMP (C)	TBD	______	TBD	______
MIN HEAT REJECTION (KW)	______	______	______	______
MAX HEAT REJECTION (KW)	______	______	______	______

SPECIAL CONSIDERATIONS:

Limits of the system will depend on the type of mission conducted, and is not dependent on the Space Station.

MRDB Form 5

DATA/COMMUNICATION

MISSION CODE: TDMX 2561, 2562, 2563, and 2063 Combined

ON-BOARD DATA PROCESSING REQUIRED: 1

If 1 (YES), this description:

Health status and start-up/shutdown commands

ON-BOARD STORAGE (MBIT): TBD

STATION DATA REQUIRED:

COMMUNICATION LINKS:

1.	From: Station To: Ground	Digital Data	Video Data	Voice
a.	Generation rate (kbps)			NA
b.	Duration (hours)			
c.	Frequency (per day)			0
d.	Delivery Time (hours)			
e.	Security (yes/no)			
f.	Reliability (%)			Yes
g.	Interactive (yes/no)			

2.	From: Ground To: Station	Digital Data	Video Data	Voice
a.	Generation rate (kbps)			NA
b.	Duration (hours)			
c.	Frequency (per day)			0
d.	Delivery Time (hours)			
e.	Security (yes/no)			
f.	Reliability (%)			Yes
g.	Interactive (yes/no)			

MRDB Form 6

3.	From: Station To: Free Flyer	Digital Data	Video Data	Voice
a.	Generation rate (kbps)	10		NA
b.	Duration (hours)	TBD		
c.	Frequency (per day)	2		
d.	Delivery Time (hours)	TBD		0
e.	Security (yes/no)	0		
f.	Reliability (%)	95		
g.	Interactive (yes/no)	YES		Yes

4.	From: Free Flyer To: Station	Digital Data	Video Data	Voice
a.	Generation rate (kbps)	10		NA
b.	Duration (hours)	TBD		
c.	Frequency (per day)	2		
d.	Delivery Time (hours)	TBD		0
e.	Security (yes/no)	0		
f.	Reliability (%)	95		
g.	Interactive (yes/no)	YES		Yes

5.	From: Station To: Platform	Digital Data	Video Data	Voice
a.	Generation rate (kbps)			NA
b.	Duration (hours)			
c.	Frequency (per day)			
d.	Delivery Time (hours)			0
e.	Security (yes/no)			
f.	Reliability (%)			
g.	Interactive (yes/no)			Yes

6.	From: Platform To: Station	Digital Data	Video Data	Voice
a.	Generation rate (kbps)			NA
b.	Duration (hours)			
c.	Frequency (per day)			
d.	Delivery Time (hours)			0
e.	Security (yes/no)			
f.	Reliability (%)			
g.	Interactive (yes/no)			Yes

MRDB Form 6 (cont'd)

7. From: Platform
To: Ground

		Digital Data	Video Data	Voice
a.	Generation rate (kbps)	10		NA
b.	Duration (hours)	TBD		
c.	Frequency (per day)	2		
d.	Delivery Time (hours)	TBD		0
e.	Security (yes/no)	0		
f.	Reliability (%)	95		
g.	Interactive (yes/no)	YES		Yes

8. From: Ground
To: Platform

		Digital Data	Video Data	Voice
a.	Generation rate (kbps)	10		NA
b.	Duration (hours)	TBD		
c.	Frequency (per day)	2		
d.	Delivery Time (hours)	TBD		0
e.	Security (yes/no)	0		
f.	Reliability (%)	95		
g.	Interactive (yes/no)	YES		Yes

9. From: Station
To: Shuttle

		Digital Data	Video Data	Voice
a.	Generation rate (kbps)			NA
b.	Duration (hours)			
c.	Frequency (per day)			
d.	Delivery Time (hours)			0
e.	Security (yes/no)			
f.	Reliability (%)			
g.	Interactive (yes/no)			Yes

10. From: Shuttle
To: Station

		Digital Data	Video Data	Voice
a.	Generation rate (kbps)			NA
b.	Duration (hours)			
c.	Frequency (per day)			
d.	Delivery Time (hours)			0
e.	Security (yes/no)			
f.	Reliability (%)			
g.	Interactive (yes/no)			Yes

COMMENTS:

MRDB Form 6 (concluded)

EQUIPMENT

MISSION CODE: TDMX2561, 2562, 2563, AND 2063 COMBINED

MODULE CODE: 1

SHARED FACILITY CODE: 0

MISSIONS: ______
______
______
______

EQUIPMENT LOCATIONS: ______ (END OF MAIN POWER BOOM ______)

DIMENSIONS (M)
- Length 2.34
- Width or diam 4.48
- Height (or blank) ______

VOLUME (M^3) 36.89

PKG DIMENSION (M)
- Length ______
- Width or diam ______
- Height (or blank) ______

PKG VOLUME (M^3) ______

LAUNCH MASS (KG) ______

ACCELERATION MAX (G) 4.0

ATTACH POINTS: ______

SET-UP CODE: 1 2 3

HARDWARE DESCRIPTION:

______
______
______
______

MRDB Form 7

CREW

MISSION CODE: TDMX 2561, 2562, 2563, AND 2063 COMBINED

INITIAL CONSTRUCTION/SET-UP: 1

TASK: Assemble, construct, and checkout on-orbit

PERIOD: 3 Days

IVA TOTAL CREW TIME: 16 Man Hours

EVA PRODUCTIVE DREW TIME: 16 Man Hours

SKILLS:

SKILL TYPE

SKILL LEVEL	1	2	3	4	5	6	7
1							
2	1						
3							

DAILY OPERATATIONS: 0

TASK: ______

IVA CREW TIME PER DAY: ______Man Hours

SKILLS:

SKILL TYPE

SKILL LEVEL	1	2	3	4	5	6	7
1							
2							
3							

MRDB Form 8

SERVICING

MISSION CODE: TDMX 2561, 2562, 2563, AND 2063 COMBINED

SERVICING: ______ (If 1 is selected,skip remainder of Form 9)

SERVICE INTERVAL (DAYS): ______

CONSUMABLES:
TYPES: ______

WEIGHTS: ______KG

RETURN: ______KG

VOLUME UP: ______M^3

VOLUME DOWN: ______M^3

POWER: ______KW

HOURS FOR POWER: ______HRS

EVA HOURS PER SERVICE: ______HRS

TYPICAL TASKS (EVA): ______

IVA HOURS PER SERVICE: ______HRS

LOCATION OF SERVICING: ______

TYPICAL TASKS (IVA): ______

SPECIAL CONSIDERATIONS:

MRDB Form 9

CONFIGURATION CHANGES

MISSION CODE: TDMX 2561, 2562, 2563, AND 2063 COMBINED

CONFIGURATION CHANGES: 2 (If 1 is selected, skip the remainder of Form 10)

INTERVAL (DAYS): 10 - 60

CHANGE-OUT EQUIPMENT:

TYPE: OMV retrieval, reboost, and replacement or modules

WEIGHT: _______KG

RETURN: _______KG

VOLUME UP: _______M^3

VOLUME DOWN: _______M^3

POWER: _______KW

HOURS FOR POWER: _______HRS

EVA HOURS PER CHANGE: _______HRS

TYPICAL TASKS (EVA): ____________________

____

IVA HOURS PER CHANGE: _______HRS

LOCATION: _______

TYPICAL TASKS (IVA): ____________________

____

SPECIAL CONSIDERATIONS:

____

____

____

____

MRDB Form 10

MISSION CODE: TDMX 2561, 2562, 2563, AND 2063 COMBINED

SCIENTIFIC AIRLOCK:

TETHER:

VACUUM VENTING:

OTHER:

MRDB Form 11 (concluded)

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MISSION DESIGN

MISSION CODE: TDMX 2561, 2562, 2563 and 2063 Combined

PAYLOAD ELEMENT NAME: **Satellite Servicing**

COUNTRY: USA NASA OAST (TDMX)

CONTACT: George McKay
PM01
Marshall Space Flight Center
MSFC, AL 35812

PHONE: 205-544-1773

STATUS: Candidate

FLIGHTS:	92	93	94	95	96	97	98	99	00	01
Equipment Up (flights)	0	0	0	0	1	0	0	0	0	0
Equip Down (# of times)	0	0	0	0	0	0	0	0	0	0
Operational Days/Flight	0	0	0	0	10	60	60	60	60	60
OTV Flights	0	0	0	0	1	3	2	2	2	2

EARLY FLIGHTS ____
LATE RETURN ____

OBJECTIVE:

To demonstrate satellite construction, maintenance, repair, refurbishment and resupply techniques; including the checkout, testing and self-diagnosis of on-orbit systems, the replacement of fluids (including fuel) and change-out of payload modules (by EVA and robotic operations) with retrieval from and return to an operational orbit.

DESCRIPTION:

The main Satellite Servicing Test Article will be an independent vehicle that will serve as a carrier for a functioning payload (primary facility Materials Processing in Space Facility, but not limited to this type of payload). Several precurser missions will be performed with small payloads that may be carried in the Shuttle.

The main Test Article will use a gaseous nitrogen propellant with a system start-up and shutdown capability. It will use a section of the Space Station type solar array and battery storage (NiH batteries) systems for power. It will use an enclosed Space Station Unpressurized Logistics Module for structural framework. It will be supported by the Space Station OMV for orbit placement and retrieval when necessary, otherwise it will be capable of orbit maintenance.

TYPE NUMBER: 15

IMPORTANCE OF SPACE STATION: 8

NON-SERVICING OMV FLIGHTS (per year): 1 - 2

ADD RESOURCES: 1

RESOURCE REFERENCE: ________________________

MISSION DESIGN

MISSION CODE: NEW COMBINED MISSION

TDMX2561 TDMX 2562, TDMX 2563 TDMX 2564

PAYLOAD ELEMENT NAME:

SATELLITE SERVICING

COUNTRY:

USA NASA OAST TDMX

CONTACT:

GEORGE MCKAY
PM01
MARSHALL SPACE FLIGHT CENTER
MSFC AL 35812

PHONE:

205-544-1773

STATUS: CANDIDATE

FLIGHTS:

	1992	1993	1994	1995	1996	1997	1998	1999	2000	2001
EQUIPMENT UP (flights)					1					
EQUIPMENT DOWN (no. of times)										
OPERATIONAL DAYS (per flight)					10	60	55	60	[illegible]	0
OTV FLIGHTS					1	3	2	2	2	2

EARLY FLIGHT: __________

LATE RETURN: __________

OBJECTIVE:

LINE

1 TO DEMONSTRATE SATELLITE CONSTRUCTION,
MAINTENANCE, REPAIR, REFURBISH-

2 MENT AND RESUPPLY TECHNIQUES; INCLUDING
THE CHECKOUT OF ON ORBIT SYSTEMS

3 , REFUELING AND FLUIDS REPLACEMENT, PAY-
LOAD MODULE CHANGEOUT, AND

4 RETRIVAL FROM AND RETURN TO AN
OPERATIONAL ORBIT

5

6

7

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DESCRIPTION:

LINE

1

2

3

4

5

6

7

8

9

10

11

12

13

14

TYPE NUMBER: 15

IMPORTANCE OF SPACE STATION: 1

NON-SERVICING OMV FLIGHTS (per year): 1-2

ADD RESOURCES: 1

RESOURCE REFERENCE: | | | | | | | | |

ORBIT

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MISSION CODE: TDMX

ORBIT: 2 (If 1 is selected, skip remainder of Form 2)

APOGEE: 400 km + 10 km, - 10 km TOLERANCE

PERIGEE: 400 km + 10 km, - 10 km TOLERANCE

INCLINATION: 28.5 deg + ____ deg, - ____ deg TOLERANCE

LOCAL TIME OF EQUATOR CROSSING NODE: ____ hr ____ min

ASCENDING OR DESCENDING: ____

SPECIAL CONSIDERATIONS (ORBIT):

LINE	
1	10 KM CO-ORBIT WITH THE SPACE STA-[illegible]
2	
3	
4	

MISSION CODE: [ ]

POINTING/ORIENTATION: TBD (If 1 is selected, skip remainder of FORM 3)

VIEW DIRECTION: TBD

If 4 selected, OTHER: [ ]

HOURS: ______

TRUTH SITES:

[ ]

POINTING ACCURACY: ______ sec

POINTING KNOWLEDGE: ______ sec

FIELD OF VIEW: ______ deg

POINTING STABILITY RATE: ______ sec per sec

POINTING STABILITY: ______ sec

PLACEMENT: ______ sec

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SPECIAL CONSIDERATIONS:

LINE	
1	POINTING CAPABILITIES ARE DEPENDE- ON THE TYPE OF MISSION -
2	CONDUCTED.
3	
4	

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POWER

MISSION CODE: TOMX

POWER: 2

	AC	DC
OPERATING (KW):		1.5
HOURS, PER DAY (OPERATING)		24
VOLTAGE:		28
FREQUENCY:		
PEAK (KW):		1.75
HOURS PER DAY (PEAK)		.1
STANDBY POWER (KW)		0

SPECIAL CONSIDERATIONS (POWER):

LINE

1 THIS IS A SELF-CONTAINED TEST ARTICLE ACTING IN A SPACE STATION

2 CO-ORBIT

3

4

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THERMAL

MISSION CODE:

THERMAL: 3

	ACTIVE		PASSIVE	
	OPERATIONAL	NON-OPERATIONAL	OPERATIONAL	NON-OPERATIO
MIN TEMP (°C)	TBD		TBD	
MAX TEMP (°C)	TBD		TBD	
MIN HEAT REJECTION (KW)				
MAX HEAT REJECTION (KW)				

SPECIAL CONSIDERATIONS:

LINE

1 LIMITS OF THE SYSTEM WILL DEPEND ON THE
TYPE OF MISSION CONDUCTED, AND

2 IS NOT DEPENDENT ON THE SPACE STATION

3

4

DATA/COMMUNICATIONS

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MISSION CODE: TDMX

ON-BOARD DATA PROCESSING REQUIRED: 1

If 1 (YES), this DESCRIPTION:

HELTH STATUS AND STARTUP/SHUTDOWN
COMANDS

ON-BOARD STORAGE (MBIT): TBD

STATION DATA REQUIRED:

LINE

1

2

COMMUNICATION LINKS:

1. From: Station
   To: Ground

	Digital Data	Video Data	Voice
a. Generation rate (kbps)			NA
b. Duration (hours)			
c. Frequency (per day)			
d. Delivery time (hours)			0
e. Security (yes/no)			
f. Reliability (%)			
g. Interactive (yes/no)			Yes

2. From: Ground
   To: Station

	Digital Data	Video Data	Voice
a. Generation rate (kbps)			NA
b. Duration (hours)			
c. Frequency (per day)			
d. Delivery time (hours)			0
e. Security (yes/no)			
f. Reliability (%)			
g. Interactive (yes/no)			Yes

3. From: Station
To: Free Flyer

	Digital Data	Video Data	Voice
a. Generation rate (kbps)	10		NA
b. Duration (hours)	TBD		
c. Frequency (per day)	2		
d. Delivery time (hours)	TBD		0
e. Security (yes/no)	0		
f. Reliability (%)	95		
g. Interactive (yes/no)	YES		Yes

4. From: Free Flyer
To: Station

	Digital Data	Video Data	Voice
a. Generation rate (kbps)	10		NA
b. Duration (hours)	TBD		
c. Frequency (per day)	2		
d. Delivery time (hours)	TBD		0
e. Security (yes/no)	0		
f. Reliability (%)	95		
g. Interactive (yes/no)	YES		Yes

5. From: Station
To: Platform

	Digital Data	Video Data	Voice
a. Generation rate (kbps)			NA
b. Duration (hours)			
c. Frequency (per day)			
d. Delivery time (hours)			0
e. Security (yes/no)			
f. Reliability (%)			
g. Interactive (yes/no)			Yes

6. From: Platform
To: Station

	Digital Data	Video Data	Voice
a. Generation rate (kbps)			NA
b. Duration (hours)			
c. Frequency (per day)			
d. Delivery time (hours)			0
e. Security (yes/no)			
f. Reliability (%)			
g. Interactive (yes/no)			Yes

7. From: Platform
To: Ground

	Digital Data	Video Data	Voice
a. Generation rate (kbps)	10		NA
b. Duration (hours)	TBD		
c. Frequency (per day)	2		
d. Delivery time (hours)	TBD		0
e. Security (yes/no)	0		
f. Reliability (%)	95		
g. Interactive (yes/no)	YES		Yes

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8. From: Ground
To: Platform

	Digital Data		Video Data		Voice
a. Generation rate (kbps)	10	:		:	NA
b. Duration (hours)	TBD	:		:	
c. Frequency (per day)	2	:		:	
d. Delivery time (hours)	TBD	:		:	0
e. Security (yes/no)	Ø	:		:	
f. Reliability (%)	95	:		:	
g. Interactive (yes/no)	YES	:		:	Yes

9. From: Station
To: Shuttle

	Digital Data		Video Data		Voice
a. Generation rate (kbps)		:		:	NA
b. Duration (hours)		:		:	
c. Frequency (per day)		:		:	
d. Delivery time (hours)		:		:	0
e. Security (yes/no)		:		:	
f. Reliability (%)		:		:	
g. Interactive (yes/no)		:		:	Yes

10. From: Shuttle
To: Station

	Digital Data		Video Data		Voice
a. Generation rate (kbps)		:		:	NA
b. Duration (hours)		:		:	
c. Frequency (per day)		:		:	
d. Delivery time (hours)		:		:	0
e. Security (yes/no)		:		:	
f. Reliability (%)		:		:	
g. Interactive (yes/no)		:		:	Yes

COMMENTS:

LINE

1

2

3

4

NASA Space Station Mission Data Base - Form 6 (concluded)

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MISSION CODE: TPMX

MODULE CODE: 1

SHARED FACILITY CODE: 0

(If 1 is selected, list mission codes of sharing missions below:)

EQUIPMENT LOCATION:

If equipment location is:	1	2	3	4	5	6	7
	INTERNAL PRESSURIZED	EXTERNAL/ATTACHED PRESSURIZED	UNPRESSURIZED	FREE FLYER (REMOTE)	FREE FLYER (CO-ORBITING)	28.5 DEGREE PLATFORM	SUN SYNC/POLAR PLATFORM
DIMENSIONS (M)							
Length					2.34		
Width or Dia.					4.48		
Height (or blank)							
VOLUME (M^3)					36.59		
PEG DIMENSION (M)							
Length							
Width or Dia.							
Height (or blank)							
PEG VOLUME (M^3)							
LAUNCH MASS (KG)							
ACCELERATION MAX (g)					4.0		

ATTACH POINTS: ______

SET UP CODE: 1 2 3

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NE

HARDWARE DESCRIPTION:

NASA Space Station Mission Data Base - Form 7 (concluded)

MISSION CODE: TDM1

INITIAL CONSTRUCTION/SET UP: 1 (If 0, skip to DAILY OPERATIONS)

TASK:

ASSEMBLE, CONSTRUCT, AND CHECKOUT ON-ORBIT

PERIOD: 3 days

IVA TOTAL CREW TIME: 16 man-hrs

EVA PRODUCTIVE CREW TIME: 16 man-hrs

SKILLS: (See last page of Form 8 for example)

Enter number of skill type/levels required:

SKILL TYPE

SKILL LEVEL	1	2	3	4	5	6	7
1							
2	1						
3							

DAILY OPERATIONS: ________ (If 0, skip to PERIODIC OPERATIONS)

TASK:

IVA CREW TIME PER DAY: ________ man-hrs

SKILLS:

Enter number of skill type/levels required:

SKILL TYPE

SKILL LEVEL	1	2	3	4	5	6	7
1							
2							
3							

PERIODIC OPERATIONS: 1 (If 0, skip to TEARDOWN AND STOW)

TASK:

MONITORING OPERATIONS

IVA OCCURRENCE INTERVAL: 0.1 days

CREW TIME/OCCURRENCE: 2 man-hrs

EVA OCCURRENCE INTERVAL: ________ days

EVA PRODUCTIVE CREW TIME/OCCURRENCES: ________ man-hrs

SKILLS:

Enter number of skill type/levels required:

ORIGINAL PAGE IS OF POOR QUALITY

SKILL TYPE

SKILL LEVEL	1	2	3	4	5	6	7
1							
2	1						
3							

TEARDOWN AND STOW: ________ (If 0, skip this section)

TASK:

PERIOD: ________ days

IVA TOTAL CREW TIME: ________ man-hrs

EVA PRODUCTIVE CREW TIME: ________ man-hrs

SKILLS:

Enter number of skill type/levels required:

SKILL TYPE

SKILL LEVEL	1	2	3	4	5	6	7
1							
2							
3							

COMMENTS:

LINE

1

2

3

4

Typical example of skill type/level matrix input:

Skill Types

1. No Special Skill Required
2. Medical/Biological
3. Physical Sciences
4. Earth and Ocean Sciences
5. Engineering
6. Astronomy
7. Spacecraft Systems

Skill Levels

1. Task Trainable
2. Technician
3. Professional

If two medical/biological professionals are required, put 2 in second column, third row. No more than 6 skill types can be used for a given task.

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NASA Space Station Mission Data Base - Form 8 (concluded)

MISSION CODE: TOMX

SERVICING: ________ (If 1 is selected, skip remainder of Form 9)

SERVICE INTERVAL (days): ________

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CONSUMABLES

TYPE:

WEIGHT: ________ kg

RETURN: ________ kg

VOLUME UP: ________ m^3

VOLUME DOWN: ________ m^3

POWER: ________ kw

HOURS FOR POWER: ________ hrs

EVA HOURS PER SERVICE: ________ hrs

TYPICAL TASKS (EVA):

IVA HOURS PER SERVICE: ________ hrs

LOCATION OF SERVICING: ________

TYPICAL TASKS (IVA):

SPECIAL CONSIDERATIONS:

LINE

1

2

3

4

NASA Space Station Mission Data Base - Form 9

MISSION CODE: TDMX

CONFIGURATION CHANGES: 2 (If 1 selected, skip the remainder of Form 10)

INTERVAL: 10-60 days

CHANGE-OUT EQUIPMENT

TYPE:

OMV RETREVIAL, REBOOST, AND REPLACEMENT OF MODULES

WEIGHT: ______ kg

RETURN: ______ kg

VOLUME UP: ______ m^3

VOLUME DOWN: ______ m^3

POWER: ______ kw

HOURS FOR POWER: ______ hrs

EVA HOURS PER CHANGE: ______ hrs

TYPICAL TASKS (EVA):

IVA HOURS PER CHANGE: ______ hrs

LOCATION: ______

TYPICAL TASKS (IVA):

SPECIAL CONSIDERATIONS:

LINE

1

2

3

4

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OF POOR QUALITY

SPECIAL NOTES

MISSION CODE:

LINE — CONTAMINATION:

1

2

LINE — STRUCTURES:

1

2

LINE — MATERIALS:

1

2

LINE — RADIATION:

1

2

LINE SAFETY:

1.

2.

LINE STORAGE:

1.

2.

LINE OPTICAL WINDOW:

1

2

LINE SCIENTIFIC AIRLOCK:

1

2

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NASA Space Station Mission Data Base - Form 11 (continued)

LINE TETHER:

1

2

LINE VACUUM VENTING:

1

2

LINE OTHER:

1

2

3

4

NASA Space Station Mission Data Base - Form 11 (concluded)

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MISSION DESIGN

MISSION CODE: TDMX 2564

PAYLOAD ELEMENT NAME: Coatings Maintenance

COUNTRY: USA NASA OAST (TDMX)

CONTACT: George McKay
PMO1
Marshall Space Flight Center
MSFC, AL 35812
PHONE: 205-544-1773

STATUS: CANDIDATE

FLIGHTS:

	92	93	94	95	96	97	98	99	00	01
Equipment Up (flights)	0	0	0	0	0	1	0	0	0	0
Equip Down (# of times)	0	0	0	0	0	0	1	0	0	0
Operational Days/Flight	0	0	0	0	0	20	100	0	0	0
OTV Flights	0	0	0	0	0	0	0	0	0	0

EARLY FLIGHTS ____
LATE RETURN ____

OBJECTIVE:

To evaluate the deterioration of several types of surfaces exposed to the environment outside the Space Station and to clean, resurface and recoat these exposed surfaces using active beam technology which includes ion beam, plasma beam, and laser beam applications.

DESCRIPTION:

Using commercially available lasers of 1.06 x 10-6 or 0.53 x 10-6 meter wavelengths (selected for solid state epitaxy capability with silicon and gallium arsenide) and the Active Cleaning Technology ion and plasma beam equipment; ground research the cleaning, resurfacing (including refurbishment of solar cells) and recoating capacity of these systems on various surfaces and develop the ability to measure the deterioration and recoat thickness of the surfaces remotely space qualifying the parts as required. An evaluation of possible EMI and particluate contamination fields will be made for each operation.

TYPE NUMBER: 17

IMPORTANCE OF SPACE STATION: 10

NON-SERVICING OMV FLIGHTS (per year): 0

ADD RESOURCES: 1

RESOURCE REFERENCE: ______________________

MRDB Form 1 (concluded)

ORBIT

MISSION CODE: TDMX 2564

ORBIT: 1 (If 1 is selected, skip remainder of Form 2)

APOGEE:	_____ KM	+	_____ KM	TOLERANCE
		-	_____ KM	
PERIGEE:	_____ KM	+	_____ KM	TOLERANCE
		-	_____ KM	
INCLINATION:	_____ DEGREE	+	_____ DEG	TOLERANCE
		-	_____ DEG	

LOCAL TIME OF EQUATOR CROSSING NODE: _____ HR _____ MIN
ASCENDING OR DESCENDING: _____

SPECIAL CONSIDERATIONS (ORBIT):

____
____
____
____

POINTING/ORIENTATION

MISSION CODE: TDMX 2564

POINTING/ORIENTATION: 1 (If 1 is selected, skip remainder of Form 3)

VIEW DIRECTION: ___
If 4 selected, OTHER_______________

HOURS: _____

TRUTH SITES:

POINTING ACCURACY: _______________arc sec

POINTING KNOWLEDGE: _______________arc sec

FIELD OF VIEW: _______________deg

POINTING STABILITY RATE: _____-____arc sec/sec

POINTING STABILITY: _______________arc sec

SPECIAL CONSIDERATIONS:

Pointing capabilities are dependent on the type of mission conducted.

POWER

MISSION CODE: TDMX 2564

POWER: TBD

	AC	DC
OPERATING (KW):	_____	_____
HOURS, PER DAY (OPERATING)	_____	_____
VOLTAGE:	_____	_____
FREQUENCY:	_____	
PEAK (KW):	_____	_____
HOURS PER DAY (PEAK)	_____	_____
STANDBY PWER (KW)	_____	_____

SPECIAL CONSIDERATIONS:

THERMAL

MISSION CODE: TDMX 2564

THERMAL: N/A

	ACTIVE		PASSIVE	
	OPER.	NON-OP	OPER.	NON-OP
MIN TEMP (C)				
MAX TEMP (C)				
MIN HEAT REJECTION (KW)				
MAX HEAT REJECTION (KW)				

SPECIAL CONSIDERATIONS:

DATA/COMMUNICATION

MISSION CODE: TDMX 2564

ON-BOARD DATA PROCESSING REQUIRED: 1

If 1 (YES), this description:

On-board data acquisition gathered remotely from the platform.

ON-BOARD STORAGE (MBIT): 3

STATION DATA REQUIRED:

COMMUNICATION LINKS:

	1. From: Station To: Ground	Digital Data	Video Data	Voice
a.	Duration rate (kbps)			NA
b.	Duration (hours)			
c.	Frequency (per day)			
d.	Delivery Time (hours)			0
e.	Security (yes/no)			
f.	Reliability (%)			
g.	Interactive (yes/no)			Yes

	2. From: Ground To: Station	Digital Data	Video Data	Voice
a.	Generation rate (kbps)			NA
b.	Duration (hours)			
c.	Frequency (per day)			
d.	Delivery Time (hours)			0
e.	Security (yes/no)			
f.	Reliability (%)			
g.	Interactive (yes/no)			Yes

	3. From: Station To: Free Flyer	Digital Data	Video Data	Voice
a.	Generation rate (kbps)	10		NA
b.	Duration (hours)	TBD		
c.	Frequency (per day)	2		
d.	Delivery Time (hours)	TBD		0
e.	Security (yes/no)	0		
f.	Reliability (%)	95		
g.	Interactive (yes/no)	YES		Yes

4.	From: Free Flyer To: Station	Digital Data	Video Data	Voice
a.	Generation rate (kbps)	10		NA
b.	Duration (hours)	TBD		
c.	Frequency (per day)	2		
d.	Delivery Time (hours)	TBD		0
e.	Security (yes/no)	0		
f.	Reliability (%)	95		
g.	Interactive (yes/no)	YES		Yes

5.	From: Station To: Platform	Digital Data	Video Data	Voice
a.	Generation rate (kbps)			NA
b.	Duration (hours)			
c.	Frequency (per day)			
d.	Delivery Time (hours)			0
e.	Security (yes/no)			
f.	Reliability (%)			
g.	Interactive (yes/no)			Yes

6.	From: Platform To: Station	Digital Data	Video Data	Voice
a.	Generation rate (kbps)	10		NA
b.	Duration (hours)	.25		
c.	Frequency (per day)	2-4		
d.	Delivery Time (hours)	TBD		0
e.	Security (yes/no)	0		
f.	Reliability (%)	95		
g.	Interactive (yes/no)	YES		Yes

7.	From: Platform To: Ground	Digital Data	Video Data	Voice
a.	Generation rate (kbps)	10		NA
b.	Duration (hours)	.25		
c.	Frequency (per day)	2-4		
d.	Delivery Time (hours)	TBD		0
e.	Security (yes/no)	0		
f.	Reliability (%)	95		
g.	Interactive (yes/no)	YES		Yes

8.	From: Ground To: Platform	Digital Data	Video Data	Voice
a.	Generation rate (kbps)	10		NA
b.	Duration (hours)	TBD		
c.	Frequency (per day)	2		
d.	Delivery Time (hours)	TBD		0
e.	Security (yes/no)	0		
f.	Reliability (%)	95		
g.	Interactive (yes/no)	YES		Yes

MRDB Form 6 (cont'd)

9.	From: Station / To: Shuttle	Digital Data	Video Data	Voice
a.	Generation rate (kbps)			NA
b.	Duration (hours)			
c.	Frequency (per day)			
d.	Delivery Time (hours)			0
e.	Security (yes/no)			
f.	Reliability (%)			
g.	Interactive (yes/no)			Yes

10.	From: Shuttle / To: Station	Digital Data	Video Data	Voice
a.	Generation rate (kbps)			NA
b.	Duration (hours)			
c.	Frequency (per day)			
d.	Delivery Time (hours)			0
e.	Security (yes/no)			
f.	Reliability (%)			
g.	Interactive (yes/no)			Yes

COMMENTS:

EQUIPMENT

MISSION CODE: TDMX 2564

MODULE CODE: 1

SHARED FACILITY CODE: 0

MISSIONS: ____________
____
____
____

EQUIPMENT LOCATIONS: _____ (END OF MAIN POWER BOOM _____)

DIMENSIONS (M)	
Length	4.3
Width or diam	4.3
Height (or blank)	2.4
VOLUME (M^3)	44.4
PKG DIMENSION (M)	
Length	4.3
Width or diam	4.3
Height (or blank)	2.4
PKG VOLUME (M^3)	44.4
LAUNCH MASS (KG)	TBD
ACCELERATION MAX (G)	4.0

ATTACH POINTS: Space Station standard platform

SET-UP CODE: 2 _____ _____

HARDWARE DESCRIPTION:
This experiment set will be conducted on the standard Space Station platform with power and data supplied and with the possible support of the thermal load.

CREW

MISSION CODE: TDMX 2564

INITIAL CONSTRUCTION/SET-UP: 1

TASK: Installation of the pallet and initial check-out

PERIOD: 1 Days

IVA TOTAL CREW TIME: 10 Man Hours

EVA PRODUCTIVE DREW TIME: 6 Man Hours

SKILLS:

SKILL TYPE

SKILL LEVEL	1	2	3	4	5	6	7
1							
2							
3	3						

DAILY OPERATATIONS: 1

TASK: Data transfer and experiment direction

IVA CREW TIME PER DAY: 1-2 Man Hours

SKILLS:

SKILL TYPE

SKILL LEVEL	1	2	3	4	5	6	7
1							
2	1						
3							

PERIODIC OPERATIONS: 0 (If 0, skip to TEARDOWN and STOW)

TASK: ______

IVA OCCURRENCE INTERVAL: ______ days

CREW TIME/OCCURRENCE: ______ man-hours

EVA OCCURRENCE INTERVAL: ______ days

EVA PRODUCTIVE CREW TIME/OCCURRENCES: ______ man-hours

SKILLS:

SKILL LEVEL	SKILL TYPE 1	2	3	4	5	6	7
1							
2							
3							

TEARDOWN AND STOW: 1 (If 0, skip this section)

TASK: Removal and storage of pallet samples. Eventual system readiness for return to Earth.

PERIOD: 1 day

IVA TOTAL CREW TIME: 8 man-hours

EVA PRODUCTIVE CREW TIME: 6 man-hours

SKILLS:

SKILL LEVEL	SKILL TYPE 1	2	3	4	5	6	7
1							
2							
3	3						

SERVICING

MISSION CODE: TDMX 2564

SERVICING: 1 (If 1 is selected, skip remainder of Form 9)

SERVICE INTERVAL (DAYS): 10-60 days

CONSUMABLES:

TYPES: Exchange of sample material for storage

WEIGHTS: _______KG

RETURN: _______KG

VOLUME UP: _______M^3

VOLUME DOWN: _______M^3

POWER: _______KW

HOURS FOR POWER: _______HRS

EVA HOURS PER SERVICE: TBD HRS

TYPICAL TASKS (EVA): ______________________________

IVA HOURS PER SERVICE: TBD HRS

LOCATION OF SERVICING: PALLET

TYPICAL TASKS (IVA): Sample removal and replacement

SPECIAL CONSIDERATIONS:

CONFIGURATION CHANGES

MISSION CODE: TDMX 2564

CONFIGURATION CHANGES: 1 (If 1 is selected, skip the remainder of Form 10)

INTERVAL (DAYS): ________

CHANGE-OUT EQUIPMENT:

TYPE: OMV retrieval, reboost, and replacement or modules

WEIGHT: ______KG

RETURN: ______KG

VOLUME UP: ______M^3

VOLUME DOWN: ______M^3

POWER: ______KW

HOURS FOR POWER: ______HRS

EVA HOURS PER CHANGE: ______HRS

TYPICAL TASKS (EVA): ____

IVA HOURS PER CHANGE: ______HRS

LOCATION: ________

TYPICAL TASKS (IVA): ____

SPECIAL CONSIDERATIONS:

SPECIAL NOTES

MISSION CODE: TDMX 2564

CONTAMINATION:
This system may produce particulate contamination that may affect other experiments and operations.

STRUCTURES:

MATERIALS:

RADIATION:
This system may produce radio frequency emissions that may require shielding of the experiment, it's parts or nearby operations.

SAFETY:
See RADIATION section

STORAGE:

OPTICAL WINDOW:

MISSION CODE: TDMX 2564

SCIENTIFIC AIRLOCK:

TETHER:

VACUUM VENTING:

OTHER:

MISSION DESIGN

MISSION CODE: TDMX 2565

PAYLOAD ELEMENT NAME: **Thermal Interface Technology**

COUNTRY: USA NASA OAST (TDMX) ______________________

CONTACT: George McKay ______________________
PMO1 ______________________
Marshall Space Flight Center ______________________
MSFC, AL 35812 ______________________

PHONE: 205-544-1773 ______________________

STATUS: Candidate

FLIGHTS:	92	93	94	95	96	97	98	99	00	01
Equipment Up (flights)	2	0	0	0	0	0	0	0	0	0
Equip Down (# of times)	2	0	0	0	0	0	0	0	0	0
Operational Days/Flight	0	0	0	0	0	0	0	0	0	0
OTV Flights	0	0	0	0	0	0	0	0	0	0

*Note: Shuttle Flights

EARLY FLIGHTS ____
LATE RETURN ____

OBJECTIVE:

To remove and replace electrical components and thermal-fill material from an on-orbit cold plate and verify the proper thermal conductivity.

DESCRIPTION:

The main testing for this TDMX will involve ground testing of thermal grease, thin metal foils, "Comerics", anodizing electrical components or abrasion contact with the cold plate, but need not be limited to these methods. Ground testing will be conducted in vacuum chambers, neutral buoyancy tanks and, possibly, man-rated vacuum chambers with manipulation tools and robotics used where possible. EVA experiments will be conducted on Space Shuttle flights.

TYPE NUMBER: 17

IMPORTANCE OF SPACE STATION: 1

NON-SERVICING OMV FLIGHTS (per year): 0

ADD RESOURCES: 1

RESOURCE REFERENCE: ____________________________

MRDB Form 1 (concluded)

ORBIT

MISSION CODE: TDMX 2565

ORBIT: 1 (If 1 is selected, skip remainder of Form 2)

APOGEE:	_____ KM	+	_____	KM	TOLERANCE
		-	_____	KM	
PERIGEE:	_____ KM	+	_____	KM	TOLERANCE
		-	_____	KM	
INCLINATION:	_____ DEGREE	+	_____	DEG	TOLERANCE
		-	_____	DEG	

LOCAL TIME OF EQUATOR CROSSING NODE: ______ HR ______ MIN
ASCENDING OR DESCENDING: ______

SPECIAL CONSIDERATIONS (ORBIT):

____
____
____
____

POINTING/ORIENTATION

MISSION CODE: TDMX 2565

POINTING/ORIENTATION: 1 (If 1 is selected, skip remainder of Form 3)

VIEW DIRECTION: ______

If 4 selected, OTHER ______________

HOURS: _____

TRUTH SITES:

POINTING ACCURACY: ______________ arc sec

POINTING KNOWLEDGE: ______________ arc sec

FIELD OF VIEW: ______________ deg

POINTING STABILITY RATE: _____-_____ arc sec/sec

POINTING STABILITY: ______________ arc sec

SPECIAL CONSIDERATIONS:

Pointing capabilities are dependent on the type of mission conducted.

POWER

MISSION CODE: TDMX 2565

POWER: N/A

	AC	DC
OPERATING (KW):	______	______
HOURS, PER DAY (OPERATING)	______	______
VOLTAGE:	______	______
FREQUENCY:	______	
PEAK (KW):	______	______
HOURS PER DAY (PEAK)	______	______
STANDBY PWER (KW)	______	______

SPECIAL CONSIDERATIONS:

THERMAL

MISSION CODE: TDMX 2565

THERMAL: N/A

	ACTIVE		PASSIVE	
	OPER.	NON-OP	OPER.	NON-OP
MIN TEMP (C)				
MAX TEMP (C)				
MIN HEAT REJECTION (KW)				
MAX HEAT REJECTION (KW)				

SPECIAL CONSIDERATIONS:

C 3

DATA/COMMUNICATION

MISSION CODE: TDMX 2565

ON-BOARD DATA PROCESSING REQUIRED: 1

If 1 (YES), this description: ____________________

ON-BOARD STORAGE (MBIT): 3

STATION DATA REQUIRED:

COMMUNICATION LINKS:

	1. From: Station To: Ground	Digital Data	Video Data	Voice
a.	Duration rate (kbps)			NA
b.	Duration (hours)			
c.	Frequency (per day)			
d.	Delivery Time (hours)			0
e.	Security (yes/no)			
f.	Reliability (%)			
g.	Interactive (yes/no)			Yes

	2. From: Ground To: Station	Digital Data	Video Data	Voice
a.	Generation rate (kbps)			NA
b.	Duration (hours)			
c.	Frequency (per day)			
d.	Delivery Time (hours)			0
e.	Security (yes/no)			
f.	Reliability (%)			
g.	Interactive (yes/no)			Yes

	3. From: Station To: Free Flyer	Digital Data	Video Data	Voice
a.	Generation rate (kbps)			NA
b.	Duration (hours)			
c.	Frequency (per day)			
d.	Delivery Time (hours)			0
e.	Security (yes/no)			
f.	Reliability (%)			
g.	Interactive (yes/no)			Yes

4.	From: Free Flyer To: Station		Digital Data	Video Data	Voice
a.	Generation rate	(kbps)	______	______	NA
b.	Duration	(hours)	______	______	______
c.	Frequency	(per day)	______	______	______
d.	Delivery Time	(hours)	______	______	0
e.	Security	(yes/no)	______	______	______
f.	Reliability	(%)	______	______	______
g.	Interactive	(yes/no)	______	______	Yes

5.	From: Station To: Platform		Digital Data	Video Data	Voice
a.	Generation rate	(kbps)	______	______	NA
b.	Duration	(hours)	______	______	______
c.	Frequency	(per day)	______	______	______
d.	Delivery Time	(hours)	______	______	0
e.	Security	(yes/no)	______	______	______
f.	Reliability	(%)	______	______	______
g.	Interactive	(yes/no)	______	______	Yes

6.	From: Platform To: Station		Digital Data	Video Data	Voice
a.	Generation rate	(kbps)	______	______	NA
b.	Duration	(hours)	______	______	______
c.	Frequency	(per day)	______	______	______
d.	Delivery Time	(hours)	______	______	0
e.	Security	(yes/no)	______	______	______
f.	Reliability	(%)	______	______	______
g.	Interactive	(yes/no)	______	______	Yes

7.	From: Platform To: Ground		Digital Data	Video Data	Voice
a.	Generation rate	(kbps)	______	______	NA
b.	Duration	(hours)	______	______	______
c.	Frequency	(per day)	______	______	______
d.	Delivery Time	(hours)	______	______	0
e.	Security	(yes/no)	______	______	______
f.	Reliability	(%)	______	______	______
g.	Interactive	(yes/no)	______	______	Yes

8.	From: Ground To: Platform		Digital Data	Video Data	Voice
a.	Generation rate	(kbps)	______	______	NA
b.	Duration	(hours)	______	______	______
c.	Frequency	(per day)	______	______	______
d.	Delivery Time	(hours)	______	______	0
e.	Security	(yes/no)	______	______	______
f.	Reliability	(%)	______	______	______
g.	Interactive	(yes/no)	______	______	Yes

MRDB Form 6 (cont'd)

9.	From: Station To: Shuttle	Digital Data	Video Data	Voice
a.	Generation rate (kbps)			NA
b.	Duration (hours)			
c.	Frequency (per day)			
d.	Delivery Time (hours)			0
e.	Security (yes/no)			
f.	Reliability (%)			
g.	Interactive (yes/no)			Yes

10.	From: Shuttle To: Station	Digital Data	Video Data	Voice
a.	Generation rate (kbps)			NA
b.	Duration (hours)			
c.	Frequency (per day)			
d.	Delivery Time (hours)			0
e.	Security (yes/no)			
f.	Reliability (%)			
g.	Interactive (yes/no)			Yes

COMMENTS:

MRDB Form 6 (concluded)

EQUIPMENT

MISSION CODE: TDMX 2565

MODULE CODE: N/A

SHARED FACILITY CODE: ______

MISSIONS: ____________
____
____
____

EQUIPMENT LOCATIONS: ______ (END OF MAIN POWER BOOM ______)

DIMENSIONS (M)
Length ______
Width or diam ______
Height (or blank) ______

VOLUME (M^3) ______

PKG DIMENSION (M)
Length ______
Width or diam ______
Height (or blank) ______

PKG VOLUME (M^3) ______

LAUNCH MASS (KG) ______

ACCELERATION MAX (G) ______

ATTACH POINTS: ________________________________

SET-UP CODE: ______ ______ ______

HARDWARE DESCRIPTION:

____
____
____
____

CREW

MISSION CODE: TDMX 2565

INITIAL CONSTRUCTION/SET-UP: ______

TASK: ______

PERIOD: ______Days

IVA TOTAL CREW TIME: ______ Man Hours

EVA PRODUCTIVE DREW TIME: ______ Man Hours

SKILLS:

SKILL TYPE

SKILL LEVEL	1	2	3	4	5	6	7
1							
2							
3							

DAILY OPERATATIONS: ______

TASK: ______

IVA CREW TIME PER DAY: ______ Man Hours

SKILLS:

SKILL TYPE

SKILL LEVEL	1	2	3	4	5	6	7
1							
2							
3							

PERIODIC OPERATIONS: _____ (If 0, skip to TEARDOWN and STOW)

TASK: ____

____

IVA OCCURRENCE INTERVAL: _____ days

CREW TIME/OCCURRENCE: _____ man-hours

EVA OCCURRENCE INTERVAL: _____ days

EVA PRODUCTIVE CREW TIME/OCCURRENCES: _____ man-hours

SKILLS:

SKILL TYPE

SKILL LEVEL	1	2	3	4	5	6	7
1							
2							
3							

TEARDOWN AND STOW: 1 (If 0, skip this section)

TASK: ____

____

PERIOD: _____ days

IVA TOTAL CREW TIME: _____ man-hours

EVA PRODUCTIVE CREW TIME: _____ man-hours

SKILLS:

SKILL TYPE

SKILL LEVEL	1	2	3	4	5	6	7
1							
2							
3							

SERVICING

MISSION CODE: TDMX 2565

SERVICING: N/A (If 1 is selected, skip remainder of Form 9)

SERVICE INTERVAL (DAYS): _____ days

CONSUMABLES:

TYPES: ______________________________

____

WEIGHTS: ______KG

RETURN: ______KG

VOLUME UP: ______M^3

VOLUME DOWN: ______M^3

POWER: ______KW

HOURS FOR POWER: ______HRS

EVA HOURS PER SERVICE: ______HRS

TYPICAL TASKS (EVA): ______________________________

____

IVA HOURS PER SERVICE: ______HRS

LOCATION OF SERVICING: ______________________________

TYPICAL TASKS (IVA): ______________________________

____

SPECIAL CONSIDERATIONS:

____

____

____

____

CONFIGURATION CHANGES

MISSION CODE: TDMX 2565

CONFIGURATION CHANGES: N/A (If 1 is selected, skip the remainder of Form 10)

INTERVAL (DAYS): ________

CHANGE-OUT EQUIPMENT:

TYPE: ____
____

WEIGHT: ______KG

RETURN: ______KG

VOLUME UP: ______M^3

VOLUME DOWN: ______M^3

POWER: ______KW

HOURS FOR POWER: ______HRS

EVA HOURS PER CHANGE: ______HRS

TYPICAL TASKS (EVA): ____
____

IVA HOURS PER CHANGE: ______HRS

LOCATION: ________

TYPICAL TASKS (IVA): ____
____

SPECIAL CONSIDERATIONS:

____
____
____
____

SPECIAL NOTES

MISSION CODE: TDMX 2565

CONTAMINATION:

STRUCTURES:

MATERIALS:

RADIATION:

SAFETY:

STORAGE:

OPTICAL WINDOW:

MISSION CODE: TDMX 2565

SCIENTIFIC AIRLOCK:

TETHER:

VACUUM VENTING:

OTHER:

MRDB Form 11 (concluded)